Laplace Plane GeoSAR Feasibility Study

Summary of the Group Design Project
MSc in Astronautics and Space Engineering 2014–15
Cranfield University

College of Aeronautics Report SP003

Dr. Stephen Hobbs
Cranfield Space Research Centre
Cranfield University
Cranfield
Bedford MK43 0AL UK
s.e.hobbs@cranfield.ac.uk
http://www.cranfieldspace.co.uk/

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Abstract

Students of the MSc course in Astronautics and Space Engineering 2014–15 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 geosynchronous synthetic aperture radar Earth observation mission using the Laplace orbit plane to reduce station-keeping propulsion demand. User applications are drawn from a wide range of sectors (agriculture, meteorology, geohazards, etc.) and are translated into system design requirements. The proposed mission design uses satellites with 13 m diameter antennas and a total electrical power demand of 6 kW.

The mission seems feasible, although further study is recommended especially for the areas of

- orbit selection with respect to user requirements, imaging performance and orbit maintenance,
- mass budget (driven largely by the propulsion system),
- user requirements, imaging performance and operational imaging modes,
- opportunities for improved imaging with a constellation.

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
SP003	2015	Laplace Plane GeoSAR Feasibility Study, Summary of the Group Design Project, MSc in Astronautics and Space Engineering 2014–15

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 (Frederick Alapa, Robert Arthur, Ezra Clarke, Henrique Daitx, Daniel Erkel, Dorian Gegout, Ilji Jang, Oscar Jennings, Pranav Keskar, Eric Li, Moreno Peroni, Jesus Quirce-Garcia, Eddie Ross, Carlos Roldan Rueda, Adrian Segura, Chris Slattery, Andrea Testore and Alex Trolley), who have each contributed about 600 hours.

An important reference for the project was provided by Prof. Andrea Monti Guarnieri (Politecnico di Milano, Italy) and Prof. Geoff Wadge (University of Reading, UK) from ESA's GeoSTARe project in which Cranfield was also a partner. Other members of staff in Cranfield's Space Research Centre, in particular Drs Jenny Kingston and Joan Pau Sanchez, and contacts in the space industry have helped students by responding to queries or providing technical information. The time spent, help provided, and general encouragement is greatly appreciated.

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Chapter 1

Introduction

This report summarizes a group project of the MSc in Astronautics and Space Engineering for the academic year 2014–15 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 2014–15 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 10 000 hours' work (about 6 years) for the academic year 2014–15.

Students are given responsibility for all technical aspects of the mission and over the 6 months of the project are required to evaluate the concept's feasibility and 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

Figure 1.2 shows the general imaging concept assumed for the mission. This involves the radar operating in spotlight, squint or stripmap mode, with integration times from a few minutes upwards.

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	Trolley (2015)
	Baseline, budgets, cost	Jennings (2015)
	Operations	Daitx (2015)
	Software	Peroni (2015)
WP2000 Mission	Launch	Ross (2015)
	Orbit	Li (2015)
	ADCS	Arthur (2015)
	End of mission	Gegout (2015)
	Constellation	Jang (2015)
WP3000 Mechanical	Configuration and structure	Roldan Rueda (2015)
	Thermal	Erkel (2015)
	Mechanisms	Quirce-Garcia (2015)
WP4000 Electrical	Power	Daitx (2015)
	OBDH	Peroni (2015)
	Communications	Segura (2015)
WP5000 Payload	Requirements, operations	Alapa (2015)
	Payload performance	Slattery (2015)
	Payload mechanical	Quirce-Garcia (2015)
	Payload data	Keskar (2015)
	Opportunity payload(s)	Clarke (2015)
	Payload constellation	Testore (2015)

A particular feature of the Lagrange Orbit GeoSAR is that the 7.5° inclination orbit has long-term stability with respect to orbit perturbations from Sun and Moon gravity. This should reduce the orbit maintenance cost significantly and permit an extended mission lifetime, which in turn promises reduced mission cost. A goal of this year's project is to investigate the feasibility of this mission concept and to understand which factors limit the mission lifetime and cost-effectiveness.

A geosynchronous SAR in the Lagrange orbit is likely to need a large antenna and precise orbit control to enable interferometry. The expected challenges include:

- Extending the satellite lifetime well beyond 15 years poses new technical challenges
- Large lightweight antennas are difficult to design to achieve high reliability
- Attitude and orbit control face challenges due to the unusual configuration (and therefore potential disturbance torques) and the demand for precise orbit control to permit interferometry
- User requirements need to be assessed and translated into appropriate system requirements while achieving a design which is versatile enough to adapt to changing requirements over the mission lifetime
- The operational concept needs to integrate the constraints of synthetic aperture radar imaging and the Lagrange orbit with user applications and an appropriate ground segment architecture.

1.2 Structure of this report

Following this Introduction, Chapter 2 and Appendix B 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

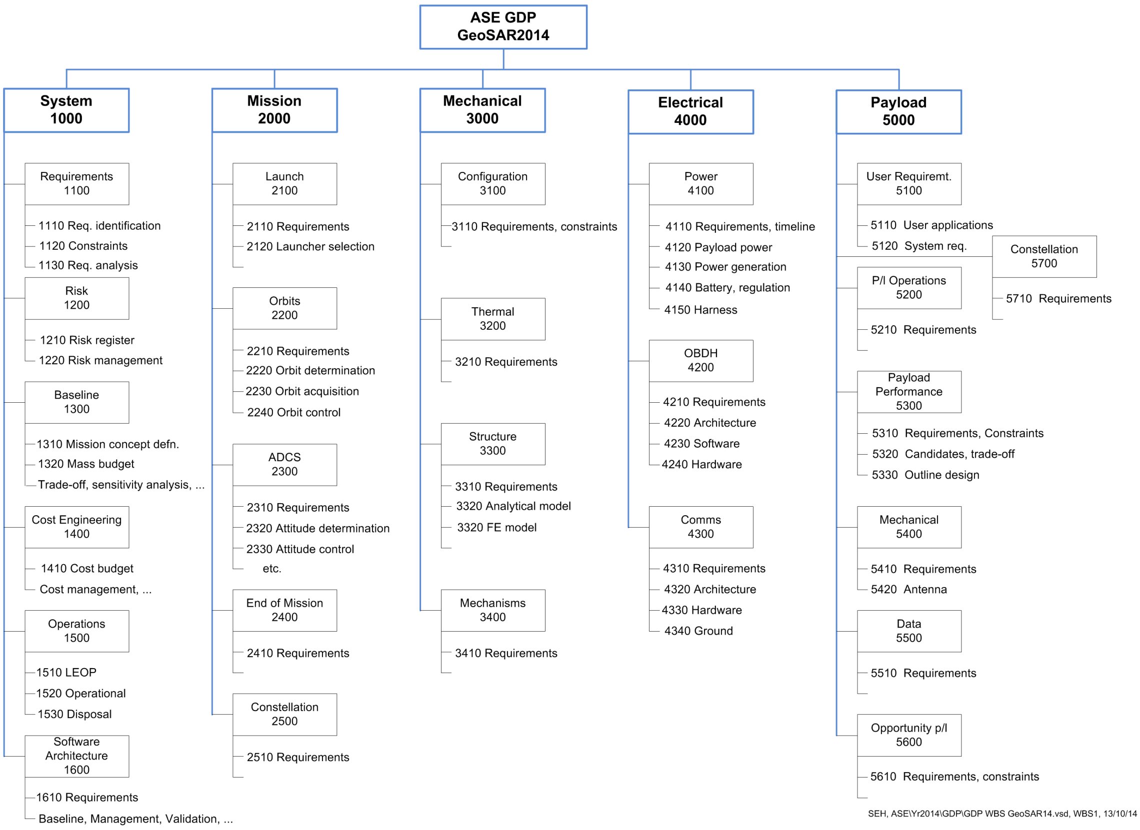


Figure 1.1: Work breakdown structure for the GeoSAR study.

