GeoSAR Feasibility Study

Summary of the Group Design Project
MSc in Astronautics and Space Engineering 2012–13
Cranfield University

College of Aeronautics Report SP002

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Abstract

Students of the MSc course in Astronautics and Space Engineering 2012–13 at Cranfield University performed a feasibility study of a geosynchronous radar mission for their group project. 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.

The mission studied is a Ku-band monostatic SAR in a small inclination (80 km relative orbit diameter) geosynchronous orbit. The total launch mass is 2 tonnes, and images with resolution 40 m to monitor land subsidence in urban areas are produced. The mission appears to be feasible within the scope of an ESA Earth Explorer proposal.

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
9603	1996	Linear mixture modelling solution methods for satellite remote sensing
9903	1999	Mars Xpress, summary of the group design project, MSc in Astronautics and Space Engineering, 1997/98
0019	2000	CUSTARD, A microsystem technology demonstrator nanosatellite. Summary of the group design project MSc in Astronautics and Space Engineering 1999-2000
9918	2000	ORWELL Demonstrator, Summary of the group design project, MSc in Astronautics and Space Engineering 1998/99
0205	2003	Lunar South pole mission, Summary of the group design project, MSc in Astronautics and Space Engineering 1996/97
0206	2003	Mustang 2001, Summary of the group design project, MSc in Astronautics and Space Engineering 2001/02
0502	2005	Mustang 0, A low-cost technology demonstration nanosatellite; summary of the group design project, MSc in Astronautics and Space Engineering 2004/05
0509	2006	GeoSAR: summary of the group design project, MSc in Astronautics and Space Engineering 2005/06
0703	2007	PRIMA, Precursor Rendezvous for Impact Mitigation of Asteroids. Summary of the Group Design Project MSc in Astronautics and Space Engineering 2006/07
1001	2010	Debris Removal from Low Earth Orbit (DR LEO), Summary of the Group Design Project MSc in Astronautics and Space Engineering 2009/10
SP001	2012	Space Weather Warning System, Summary of the Group Design Project, MSc in Astronautics and Space Engineering 2011–12
SP002	2013	GeoSAR Feasibility Study, Summary of the Group Design Project, MSc in Astronautics and Space Engineering 2012–13

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 (Marc Dayas, Pierre Debusschere, Aston French, Nathan Hara, Guillaume Lengliné, Ye Chun Lo, Ross McDonald, Luc Meyer, Nicolas Petitpas, Guillaume Pineau, Adam Scott, Colin Shirran and Nicolas Vinikoff), 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.

Contents

List of Figures

List of Tables

Chapter 1

Introduction

This report summarizes a group project of the MSc in Astronautics and Space Engineering for the academic year 2012–13 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 2012-13 was a geosynchronous SAR 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 7800 hours' work (about 5 man-years) for the academic year 2012-13.

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 design. 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

The aim of the project was to develop a credible mission baseline for a geosynchronous SAR mission which would be compatible with the European Space Agency (ESA) Earth Explorer (EE) guidelines. Specific objectives or requirements beyond this were not stated explicitly, although the GEO SAT mission proposed to ESA was referred to as indicative of the performance which might be achieved. The EE guidelines were an important input to the project.

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

WP	Description	Student
WP1000 System	Requirements, Risk	Lo (2013)
	Baseline, budgets, cost	McDonald (2013)
	Operations, S/ware	Dayas (2013)
WP2000 Mission	Orbit, orbit perturbations	Meyer (2013)
	Orbit, orbit determination	Debusschere (2013)
	Orbit control	Pineau (2013)
	Launch, propulsion	French (2013)
	ACS	Vinikoff (2013)
WP3000 Mechanical	Configuration and structure	Lengliné (2013)
	Mechanisms	Scott (2013)
	Thermal	Petitpas (2013)
WP4000 Electrical	Power	Shirran (2013)
	OBDH	Dayas (2013)
	Communications	Hara (2013)
WP5000 Payload	Requirements	Hara (2013)
	Payload mechanical	Scott (2013)
	Payload electrical	Shirran (2013)

Figure 1.2 shows the general imaging concept assumed for the mission. This involves the radar operating in spotlight mode, i.e. staring continuously at the target area, for a period of minutes to hours, and then steering the beam to another target area to form the next image. More conventional stripmap imaging is also possible, but this is only appropriate for extremely large systems which almost certainly do not meet ESA's EE guidelines.

To enable full mission design, the team needed to translate the high-level project aim into specific requirements. Some of the mission aspects to be identified early in the project included:

- Imaging accuracy (spatial and temporal resolution)
- Temporal repeat frequency
- Area to be covered, types of target area
- Mission concept of operation and lifetime

One of the practical difficulties for the project was that the basic imaging principles for geosynchronous SAR are still being explored in detail. Although the general principle seems sound and has been partially validated through analysis and experiment, there are details of the image formation process which are not yet fully understood. Despite this difficulty, the team identified some credible requirements from which the rest of the mission design could proceed.

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.*

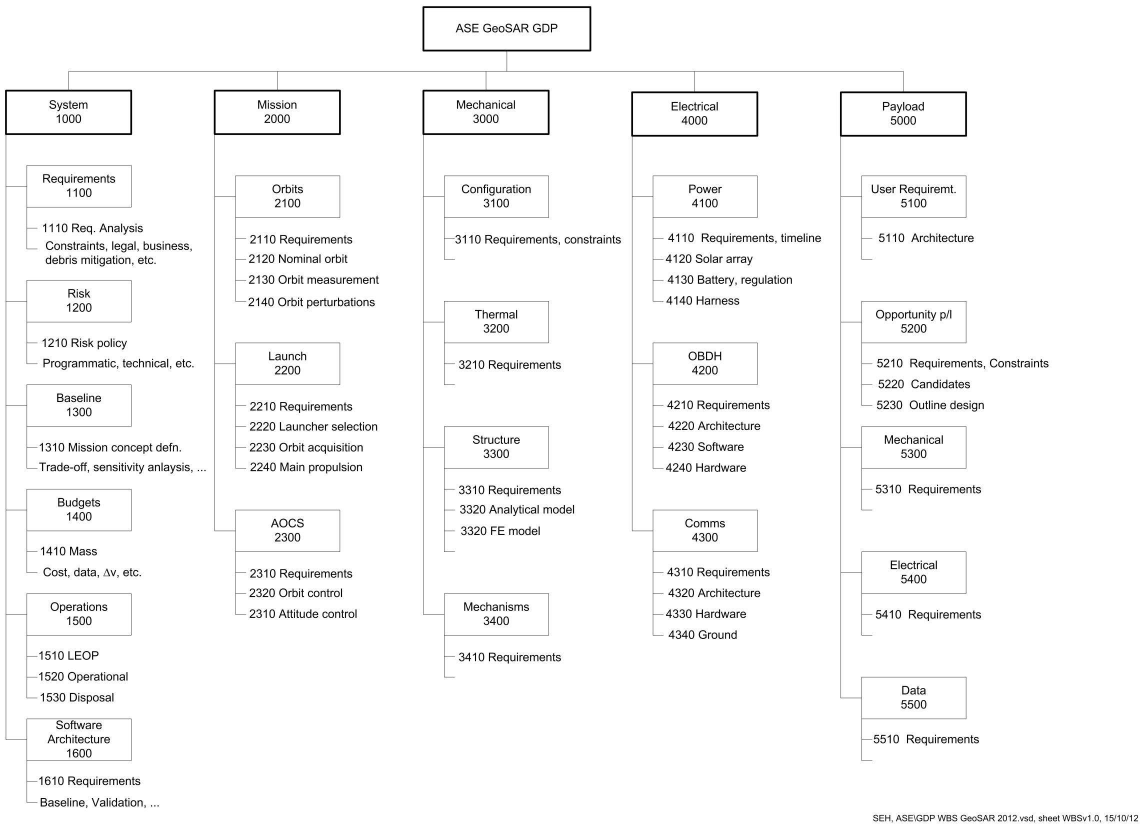


Figure 1.1: Initial work breakdown structure for the GeoSAR study.

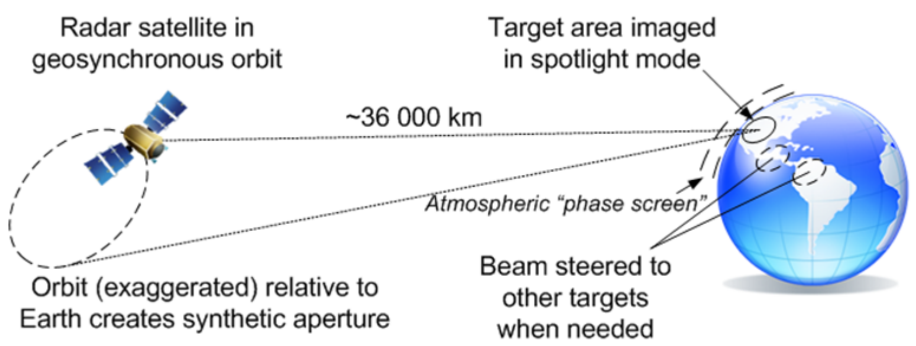


Figure 1.2: Overview of the mission concept for geosynchronous SAR (spotlight mode imaging).

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

Several aspects of the project can be analysed in outline without knowing much detail of the mission. Many satellites have been designed and operated in geosynchronous orbit, so it is reasonable to assume that problems of thermal design, radiation tolerance, etc. can be solved using standard solutions. The design challenges relate especially to those aspects of the mission which are innovative.

Orbit for Radar Imaging

Geosynchronous orbit is used by most communication satellites and is well understood. It is tightly regulated to enable efficient use by the International Telecommunications Union (ITU) so that satellites must maintain their orbit within specific limits. This station-keeping requires periodic orbit corrections of approximately 50 m s^{-1} yr^{-1} (Fortescue, Swinerd and Stark, 2011). Launch is typically to geostationary transfer orbit (GTO) and then orbit circularisation requires approximately 1.5 km s^{-1}. These figures summed over a typical GEO mission duration of 10–20 yr suggest that propulsion mass may be significant for the design.

Challenges for a GEO SAR mission are likely to involve the compatibility of SAR imaging with the standard communication satellite operations and the fact that a specific orbit, very slightly perturbed from exactly geostationary, is required. Orbit maintenance needs to be good enough to allow interferometry, so the track should repeat with a displacement of a small fraction of the critical baseline.

$$b_{\perp crit} = \frac{R\lambda \tan\theta}{c\tau} = \frac{R\lambda \tan\theta}{2\Delta_R} \tag{2.1}$$

For typical values of $R = 40\ 000$ km, $\theta = 60°$, $c\tau = 100$ m, $\lambda = 0.1$ m, the critical baseline is 69.3 km. For useful imaging, the baseline should not exceed about 20% of this, i.e. 13.9 km. For shorter wavelengths this reduces in proportion to the wavelength. This is an important but not insurmountable constraint.

Payload

Synthetic aperture radar (SAR) mission design is not trivial since so many parameters are interrelated. In the case of GEO SAR there is also the question of which wavelength to choose. This decision is strongly related to the desired application, and is constrained by both radar design principles and the atmospheric perturbations which can affect radar imaging.

For the spacecraft mechanical design, the antenna will be a significant item since most GEO SAR mission concepts use medium to large antennas. Technology for large lightweight structures is therefore likely to be necessary.

Operations

Imaging from GEO brings the possibility of far more versatile imaging modes than are possible from low Earth orbit. The operations task therefore has to consider novel ways of using the satellite, while meeting user needs (perhaps especially demanding for timeliness of data delivery) and satisfying constraints of orbit and attitude maintenance, as well as image quality (and therefore signal integration time).

A weekly operations cycle is convenient for practical personnel reasons.

Since the orbit period is 1 sidereal day (23 hr 56 min) but operations have a period of one solar day (24 hr), the orbit cycle is slightly out of phase with operations. Over a year this means that the time of, say, the most northerly point of the orbit drifts through a whole day. Since to permit interferometry a given target needs to be imaged from the same orbit segment, the local (solar) imaging time for a given target will drift through the year.

Some of the potential applications require timely data. To reduce the time delay before a user receives a usable image, the ground segment and operations need to be designed carefully to keep delays to an acceptable level.

Attitude Control

The spotlight imaging mode expected entails staring at one target area for a period and then switching to another. Since the maximum latitude imageable from GEO is around 60°, the maximum slew from one target to another will be equivalent to that between 60° N and 60° S. The corresponding angle from GEO is $2 \times 8.1°$, and more typically a slew will be of perhaps 2° (equivalent to approximately 1250 km on the ground). These slews may need to be performed every hour or so.

2.2 Technical

Appendix A gives an overview of the mission baseline developed. The general configuration (stowed and deployed) is shown in Figure 2.1. Some significant features of the mission baseline include:

- Electric propulsion is used for orbit raising and for station-keeping,
- A large lightweight antenna is deployed at the end of a lightweight boom, raising questions of mechanical stability and its dynamic response to manoeuvres.

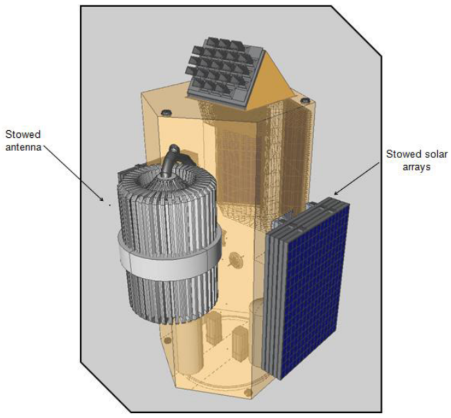
The constraints assumed have influenced the design significantly. Although these have been reviewed by the students, they were largely imposed on the project and are therefore outside the scope of the students' responsibility. The two main factors were

- Develop a mission concept compatible with ESA's Earth Explorer guidelines,
- The target application suggested was subsidence monitoring focussed on urban areas.

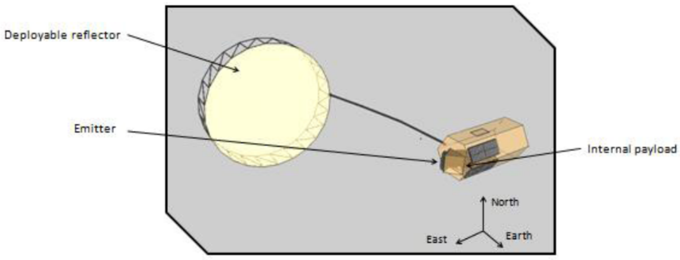
The Earth Explorer guidelines require some experience to interpret fully. It implies designing the mission so that it will be attractive to the reviewers appointed by ESA, which encompasses aspects such as technical challenge, exciting new measurement capabilities coupled with applications, and credible development, cost, and project management planning.

The target application of urban subsidence monitoring influenced the choices of wavelength and temporal and spatial resolution, which then feed into many other aspects of mission design. The main focus of the MSc course is space engineering, therefore the choice of application has to some extent been secondary to the main focus on developing a credible baseline mission design traceable to the original requirements. A plausible alternative target application is atmospheric monitoring: this would have led to a different mission design in several ways (e.g. choices of wavelength and spatial resolution).

The technical work packages summarised in Appendix B should be read taking these factors into account.



(a) Stowed



(b) Deployed (excl. solar arrays)

Figure 2.1: GeoSAR spacecraft configuration

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 mission baseline developed is broadly credible and provides a useful baseline for further work. It is also useful to have this baseline to compare with the study done in 2005/06 for a passive bistatic GEO SAR mission (Hobbs, 2006). Perhaps the next step apart from the points listed below for further work is to develop a baseline design for a longer wavelength (C, S or L-band) which may be better suited to atmospheric remote sensing.

3.2 Conclusions

The GEO SAR mission concept seems feasible both from an engineering viewpoint and also in terms of compatibility with the ESA Earth Explorer guidelines. The project sets a valuable baseline for a monostatic Ku-band GEO SAR mission. The baseline presented is not optimal however, and a further design iteration is recommended before it is taken as a stable mission proposal.

3.3 Future Work

A feasibility study always raises many further questions about the mission design. As well as a general review of the whole mission design based on the lessons learned from this first iteration, the following points are recommended for specific investigation.

- Review the system design: a mass reduction towards as little as half the current value may be possible, since the power, structure, thermal and propulsion sub-systems account for 1040 kg of the current dry mass and all these should reduce significantly if only 1 engine is assumed instead of 2.

- Review the choice of target application and wavelength. This requires further study of the surface properties (especially the temporal stability) as well as of potential applications. It would especially interesting to see how using a longer wavelength affects mission design.

- Check the dynamics of the coupled satellite / antenna structure since it may be prone to flexible body motion which will interfere with signal focussing and associated phase perturbation.

- The spacecraft configuration should be studied to see whether it is possible to reduce the time needed to make station-keeping manoeuvres (i.e. by avoiding the need for large slew manoeuvres).

- The mission compatibility with European launchers should be investigated.
- Review the signal-to-noise calculations to ensure that a good compromise between antenna size, transmitter power and beam coverage has been achieved.

In addition to these main items, an initial study like this should be reviewed in full. A subject of specific interest for GeoSAR is the scheduling of image acquisitions. The options are much more varied than for conventional LEO missions although there are specific constraints which may narrow the options, such as:

- Interferometry requires repeating the orbit track within a tight tolerance, e.g. a few km, so that a particular segment of the orbit arc should be re-used, specific to each target area,
- The orbit period is not the same as a solar day, so during the year the time corresponding to each arc will change, and therefore repeat imaging at the same solar time is not appropriate for interferometry.

Image scheduling needs to be planned carefully so that opportunities for interferometry and the usefulness of other data products are optimised. Another area for study is the final spacecraft disposal. Procedures for end-of-life operations with electrically-powered spacecraft have yet to be established; estimating the amount of remaining propellant is likely to be a key task in this process.