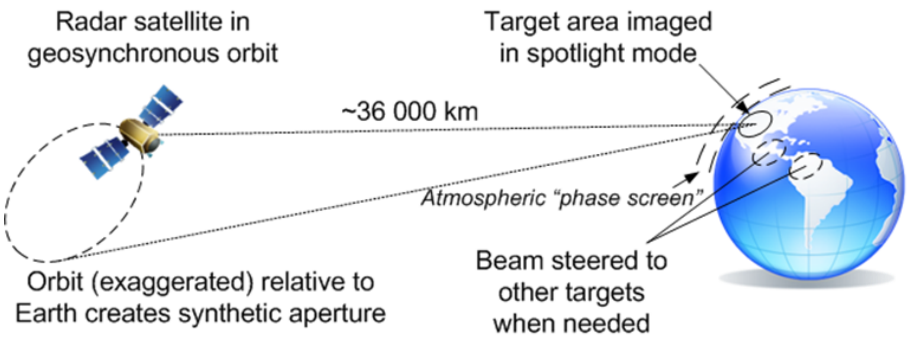


Figure 1.2: Overview of the mission concept for geosynchronous SAR (spotlight mode imaging).

further work. The main content of the report is Appendix C 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 course administrator for the MSc in Astronautics and Space Engineering, Cranfield University, and are summarised in Appendix C. *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.*

Chapter 2

Technical Discussion

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

2.1 Requirements Analysis

The initial set of requirements for the project was:

- MR1 Perform SAR and InSAR imaging from geosynchronous orbit
- MR2 Mission lifetime is 50 years
- MR3 Operational user requirements (specific requirements Wadge, 2014)
- MR4 Cost-effective with respect to ESA Sentinel missions

Constraints imposed on the mission are:

- MC1 Use the drift-free geosynchronous orbit with inclination of 7.5°
- MC2 European technology should be used where available
- MC3 Use currently available technology or technology at high TRL
- MC4 Consistent with current space debris mitigation guidelines

Requirements relate directly to the service provided by the system to the user; constraints are other requirements which tend to restrict the designer's freedom.

2.1.1 User Requirements

Table 2.1 from Wadge (2014) summarises the user requirements for the GeoSTARe study recently carried out for ESA. These form the core requirements assumed for the Laplace plane GEO SAR mission. Some additional applications considered relate to monitoring extraction processes such as in oil fields and for ground water.

These user requirements are translated into radar payload performance requirements using two key relationships. The first relates azimuth resolution to synthetic aperture length, which itself is determined by the integration time and orbit speed. The second relates measurement quality (e.g. InSAR phase uncertainty) to radar image SNR, number of looks and spatial resolution, to then estimate parameters such as transmitter power and antenna diameter. Once this is done (often iterative, to find a "reasonable" set of parameters), figures such as Figure 2.1 can be drawn to visualise power demand as a function of time for each application, and potential for using several spot beams. The total power demand is the highest individual demand rather than the sum, since one set of raw data can be processed to different spatial resolutions, to be used for multiple applications. The orbit speed is quite high in the Laplace plane, so that integration time is short and many spot beams can be used in sequence to provide much greater coverage (Figure 2.2).

2.1.2 Requirements Discussion

MR1 and MC1 overlap and are related to the constraint to use the drift-free geosynchronous orbit. The drift-free orbit offers the possibility of very low fuel consumption and therefore an extended mission lifetime. A mission life longer than the 15 years usually designed for in GEO should improve the mission economics. Simple estimates of the expected fuel consumption (say 10 m s^{-1} per year instead of 50 m s^{-1} per year for a geostationary orbit) suggest that tripling the mission lifetime from 15 yr to around 50 yr is possible (hence MR2). As for most requirements in a feasibility study, the mission lifetime is negotiable.

However, a long lifetime implies several other mission aspects which need to be evaluated. These include:

- A long duration mission should provide an operational service: it is not suitable for technology demonstration
- Sub-systems and components need to be able to function for extended periods, including mechanisms, electronics, power raising
- Designing the mission for graceful degradation seems wise, as a means to manage the risk of sub-system failures
- The mission concept needs to be adaptable to changing demands: it is unlikely that there will be no change in operational needs over periods of several decades

However, the main aim of the study is to evaluate the feasibility of the "drift-free" inclined geosynchronous orbit and therefore the mission design should exploit this. Its main advantage is the low propulsion demand and so any other mission features which negate this (e.g. high propulsion demand for station-keeping or orbit reconfiguration) should be challenged.

2.1.3 Final Mission Requirements

The set of requirements which was finally agreed for the mission is given in Appendix B.

2.2 Operational Concept

The operational concept links useful applications with the constraints of the engineering solutions to provide the best service to users.

Figure 2.2 shows a representative instantaneous beam footprint for an array of 19 spot beams (19 beams fill the region within 2 beamwidths of the antenna axis, antenna feed displacements should be much less than the focal length to avoid distortions, Skolnik, 1970, p 10-10). Only one or a few of these beams may be in use at any time, to manage the power and data bandwidth requirements. These beams, especially in X-band, need to be steered to give useful coverage of areas of interest. It may be that the L-band beams can be steered simply by switching electronically between the available spot beams.

2.2.1 Atmospheric Phase Screen Compensation

The integration time may be a few minutes or more to achieve the required azimuth resolution. If t_{int} is longer than 2–3 minutes then atmospheric phase screen correction is likely to be needed to focus the image. This adds to the complexity of data processing (on the ground) but is not a fundamental problem. For simplicity, the current mission design assumes that images are acquired fast enough that no compensation of atmospheric phase changes is needed. This is a significant assumption and should be reviewed in the next design iteration.

2.2.2 Multiple Satellite Options

To achieve a 50 year mission lifetime the most likely scenario seems to be that individual satellites last for 20–30 yr, and so several overlapping satellite missions would be needed (Figure 2.3). This raises a wide range of options for collective use of the satellites.

- Global coverage
- Local cluster to improve area coverage
- Tight local cluster to operate cooperatively
- Temporary (potentially) constellation formed as one satellite is deployed to take over from another

These options have not been studied in depth but they raise a surprising number of different mission scenarios.

2.3 Mission Design Assumptions

In addition to the defined user requirements, some assumptions were made early in the project to enable the study to proceed. Having completed one design iteration, these assumptions should be reviewed since they have had a significant impact on the design. Two assumptions in particular deserve further consideration:

- Atmospheric Phase Screen compensation The choice to require image integration time to be no more than a few minutes implied a need for high orbit speed, and thus relatively high orbit eccentricity. This in turn increases the propellant demand for station-keeping and end-of-life disposal. If active phase compensation were accepted then the propellant demand would be reduced, which would reduce mass and / or extend mission lifetime, both of which tend to lower mission cost. The additional complexity of phase compensation affects the ground data processing and is not expected to be a fundamental problem for the mission. Phase estimation is required anyway for some of the data products, so it is likely that it will be used whether or not it is required to form the primary image products.
- Chemical vs Electric propulsion To reduce the delay from launch to mission deployment, the decision was made to use chemical rather than electric propulsion. Chemical propulsion has the advantage of being relatively reliable also, but it does imply a large mass of propellant. Since the satellite will have large solar arrays for the radar, there seems little negative impact for using electric propulsion apart from the slower deployment and there would almost certainly be a significant mass reduction. It may also be possible to design the insertion manoeuvre so that a shared launch to GTO could be used (lower cost than a dedicated launch) to be followed by a low thrust GEO insertion which also manages the inclination change to the Laplace plane.

Table 2.1: User requirements based on the GeoSTARe study

(a) L-band, phase

Application	L / m	t_{repeat} / hr	δl / mm	Rank	Comment
APS	2000	0.25	10	1–3	
Snow mass	200	2	10	4	
EQ interseismic	100	12	10	6	
EQ response	100	6	10	7	

(b) L-band, backscatter

Application	L / m	t_{repeat} / hr	NEσ^0 / dB	Rank	Comment
Snow cover	200	12	-23	11b	
Agriculture	100	12	-18	12b	
Hydrology	1000	1	-18	13	

(c) L-band: Coherent change detection

Application	L / m	t_{repeat} / hr	Rank	Comment
Volcano,intra	100	12	5	
EQ response	100	6	7	

(d) X-band, phase

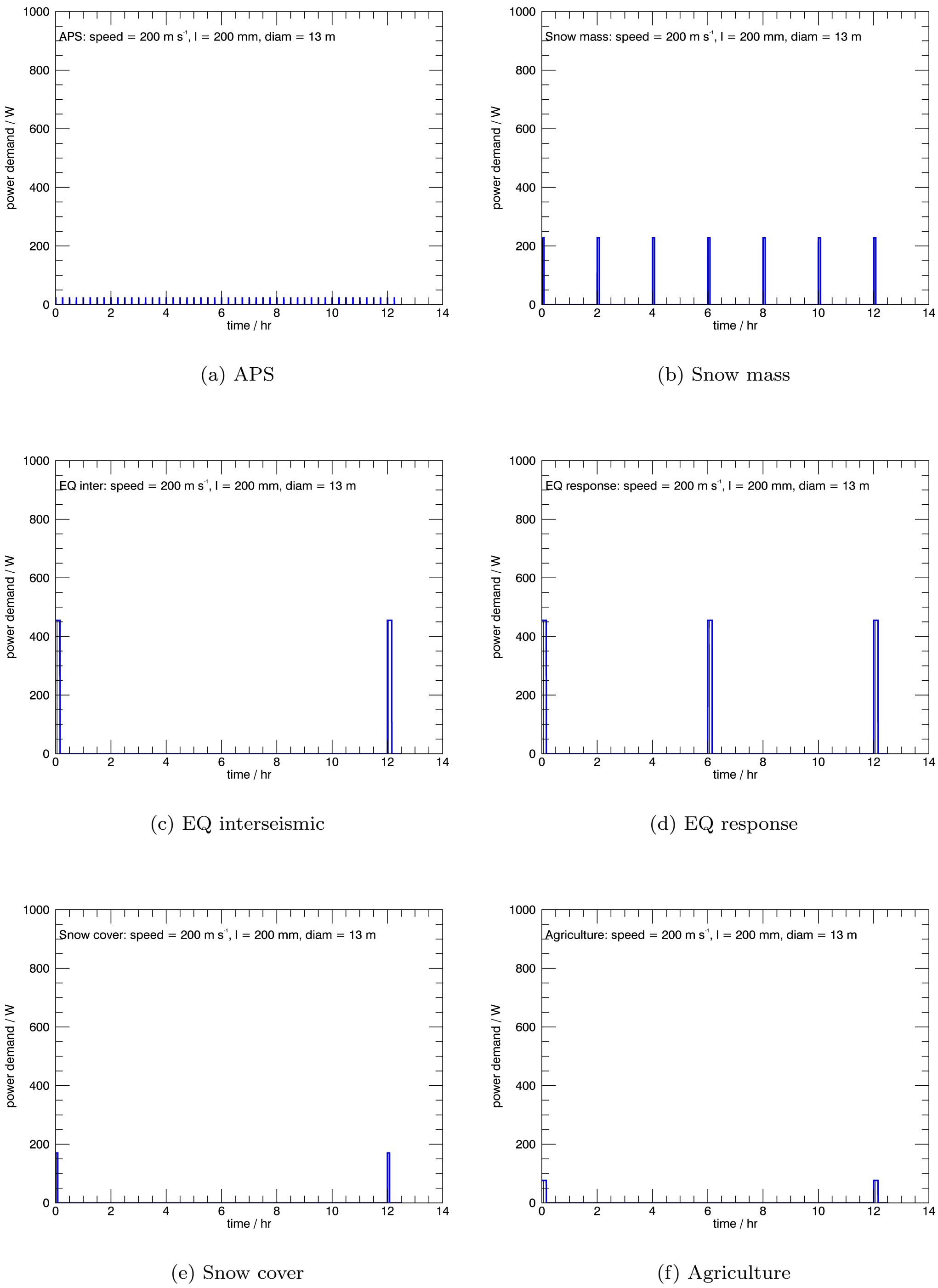
Application	L / m	t_{repeat} / hr	δl / mm	Rank	Comment
Volcano (intra)	20	3	10	5a	
Landslide (triggered)	20	6	2	10	
Subsidence	10	12	2	15	
Glacier	20	12	1	16	

(e) X-band, backscatter

Application	L / m	t_{repeat} / hr	NEσ^0 / dB	Rank	Comment
Flooding	30	2	-14	8	
Snow cover	50	2	-14	11a	
Agriculture	50	3	-14	12a	

(f) X-band: Coherent change detection

Application	L / m	t_{repeat} / hr	Rank	Comment
Flooding	30	2	8	
Volcano, hazard	20	3	9	
Landslide, triggered	30	6	10	



(a) APS

(b) Snow mass

(c) EQ interseismic

(d) EQ response

(e) Snow cover

(f) Agriculture

Figure 2.1: Example RF power demands and duty cycle for each application with 3 spot beams in sequence (L-band, 13 m diameter antenna, orbit speed 200 m s^{-1}). These figures suggest that more spot beams could be used and that the mean power demand should be modest.

(a) X-band ($\lambda = 3$ cm)

(b) L-band ($\lambda = 23$ cm)

Figure 2.2: Representative instantaneous beam footprints for a 13 m diameter antenna in geosynchronous orbit.

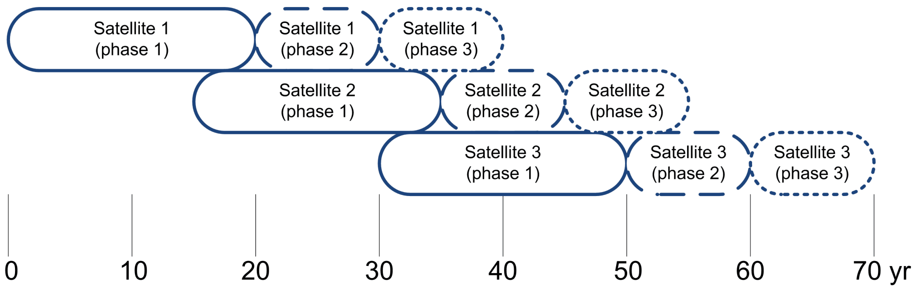


Figure 2.3: Hypothetical mission phasing showing overlap between satellites with different capabilities (solid border = fully functional, dash = partial transmitter failure, fine dash = full transmitter failure).

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

Orbit choice is not a trivial decision. There is a clear trade-off between increasing eccentricity to have a higher East-West speed and therefore shorter integration times, and the cost of orbit maintenance. If it is accepted to use active phase compensation then longer integration times can be used and less eccentric orbits are usable. Other aspects of the orbit choice are (a) a shared launch to a Laplace plane orbit is unlikely since there are few other users of this orbit, and (b) the expected mission lifetime relates to the annual station-keeping fuel budget and thus the choice of orbit.

The applications identified in the GeoSTARe study (Wadge, 2014) cover a broad range. Now that an outline mission design has been achieved, the application requirements should be reviewed to see whether a better match between requirements and system performance can be obtained at lower cost.

The assumptions made to allow the study to proceed should be reviewed. Two significant assumptions relate to (a) the choice of chemical rather than electric propulsion, and (b) the requirement for a short enough integration time to avoid the need for routine atmospheric phase screen compensation.

Constellation design is a large topic only touched on in this study. Its potential benefits and the wide range of options available suggest that more work in this area is needed.

3.2 Conclusions

From the study presented it seems that the mission concept is broadly feasible, although it is clearly only a first iteration.

The large number of applications is a strength, but to progress the mission design it is recommended that stronger prioritisation is used. Mature application proposals should be developed around the most promising applications so that a strong, coherent mission proposal can be developed. Once the priority applications are demonstrated it is likely that others will follow, including quite possibly several which have not yet even been imagined.

3.3 Future Work

Several topics are suggested as priorities for the next iteration of the mission design:

- Optimisation of the orbit choice,
- Use of electric propulsion as an alternative to conventional chemical propulsion,

- Review (prioritised) user applications and mission design to find improved design solutions,
- Consider the costs and benefits of accepting atmospheric phase compensation for routine image focussing,
- More thorough understanding of the satellite lifetime / mission design trade-off,
- Evaluation of the options for constellation design if two or more spacecraft (perhaps with different capabilities) are available in the Laplace plane orbit.

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Appendix A

Spot Beam Footprint Geometry

This appendix outlines the geometry for calculating the footprints of multiple spot beams (e.g. Figure 2.2).