This study makes a contribution to research into the feasibility of geosynchronous radar imaging. It will be fascinating to see whether issues identified in this study influence the design of operational missions.

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Appendix A

Mission Baseline Technical Summary

This Appendix provides the common general baseline in which the GeoSAR GDP Team member has arrived at. The information collected in this appendix goes from the top level perspective presenting the Aim of the GDP, the Mission Objective and the Top Level Requirements to the subsystem level presenting budgets, timelines, explanatory figures and satellite drafts.

A.1 Top Level Statements

A.1.1 GDP Aim

The aim of the GeoSAR GDP Team 2012-2013 is: “To explore the feasibility of employing space-borne Synthetic Aperture Radar to acquire useful scientific data from Geosynchronous orbit”

A.1.2 Mission Objective

The objective of the GeoSAR Mission chosen to fulfil the GDP Aim is: “To provide scientific data of subsidence events over Europe and North Africa within ESA Earth Explorer guidelines”

A.1.3 Top Level Requirements

The top level requirements that were derived from the GeoSAR Mission Objective are:

- R1 The GeoSAR System shall operate in GEO with interferometric SAR technology
- R2 The GeoSAR Mission shall be accomplished within ESA EE guidelines
- R3 The GeoSAR System shall select the appropriate wave band for imaging performance
- R4 The GeoSAR System shall generate imagery products with fine spatial and temporal resolution
- R5 The GeoSAR System operational lifetime shall be, at least, 15 years

A.2 Mission Timelines and Schedules

A.2.1 General Mission Timeline

The mission timeline is summarised in Figure A.1.

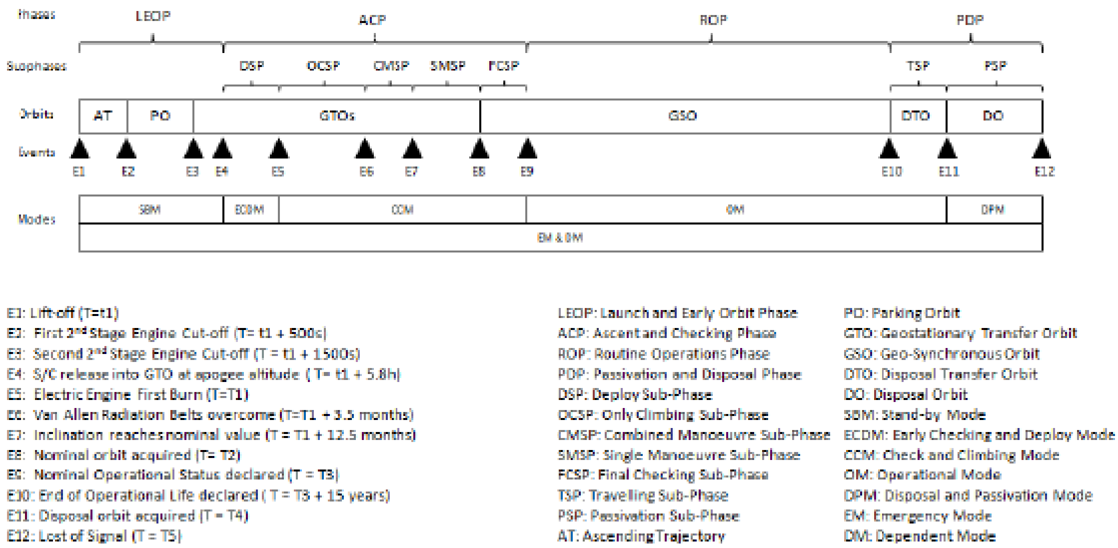


Figure A.1: Proposed mission timeline

A.2.2 Routine Operations (ROP)

The mission operations are summarised in Figure A.2. Imaging within the ROP framework is shown in Figure A.3 and the regular orbit maintenance manoeuvre timing is shown in Figure A.4.

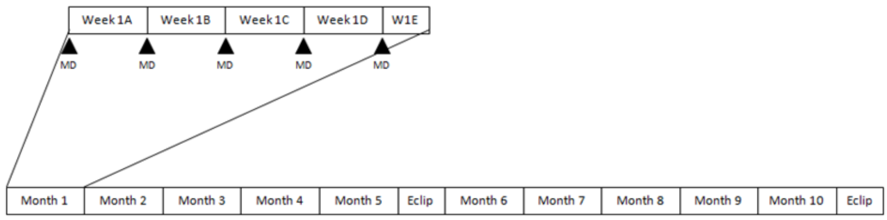


Figure A.2: ROP general architecture

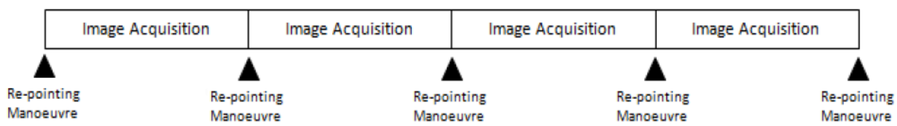


Figure A.3: ROP Imaging Activities Architecture

A.3 Configuration

Figures A.5 to A.12 show the configuration of the GeoSAR spacecraft.

A.4 Budgets

Tables A.1 and A.2 summarise the mass and ΔV budgets and specify the nominal orbit parameters.

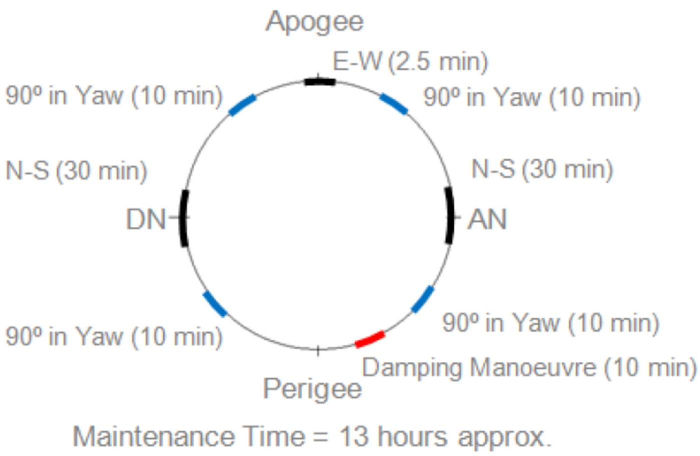


Figure A.4: ROP Maintenance Activities Architecture

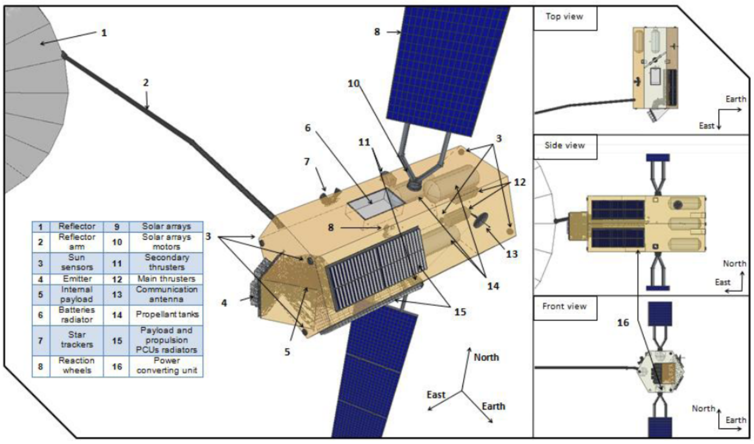


Figure A.5: Spacecraft configuration (deployed)

A.4.1 Launcher and Propulsion

Launcher

- Primary Launcher: Falcon 9, SpaceX, Cape Canaveral.
- Second Launcher: Long March 3B/E, China Great Wall Industry Corporation, Xichang Satellite Launcher Center.

Propulsion

- Primary Propulsion: 2x SPT-140, OKB Fakel, 4.5 kW maximum Hall Effect Thruster.
- Secondary Propulsion (ACS): 2x RIT-10, EADS Astrium, 460W maximum Ion Thruster.

Propellant: 350 kg of high-purity Xenon.

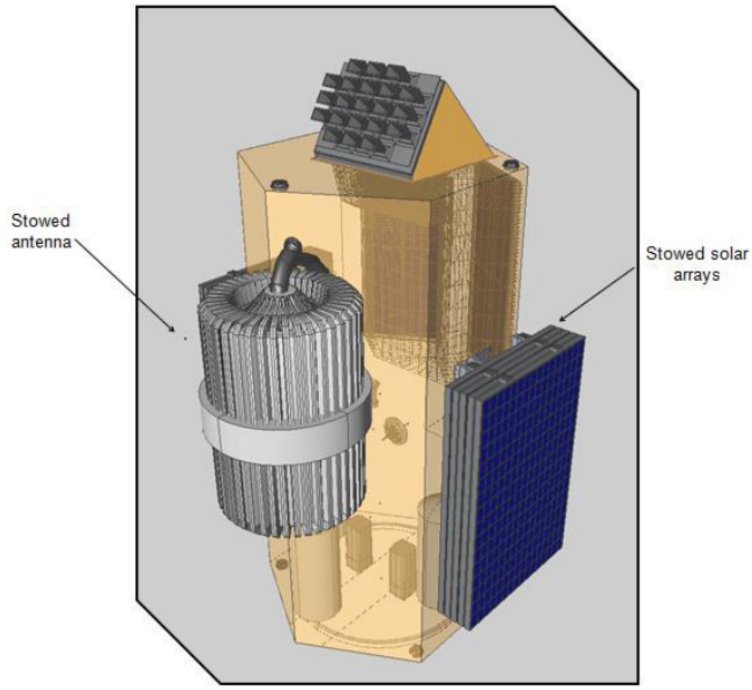


Figure A.6: Spacecraft configuration (stowed)

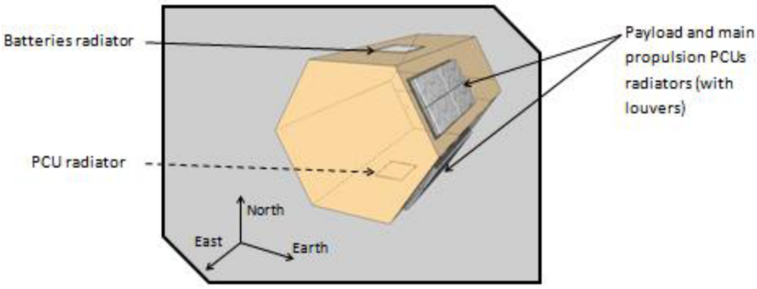


Figure A.7: Spacecraft configuration: thermal sub-system components

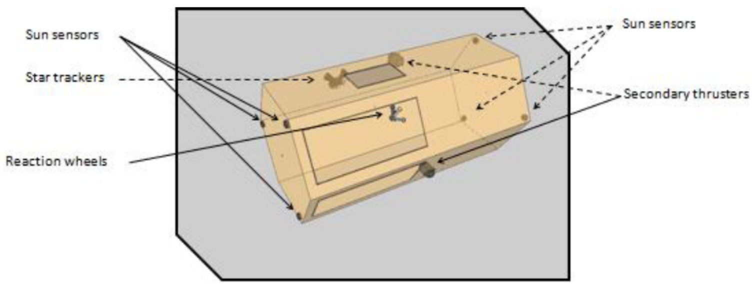


Figure A.8: Spacecraft configuration: ADCS sub-system components

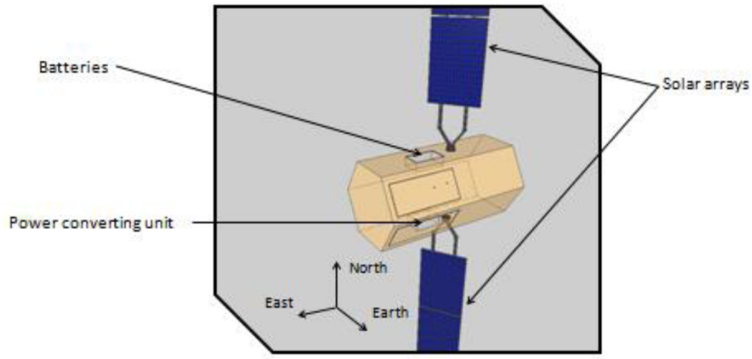


Figure A.9: Spacecraft configuration: power sub-system components

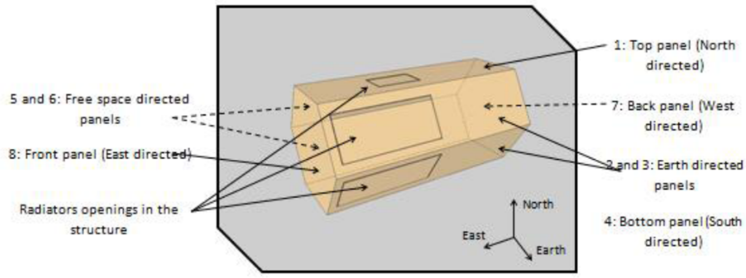


Figure A.10: Spacecraft body

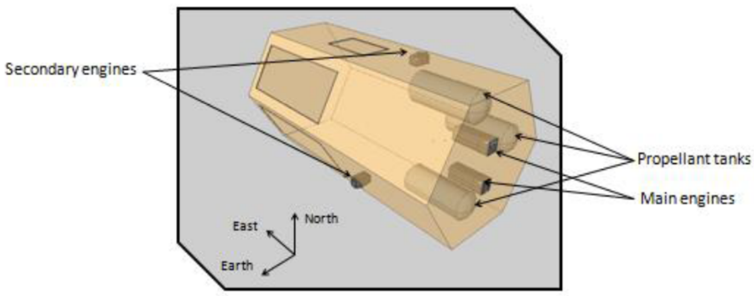


Figure A.11: Spacecraft configuration: propulsion sub-system components

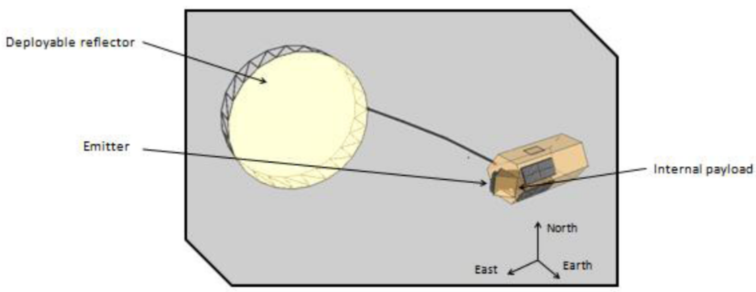


Figure A.12: Spacecraft configuration: payload sub-system components

Table A.1: Mass budget.

Subsystem	Mass (kg)
Payload (Electrical)	371
Bus Structure and Mechanical	400
Thermal	57
Power	426
TT & C (including OBDH)	40
Payload (Structural)	110
Propulsion/AOCS Engines	159
Initial Total	1,563
10% Margin	156
Subtotal	1,719
Propellant for Electrical Prop. System.	350
TOTAL	**2,069**

Table A.2: Orbit and ΔV budget.

Delta-V Budget

	Orbit Insertion	Station-keeping	Graveyard	Desaturation	TOTAL
Required Delta-V [m/s]	2,060	780	13	<2	2,855

Orbit parameters Budget

Parameter	Value
Semi-major axis	42,164 km
Inclination	0.054 deg
Eccentricity	0.00047
Argument of perigee	270 deg
Longitude of the Ascending Node	8.05 deg
Epoch	J2000 True of Date Coordinate System

A.4.2 Power Sub-system

Tables A.3 to A.7 summarise the power sub-system.

Table A.3: Total Maximum Power of All Systems.

System	Power (W)
Payload	3,000
AOCS	1,536
Communications	35
OBDH	36
Thermal	891
Propulsion	8,400
Power	732

Table A.4: Worst case power budget.

Mission Phase	Power (W)
Launch (Stand By)	881.4
Launch (De-Tumbling)	1,893 (On Thrusters), 1360 (Reaction Wheels)
Launch (Sun Acquisition)	1,893 (On Thrusters), 1360 (Reaction Wheels)
Ascent to GEO (During Burn)	8,831 (At BOL output from arrays)
Operational Phase (Ave. Power Sunlight)	4,163
Operational Phase (Ave. Power Eclipse)	4,169
Operational Phase (Ave. Power Outside of Eclipse Season	3,838
Orbit Maintenance (N/S and E/W Station Keeping)	8,766 (At BOL output from arrays)
Orbit Maintenance (Reaction Wheel Unloading)	1,378
Disposal Orbit (Using Main Propulsion)	6,226 (At EOL output from arrays)

Table A.5: Power system mass budget.

Component	Mass (kg)
Solar Array	104
Battery	119
Solar Array Drives	12
PCDU	<60
Miscellaneous (Wiring etc.)	48
Total	**426**

A.4.3 Communications Link Budget

Table A.8 summarises the communications link budget.

A.4.4 Thermal Hardware Budget

Table A.9 summarises the thermal sub-system.

A.4.5 Cost Budget

Table A.10 summarises the mission costing.

A.4.6 Software Budget

Table A.11 summarises the software sizing process.

Table A.6: Power system summary.

Type	Triple Junction GaInP/GaAs/Ge
Area	$27m^2$
Power BOL (23.5 sun angle)	8.5kW
Power EOL (23.5 Sun Angle)	6.3kW
Cost Estimate	$2.5M

Table A.7: Battery characteristics.

Type	Li-Ion
Cell voltage	3.6V
Cell Capacity	39Ahr
Total Number of Cells	96
Total Battery Capacity	13.5kWhr
Nominal Battery Voltage	86.4V
Dimensions of One Battery Pack	1.3m x .11m x .25m
Cost Estimate with a 20% margin ($3000/kWhr)	$48.6k

Table A.8: Link budget summary.