Assume the central beam unit vector is $\mathbf{e}_0$ from the satellite towards the Earth and a unit vector parallel to the Earth's spin axis is $\mathbf{e}_N$. From these, orthogonal unit vectors perpendicular to the beam axis can be defined: $\mathbf{e}_1, \mathbf{e}_2$.

$$\mathbf{e}_1 = \alpha(\mathbf{e}_N - \mathbf{e}_0(\mathbf{e}_N \cdot \mathbf{e}_0)) \tag{A.1}$$

$$\alpha = 1/\sqrt{1 - (\mathbf{e}_N \cdot \mathbf{e}_0)^2} \tag{A.2}$$

$$\mathbf{e}_2 = \mathbf{e}_0 \times \mathbf{e}_1 \tag{A.3}$$

These can be used to calculate unit vectors for the axis of each of the n_B spot beams offset by angle a from the central beam axis.

$$\mathbf{e}_{ai} = \cos a\ \mathbf{e}_0 + \sin a \left(\cos(\theta_0 + i\frac{2\pi}{n_B})\mathbf{e}_1 + \sin(\theta_0 + i\frac{2\pi}{n_B})\mathbf{e}_2 \right) \tag{A.4}$$

This new beam axis can be used with the unit North vector to define another set of orthogonal unit vectors for each spot beam. Equation A.4 then gives a set of directions around this axis. Figure A.1 shows the geometry used to calculate the point at which a given direction from the satellite intercepts Earth's surface.

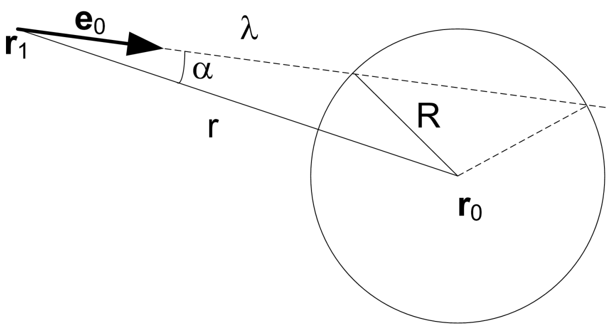

Figure A.1: Geometry used to calculate the point at which a given direction intercepts Earth's surface.

At the interception point, the position vector $\mathbf{r}_1 + \lambda\mathbf{e}_0$ is distance R from Earth's centre. Writing $r = |\mathbf{r}_0 - \mathbf{r}_1|$ for the separation of the satellite from Earth's centre the parameter λ can be calculated (the smaller root gives the nearer interception point).

$$R^2 = \lambda^2 + r^2 - 2\lambda r \cos\alpha \tag{A.5}$$

$$\cos\alpha = \mathbf{e}_0 \cdot (\mathbf{r}_0 - \mathbf{r}_1) \tag{A.6}$$

$$\lambda = r\cos\alpha \pm \sqrt{R^2 - r^2 \sin^2\alpha} \tag{A.7}$$

A.1 Beam Shape Approximations

Table A.1 lists significant values for the Bessel function which models the beam shape (1-way power) from a circular aperture. The conventional beamwidth is the angle from the beam axis to its first minimum, i.e. $\theta = 1.2197..\,\lambda/d \simeq 1.22\lambda/d$. By coincidence, the angle from beam axis to the point at which the power falls to $1/e$ of its maximum is almost exactly half this angle ($0.6096\lambda/d$): the side-to-side beamwidth between these $1/e$ points is thus also $1.22\lambda/d$.

Table A.1: Significant values for the Bessel function model ($\left(\frac{2J_1(x)}{x}\right)^2$) of a circular aperture's one-way radiation intensity variation with angle from beam centre ($x = ka\sin\theta$).

x	$(2J_1(x)/x)^2$	$x/\pi = \sin\theta/(\lambda/d)$
1.16027	$1/\sqrt{2}$	0.3693
1.61633	0.5	0.5145
1.91499	$1/e$	0.6096
2.58383	$(1/e)^2$	0.8225
3.83171	0	1.2197

A Gaussian beam profile ($e^{-(\theta/\theta_0)^2}$) is often used analytically for simplicity and can be matched to the Bessel function at chosen points (e.g. half-power width) by choosing an appropriate Gaussian beamwidth θ_0.

Appendix B

Mission Baseline Technical Summary

This appendix summarises the overall mission baseline (including budgets, images and key information). The mission is summarised using the organisation of the project work breakdown structure, following an initial statement of the mission requirements.

B.1 Mission Objectives

The mission objective is to produce a feasibility study outlining the system design of a long life GEO SAR mission to propose to ESA.

B.1.1 Mission Requirements

MR.01	The mission shall be in operation for users no later than 31/12/2020
MR.02	The mission shall, in a safe and responsible manner, last for 50 years
MR.03	The mission shall implement SAR and inSAR from geosynchronous orbit
MR.04	The mission shall be cost effective with respect to the ESA Sentinel missions

B.1.2 Functional Requirements

SR.01	All equipment, services and construction materials required to design, build and implement the system shall be European where available
SR.02	The system shall provide spatial resolution down to 10m
SR.03	The system shall provide imaging capability within 12 hours after a natural disaster

B.1.3 Operational Requirements

SR.04	The system shall provide images of at least the European Union
SR.05	The system shall satisfy all the applications listed in the WP5100 report
SR.06	The data produced shall be delivered to the customers within scheduled times

B.1.4 Constraints

SR.07	The system including RDT&E, ground station operations and launch shall not exceed a budget of € 2.11 billion
SR.08	The system including ground station operations shall comply with ITU regulations
SR.09	The system shall only use technology with a high TRL
SR.10	The mission shall comply with IADC recommendations

B.2 Baseline

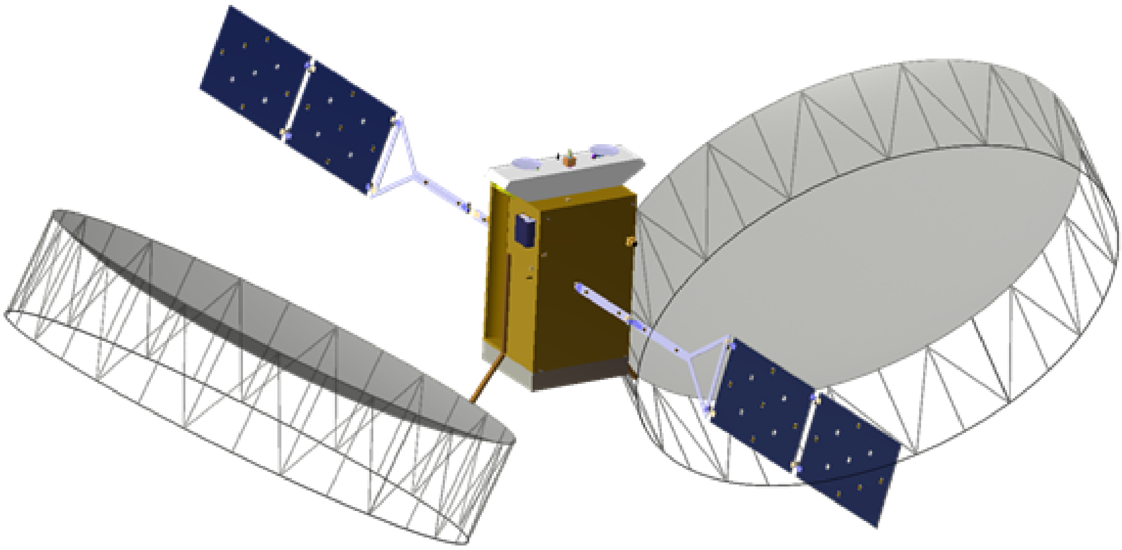

Figure B.1: Deployed Satellite Representation

The satellite consists of 2 large mesh antennas with an effective diameter of 13 m. These antennas will image the earth with L-band and X-band radar to form a Synthetic Aperture Radar (SAR). The system will be powered by 30 m2 of solar arrays that will provide 6.2 kW end of life power output. The satellite has a dry mass of 2153 kg.

The antennas are fed by 36 feed horns. 6 for L-band, 30 for X-Band.

The propulsion system is a monomethylhydrazine and nitrogen tetroxide bipropellant system. There is one 500 N apogee kick motor for orbit raising, as well as eight 22 N thrusters for station keeping manoeuvres all using a total of 3087 kg liquid bipropellant. The tanks are kept pressurized by liquid helium.

There are 4 reaction wheels in a pyramid structure, and a total of 13 attitude sensors.

The On Board Data Handling is carried out by 2 buses linked to 3 CPUs, and 4 FPGAs. This system is complemented by an identical system separated by optical insulators. The whole system has 2 TB of storage and is protected by 6 mm thick aluminium shielding.

The data transmission to earth is handled by two 0.75 m antennas pointing at ground stations in Kiruna and Maspalomas which can maintain a downlink speed of 300 Mbps. Telemetry, Tracking and Control orders are received by 4 conical low gain antennas.

The bus carries 3 secondary payloads: and optical sensor, a laser communications unit, and a radiation detector.

Thermal control of the satellite involves 2 radiators to disperse heat, as well as 120 small patch heaters to keep systems operating during eclipse periods, or launch. The propellant tanks, boom arms, and satellite are all covered by Multi-Layered Insulation blankets.

The satellite has a secondary payload consisting of a radiation detector, an optical imager, and laser communications unit.

Mission Operations

1. The satellite will launch from Kourou on the Ariane 5 ECA with a launch mass of 6000kg.

2. The satellite will use a bipropellant AKM to raise to a geosynchronous orbit with an inclination of 7.5°, and an eccentricity of 0.089. This will take 7 burns and 2.15 days.

3. The satellite will begin nominal operations and remain in space for an estimated 20 years.

4. A second satellite will launch 15 years after the initial launch, and carry out a similar launch

operation. This satellite will then work in a constellation with the first to provide a higher performance.

5. When the performance of the first satellite drops below a point, it will carry out an end of life burn to raise its perigee, and pacify its propellant system. This manoeuvre costs a ΔV of 134.2 $m.s^{-1}$.

6. The second satellite will continue GeoSAR operations on its own.

7. 30 years after the first launch, a third satellite will undergo a similar launch. And join the second in a constellation.

8. Once the performance of the second satellite drops past a point, then it will carry out an end of life burn to raise its perigee, and pacify its propellant system.

9. The third satellite will continue GeoSAR operations until the end of its life where it will carry out an end of life burn to raise its perigee, and pacify its propellant system.

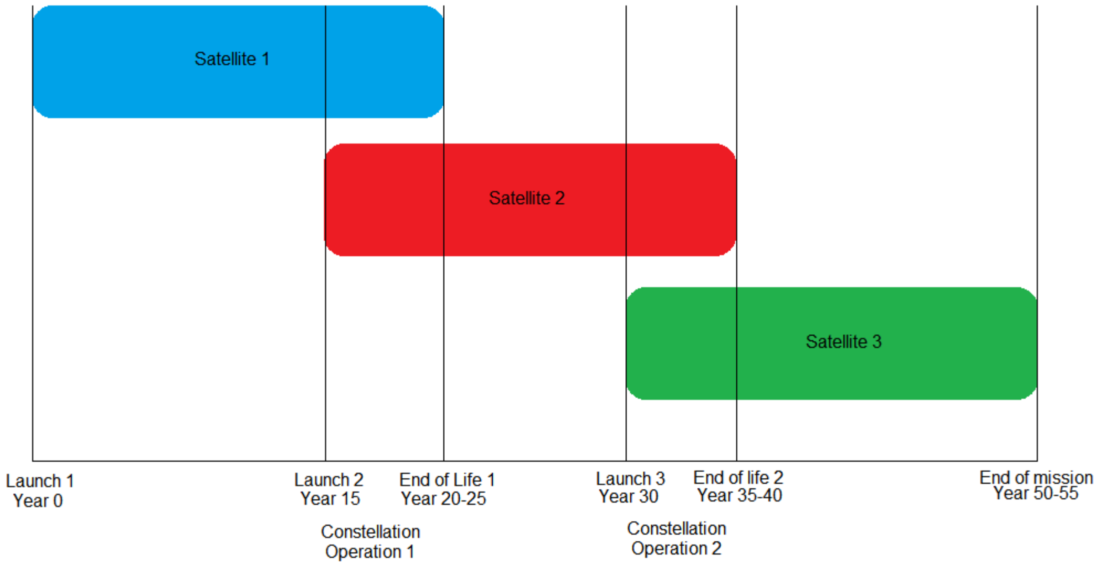

Figure B.2: Timeline of Satellite Overlap

B.2.1 Applications

Applications are summarised in Table B.1. The applications are ranked in priority order in Table B.2.

Civil defence	Azimuth Res. (m)	Time Res. (/hr)	Delivery Time (hr)	Wavelength (m)	Mode
Flooding	30	2		0,03 (X)	Backscatter
Earthquake	10	12	1	0,03 (X)	InSAR-Cohe
Earthquake	100	6	6	0,3 (L)	InSAR-motion
Volcano	20	3	2	0,03 (X)	Phase-cohe
Volcano	200	3	2	0,3 (L)	Phase-cohe
Landslide	30	6	12	0,03 (X)	InSAR-Cohe

Hydrology	Azimuth Res. (m)	Time Res. (1/hr)	Delivery Time (hr)	Wavelength (m)	Mode
Snow Mass	200	2	12	0,3 (L)	InSAR
Snow Cover	200	2	24	0,03 (X)	Backscatter
Snow Cover	200	24	24	0,3 (L)	Backscatter
Hydrology	1000	1	1	0,3 (L)	Backscatter

Weather	Azimuth res. (m)	Time Res (1/hr)	Delivery Time (hr)	Wavelength (m)	Mode
APS - Geo	2000	0,25	2	0,3 (L)	InSAR
APS - Leo	2000	0,25	2	0,3 (L)	InSAR
NWP	2000	0,25	0,3	0,3 (L)	InSAR

Human Activities	Azimuth Res. (m)	Time Res (1/hr)	Delivery Time (hr)	Wavelength (m)	Mode
Oilfield subsidence	50	24	N.D.	0,03 (X)	InSAR
Urbanisation control	10	720	24	0,03 (X)	Backscatter
Agriculture	100	24	N.D.	0,3 (L)	Backscatter
Agriculture	50	3	N.D.	0,03 (X)	Backscatter
Cultural Heritage	10	720	N.D.	0,03 (X)	InSAR

Climate change	Azimuth Res. (m)	Time Res. (1/hr)	Delivery Time (hr)	Wavelength (m)	Mode
Chemical (CH4)	100	36	N.D.	0,3 (L)	Backscatter
Deforestation Biomass	20	2880	N.D.	0,03 (X)	backscatter
Deforestation Tree Height	20	2880	N.D.	0,03 (X)	InSAR

Table B.1: List of User Applications

1	Volcano eruption hazard
2	Earthquake response
3	Flooding
4	Landslide triggered
5	APS - NWP
6	Hydrology
7	Snow Mass
8	Snow Cover
9	Agriculture
10	Oilfield subsidence
11	Cultural Heritage
12	Urbanisation Control
13	Chemical (CH4)
14	Deforestation

Table B.2: User applications in order of imaging priority

B.2.2 Payload Specifications

The payload specification is summarised in Table B.3.

	L-Band	X-Band
Centre Frequency (GHz)	1.2175	9.515
Gain (dB)	59.64	41.80
Bandwidth (MHz)	5	30
Maximum Spatial Resolution (m)	100	10
Minimum Spatial Resolution (m)	2000	200
Maximum Spot Power (W)	1740	2076
Minimum Spot Power (W)	271	235
Spot Area (km2)	996856	9961

Table B.3: Payload Specifications

B.2.3 Data storage on earth

Data storage required is summarised in Table B.4.