Parameter	**Value**	**Value in dB**
Maximum distance (km)	40,000	-
Data rate (kb/s)	10	-
Antenna size (m)	0.043	-
Ground station antenna size (Redu) (m)	13	-
Frequency (GHz)	2.2	-
Transmitter gain	-	-2.6
Ground station gain	-	46.9
Free space loss	$5.0\ 10^{-17}$	-191.3
System noise temperature (K)	250	24
Boltzmann constant	$1.38\ 10^{-23}$	-228
Other losses	-	-5
Link margin	-	-6
Transmitted power (W)	4.55	3.45
Beamwidth (degree)	181.8	-

Table A.9: Thermal system summary.

Thermal component	**Mass (kg) (+20% margin)**	**Max. Power (W) (+20% margin)**
Body external MLI	29.94	0
Body internal MLI	3.50	0
Batteries radiators	1.10	0
Payload radiator 1	1.57	0
Payload radiator 2	1.57	0
PCU radiator	0 (structure)	0
Heaters	6.08	240
Total	56.61	240

Table A.10: Cost budget summary.

Subsystem	**Cost (million FY2010$)**
Structure/Thermal	43.5
AOCS/Propulsion	48.6
Power	28.9
TT&C	26.9
IAT	0.34
Antenna	10.0
Programming	0.14
AGE	0.17
Launch	35.0
TOTAL	**(€150M FY2013 app.)**

Table A.11: Software sizing summary.

Function		Size		
		Code (Bytes)	Data (Bytes)	Throughput (KIPS)
Communications				
	Command Processing	2,048	8,192	7
	Telemtery Processing	2,048	5,120	3
	TOTAL	4,096	13,312	10
Attitude Sensor Processing				
	IMU	1,638.4	1,024	9
	Sun Sensor	1,024	204.8	1
	Star Tracker	4,096	30,720	2
	TOTAL	6,758.4	31,948.8	12
Attitude Determination & Control				
	Kinematic Integration	4,096	409.6	15
	Error Determination	2,048	204.8	12
	Precession Control	6,758.4	3,072	30
	Thruster Control	1,228.8	819.2	1.2
	Reaction Wheel Control	2,048	614.4	5
	TOTAL	16,179.2	5,120	63.2
Payload				
	SAR	43,941.888	43,950.08	1,132.076
Autonomy				
	Complex Autonomy	30,720	20,480	20
Fault Detection				
	Monitors	8,192	2,048	15
	Fault Correction	4,096	20,480	5
	TOTAL	12,288	22,528	20
Other Functions				
	Power Management	2,457.6	1,024	5
	Thermal Control	1,638.4	3,072	3
	TOTAL	4,096	4,096	8
Operating System				
	Executive	7,168	4,096	29.433
	Run-time Kernel	16,384	8,192	Inc. in other
	I/O Device Handler	4,096	1,433.6	40
	Built-in Test and Diagnostic	1,433.6	819.2	0.5
	TOTAL	29,081.6	14,540.8	69.933
ESTIMATED COMPUTER REQUIREMENTS		**163,340.288**	**161,095.68**	**1,272.009**

A.4.7 ADCS Budget

Tables A.12 to A.15 summarise the ADC system sensors and actuators.

Table A.12: ADC sensors: GEO transfer mode.

GEO Transfer Mode							
Component	Baseline	In Use	Continuous use	Mass (per unit) [kg]	Assembly Mass [kg]	Power (per unit) [W]	Assembly Power [W]
Star trackers	3	0	NO	1.25	3.75	11	0
Star tracker Electronic Unit	1	0	NO	1.75	1.75		
Sun sensors	6	6	YES	0.4	2.4	1	6
IMUs	2	1	YES	7.5	15	24	24
N.B. 1: Star trackers are not used for failure risk mitigation and data handling reasons N.B. 2: The IMU is used for detumbling implementation and attitude control during firing N.B. 3: Sun sensors are used both as sensors for Sun tracking and as normal attitude determination sensors						GTO Power [W]	30

Table A.13: ADC sensors: on-station mode.

On-Station Mode							
Component	Baseline	In use	Continuous use	Mass (per unit) [kg]	Total Mass [kg]	Power (per unit) [W]	Assembly Power [W]
Star trackers	3	1	YES	1.25	3.75	11	11
Star tracker Electronic Unit	1	1	YES	1.75	1.75		
Sun sensors	6	6	NO	0.4	2.4	1	6
IMUs	2	1	YES	7.5	15	24	24
N.B. 1: Sun sensors are supporting Sun tracking only; Star trackers are the attitude determination instruments (attitude knowledge requirement) N.B. 2: The IMU is used to propagate the attitude between star tracker updates and naturally to provide the body rates data required for control N.B. 3: Sun sensors can be put in stand-by when the S/C is in eclipse that is why it is indicated that they are not continuously operating							
						OSM Power - No Eclipse [W]	35
						OSM Power - Eclipse [W]	41

A.4.8 Mechanisms Budget

Tables A.16 to A.19 summarise the GeoSAR spacecraft mechanisms.

A.4.9 Payload Budget

Table A.20 lists the SAR parameters affecting system design. Figure A.13 summarises the dependence of some of the payload performance / design parameters on target latitude.

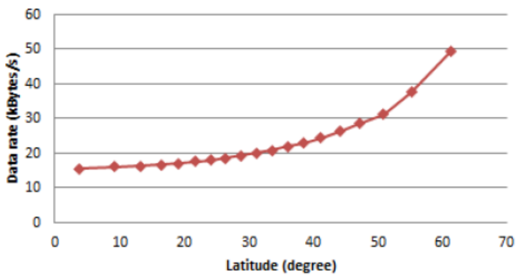
Table A.14: ADC system summary: GEO transfer mode.

(a) De-tumbling

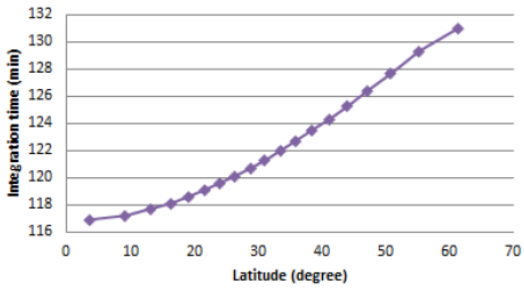
DETUMBLING							
Component	Baseline	In Use	Continuous use?	Mass (per unit) [kg]	Assembly Mass [kg]	Peak Power (per unit) [W]	Assembly Peak Power [W]
Reaction Wheels	4	3	YES	5.2	20.8	145	435
Thrusters (Astrium RIT-10)	2	2	NO	1.8	3.6	460	920
N.B. 1: Depending on the X-axis body rate. the thrsuters will be used or not							
N.B. 2: Reaction wheels are used to drive the Y- and Z-axis body rates to zero when it has been done for the X-axis							
				GTM Power - Detumbling [W]		1210	

(b) Orbit raising

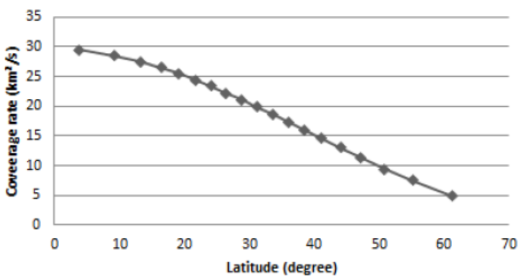
ASCENT PHASE							
Component	Baseline	In Use	Continuous use?	Mass (per unit) [kg]	Assembly Mass [kg]	Steady-State Power (per unit) [W]	Assembly Steady-State Power [W]
Reaction Wheels	4	3	YES	5.2	20.8	16.3	48.9
Thrusters (Astrium RIT-10)	2	0	NO	1.8	3.6	460	0
						GTM Power - Ascent Phase [W]	48.9



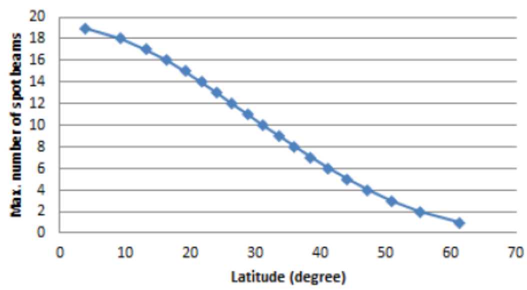
(a) Data rate



(b) Integration time



(c) Coverage rate



(d) Maximum number of spot beams

Figure A.13: Payload system parameters as a function of latitude

Table A.15: ADC system summary: on-station mode.

(a) Imaging

IMAGING							
Component	Baseline	In Use	Continuous use?	Mass (per unit) [kg]	Assembly Mass [kg]	Steady-State Power (per unit) [W]	Assembly Power [W]
Reaction Wheels	4	3	YES	5.2	20.8	16.3	48.9
Thrusters (Astrium RIT-10)	2	0	NO	1.8	3.6	460	0
N.B: Reaction wheels compensate for the external disturbances (periods of 2 hours)							
						OSM Power - Imaging [W]	48.9

(b) Re-pointing

RE-POINTING & STATION-KEEPING							
Component	Baseline	In Use	Continuous use?	Mass (per unit) [kg]	Assembly Mass [kg]	Peak Power (per unit) [W]	Assembly Power [W]
Reaction Wheels	4	2	YES	5.2	20.8	145	290
Thrusters (Astrium RIT-10)	2	0	NO	1.8	3.6	460	0
N.B. 1: Minimum of one operating RW and maximum of three; The median value is taken							
N.B. 2: Minimum duration of re-pointing: 30 s; Station-keeping manœuvre (one 90-degree rotation out of three): 7 minutes							
						OSM Power - Manoeuvres [W]	290

(c) Momentum dumping

REACTION WHEELS DUMPING							
Component	Baseline	In Use	Continuous use?	Mass (per unit) [kg]	Assembly Mass [kg]	Peak Power (per unit) [W]	Assembly Power [W]
Reaction Wheels	4	0	NO	5.2	20.8	145	0
Thrusters (Astrium RIT-10)	2	2	NO	1.8	3.6	460	920
N.B. 1: Simultaneous desaturation of X- and Y-wheel in 6 minutes; No RWs are used							
N.B. 2: Recovering from the introduced attitude error using the RWs in 3 minutes; No thrusters are used							
						OSM Power - Dumping [W]	920

Table A.16: Astromesh mechanism summary.

Astromesh RBA Summary	
Characteristic	**Value**
Diameter	7m from Earth
Mass	80 kg
Stowed Dimensions	1m dia x 1.5m reflector 4m x 0.3m dia for the stowed boom
Surface Accuracy	+/-0.5 mm
Cost (est)	15 million Euros

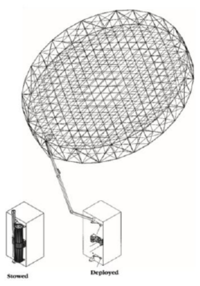


Table A.17: Feed horn array mechanism summary.

Feed Array Summary	
Characteristic	**Value**
Feed Horns	ATM WR62 (21)
Mass (kg)	10 kg
Dimensions (mm)	260 length x 300 depth x 200 height
Cost	Unknown, but not anticipated to be significantly expensive (thousands to tens of thousands)

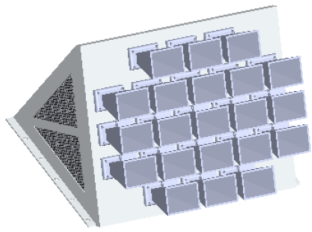


Table A.18: Feed horn mechanism summary.

WR62 Summary	
Frequency Range	12.4-18 GHz
Mass	68g
Dimensions	42x33x62mm
Beam Width	~30 degrees

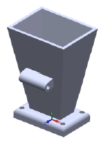


Table A.19: Solar array mechanism summary.

Solar Array Summary	
Wings	2
Panels per Wing	4
Panel size (length x width)	2.7 x 1.3 m
Yoke size (length x width)	1.5 x 0.9 m
Number of Holddowns per wing	5
Partial Deployment Possible	Yes

Table A.20: List of the SAR parameters affecting the system design.

Orbit/ geometry	Shape	Circular.
	Maximum synthetic aperture S_{max}	80.9 km
	Synthetic aperture S	20.3 km at lowest latitude observed (3.8°)) up to 23.2 km at highest latitude (61.4°)
	Range R	35800 at lowest latitude, 40300 km at highest latitude
	Looking angle θ	4.3,° at lowest latitude to 69°
Coverage	Total area covered A_t	32% of europe total area (mean coverage rate multiplied by the revisit time)
	Temporal resolution or revisit time T_{tot}	48h
	Number of spot beams n	Depends on the latitude (see figure A)
	Range of latitudes observed $[\phi_{min}, \phi_{max}]$	3.8° to 61.4°
	Backscatter coefficient σ_0	-10 dB
Image quality	Along-track resolution L_{al}	20 m
	Across-track resolution: L_{ac}	20 m (on the output image)
	SNR	20 dB
Antenna	Shape	Elliptical
	Dimensions of the antenna d, d'	7 m cross-section
System	Type	Monostatic
	Type Power of the transmitted electromagnetic wave(peak power) P_t	Active 10 kW
	Duty cycle fraction f_T	0.1
	System noise figure F_n	5
	Range bandwidth B_R	100 MHz at lowest latitude, 8 MHz at high latitude
	Pulse repetition frequency n_{prf}	between 0.29 Hz and 85.5 Hz
	Integration time T	116.9 min at lowest latitude, 131.1 min at highest latitude (see figure A)
	Mass m	Estimated at 420 kg
	F-number: F	1.1

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.

Table B.1: Sub-system responsibilities for each student *(no summary report is available for Structure and configuration)*

Student	Work area
Marc Dayas	Operations, software
Pierre Debusschere	Nominal orbit, orbit determination
Aston French	Launch and propulsion
Nathan Hara	Payload, communications
Guillaume Lengline	*Configuration, structure*
Ye Chun Lo	System engineering, risk management
Ross McDonald	Budgets, baseline definition
Luc Meyer	Nominal orbit, orbit perturbations
Nicolas Petitpas	Configuration, thermal
Guillaume Pineau	Orbit control
Adam Scott	Mechanisms, payload mechanical
Colin Shirran	Power, payload electrical design
Nicolas Vinikoff	AOCS

B.1 Operations, OBDH, Software: Marc Dayas

This summary covers the operations, On Board Data Handling (OBDH) and SoftWare (S/W) work packages of the GeoSAR Group Design Project (GDP). This project is a phase A study performed by 13 students held under the supervision of Dr. Steve E. Hobbs and it is included within the frame of the MSc course in Astronautics and Space Engineering 2012-2013 taught in the School of Engineering of Cranfield University. The aim of this project is to explore the feasibility of employing space-borne Synthetic Aperture Radar (SAR) to acquire useful scientific data from geosynchronous orbit (GSO) and the mission chosen to evaluate the feasibility of SAR in GSO from the technical and the operational point of view has, as the principal objective, to provide scientific data of subsidence events over Europe and North Africa accomplishing with European Space Agency (ESA) Earth Explorer (EE) guidelines. The fulfilment of the ESA EE guidelines ensures that the mission of the project accomplishes with the aim of the project of exploring applications that acquire useful scientific data.

B.1.1 Operations

The aim of the operations WP is to design a candidate mission architecture (ground segment and interactions between the satellite, ground station and final user), mission schedule (timelines and activities) and spacecraft overall modes for the GeoSAR spacecraft. Parallel to that, the operations WP is being included into a major WP called Systems Engineering WP. Hence, the secondary objective of the operations WP is to ensure all parts of the mission are internally coherent by dealing with subsystem level conflicts and solving them according to reliability, availability and cost-effectiveness criteria. The major constraints that are derived from other WPs are the mission lifetime, the type of targets to steer when imaging, the type of orbit it must be attended to describe the synthetic aperture needed and the nature of the mission operators.

Ground Segment

The GeoSAR ground segment is going to be operated by ESA staff in the European Space Operations Centre (ESOC), located in Darmstadt, Germany. However, during the launch period, the mission will be managed by Space Exploration Technologies Corporation (SpaceX) Launch Operations Control Centre, located in Cape Canaveral, Florida, US. The reason for having a completely different operator at launch is that, for cost-effective purposes, a SpaceX Falcon 9 launcher was chosen. The ground stations that are going to be used are located in Redu, Belgium and, as a back-up station, Villafranca, Spain. Both belong to the ESA Tracking Network ESTRACK. Regarding the launch period, SpaceX will use NASA Near Earth Network as it will hold a partnership contract.

Mission Timeline

The GeoSAR mission is split into four major phases: Launch and Early Orbit Phase (LEOP), Ascent and Checking Phase (ACP), Routine Operations Phase (ROP) and Passivation and Disposal Phase (PDP). Starting with LEOP, this phase, which will take 5.8 hours to be performed, is going to be comprised between the lift-off and the release of the GeoSAR satellite from the launcher into a Geostationary Transfer Orbit (GTO). Following that event, ACP will take 12.5 months in raising the first GTO orbit up to the final GSO to be attended. During this phase, several stages will be considered to clear as soon as possible the inner Van Allen Belt region where environmental radiation reaches peak levels and to get the final GSO as quick as possible. The ACP phase will be ended once Nominal Operational Status is declared and then ROP will take place in the mission. The ROP is the longest phase of the mission as it takes 15 years to be done and its principal aim is to accomplish with final user request during that time. The activities the GeoSAR satellite will have to do are classified into two categories: Imaging activities and Maintenance activities. While imaging activities occupy most of the time along the whole ROP, maintenance activities are concentrated in one day once every week. These maintenance activities include software updates, momentum damping manoeuvres, station-keeping manoeuvres and internal checking procedures.