Total Archiving Capacity					
Payload Type		Most Challenging Data rates Mbps	With 20% Margin applied Mbps	Most Demanding duty hours/month	Total Archiving Capacity (25 operational yrs) TB
Radar	L Band	9	10.8	61	74.115
	X Band	175	210	109	2575.125
Secondary payload	Imager	21	25.2	744	2109.24
	Radiation detector	0.0003	0.00036	744	0.03
				Total Capacity (PB)	5

Table B.4: Data Handling

B.2.4 Coverage

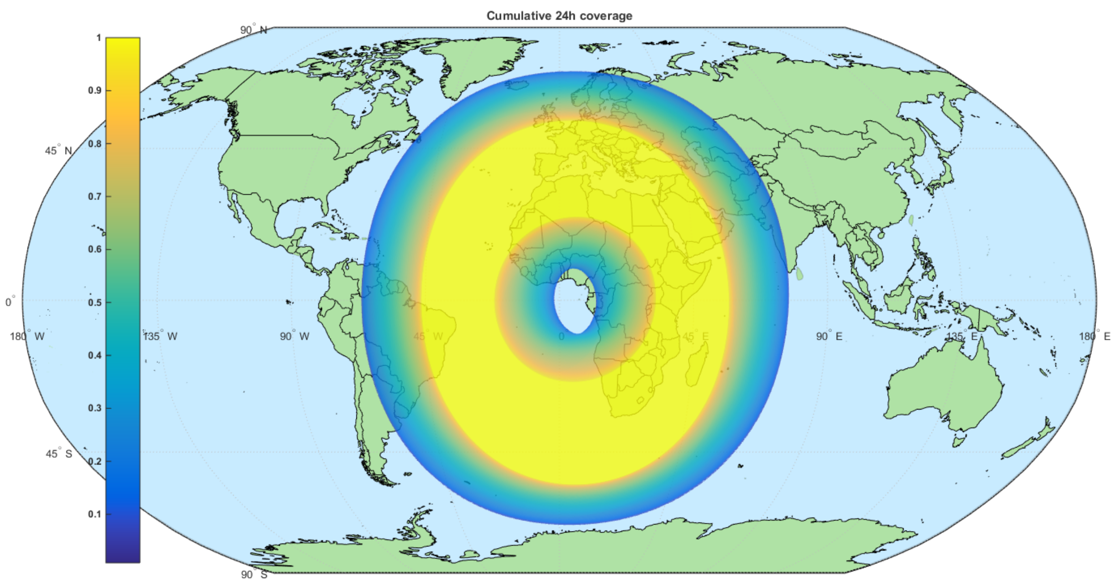

Figure B.3: Satellite Coverage Diagram

B.3 Budget

B.3.1 ΔV

Orbit Insertion	1461.3 m s^{-1}
Eccentricity Correction	2.8 m $s^{-1}year^{-1}$
Inclination Correction	8.76 m $s^{-1}year^{-1}$
Longitudinal Correction	1.6 m $s^{-1}year^{-1}$
End Of Life Manoeuvres	134.2 m s^{-1}
Total Lifetime Budget	1924.5 m s^{-1}

Table B.5: ΔV Budget

B.3.2 Mass

	Mass (kg)	**Dry Mass Breakdown**
Payload	627	29.1%
Structure	524	24.3%
Thermal	90	4.2%
Power	493	22.9%
TT& C	61	2.8%
OBDH	130	6.0%
ADCS	70	3.3 %
Propulsion	158	7.3%
Total Dry Mass	2153	
Propellant	3087	

Table B.6: Mass Budget

B.3.3 Power

	Power (W)	Power Breakdown
Payload	4464	72%
Structure	0	0%
Thermal	650	10.1%
Power	236	3.8%
TT& C	198	3.2%
OBDH	360	5.8%
ADCS	130	2.1 %
Propulsion	74	1.2%
Power Demand	6112	98.2%
Power Raising Capability	6200	

Table B.7: Power Budget

B.3.4 Cost

Satellite Development	€ 144M
Satellite Construction	€ 240M per Satellite
Launch	€ 125M per Launch
Operations	€ 12.5M per Year
Mission total	€ 1.86B for 50 years

Table B.8: Cost Budget

B.3.5 Link

	Downlink		Uplink	
	X-Band	**S-Band**	**X-Band**	**S-Band**
Frequency (Hz)	8.30E+09	2.29E+09	8.10E+09	2.08E+09
Data Rate (bps)	3.00E+08	6.00E+04	2.00E+03	2.00E+03
Lambda (m)	0.0361	0.1311	0.0370	0.1442
S/C Antenna Gain (dB)	34.06	-1.00	33.85	-1.00
S/C Antenna BW (deg)	3.37	200.00	3.46	200.00
G/S Antenna Gain (dB)	60.09	48.89	59.87	48.07
G/S G/T (dB/K)	37.08	25.88	36.86	25.05

Table B.9: Link Budget

		Downlink		Uplink	
		X-Band	**S-Band**	**X-Band**	**S-Band**
Maspalomas	**Min. Elevation Angle (deg)**	48.1712			
	Max. Distance (km)	40242.85			
	Path Losses (dB)	204.0528	192.2899	203.84	191.46
	Power Required (dBw)	4.0827	0.0222	-34.89	-0.60
	Power Required (dBm)	34.0827	30.0222	-4.89	29.40
	Power Required (W)	2.5602	1.0051	0.0003	0.8707

		Downlink		Uplink	
		X-Band	**S-Band**	**X-Band**	**S-Band**
Kiruna	**Min. Elevation Angle (deg)**	13.6018			
	Max. Distance (km)	43082.76			
	Path Losses (dB)	206.0578	193.3404	205.85	192.51
	Power Required (dBw)	6.0878	0.6145	-32.88	0.45
	Power Required (dBm)	36.0878	30.6145	-2.88	30.45
	Power Required (W)	4.0624	1.1520	0.0005	1.1090