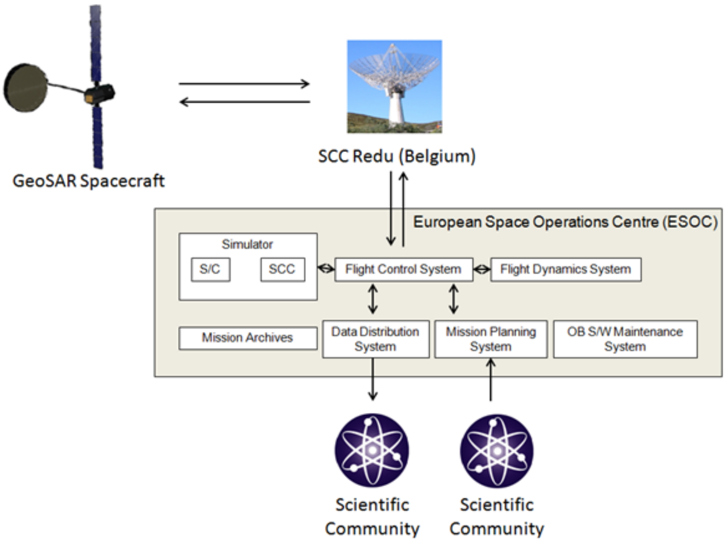


Figure B.1: Ground segment architecture

It shall be said that, during ROP, even though eclipse periods of 44 days are going to appear twice every year, GeoSAR satellite is prepared to be fully operable during these periods.

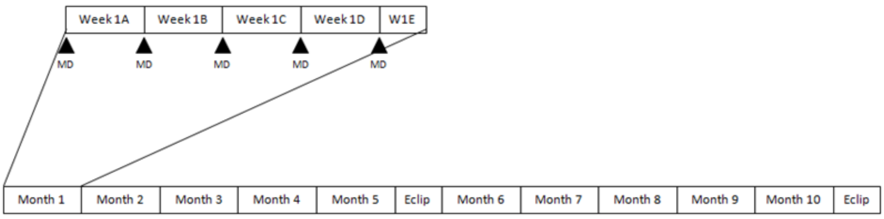


Figure B.2: Mission timeline

Once ROP finishes when end of operational life is declared, PDP will be run. During this phase, the spacecraft will climb up to a graveyard orbit that accomplishes with the Inter-Agency Space Debris Coordination Committee (IADC) guidelines and will execute passivation and disposal procedures until loss of signal is declared.

Spacecraft Modes

The GeoSAR spacecraft will have 7 operational modes strongly influenced by the mission timeline. This modes are the Stand By Mode, the Early Check and Deploy Mode, the Check and Climbing Mode, the Operational Mode, the Passivation and Disposal Mode, the Emergency Mode and the Dependence Mode. These spacecraft modes are going to turn on according to the following scheme:

Mission phase	Mode(s)
During LEOP	1. Stand-By Mode
During ACP	1. Stand-By Mode
	2. Early Check and Deploy Mode
	3. Check and Climbing Mode
During ROP	1. Operational Mode
During PDP	1. Passivation and Disposal Mode

Notice that the Emergency Mode and the Dependence Mode are going to be ready to be switch on in all the phases of the mission.

B.1.2 OBDH and S/W

The aim of the OBDH WP and the S/W WP is to design from a top level perspective the GeoSAR on-board computing system at a hardware level and at a software level. Since both WPs are combined into a single one and the software architecture WP belongs to the Systems Engineering major WP, this double WP also have greater input from the whole system and interactions between it and operations WP are evident when defining what type of activities the on-board computing system must perform along the mission timeline. The major constraints these two WPs receive from external WPs are related with radiation hardening levels and operational capability of the system. To tackle the design of this subsystem, robustness, low-cost and autonomy criteria were followed.

System State Transition Diagram

The tasks the on-board computing system must execute can be arranged into system states which are connected one each other with strict boolean conditions to avoid bit errors to cause hard failure of the system.

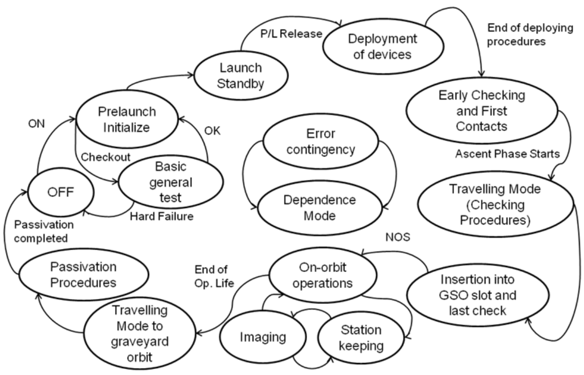


Figure B.3: Mission state transition diagram

The Error contingency and Dependence Mode states are connected with all other states except for OFF, Prelaunch Initialize and Basic general test states and these connections are not shown here due to clarity purposes.

On-Board Computer Architecture

The GeoSAR on-board computing architecture can be divided into a Serial-Parallel hardware architecture supported with a modular software architecture potentially compatible with oriented programming languages. The hardware components that form the architecture are the same embedded components normal personal desk computers have: Central Processing Unit (two of them for redundancy), Mass Storage Memory, Main Memory, Bus and Inputs and Outputs. It is understood as Inputs and Outputs, within this configuration, the rest of the physical components that need connection with the on-board computer. The interaction these Inputs and Outputs have with the Central Processing Unit follows an interruption scheme.

A baseline that describes the on-board computing system is the following:

Table B.2: Computer baseline

OB Computer Component	Size	Constraints
CPU	5 MIPS and 0.33 Mrads	SR1 and S/W sizing
Main Memory	1.33 MB	S/W sizing
Mass Storage Memory	2.5 GB	Payload Data Rate

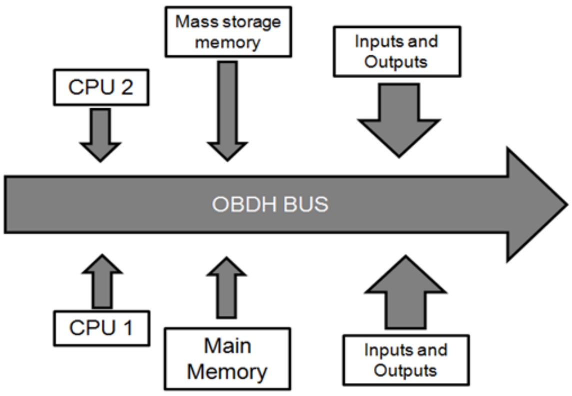


Figure B.4: On-board data handling architecture

B.2 Orbit, Orbit Measurement: Pierre Debusschere

B.2.1 Nominal orbit

The aim of the nominal orbit is to make the satellite run through a synthetic aperture (SA) for payload reasons (providing along-track resolution). It is wanted to get a spatial resolution of 20m, at Ku band, from GEO altitude. From the diffraction formula, $d = h\lambda/(2l)$, and at 60° of latitude (maximum observable latitude because of atmosphere reflection), a SA of 20km is necessary, as shown by the diagram below.

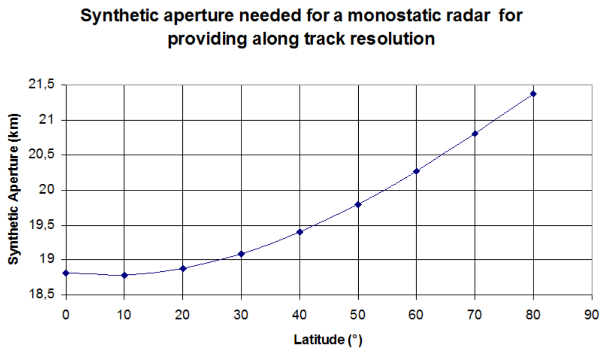


Figure B.5: Synthetic aperture size required

The SA is the chord of the orbit track. It is wanted to image (using the synthetic aperture) during 2hrs. As a circular orbit track (and then also ground track) is required to be able to steer at any place in the target area (Europe and North Africa), the required diameter of the orbit track has been calculated equal to 80km.

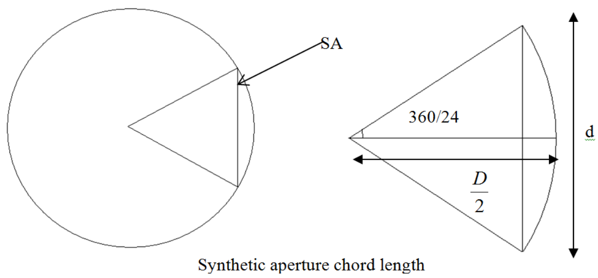


Figure B.6: Synthetic aperture chord length

The orbit parameters are calculated to get this 80km diameter circular orbit track. The inclination and eccentricity are affecting the size and shape of the orbit track as shown below. The argument of perigee has to be equal to 90° or 270° to get a circle. 270° is chosen to spend a fraction of time more in the northern hemisphere.

$$D = \frac{\lambda\sqrt{R_E^2 + R_G^2 + 2R_E R_G \cos(\phi - i)}}{2l} / \sin\left(\frac{2\pi}{24}\right) \tag{B.1}$$

The calculated nominal orbit parameters are given in Table B.3.

These parameters are valid at J2000 epoch and using a True of Date coordinate system. From the beginning, it has been kept in mind that the spacecraft orbit must abide by the ITU laws, and especially staying inside a geostationary slot of 147 km width. This requires weekly station keeping for tackling orbit perturbations.

B.2.2 Orbit Measurements

As being almost a geostationary satellite, the GeoSAR spacecraft stays over the same point of the surface of the earth, and then, there is no need for on board or semi-autonomous navigation, and

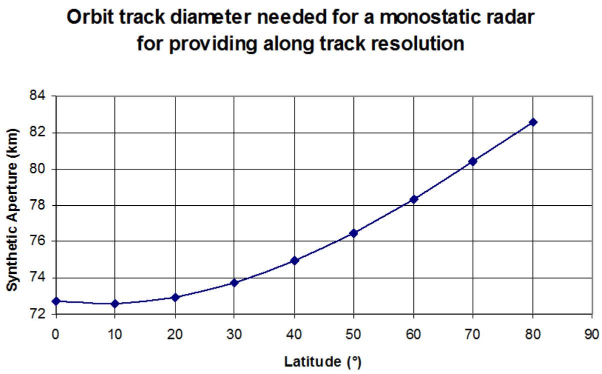


Figure B.7: Orbit track diameter needed

Table B.3: Nominal orbit parameters

42164 km	Semi-major axis
0.054°	Inclination
0.00047	Eccentricity
270°	Argument of perigee
8.05°	Longitude of ascending node
-	True Anomaly

trajectory control. The measurements as well as the station keeping commands can be handled by only one ground station.

The network which has been chosen is the ESA ESTRACK network to be coincident with the starting aim of the project which was to follow ESA Earth Explorer call guidelines. The chosen ground station is Redu Ground station, which is specially equipped with facilities dedicated to the assessment of geostationary satellites performances. This ground station is linked with ESOC center in Germany, so that commands can also be send from the same ground station, for example for station keeping manoeuvres.

B.2.3 Key references

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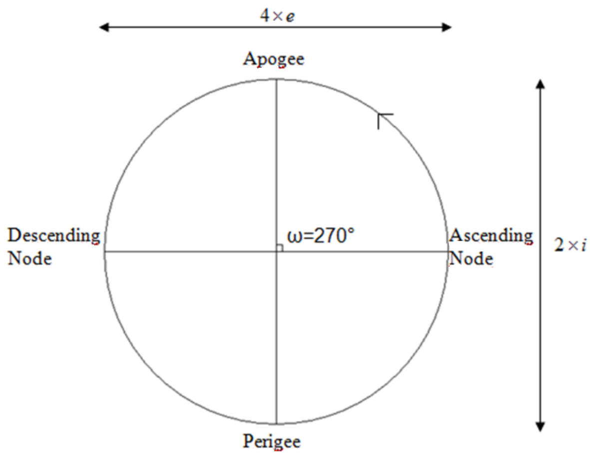

Figure B.8: Orbit sizing

Figure B.9: ESA ESTRACK network (ESA)

B.3 Launch and Propulsion: Aston French

B.3.1 Launch

Introduction

This is a brief summary of the work done for the launch segment of the GeoSAR project, and the final choices made. GeoSAR is a proposed Geosynchronous Synthetic Aperture Radar (SAR) utilising interferometry for the purposes of earth observation. The entire proposal has been undertaken under the framework of the ESA Earth Explorer call. In this summary, the requirements will first be identified, and then the most suitable launch vehicles for the mission requirements will be selected. At the end of this, a brief conclusion will outline the final launch scheme.

Requirements

The requirements affecting the launch segment are a combination of direct-derivatives of the top level requirements, and some requirements derived from other subsystems.

Top Level Requirements

1. The GeoSAR System shall operate in GEO with interferometric SAR technology
2. The GeoSAR Mission shall be accomplished within ESA EE guidelines
3. The GeoSAR System shall select the appropriate wave band for imaging performance
4. The GeoSAR System shall generate imagery products with fine spatial and temporal resolution
5. The GeoSAR System operational lifetime shall be, at least, 15 years

The only requirements directly affecting the launch segment are the first, second and fifth. The first requirement makes it clear that the launch vehicle either needs to inject the satellite directly into GEO, or that some provision needs to be made for propulsion to allow transferring to GEO from GTO. The second requirement implies a price limit on the launch segment, as the total Earth Explorer budget for a mission is only € 200M. The fifth requirement implies the need for a propulsion system, and the satellite lifetime will therefore directly impact the amount of propellant required and therefore the mass.

Derived Requirements

1. Launch vehicle must have sufficient mass capability to deliver the satellite to a suitable orbit. This is a derived requirement from the first top-level requirement but balanced against the capability of the Propulsion segment.
2. Payload fairing enclosure size must be large enough to contain the satellite.
3. Launch environment (vibrational, acoustic etc.) must be acceptable and ensure satellite health.
4. Launch-related risks must be acceptable.
5. Orbital injection accuracy must be within acceptable levels.

The first requirement merely explicitly restates the analysis made in the previous section. In terms of the other requirements, the variation between different launch vehicles is comparatively far smaller and therefore a secondary consideration at this stage. A partial exception to this is the second requirement as a suitably large payload fairing is required if a multiple-payload launch is to be possible, with the attached possibility of sharing the cost of launch with another satellite.

Table B.4: Table of example suitable launch vehicles, with all information primarily taken from [1] and confirmed against the relevant launch vehicle's User Manual where possible.

Launch Vehicle	Cost	GTO Capability	Payload Cost/Mass	Reliability		
	\$M	kg	\$ kg^{-1}	Success	Partial	Failure
Atlas V 401	100	4,950	20,202	17	1	0
Delta IV M+	100	6,822	14,658	12	0	0
Falcon 9	54	4,850	11,134	4	1	0
H-IIA 202	100	4,100	24,390	11	0	0
Long March 3B/E	70	5,500	12,727	13	0	0
Proton M	85	6,150	13,821	63	1	6

Potential Launch Vehicles

A table of launch vehicles with a capacity of 4,000 to 7,000 kg to GTO was compiled into a table as can be seen in Table B.4, with the mass range chosen so as to facilitate a co-manifest launch with an appropriate partner satellite. Note that this is only a subset of all available options, with a maximum of one variant from a family of launch vehicles included.

Selected Launch Vehicles

Following analysis, the Falcon 9 was chosen as the primary launch vehicle, and the Long March 3B/E as the backup launch vehicle in the case of there being a significant issue with the primary launcher or other unforeseen circumstances that preclude its use.

Falcon 9 The Falcon 9 [2] was chosen as the primary launch vehicle as a result of its excellent cost per kilogram, more-than-adequate launch mass capability and substantial fairing volume that facilitates a multiple-payload launch. As an additional point of benefit, SpaceX are far more open about their launch vehicle and company capabilities, and work closely with the customer to ensure that the satellite and launch vehicle are compatible from an early stage. Finally, they have a substantial contract with NASA that drives a launch scheme that should help build the Falcon 9 heritage rapidly, as well as funding further research and development of the launch vehicle and supporting systems. Indeed, by the earliest possible points of launch, the next upgrade of the Falcon 9 should have become active giving even better performance.

The Falcon 9 is the second generation of launch vehicles from SpaceX, being preceded by the Falcon 1. The Falcon 9 uses 9 upgraded Merlin engines in the first stage, and a single Merlin engine in the second stage. Unlike most other multi-stage launch vehicles with entirely different lower and upper stages the Falcon 9 utilises the same engine for both, with the simple modification that the single engine of the second stage has an extended nozzle to improve the vacuum Isp. This allows flight time and data on the engine performance to be built far more rapidly than might otherwise be, and also allows for far more rapid development of the engine and common systems due to removing the need for multiple parallel projects.

Long March 3B/E The Long March 3B/E [3] was chosen as the secondary (or backup) launch vehicle due to having a cost per kilogram that was nearly as good as the Falcon 9 and a slightly larger launch mass capability. However, it suffers the weakness that its payload fairing is significantly shorter than the Falcon 9 (roughly half as long), and is also narrower. As a result, despite its greater launch mass capability it is a far weaker candidate for a multiple-payload launch.

The Long March 3B/E is the enhanced version of the Long March 3B, which itself is an enhanced Long March 3A with 4 radially mounted liquid rocket boosters. Unlike the Falcon 9, each stage of the Long March is different and each utilises a different engine and configuration. Despite this, it has proven to be a fairly reliable vehicle - with a few notable exceptions - and the 3B and 3B/E have made a combined total of 23 launches since 1996.

Conclusion

In conclusion, all requirements have either been achieved or are currently being explored with no expectation of failure. The chosen primary and secondary launch vehicle each have the capability to launch a pair of satellites to GTO, and have the best cost-per-kilogram of the suitable launch vehicles. They represent the best value for the mission while still exceeding the stated requirements, and as a result also the most robust in the face of future alterations to the satellite or mission.

Future work will focus on improving the knowledge of supporting launch procedures and capabilities such as the exact situation regarding co-manifest launches and their cost, and the specifications of a Falcon 9-compatible adapter to allow multiple satellites to be encapsulated on the same launch.

B.3.2 Propulsion

Introduction

This is a brief summary of the work done for the propulsion segment of the GeoSAR project, and the final choices and design. GeoSAR is a proposed Geosynchronous Synthetic Aperture Radar (SAR) utilising interferometry for the purposes of earth observation. The entire proposal has been undertaken under the framework of the ESA Earth Explorer call. In this summary, the requirements will first be identified and the type of propulsion chosen, and then specific thrusters and support equipment will be selected. At the end of this, a brief conclusion will outline the final propulsion design.

Requirements

The requirements affecting the propulsion segment are a combination of direct-derivatives of the top level requirements, and some requirements derived from other subsystems.

Top Level Requirements

1. The GeoSAR System shall operate in GEO with interferometric SAR technology
2. The GeoSAR Mission shall be accomplished within ESA EE guidelines
3. The GeoSAR System shall select the appropriate wave band for imaging performance
4. The GeoSAR System shall generate imagery products with fine spatial and temporal resolution
5. The GeoSAR System operational lifetime shall be, at least, 15 years

The first requirement implies that some form of propulsion system capable of transferring the satellite from GTO to GEO will be necessary. In combination with the fifth requirement, this then also implies that on the order of 2,600 m $s^{-1}$of ΔV will be required for the satellite's ascent and lifetime operations. The second requirement places a constraint on the propulsion system requiring that it trade cost against system mass due to the Earth Explorer mission budget and the cost interlink with the launch segment, in terms of the need for a method of transferring from GTO to GEO.

Derived Requirements

1. The propulsion system must be capable of transferring the satellite from GTO to the final GEO orbit.
2. There must be sufficient capability to provide 15 years of station-keeping and move the satellite to a "graveyard orbit" at the end of lifetime.
3. The wet mass of the satellite must be conducive to a multiple-payload launch in an affordable launch vehicle, e.g. less than 3,500 kg.

4. Provision must be made for supporting the normal operation of the ACS segment, namely by providing facility for the unloading of reaction wheels.

5. The satellite must have some form of fail-soft capability in the event of a main-engine failure.

6. The propellant must be stored in such a way so as to reduce “fuel slosh”.

7. The power requirement of electric propulsion must be sized to the existing power systems.

The first and second requirements are effectively just an explicit restatement of top level requirements, but they effectively size the ΔV budget for the mission. The third requirement is linked with the launch segment, as sharing a launch has the potential to save a lot of money. The seventh requirement limits the potential usage of electric propulsion to a level that minimises the disruption to the rest of the design. The other requirements essentially specify particular solutions that can be met by appropriate choice of components and design.

Propulsion System Selection

Designing the propulsion system requires the choice of an appropriate family of propulsion (solid, liquid, electric), then choosing specific thrusters and finally the supporting equipment to allow their nominal operation.

Propulsion Family A trade-off was performed between the different families of propulsion. Due to the prime driver being the cost of the mission, the significant reduction to launch mass that is associated with electric propulsion made it clearly the most suitable option for the propulsion. Additionally, after analysing the relationship between the cost of a Falcon 9 shared launch as a function of Isp via launch mass, it was apparent that a propulsion system with an Isp of 1,500 to 10,000 s had the potential to approximately halve the cost of launch. As such, the thrusters that were examined were all of the Hall Effect or Ionic type.

Primary and Secondary Thruster Selection Having narrowed down the type of thrusters that was desired, a list of appropriate thrusters for the primary propulsion system was compiled from [4] and [5], and can be seen in Table B.5.

Table B.5: Compiled table giving basic information on a range of high-performance Ion and Hall Effect Thrusters.

Manufacturer, Thruster	Isp (s)	Thrust (mN)	Input Power (W)	Thrust per Watt (mN W^{-1})
QinetiQ, T6	4,700	145	5,000	0.029
Astrium, RIT-XT	4,500	150	5,000	0.030
ETI, XIPS-25cm	3,500	165	4,500	0.041
Busek, BHT-8000	1,900	512	8,000	0.064
Aerojet, BPT-4000	1,770	290	4,500	0.064
Fakel, SPT-140	1,750	300	4,500	0.067

The SPT-140 produced by Fakel was chosen as the most suitable option due to it having an acceptable Isp, while also having a comparatively high thrust and a power requirement that was conducive to utilising a pair in parallel with the existing GeoSAR power systems.

Secondary thrusters were also selected from a second list of smaller thrusters, and the RIT-10 produced by Astrium was chosen as the most suitable option for supporting the ACS segment due to the large range through which the thrust could be varied (0.3 to 41 mN) allowing for greater fidelity of control.

Final System Overview Table B.6 summarises the propulsion system mass budget.

Table B.6: Propulsion system mass budget

Component	Mass (kg)	Quantity
Fakel, SPT-140	8.00	2
Astrium, RIT-10	1.80	2
Fakel, PPU-140	13.70	2
Astrium, RFG	2.50	2
ATK, 80458-101 Xenon Tank	19.05	3
RUAG, ITAM Gimballed Mount	4.30	2
Moog, XFC Xenon Flow Controller	0.53	2
Wiring and piping etc. (estimate)	15.00	-
TOTAL DRY MASS	133.81	-
High-Purity Xenon Propellant	350	-
TOTAL SYSTEM MASS	483.81	-

Conclusion

In conclusion, the final system fulfils all the requirements of the propulsion segment. The selection of Hall effect thrusters with their high Isp means that the satellite launch mass is reduced from approximately 4,500 kg to 2,069 kg (with a predicted saving of $26M in launch costs) from a typical chemical propulsion system. The satellite will follow a modified transfer between GTO and GEO, with multiple intermediate orbits as necessitated by the low thrust levels of electric propulsion.

Future work will focus on creating a valid simulation of the ascent phase so as to validate the assumptions made so far and to get a more reliable ΔV estimate.

B.3.3 References

1. Isakowitz, S, Hopkins, J and Hopkins Jr, J. International Reference Guide to Space Launch Systems (4th Ed). Reston, VA : AIAA, 2004.

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3. China Academy of Launch Vehicle Technology. LM-3B User's Manual. s.l. : China Academy of Launch Vehicle Technology, 1999.

4. Wertz, J, Everett, D and Puschell, J (Eds.). Space Mission Engineering: The New SMAD. Hawthorne, CA : Microcosm Press, 2011. 978-1-881-883-15-9.

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B.4 Payload and Communications: Nathan Hara

B.4.1 SAR Payload

The SAR parameters are very interconnected and there is no straightforward approach to define them from the requirements. According to the payload engineering method proposed in the new SMAD, the orbit must be defined first. However, the size of the antenna also affects the signal to noise ratio such that the orbit cannot be determined without an idea of its size. The solution adopted was to define a range of possible diameters for the antenna and define the orbit first. Then the antenna has been sized with more precision. This illustrates the method used for the payload design. When a value was needed to size a part of the system but required other decisions that had to be decided precisely afterwards, ranges were defined. The sizing order was defined as:

1. The first step has been to make assumptions on the radar. It has been supposed active and monostatic to allow the building of a model. Also, as the spotlight mode is the most efficient in terms of resolution, which is fixed by the top-level requirements. Indeed, the radiometric resolution is fixed via the signal to noise ratio and the spatial resolution was fixed. The system must achieve the requirements at least in spotlight mode. According to the technology review by Adam Scott, the antenna has been supposed circular and the diameter was supposed between six and eight metres. Double-pass was chosen. The Ku-band was chosen due to its low sensitivity to atmospheric perturbations. The average transmitted power has been supposed equal to 1 kW. Finally, the maximum bandwidth as been assumed 100 MHz considering the TerraSAR-X achieves a 150 MHz bandwidth.

2. The principle of the steering strategy has been chosen at that stage. As the along-track resolution is given, the size of the synthetic aperture is known. There are several manners of achieving a synthetic aperture of a given length. It might not be necessary to use all the available time for emission (the duty cycle multiplied by the duty cycle fraction) to achieve a given signal to noise ratio. Here the signal to noise ratio is required to be 20 dB. The maximum number of spot beams that still achieve that requirement will automatically be chosen (this number depends on the latitude) in order to increase the coverage rate.

3. These assumptions allowed to choose the orbit. The orbit shape has first been determined in order to have equal opportunities of coverage on east, west, north and south. The principle of the double-pass interferometry is to take images of the same region at different instants. The distance between the two apertures where the images were performed is limited by the critical baseline. It has then be chosen to have a repeating pattern of synthetic aperture. In other words, the synthetic apertures performed in twenty four hours must form a regular polygon. Therefore, the integration time must be a sub-multiple of 24h. Then, there is a discrete set of inclinations that allow 1, 2, 3, ... synthetic apertures to be performed in 24h. Coverage rate does not depend a lot on the orbit. The choice of the orbit is then mainly driven by orbit maintenance issues, steering strategy and latitudes to cover.

4. Besides, the interferometry has stringent constraint on the orbit knowledge and on the correction of atmospheric phase screen correction, in particular due to the long integration times in geosynchronous orbit. These constraints were determined at that stage.

5. A trade-off with the structure segment was performed in order to size precisely the antenna. Two technologies were considered: Astromesh and electronically steerable antennas. Also, two shape were studied: circular or rectangular, as the state of the art showed that other shapes were rare and require a specific design. The parameters to choose the technology and the size were:

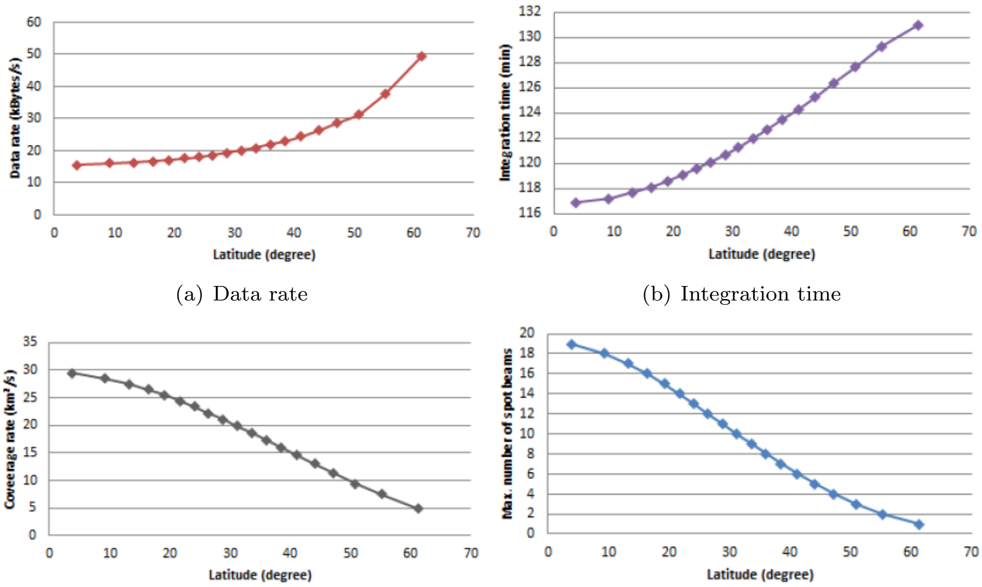
    - Mass
    - Configuration: compatibility with the launcher (size when stowed), complexity
    - Performances: resolution, SNR and coverage (increasing the diameter of the antenna increases the resolution and the SNR but reduces the coverage rate)

Also the configuration was chosen. The f-number was maximized as the larger it is the better the performances of the payload are. As increasing the f-number requires a boom, it must however be checked that the oscillations are not too important to achieve the required relative position accuracy between the antenna and the emitters, and that the risks of failures are acceptable. The emitters or the antenna can be at the end of a boom. The two opportunities were assessed and it has been chosen to put the antenna on the end of the boom. The cross-sectional area degradation was also determined.

6. At that stage, the type of radar hardware was chosen with Adam Scott (antenna, mechanisms)and Colin Shirran (power subsystem). Two opportunities were considered: electronic arrays and feed horns. The latter was finally chosen.

7. The steering strategy is a complex issue that could not be optimized within the project duration. However, some trends and some concepts were found.

8. The payload main characteristics and performances were determined, which then allowed to give precise requirements for other subsystems.



(a) Data rate

(b) Integration time

(c) Coverage rate

(d) Maximum number of spot beams

Figure B.10: Payload system performance as a function of latitude

B.4.2 Communication Sub-system

Basic links budgets have been calculated for the omnidirectional and directional antennas (Tables B.8 and B.9).

The value obtained for the power seems very small compared to other TT&C systems. In a representative TT&C budget, the power required by the S-band system is 44 W and 60 W for the X-band antenna.

Table B.7: List of the SAR parameters affecting the system design.

Orbit/ geometry	Shape	Circular.
	Maximum synthetic aperture S_{max}	80.9 km
	Synthetic aperture S	20.3 km at lowest latitude observed (3.8°)) up to 23.2 km at highest latitude (61.4°)
	Range R	35800 at lowest latitude, 40300 km at highest latitude
	Looking angle θ	4.3,° at lowest latitude to 69°
Coverage	Total area covered A_t	32% of europe total area (mean coverage rate multiplied by the revisit time)
	Temporal resolution or revisit time T_{tot}	48h
	Number of spot beams n	Depends on the latitude (see figure A)
	Range of latitudes observed $[\phi_{min}, \phi_{max}]$	3.8° to 61.4°
	Backscatter coefficient σ_0	-10 dB
Image quality	Along-track resolution L_{al}	20 m
	Across-track resolution: L_{ac}	20 m (on the output image)
	SNR	20 dB
Antenna	Shape	Elliptical
	Dimensions of the antenna d, d'	7 m cross-section
System	Type	Monostatic
	Type	Active
	Power of the transmitted electromagnetic wave(peak power) P_t	10 kW
	Duty cycle fraction f_T	0.1
	System noise figure F_n	5
	Range bandwidth B_R	100 MHz at lowest latitude, 8 MHz at high latitude
	Pulse repetition frequency n_{prf}	between 0.29 Hz and 85.5 Hz
	Integration time T	116.9 min at lowest latitude, 131.1 min at highest latitude (see figure A)
	Mass m	Estimated at 420 kg
	F-number: F	1.1

Table B.8: Link budget calculation for the omnidirectional antennas. The values used are taken accordingly to the Space System Engineering course.

Parameter	**Value**	**Value in dB**
Maximum distance (km)	40000	-
Data rate (kb/s)	10	-
Antenna size (m)	0.043	-
Ground station antenna size (Redu) (m)	13	-
Frequency (GHz)	2.2	-
Transmitter gain	-	-2.6
Ground station gain	-	46.9
Free space loss	5.0×10^{-17}	-191.3
System noise temperature (K)	250	24
Boltzmann constant	1.38×10^{-23}	-228
Other losses	-	-5
Link margin	-	-6
Eb/No (using BPSK, probability error rate of 10−6)	-	10
Transmitter power (W)	**4.55**	3.45
Beamwidth (degree)	181.8	-

Table B.9: Link budget calculation for the directional antennas. The values used are taken accordingly to the Space System Engineering course.

Parameter	**Value**	**Value in dB**
Maximum distance (km)	40000	-
Data rate (kbps)	400	-
Antenna size (m)	0.3	-
Ground station antenna size (Redu) (m)	13	-
Frequency (GHz)	8.45	-
Transmitter gain	-	-2.6
Ground station gain	-	46.9
Free space loss	5.0×10^{-17}	-191.3
System noise temperature (K)	250	24
Boltzmann constant	1.38×10^{-23}	-228
Other losses	-	-5
Link margin	-	-6
Eb/No (using BPSK, probability error rate of $10^{-}6$)	-	10
Transmitter power (W)	**0.13**	3.45
Beamwidth (degree)	6.78	-

B.5 Configuration and Structure: Guillaume Lengliné

No project summary is available.

B.6 System Engineering and Risk Management: Ye Chun Lo

B.6.1 Introduction

In the past few decades, we have seen an increase in public concern about the Earth, its environment and mankinds impact upon it. Global changes like climate warming, stratospheric ozone depletion, tropospheric pollution and more recent regional abnormalities such as the earthquake and tsunami in Japan, the wildfires in North America, the floods in Southeast Asia, and the coldest March in UK for 51 years, have aroused the public more alarm than ever about the need both to understand, monitor and predict the Earths environment. It is vital for human to sustain the long term habitability of our living planet.

In order to manage its own environment better and fulfil its international obligations and responsibilities, Europe has taken its active part through the introduction of the Living Planet Programme as ESAs new strategy for Earth observation in 1997. This programme is a result from a joint effort among ESA, the European Commission and Eumetsat, which has put forward a Proposal for a European Policy for Earth Observation from Space in 1995. In this respect, Europe could avoid relying entirely on other countries to obtain the required data to sustain its independent position globally, to fulfil its international obligations fully, and to manage its own environment satisfactorily. The aim of the GeoSAR Group Design Project is:

- To explore the feasibility of employing space-borne Synthetic Aperture Radar to acquire useful scientific data from Geosynchronous orbit For the purpose of carrying out this feasibility study, a mission concept has been selected by the GeoSAR group at the beginning of the project. And the mission objective is then defined.
- To provide scientific data of subsidence events over Europe and North Africa accomplishing with ESA Earth Explorer guidelines

In this project, we focus on monitoring the land subsidence in urban areas which are usually caused by excavation for construction, ground shrinkage due to water absorption by trees, groundwater extraction and resources extraction. The main target of this mission is to meet the Challenge 3: "Understand the pressure caused by anthropogenic dynamics on land surfaces and their impact on the functioning of terrestrial ecosystems", under Land Surface category put forward by ESA. We hope we can contribute to observe the physical state of the top few metres of soil and dynamical interactions with the Earth's interior.