Table B.10: Link power requirements to ground stations

B.4 Configuration

B.4.1 Launch

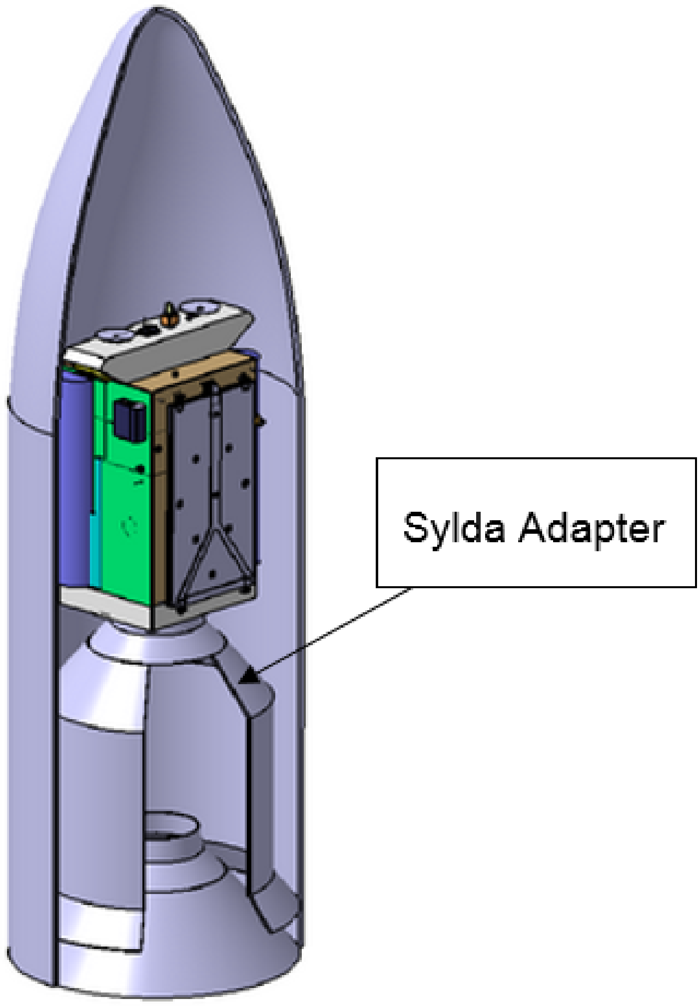

Figure B.4: Satellite Configuration in Launch Housing

B.4.2 Bus

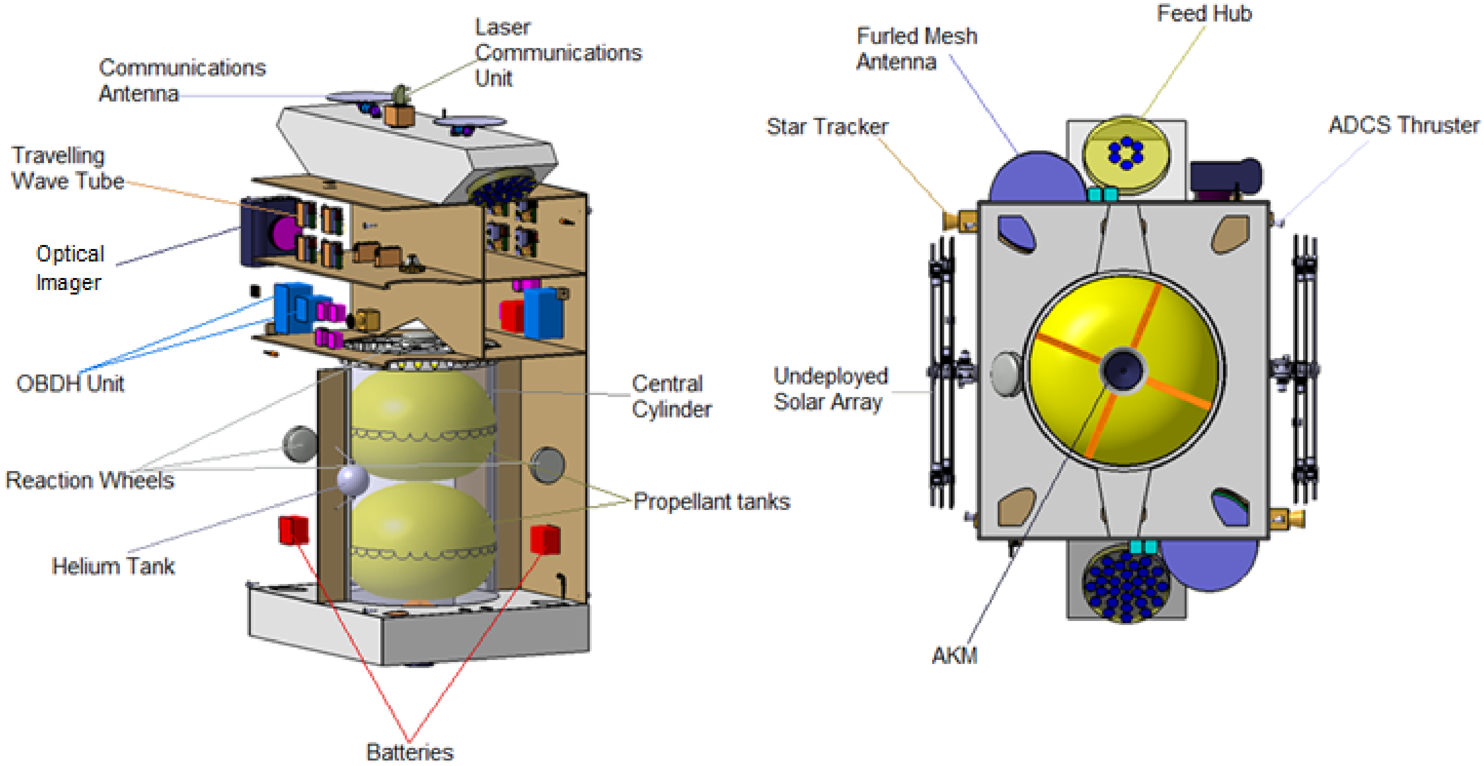

Figure B.5: Satellite Bus Configuration

B.5 Risk Register

#	Subsystem /Unit	Cause		Effects		Hazd.	Prob.	Risk	Class	Resultant Action	
		Source	Cat.	Local	Global					1	2
1	External surfaces	Solar UV radiation	Enviro. (rad.)	Darkening of incident surfaces	Disruption to thermal control system	2	2	4	LOW	Shielding	Material selection
2	External surfaces	Plasma environment	Enviro. (rad.)	Surface charging and arcing	Degradation to external electronics e.g. SA	3	2	6	MED	Appropriate grounding paths	Shielding
3	Electronics (incl SA)	SPE	Enviro. (rad.)	Displacement damage and ionisation	Total dose effects	4	4	16	V. HI	Shielding	Redundancy/ SA sizing
4	OBDH/ Computer	Cosmic rays	Enviro. (rad.)	Mass ionisation	Single event upset, latchup or burn out.	3	2	6	MED	Detection and correction measures	
5	SA/SAR antenna	Debris collision (< 1cm)	Enviro. (deb.)	Subsystem degradation	SA/antenna degradation	2	1	2	V. LO	Shielding	Redundancy/SA sizing
6	Satellite	Debris collision (1cm - 10cm)	Enviro. (deb.)	Subsystem damage	Subsystem/mission retirement	3	3	9	HIGH	Shielding	Redundancy/SA sizing
7	Satellite	Collision with debris > 10cm	Enviro. (deb.)	System damage	Mission retirement	4	1	4	LOW	Evasive manoeuvre	
8	External surfaces	Exhaust from thrusters	Design	Covering of external surfaces	Disruption to thermal/ power control systems	2	3	6	MED	Appropriate thruster orientation	
9	SA/SAR antenna	Rapid SAR ant. pointing	Ops	Increased torques and vibrations	Damage to extrusions/ reduced image cap.	3	3	9	HIGH	Use electric pointing	Use mechanical pointing
10	User apps	Slow SAR ant. pointing	Ops	Increases maximum slew time	Does not meet user requirements	2	4	8	HIGH	Use electric pointing	Use mechanical pointing
11	Launch	Adverse weather	Enviro. (earth)	Launch delay	Mission delay	1	3	3	LOW	Consult weather predictions for decisions on launch date	
12	OBDH/ Computer	Software code error/bug	Human	Operations/AOCS control malfunctions	Unexpected fuel use/incorrect user data	1	2	2	V. LO	Testing and simulation prior to launch/upload	