B.6.2 Project Management

Project Organisation

The project ran for six months, from October 2012 to March 2013. In the GeoSAR group, we had a team of thirteen members working together throughout this period. At the beginning of the first period, each member started with defining our own responsibilities. Then, as a group we all contributed in the process of generating the top-level requirements by giving our own ideas. After that, each team member did research on our own work package by searching relevant literature and share with each other. Next, we began to brainstorming for possible alternative concepts to answer the requirements needed to be met. Afterwards, we did a trade-off to select the best solution and identified the initial baseline. Finally, the first term was ended with a System Requirements Review presentation to the lecturers and the fellow students.

System Engineer Role

The system engineers for this project take an important role in fully discharging our duties as well as monitoring and guaranteeing the smooth flow for the whole project. Basically, the system engineer has been assigned with the following tasks and responsibilities:

- Project Management

- System Requirements
- System Trade-offs
- System Baseline Definition
- System Budgets
- Mission Phases
- Risk Management

B.6.3 Top Level Requirements

The top level requirements for this mission broadly fall into three categories; performance, operational and constraints. Each is summarised below.

Performance

R1 - The GeoSAR System shall operate in GEO with interferometric synthetic aperture radar technology.

R2 - The GeoSAR System shall serve as a reply to the ESA Earth Explorer 8 call for Land-Surface Processes and Interactions Mission.

R3 - The GeoSAR System shall select the appropriate waveband for the imaging performance.

R4 - The GeoSAR System shall generate imagery products with fine (at least 40 m) spatial resolution and (48 hour) temporal resolution.

Operational

R5 - The GeoSAR System operational lifetime shall be at least 15 years.

Constraints

- Cost Budget < € 100M
- Schedule: For EE8, would need to launch mission in 2019
- Regulations: ITU Regulations, IADC Regulations

B.6.4 Risk Management

Methodology

Risk management has been performed with care in this project. It was used to manage the risk that might threaten the achievement of the required performance. It was based on a given set of performance requirements and decision made stated tolerance levels, analysing identified risk scenarios with possible mitigations and with follow-up monitoring and communications.

The risk management process is a five cyclical function of Identify, Analyse, Plan, Track and Control, supported by comprehensive Communicate and Document functions. In this project, the asset taxonomy was used as team members from different subsystems could assist in forwarding and advising risks that they had been concerned with.

Summary

There are a total of one hundred and eleven individual risks identified, with no risk under high risk category, “red zone”, fifty-two risks under moderate risk category, “yellow zone”, and fifty-nine risks under low risk category, “green zone”. Resources (schedule, cost) should be prioritised to handle those risks in moderate risk category first because resources would be used more efficient in this way in minimizing the overall mission risk.

B.6.5 Conclusions

This report concluded with a summary of the top level requirements and sub-level requirements for all subsystems. It also encompassed a full list of possible risks that may be encountered in the GeoSAR mission if the mission continuously evolves. This report serves as the overview of all other subsystem reports.

The project has been accomplished with great success with all subsystems achieving a satisfactory preliminary level of details and meeting the pre-set level of requirements of each subsystem in the final design. It paves a way for any background researches and future development into related GeoSAR studies.

B.6.6 References

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B.7 System, Baseline, Budgets: Ross McDonald

This appendix includes a summary of the project itself, and a detailed general project baseline for each subsystem as a collaborative effort from all members of the project.

B.7.1 Introduction

A geosynchronous orbit (GEO) is an orbit around the Earth with a period of exactly one sidereal day. This matches the rotation period of the Earth itself. Synthetic Aperture Radar (SAR) is a radar method whereby a range of several radar images are collected while an emitter and receiver move along a set path. The overall objective for this GDP is to develop a design for a GEO satellite that would use SAR for a particular Earth-observation related purpose, chosen by the group. This satellite would ideally fit in the framework of the ESA Earth Explorer programme, in terms of function and cost.

This led to two derived objectives that formed the basis of our requirements:

1. To provide scientific information on Earth subsidence events

2. To demonstrate the feasibility of employing SAR in GEO From these objectives, a set of five top-level requirements were set:

3. The GeoSAR satellite shall operate in GEO with interferometric synthetic aperture radar technology.

4. The GeoSAR satellite shall satisfy the Earth Explorer programme guidelines.

5. The GeoSAR satellite shall select the optimal waveband for the best imaging performance.

6. The GeoSAR satellite shall generate image products with appropriate spatial and temporal resolutions.

7. The GeoSAR satellite shall operate with a lifetime similar to that of existing geosynchronous satellites (c.15 years).

The GeoSAR satellite is to be launched using a Space-X Falcon 9 rocket in order to attain geosynchronous transfer orbit (GTO). Once in GTO the electric propulsion system on the spacecraft will perform a series of burns to raise the craft into a circular geosynchronous orbit. This phase of the operation is planned to take approximately 12.5 months depending on various circumstances.

Once GEO is attained, an equipment check will be performed to ensure each of the spacecraft systems are nominal. Once this is complete, imaging shall begin. This imaging phase will continue for as long as the power remains to support it - based on other geosynchronous communication satellites already orbiting, this is estimated to be around 15 years. When degradation of the power subsystem becomes so much that that operations can no longer be carried out, the disposal phase shall begin.

In this final phase, the electric propulsion system will be used to place the spacecraft into a "graveyard orbit" approximately 200 km above the geosynchronous orbit altitude, where all contact will cease.

B.7.2 Systems Engineering

The communication and interaction between subsystems was facilitated by a structure of regular meetings, as well as a shared Internet-based computer drive SkyDrive being procured for the easy exchange of data. This computer drive became a simple way of accessing the work of the other project members for reference purposes and being able to make your own available for reference too. As well as these two factors, regular progress reports and communication between each subsystem either verbally or via email was encouraged and carried out.

After initial discussion, a list of three different variables within the spacecraft were considered for analysis on which to base various concepts: wavelength, inclination and self-launched/hosted payload. Trade-off analysis of the ESA Earth Explorer guidelines and early payload research led to observation of land subsidence being chosen as the purpose of the spacecraft with the antenna transmitting in the Ku electromagnetic band. A further trade-off included a concept where the inclination would either be high or low, however subsequent analysis a drastic and unacceptable increase in the required power of the satellite.

As a result of this, the final quantitative trade-off was run between only two different mission concepts: one that would be self-launched, and another that would be launched as part of a host payload. The result of the trade-off was the choice of a self-launched payload.

B.7.3 Mass and Power Budgets

In the initial estimations of mass and power, a considerable amount of conjecture had to be used as there are no existing spacecraft of the same type for comparison.

At the initial stage of the budgeting, a blanket mass budget margin of 25% was placed on every subsystem, due to the unique nature of the mission and therefore the considerable amount of guesswork involved in initial planning. To ascertain a hard limit for the total dry mass that could be used for initial subsystem mass budgeting, a value of 2750 kg was chosen. From the initial estimates to the final mass budget, the margin for all subsystems dropped from 25% to 10%, due to increasing reliability in mass figures from subsystems. There was also some change in subsystem mass allocation naming during the project. The final mass budget is given below in Table B.10.

Table B.10: Mass budget summary

Subsystem	Mass (kg)	Mass (% Total Mass, no margin)
Payload (Electrical)	371	19.39
Bus Structure and Mechanical	400	20.91
Thermal	57	2.98
Power	426	22.27
TT&C (inc OBDH)	40	2.09
Payload (Structural)	110	5.75
Propulsion/AOCS Engines	159	8.31
Initial Total	1563	81.70
10% Margin	156	
Subtotal	1719	
Propellant for Electrical Propulsion System	350	18.30
TOTAL	2069	100

The final mass of the spacecraft with a 10% margin comes to around 2070kg, which is significantly lower than the hard limit of 2750kg proposed early in the project. With this launch mass, a shared launch using a Space-X Falcon 9 to GTO is eminently possible which would represent a significant financial advantage. The choice of using electric propulsion for the entire mission as opposed as opposed to liquid propellant of any kind represented a significant saving of mass, and a lightweight Astromesh antenna reduced the projected mass of the payload structure considerably. Further work includes work on reducing mass budget margins further by receiving more and more comprehensive information from subsystems, and attempting to find information regarding masses of GEO satellites closer to the present day, as the information found thus far is older than desired.

B.7.4 Costing

To cost this mission, the subsystems were divided along lines broadly similar to that of the mass budget. Once this was completed a parametric Cost Estimate Relationship (CER) was used to estimate the cost of each subsystem by either its individual mass, a fixed value or other parameter. For this project, the Unmanned Space Vehicle Cost Model version 8 was used to give an initial projection of cost for the mission.

The final cost of the mission using the USCM8 and collecting all the individual subsystem costs came to $186 million (FY2010). This equates to around € 143 million, as of March 2013. When model error is taken into account, the final projected cost of the mission is € 143 million +/- € 53.8 million. To satisfy the conditions of the ESA EE Program, the project must cost less than

€ 100 million for the space segment and certain ground segments only. If the cost of the launch - \$35 million - is removed from consideration leaving only spacer, then the final cost is around € 117 million +/- € 44 million.

Using the USCM8 gave a very, very crude estimation of the cost of this mission, and the final cost quoted when error bounds are added falls within the acceptable cost of the ESA Earth Explorer requirements when only space and mission-specific ground segments are considered. However due to several factors its accuracy in this case must be called into question.

In light of these factors, it can be said that while the costing figure given is a reasonable starting approximation, further study is clearly required to gain better reliability in the final cost figure for this mission. This would include increased research into costing for individual subsystems to establish a possible custom CER for this mission given its unique nature, producing a model for inflation with regard to the mission timeline, and applying this to the construction phase of the mission, and a more comprehensive check of calculations for potential errors.

B.8 Nominal Orbit and Orbit Perturbations: Luc Meyer

B.8.1 Introduction

The aim of the GeoSAR Group Design Project is to study the feasibility of the GeoSAR mission: employ a Synthetic Aperture Radar to acquire useful scientific data from a satellite in a geosynchronous orbit.

- **Geosynchronous orbit**

  A geosynchronous orbit is an orbit with a period of a sideral day, which means a semi-major axis of 42,164 km.

- **Synthetic Aperture Radar (SAR)**

  In all radars, the more the aperture is, the more the resolution is. Thanks to the Synthetic Aperture Radar technology (SAR), the aperture can be increased without changing the size of the radar. The SAR uses the advantage of the relative motion of the satellite with respect to the area to image, in order to synthesise a higher aperture, and thus to increase the resolution.

- **GeoSAR**

  For classical radar, a trade-off between the resolution (better in low orbit) and the coverage (better in high orbit) has to be done. The advantage of the GeoSAR concept is to have a good coverage (thanks to the high orbit) with a good resolution (thanks to the Synthetic Aperture Radar).

  Among all the possible missions that such radar can realise, we decided to choose the following mission: "provide scientific data of subsidence events over Europe and North Africa accomplishing with ESA Earth Explorer guidelines".

In the context of this project, I was in charge of the mission, and particularly of the choice of the Nominal Orbit and of the study of the Orbit Perturbations.

B.8.2 Nominal Orbit

In this section, the orbital parameters of the satellite are chosen.

Orbit Requirements

The requirements taken into account for the choice of the orbital parameters are listed in the following table.

Table B.11: Summary of the orbit requirements

General requirement	Geosynchronous orbit: the period has to be one sideral day
Mission Zone to cover:	Europe and North Africa
Payload	Circular orbit track with a diameter of 80 km
Launch	Minimise the propellant and the time needed to reach the orbit
AOCS	Minimisation of the perturbations
Law	Respect of the ITU regulations

Orbit requirements From these requirements, we selected the critical ones (the requirements that have to be satisfied). They are summarized in the following table.

Determination of orbital parameters

From the previous critical orbit requirements, we deduced the following orbital parameters.

Table B.12: Summary of the critical orbit requirements

General requirement	Geosynchronous orbit: the period has to be one sideral day
Mission Zone to cover:	Europe and North Africa
Payload	Circular orbit track with a diameter of 80 km
Law	Respect of the ITU regulations

Table B.13: Orbital parameters chosen for the nominal orbit

Orbital parameter	Value
Semi-major axis	42,164 km
Eccentricity	0.000474
Argument of perigee	270 deg
Inclination	0.054355 deg
Right Ascension of the Ascending Node (RAAN)	18.45 deg
Epoch	J2000

B.8.3 Orbit Perturbations

In this section, the effects of the orbit perturbations (all the forces acting on the spacecraft except the central body force) on the orbital parameters are studied.

We first take into account the following perturbations:

- non spherical Earth (including tidal effects)
- gravitational forces of other bodies (Sun and Moon)
- solar radiation pressure
- relativistic effects
- atmospheric effects

However, in the case of a geosynchronous orbit with a low eccentricity, as in our case, we can take into account only the following perturbations:

- Non spherical Earth: oblateness and triaxiality
- Gravitational force of the Sun and the Moon
- Solar Pressure Radiation

The effects of these perturbations on the orbital parameters are summarized in the following table.

Table B.14: Effects of the perturbations on the orbital parameters of the satellite

Orbital parameter	Secular effect (per orbit)
Semi-major axis	$\Delta a = 0$ m
Eccentricity	$\Delta e = 2.05 \times 10^{-7}$
Argument of perigee	$\Delta \omega = 6.43 \times 10^{-3}$ rad
Inclination	$\Delta i = 1.15 \times 10^{-5}$ rad
Right Ascension of the Ascending Node	$\Delta \Omega = -2.3 \times 10^{-3}$ rad

B.8.4 Mission Analysis

In this part, we discuss the issues relative to the global mission. At the beginning of the mission, the spacecraft has to be launched, and then put into the right orbit. During station operations, other issues have to be taken into account: station-keeping, attitude control, and maximum time spend in eclipse. Finally, the spacecraft has to be transferred in a graveyard orbit after the end of his life.

Mission life-time: the mission life time is 15 years.

In the following table, the Delta-V budget is summarized.

Table B.15: Summary of the Delta-V budget

Manoeuvre	Delta-V needed
Nominal Orbit Acquisition	2060 m/s
Longitude Station-Keeping (per year)	2 m/s
Latitude Station-Keeping (per year)	50 m/s
Total Station-keeping (per year)	52 m/s
Total Station-keeping (15 years)	780 m/s
End-of-life orbit acquisition	10 m/s
TOTAL Delta-V	2850

Thus, the total Delta-V budget is Delta-V = 2850 m/s.

Maximum eclipse time: the maximum time spend in eclipse during the same day is 70 minutes.

B.8.5 Conclusion

From the mission analysis point of view, there is no critical issue that cannot be solved. So, we can conclude that, from this point of view, the GeoSAR concept is feasible.

Finally and thanks to the work done by the whole team, we have shown that the GeoSAR is a feasible concept and we have provided a General Baseline.

B.8.6 References

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B.9 Configuration and thermal: Nicolas Petitpas

This is a short summary of the configuration subsystem and the thermal subsystem for the GeoSAR group design project.

B.9.1 Configuration subsystem

The configuration of the GeoSAR was strongly driven by the payload subsystem, mainly because of the reflector needed as shown in Figure B.11. The structure, propulsion, power, ADCS and thermal also had an important influence on the configuration design.

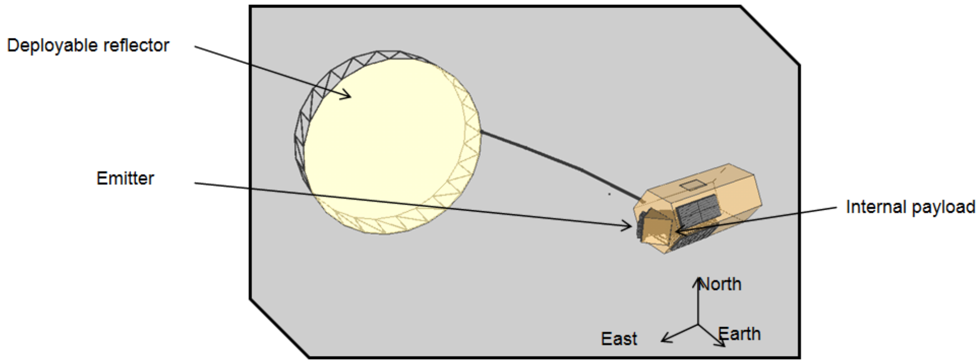
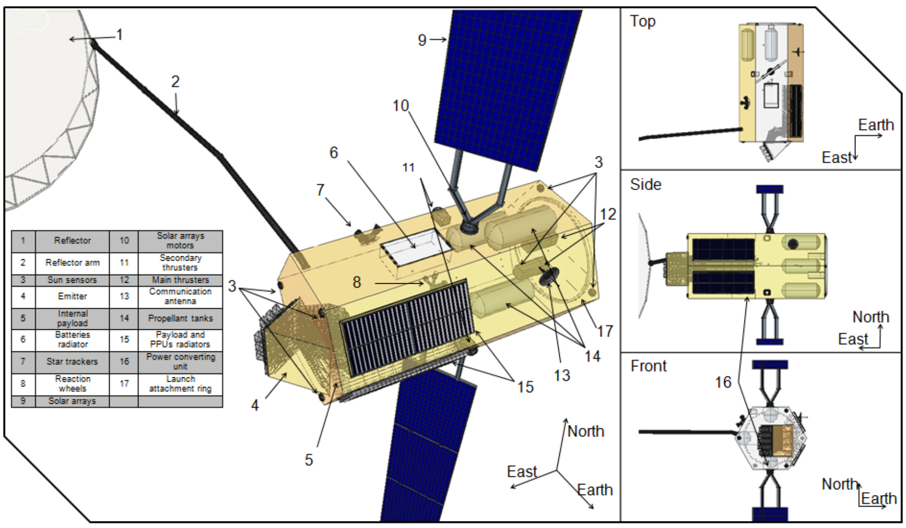


Figure B.11: Configuration of the payload

The Figure B.12 presents in more details the external and internal configuration of the body.



1	Reflector	10	Solar arrays motors
2	Reflector arm	11	Secondary thrusters
3	Sun sensors	12	Main thrusters
4	Emitter	13	Communication antenna
5	Internal payload	14	Propellant tanks
6	Batteries radiator	15	Payload and PPUs radiators
7	Star trackers	16	Power converting unit
8	Reaction wheels	17	Launch attachment ring
9	Solar arrays		

Figure B.12: Configuration of the complete GeoSAR spacecraft

B.9.2 Thermal subsystem

The thermal subsystem was mainly driven by the thermal requirements presented in Table B.16, and by the heat dissipation cycles of the components presented by Figure B.13. Figure B.14 presents the final thermal design.

Table B.16: Component thermal requirements

Component	Number	Maximum total heat power emitted (W)	Survival temperature Range (°C)		Operational temperature Range (°C)		Maximum time of operation
Batteries	2	503.7			10	40	72 minutes (discharge)
Payload	2	997.1			-30	75	Continuous
Propulsion PPUs	6	800.0	-40	70	-20	50	6 hours
PCU	1	264.0	-40	70	-20	50	Continuous
Sun Sensors	6	<1			-40	70	Continuous except during eclipses
Star trackers Heads	3	<1	-40	70	-30	60	Continuous
Star trackers Electronic Unit	1	8	-40	70	-30	60	Continuous
Inertial measurement Unit	1	24			-30	70	Continuous
Reaction wheels	4	50.7	-30	60	-15	60	10 minutes at full speed
Onboard computers	2	12	-40	75	-20	60	Continuous
Propellant tanks	3	0	20	50			None
Solar arrays	2	0	-200	130	-150	110	None

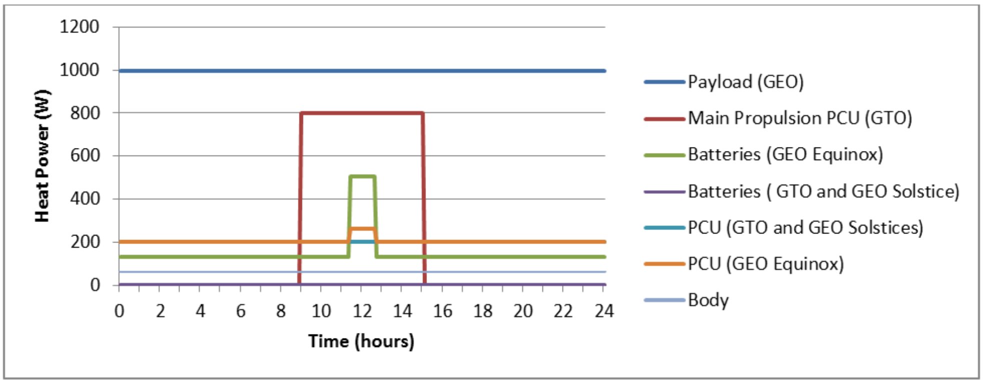


Figure B.13: Heat dissipation cycles, summary

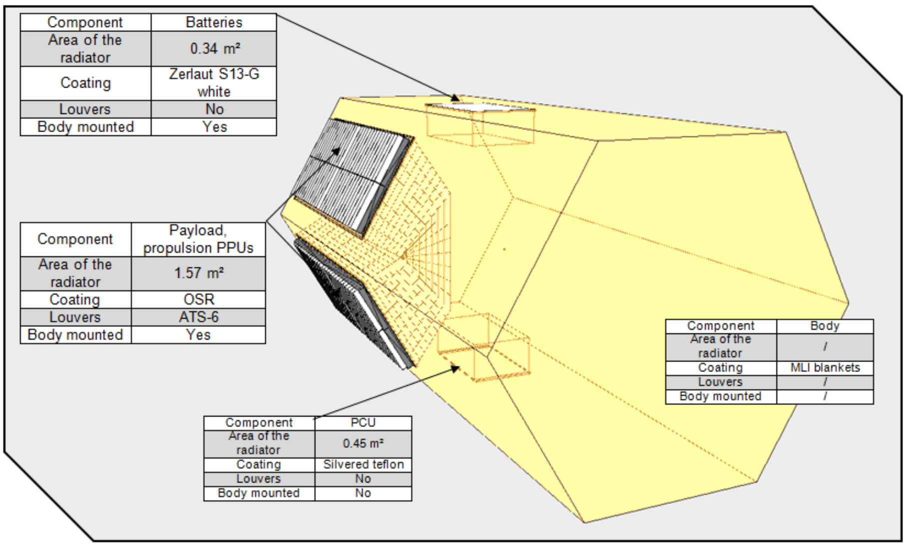


Figure B.14: Thermal subsystem, summary

B.10 Orbit Control: Guillaume Pineau

The synthetic aperture radar (SAR) technique has been used on low-Earth orbit (LEO) satellites to achieve a useful spatial resolution at Earth's surface. The main constraint of using this radar technology on LEO is the ground coverage. Because of orbit dynamics, a satellite with a global coverage cannot pass over a same point on the Earth's surface at intervals shorter than few days. By using the SAR technique aboard a geosynchronous orbit (GEO), imaging a same point on Earth's surface can be done every few hours.

The aim of this project is to explore the feasibility of employing space-borne SAR to acquire data from Geo-synchronous orbit through a mission concept.

The orbit control is required when the orbit elements needs to remains in a precise configuration. That is why this part of the project has been done with a lot of interactions with the nominal orbit part and the orbit perturbations part. This orbit allows using the SAR technology from a geosynchronous orbit. But because of orbit perturbations, those orbit elements evolve, and we need to keep them close to their nominal values.

B.10.1 Derived requirements

From ITU regulations about geostationary ring and from the SAR technology, we obtain the following derived requirements for the station keeping:

1. The orbit track shall remain within $\pm 0.9°$ of the longitude of the nominal position
2. The orbit track shall have a diameter of at least 80 km
3. The shape of the orbit track shall remain roughly circular
4. The drift of each position of the orbit track shall remain below 15 km
5. The duration and the frequency of station keeping manoeuvres shall optimise the time available for payload processing.

B.10.2 Orbit perturbations

The nominal orbit is defined considering a Kepler's model. In this model, the Earth is the only body acting on the satellite and is considered as a spherical and uniform mass. But other forces are acting on the spacecraft that cannot be neglected regarding the precision needed for the station keeping.

The main perturbations considered to evaluate orbit control are the Sun and Moon third body effect, the Earth triaxiality and the solar radiation pressure. The action of atmospheric drag on a spacecraft orbiting in GEO is too small compare to other forces to have a significant impact.

Luni-solar effect

The Luni-solar effect is due to the action of the gravitational field of the Sun and the Moon on the spacecraft. It affects the orbital plane, and leads to increase the latitude reached by the orbit track. The average secular change in inclination is approximately 0.9 degrees per year.

Earth triaxiality

The non-uniform distribution of mass of the Earth creates a longitude drift acceleration of the spacecraft. The value of this longitude drift acceleration is 1.21×10^{-3} ° day^{-2}.

Solar radiation pressure

The solar radiation pressure appears when electromagnetic radiations from the Sun interact with the surfaces of the spacecraft. Thus it creates a small acceleration of the satellite and change the eccentricity of the orbit. The perturbation is relatively small: $\Delta e = 0.005$ per orbit. Compare to the previous perturbations, the solar radiation pressure is small and will not have a major influence on the station keeping.

B.10.3 Orbit parameters control

Since the perturbations affect the orbit, the orbit track will not remain inside the boundaries defined by the requirements. Hence, a control of the orbit has to be performed to limit perturbation effects and keep the orbit track meeting the requirements.

Orbit elements and orbit track

The longitude and the latitude parameters of the orbit track are linked to the orbit parameters. So to control and keep the orbit track inside boundaries, we need to control the orbit elements.

Orbit parameters control

To control the orbit we need to control the inclination. To change the inclination, a burn is done perpendicularly to orbital plane, on the ascending or descending with a ΔV linked to change of inclination.

The control of the longitude uses the perturbation to perform a cycle and then remains inside boundaries with one single burn. The period of the cycle depends on the longitude drift acceleration and the longitude slot available.

Types of manoeuvre

The previous manoeuvre can be categorised in two types. The North-South manoeuvre controls the latitude drift of the orbit track, and the East-West manoeuvre controls the longitude drift of the orbit track.

B.10.4 Station keeping manoeuvres

Manoeuvre frequency

The manoeuvre frequency depends on the drift speed and on the slot available. Thus the maximum time interval allowed for North-South manoeuvre is 8 days, and the maximum time interval allowed for East-West manoeuvre is 12 days. We choose to perform both manoeuvres the same day every 8 days.

Types of propulsion

The propulsion used to make the manoeuvres is not a liquid but an electric engine. The low thrust can be used for station keeping because the ΔV per manoeuvre remains small enough.

Station keeping strategies

Since an electric engine is used, two strategies can be done. In the first one the manoeuvre frequency is the minimum allowed, which means 8 days. In the second one, the frequency is increased to keep a ΔV per manoeuvre small enough to consider an impulsive burn.

B.10.5 Summary

The final station keeping plan is:

Table B.17: Summary of station-keeping manoeuvres

Manoeuvre	Frequency	Drift range	ΔV per manoeuvre	ΔV m $s^{-1}yr^{-1}$	Burn duration per manoeuvre
North-South	8 days	0.018° (13.5 km)	1.1 m/s	45	50 min or 2*25 min
East-West		0.01° (8 km)	0.04 m/s	1.25	2.5 min

Figures B.15 and B.16 show that the manoeuvres can be performed on the same day and the orbit track over 8 days.

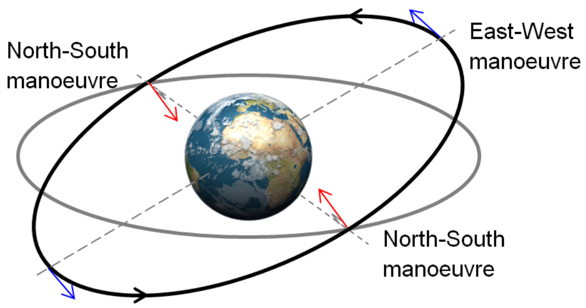


Figure B.15: Manoeuvre timing for orbit maintenance

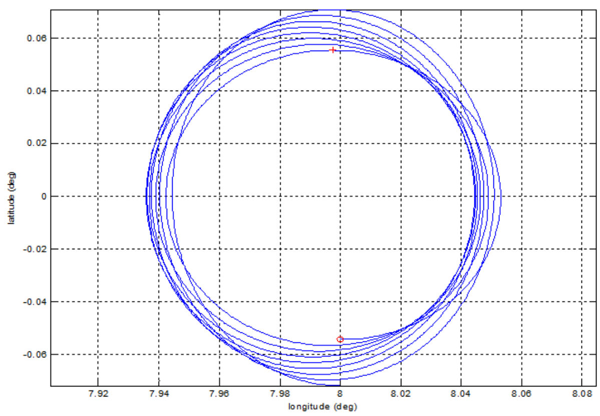


Figure B.16: Orbit drift over 8 days

B.11 Mechanisms & Payload Mechanical: Adam Scott

B.11.1 Introduction

The GEOSAR project was a feasibility study performed for a Group Design Project as part of the MSc in Astronautics & Space Engineering at Cranfield University. The concept was to determine the feasibility of an active SAR system from a geosynchronous orbit, additional to the 2006 study which used a passive SAR system.

The mechanisms & payload subsystem were investigated for this report, a mechanical configuration was determined and useful future work was highlighted.

Requirements and Derived Mechanism Design

The key requirements on these two subsystems derived from the top level requirements were the following;

- Operational Requirements

  Duration R5 15 years mission lifetime
  This means that the subsystems should be able to survive this duration at the orbit specified (GEO). Directly, this implies that (assuming sun-tracking solar arrays) the solar arrays must rotate 5474 times, and last for 15 years
  Data Content, Form and Format R2, R3 Land Surfaces Processes and Interaction
  This requirement was interpreted by the Payload subsystem into a wavelength of 2.3 cm (Hara, 2013). Based on this, a 7m diameter reflector was specified for mechanical design, and a surface accuracy requirement will be discussed.

- Constraints

  Cost R2 Under € 200M
  This constraint implies limitations of bespoke, one-off systems to save cost, as the cost of a launch is already nearly half of the allotted budget
  Schedule R2 ESA Earth Explorer Call
  Approximately 5 years for Research & Development from past missions, a high TRL is required. Additional to the requirements derived from top-level, the following requirements were specified by other subsystems

- Power

  Solar Array Area 27 m2
  Triple-Junction Solar Cells, generating 8.5 kW of Electrical Power and tracking the sun for MPP This requirement allows existing solar arrays to be analysed and a similar system to be specified

- Payload

  Reflector 7m diameter
  Ku-Band Reflector with Feed Horn Array This requirement specifies the minimum required density of the reflective surface and the surface accuracy (0.3-0.5mm RMS). This allows a reflector and feed array appropriate for the mission to be investigated

- Launch

  Launch Fairing Primary Falcon 9
  This determines the maximum mass, volume and dimensions for the spacecraft, as every part of the spacecraft (and payload) must stow into this envelope. It also means that any applicable mating hardware must be acceptable for this launch vehicle. The requirements were investigated and it was determined that each is feasible, the most difficult to meet will be the Ku-band surface accuracy requirement with a 7m diameter reflector, so most of the research was focused on reflectors.

B.11.2 Mechanisms

Clamp Band

A clamp band was simply specified for completeness, a 937 mm RUAG design was selected, which has a 100% successful space heritage and is compatible with the primary launch vehicle (Falcon 9).

Solar Arrays

Solar arrays were briefly investigated for feasibility, as the spacecraft will not require more power than a large communication satellite, a system from an existing satellite was specified to save design time and cost. This system is from the SICRAL spacecraft, and the length of the panels was required to be extended slightly to accommodate enough solar cells. This was expected to come with appropriate drive motors (SICRAL is in GEO currently), but RUAG SEPTA24 drive motors were referred to in case they are not included, weighing 6kg each.

B.11.3 Payload Mechanical

This focused on finding a reflector, which could meet the surface accuracy requirements.

Inflatable Antenna Experiment/Echo Balloons

These were determined not to have an appropriate surface accuracy in the current state, when rigidization methods have been flight-proven they should be reassessed.

Hybrid Inflatable-Fixed

Hybrid inflatable antennae show promise, however they suffer from the same issues as other inflatables to do with rigidization. Also, this mission requires a large reflector, a small one will not be able to perform the desired science; the concept is more suited for communications, where deployment failure will not end the mission.

Harris/Astromesh

This was the selected design, due to the high TRL and the availability of information and deployment reliability. Further details are provided below

Solid Folding

Solid reflectors enjoy very high surface accuracy, however this usually comes with a high mass as well. Unfortunately, current solid reflectors suffer somewhat from thermal effects and apertures of the size required have not been flight proven in the GEO environment.

B.11.4 Baseline

The baseline for the payload mechanical subsystem is shown below.

Table B.18: Astromesh Reflector and Boom Assembly

Characteristic	Value
Diameter	7m from NADIR
Mass	80 kg
Stowed Dimensions	1m dia x 1.5m reflector
	4m x 0.3m dia for the stowed boom
Surface Accuracy	+/-0.2 mm
Cost (est)	€ 15 million

Table B.19: Feed Array Summary

Characteristic	Value
Feed Horns	ATM WR62 (21)
Mass	10 kg
Dimensions (mm)	260 length x 300 depth x 200 height

B.11.5 Conclusion

This report concludes that from a mechanical point of view, SAR can be performed from a Geosynchronous orbit in order to monitor land subsidence events as defined by the requirements.

B.12 Power System and Payload Electrical Design: Colin Shirran

The project being discussed within this report is the utilisation of a Synthetic Aperture Radar (SAR) in a geosynchronous orbit to monitor land surface deformations in urban areas over Europe and North Africa. This project will also be designed to be compatible with the ESA Earth Explorer program.

The aim of the GeoSAR mission: "To explore the feasibility of employing a space-borne Synthetic Aperture Radar to acquire useful scientific data from Geosynchronous orbit"

As I was involved in two separate work packages, this report will cover both the electronic power system design and the SAR payload electrical design. Since the payload will have a large impact on the power system design, it shall be described first.

B.12.1 SAR Payload Electronics

Requirements for the design of the payload electronics have been given to the system from the payload requirements group. They are:

- An average transmitted power of 1 kW
- A duty Cycle of 10%
- Relatively low cost and therefore a high TRL
- 19 spot beams

Using the duty cycle, the peak power required form this system is 10 kW. This will have to be raised by using components of a large flight heritage and therefore a high TRL.

Types of SAR Systems

There have been two main types of SAR systems researched in this project. Those which use a planar active phased array antenna, type 1, and those which use an emitter that illuminates a parabolic reflector, type 2. However, the first system type is incompatible with GeoSAR and a parabolic antenna is needed. However, there is a design which utilises a planar phased array to illuminate a parabolic antenna instead of the conventional feed horn array and TWTA configuration. The two contending system types can be seen illustrated in Figure B.17.

After the two systems were considered a trade-off analysis was performed to determine the best option for GeoSAR. This trade-off was accompanied by a sensitivity analysis which showed that the best option for GeoSAR was the parabolic antenna being illuminated by a feed horn array, in any case.

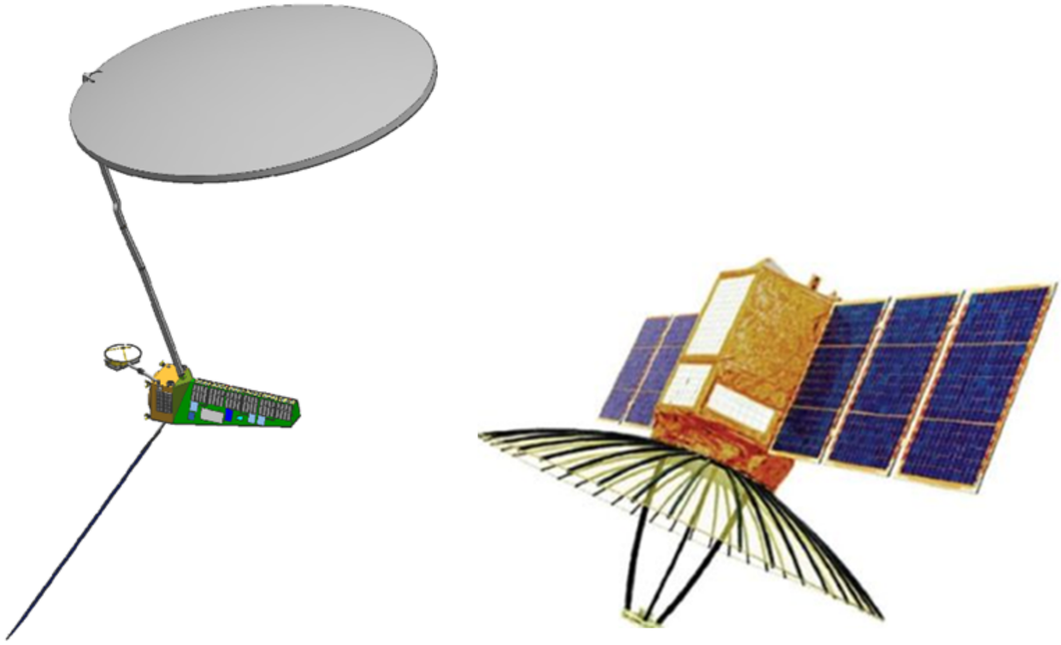
The final power and mass estimate of the system was then shown to be:

- Power = 2500 W
- Mass = 371 kg

B.12.2 Electrical Power System Design

The requirements for the power system have been derived from the top level requirements and other system requirements. The Power System Shall:

1. Be capable of supporting the power requirement from the SAR payload during its operation, including during the eclipse
2. Be capable of supporting electronic propulsion during its operation
3. Support the power requirements of the spacecraft during each phase of the mission
4. Be capable of surviving a 15 year lifetime



(a) DesDynI (b) TecSAR

Figure B.17: Two example recent SAR mission designs

Solar Array

As it has been decided that GeoSAR will use electronic propulsion, there is quite a high power demand put on the system. This means that there is a large difference between the power demand in the operational phase and the power demand during the ascent and orbit maintenance phases.

- Sunlight portion during eclipse season = 4162.5 W
- Eclipse portion during eclipse season = 4169.4 W
- Ascent to GEO = 8.8 kW
- Orbit Maintenance = 8.7 kW

This use of electric propulsion also introduces another constraint on the system in the form of power production. As the engines can only be fed power from the solar arrays during orbit raising manoeuvres, due to the time involved at such a high power, the batteries cannot be designed to make up the difference between solar array output and power demand during this time. Therefore an array capable of producing 8.8 kW at BOL is needed.

This has therefore led to a 27 m^2 array design which uses multi junction GalnP/GaAs/Ge at 28% efficiency, which is oversized for the majority of its lifetime.

- BOL Power (23.5° Sun Angle) = 8.3 kW
- BOL Power (0° Sun Angle) = 9.3 kW
- EOL Power (23.5° Sun Angle) = 6.3 kW

Batteries

The energy storages system is a Li-Ion battery with a total capacity of 13.5 kWh. This will then be split into two packs of each being half of that capacity. This battery capacity is oversized for the operational phase and results in a DOD of 33% after eclipse. The main reason for the over sizing is because of the battery configuration supplied by SAFT. However, it allows for cell degradation, complying with the redundancy requirement to last the 15 years at GEO. It will be made up of 96 cells and weigh 119 kg with a 20% margin.

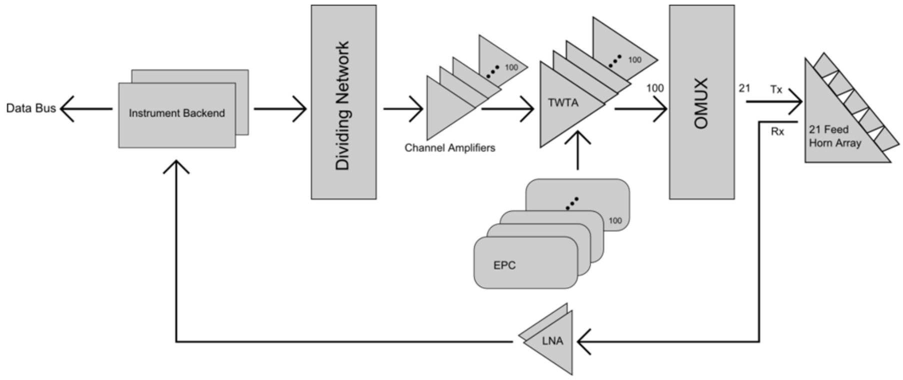


Figure B.18: Schematic of the final payload electrical design

Power System Architecture

It has been decided that the power system will use a fully regulated bus using the direct energy transfer method. This has been found to be the most efficient choice for large power satellites with long lifetimes. The bus will be 100 V to supply power to the high voltage components and also a 28 V bus to power the lower voltage components.

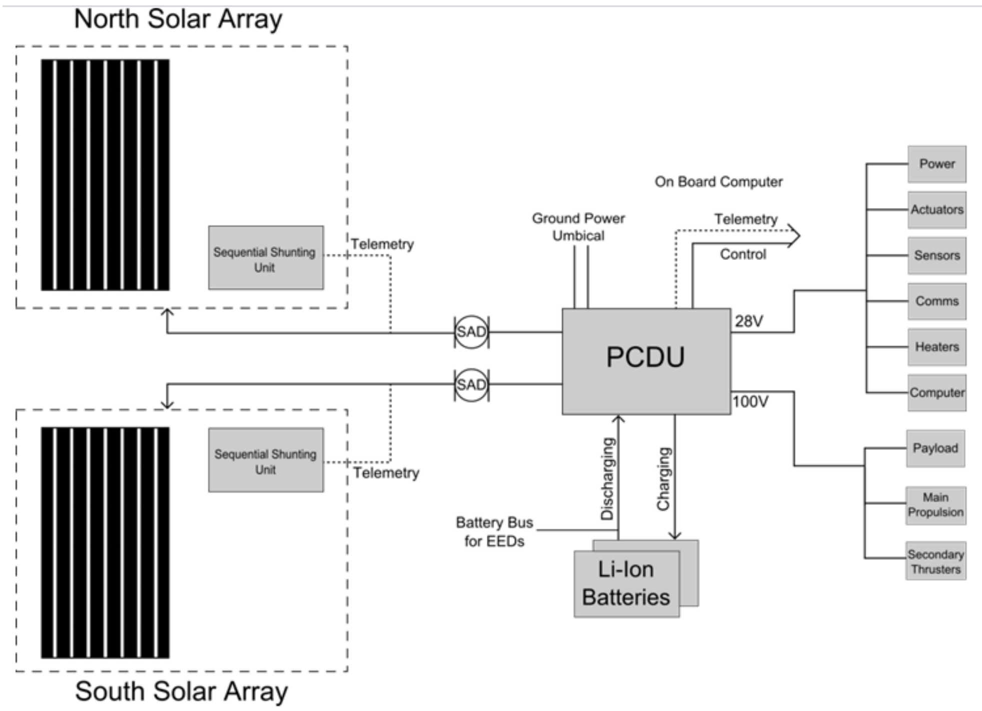


Figure B.19: Power sub-system overview

B.13 AOCS: Nicolas Vinikoff

The GeoSAR project ADCS workpackage has been first concerned with requisites investigation and derivation. The second step has been to evaluate the available options to an ADCS engineer in terms of control strategies and hardware with respect to the mission demands and to select the appropriate combination of actuators and sensors (part of the picked control scheme implementation). Finally, the arrangement of these instruments has been considered.

B.13.1 ADCS Modes and Requirements

The different identified ADCS modes are

- **GEO Transfer Mode (GTM)**
- **Attitude Acquisition Mode (AAM)**
- **On-Station Mode (OSM)**
- **Emergency Mode (EM)**

The driving modes are the GTM and the OSM and their requirements are presented

GTM

- Attitude error within $\left[\mathbf{1.8 \times 10^{-2} deg}, \mathbf{3.4 \times 10^{-2} deg}\right]$ for firing periods
- Remove the initial body rates in less than one hour
- $\mathbf{2.56}$ **deg** accuracy for Sun tracking

OSM

- Pointing accuracy of at least $\mathbf{2.6 \times 10^{-2}}$ **deg**
- Attitude knowledge between **8.05 arcsec and 14.95 arcsec**
- Slew rate of at least $\mathbf{5 \times 10^{-3}\ deg.s^{-1}}$

B.13.2 ADCS Hardware

After control methods evaluation with respect to the demanded accuracy levels, a 3-axis zero-momentum stabilising strategy has been selected. The hardware (sensors and actuators) to support this scheme is presented in Table B.20.

B.13.3 ADCS Configuration and Hardware Use

Actuators: One wheel per axis and one redundant unit tilted to give equal torque generation over all three axes. They are used in GTM for detumbling and attitude control (during the Ascent Phase); used in OSM for attitude control during imaging and for manoeuvres (re-pointings and SK) Two thrusters generating simultaneously a torque about the X- and Y-axis. Used for detumbling and reaction wheels unloading.

Sensors:

- Six Sun sensors in a complete 4π sr coverage layout for attitude determination and Sun tracking in GTM and OSM.
- Three star trackers (2 heads out of three can be bloomed) in an arrangement allowing for blooming management and attitude determination.
- Two IMUs with one being a redundant unit.

B.13.4 ADCS Results

The results have been obtained by simulation for the detumbling operations and by calculations for manoeuvres (SK and re-pointings) and reaction wheels dumping

- **The slew rate shall be at least 5×10^{-3} deg.s^{-1} for SK** The result is a 90-degree yaw rotation in 7 minutes i.e. a slew rate of 2.1×10^{-1} $deg.s^{-1}$
- **The re-pointing shall be performed in less than 1 minute** The result is presented in Figure B.23. It can easily be noticed that the times are meting the requirement
- **The detumbling shall be completed in less than one hour** Simulation shows that detumbling can be performed in 800 s even with 10% of the maximum torque capability.

B.13.5 Conclusion

The objective of this report has been to prove the mission feasibility from the attitude and determination control perspective. The proposed ADCS is then only a preliminary design in its infancy and further work remains to be done. It includes flexible appendages dynamics and behaviour detailed analysis, environmental disturbances and requirements better characterisation. It would be beneficial to the mission design and lead to a detailed design.

Space-borne geosynchronous Synthetic Aperture Radar for interferometric urban subsidence monitoring appears to be achievable though. A sensors and actuators suite has been selected, sized and arranged and it is believed that this selection can meet its specific requisites.

The major aspect to consider for future work relates to manoeuvres impact on imaging resumption due to the flexible payload structure.

Sensors	Actuators
Sun sensors (×6)	Reaction wheels (×4, 1 redundant)
Star trackers (×3)	Thrusters (×2)
IMUs (×2, 1 redundant)	

Table B.20: ADCS hardware.

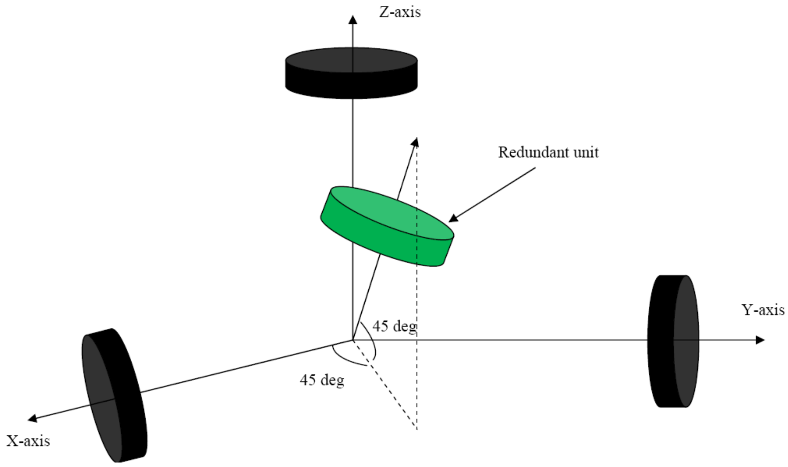
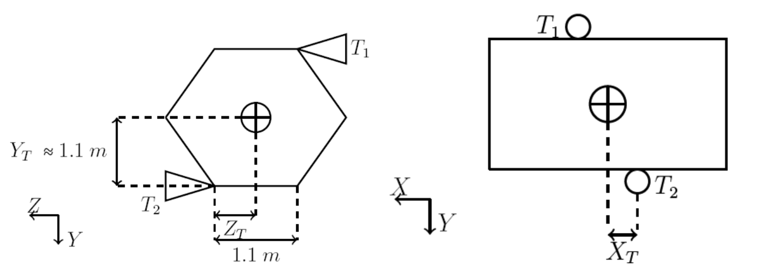


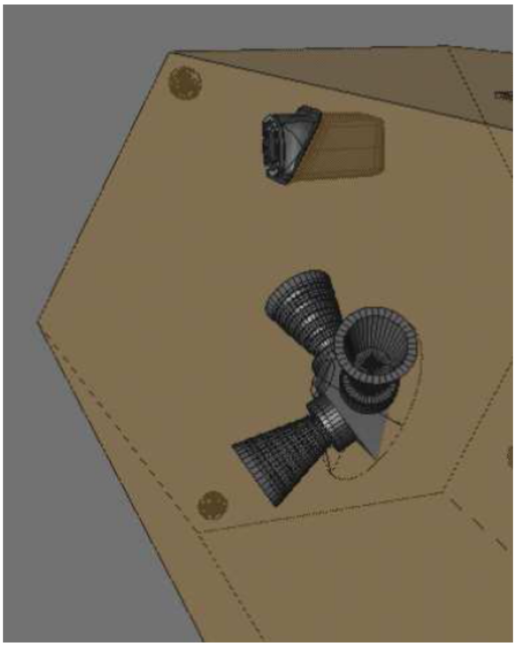
Figure B.20: Reaction wheel arrangement.



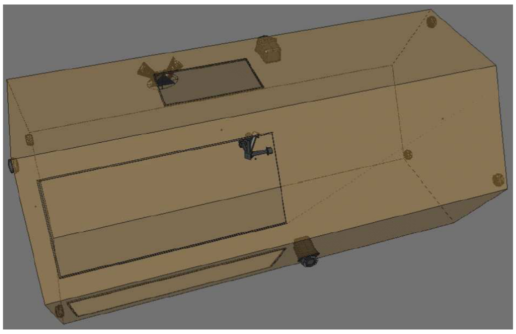
(a) Thruster configuration (S/C back view).

(b) Thruster configuration (S/C side view).

Figure B.21: Thruster configuration.



(a) Star trackers.



(b) Sun sensors.

Figure B.22: Sensors layout (courtesy of Nicolas Petitpas).

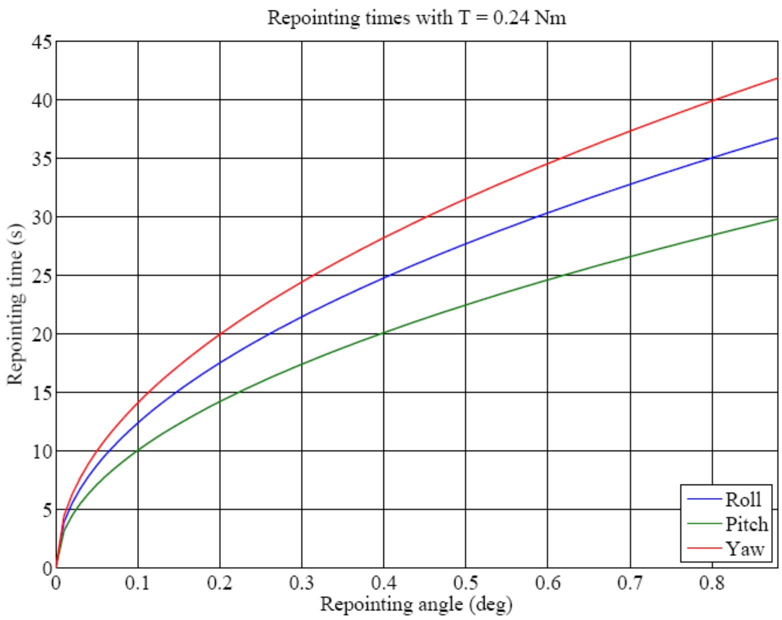


Figure B.23: Time for 0.88 deg slew with 0.24 $N.m$ of torque.