Figure B.6: Risk Register

B.6 Work Package Structure

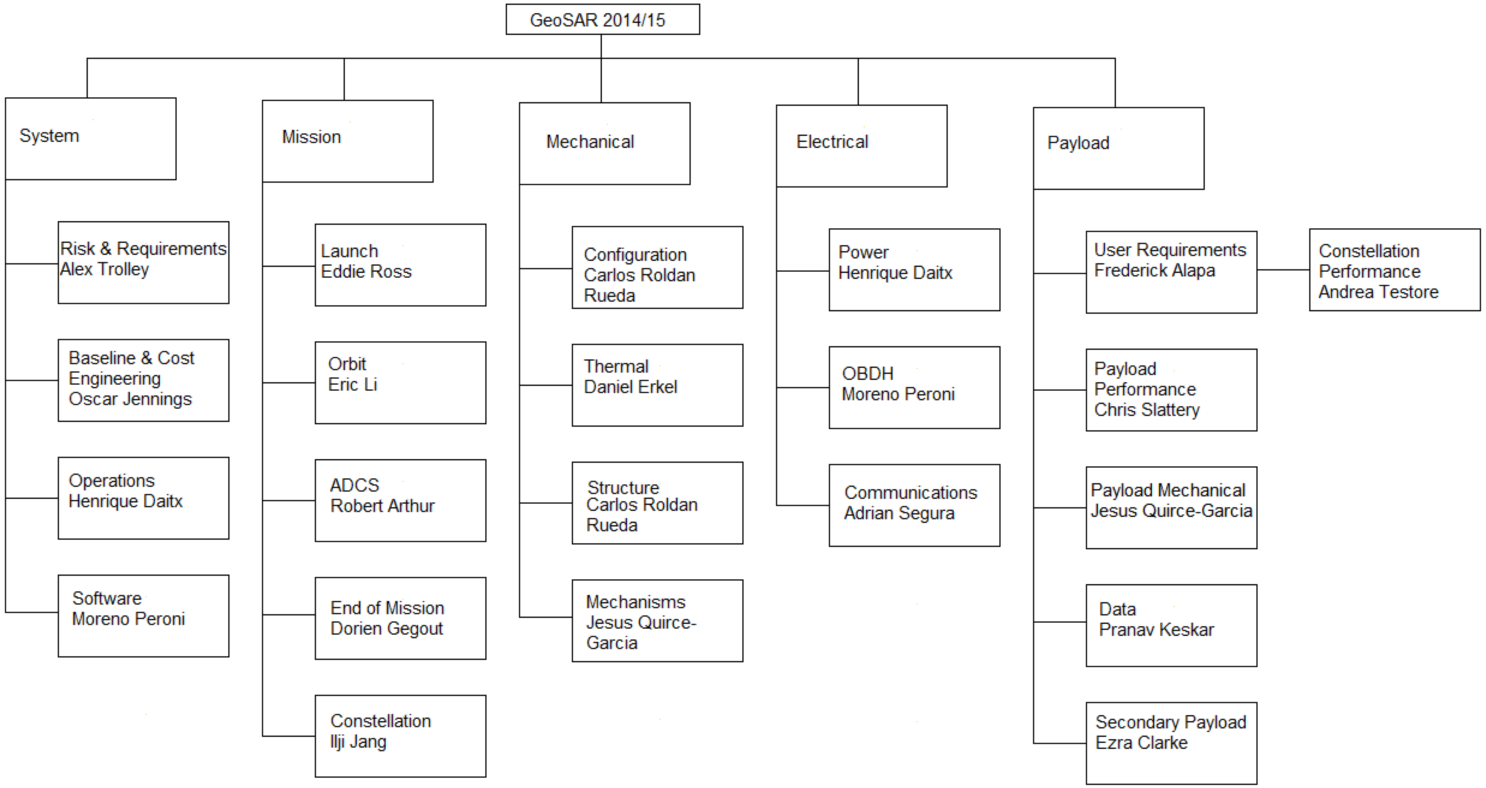

Figure B.7: Group work package structure

Appendix C

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 Space Research Centre, 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 together with Table C.1 gives students' individual responsibilities within the project.

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

Student	Work area(s)
Frederick Alapa	WP 5100, 5200 Payload requirements, operations
Robert Arthur	WP 2300 ADCS
Ezra Clarke	WP 5600 Opportunity payload(s)
Henrique Daitx	WP 1500, 4100 Operations, Power
Daniel Erkel	WP 3200 Thermal
Dorian Gegout	WP 2400 End of mission
Ilji Jang	WP 2500 Constellation orbits
Oscar Jennings	WP 1300, 1400 Baseline, Cost engineering
Pranav Keskar	WP 5500 Payload data
Eric Li	WP 2200 Orbits
Moreno Peroni	WP 1600, 4200 Software, OBDH
Jesus Quirce-Garcia	WP 3400, 5400 Mechanisms, Payload mechanical
Carlos Roldan Rueda	WP 3300 Structure
Eddie Ross	WP 2100 Launch
Adrian Segura	WP 4300 Comms
Chris Slattery	WP 5300 Payload performance
Andrea Testore	WP 5700 Payload constellation
Alex Trolley	WP 1100, 1200 Requirements, Risk

C.1 Payload User Requirements and Operations: Frederick Alapa

C.1.1 Overview

This overview serves as executive summary to the GeoSAR group project in Astronautics and Space Engineering, MSc programme for the 2014-2015 session at Cranfield University, covering in particular the work that was done with respect to the following work packages:

- WP5100: Payload user applications
- WP5200: Payload Operations

Both work packages are closely related. The essence of WP5100 is to determine which of the proposed user applications as listed in Table B.1 are feasible depending on requirements specified by the user(spatial resolution for example) and the constraints imposed on GeoSAR by some orbital parameters.

The first step is defining the azimuth resolution of each application and determining the dwell time needed by the radar. The standard equation for the dwell time, sometimes referred to as integration time is given by

$$T = \frac{cR}{2vf\Delta y} \tag{C.1}$$

Where T = integration time, c = velocity of electromagnetic wave R = Slant range, f = frequency, Δy = azimuthal resolution, v = velocity The ratio velocity to frequency is an indication of the wavelength of the band.

The daily load of the GeoSAR is calculated. Finally, its orbital velocity is determined with respect to the ground track to determine which applications are feasible since there must be relative motion between the Radar and the target on ground to form a synthetic aperture.

It is worthy to state here that there is a degradation of azimuthal geometry every twelve hours because the satellite velocity is zero at half way when it slows down at the apex. It means that not all satellite velocity is useful to perform SAR or InSAR operation all the time.

C.1.2 Analysis of orbital velocity

The analysis of the orbital velocity as it affects the feasibility of applications draws from the conclusion in GeoSTARe technical document that a time frame of not more 180 seconds is required to perform applications that does not require Atmospheric Phase Screening or correction. So, for a given eccentricity, a variation of orbital velocity is done and the proportion of application feasible without APS correction is evaluated. The eccentricity is subsequently changed and the GeoSAR velocity varied as before to determine what proportion of applications can be done. The result is the effect of eccentricity change.

C.1.3 Conclusion

It has been shown that optimum performance of the GeoSAR depends on the choice of orbit and the spacecraft velocity among others. It has been shown from the analysis that most user applications are feasible with GeoSAR velocity starting from 320 m s^{-1}, or at a lower velocity with eccentricity increase to 0.089. A careful choice of the ground track eccentricity vis-a-vis the velocity has a direct bearing on the amount of users application that can be performed. The imaging mode shall be clusters of spot beams.

Application							
		60	110	240	290	310	320
2b.	Earthquake response	468.83	255.72	117.21	97.00	90.74	87.90
3b.	Volcano eruption hazard	468.83	255.72	117.21	97.00	90.74	87.90
2a.	Agriculture	468.83	255.72	117.21	97.00	90.74	87.90
1.	Snow Mass	468.83	255.72	117.21	97.00	90.74	87.90
2b.	Snow Cover	468.83	255.72	117.21	97.00	90.74	87.90
3.	Hydrology	93.77	51.14	23.44	19.40	18.15	17.58
	APS - Geo	46.88	25.57	11.72	9.70	9.07	8.79
	APS - Leo	46.88	25.57	11.72	9.70	9.07	8.79
	NWP	46.88	25.57	11.72	9.70	9.07	8.79
1.	Chemical (CH4)	468.83	255.72	117.21	97.00	90.74	87.90
1.	Flooding	312.55	170.48	78.14	64.67	60.49	58.60
2a.	Earthquake response	937.65	511.45	234.41	194.00	181.48	175.81
3a.	Volcano eruption hazard	468.83	255.72	117.21	97.00	90.74	87.90
4.	Landslide triggered	312.55	170.48	78.14	64.67	60.49	58.60
2b.	Agriculture	187.53	102.29	46.88	38.80	36.30	35.16
2a.	Snow Cover	46.88	25.57	11.72	9.70	9.07	8.79
1.	Oilfield subsidence	187.53	102.29	46.88	38.80	36.30	35.16
2a.	Deforestation Biomass	468.83	255.72	117.21	97.00	90.74	87.90
2b.	Deforestation Tree height	468.83	255.72	117.21	97.00	90.74	87.90
	Cultural Heritage	937.65	511.45	234.41	194.00	181.48	175.81
1.	Urbanization control	937.65	511.45	234.41	194.00	181.48	175.81

Table C.2: Variation of orbital speed

Figure C.1: Potential spot beam GeoSAR imaging mode, showing representative instantaneous beam footprints (L-band, 23 cm) for a 13 m diameter antenna in geosynchronous orbit.

Applications	Useful Speed e=0.0645	Useful Speed e=0.089	Increment
Earthquake response	0.00%	35.42%	35.42%
Volcano eruption hazard	0.00%	71.88%	71.88%
Agricolture	0.00%	35.42%	35.42%
Snow Mass	60.42%	71.88%	11.46%
Snow Cover	60.42%	71.88%	11.46%
Hydrology	93.75%	94.79%	1.04%
APS - Geo	96.88%	97.92%	1.04%
APS - Leo	96.88%	97.92%	1.04%
NWP	96.88%	97.92%	1.04%
Chemical (CH4)	60.42%	71.88%	11.46%
Flooding	83.33%	87.50%	4.17%
Earthquake response	41.67%	61.46%	19.79%
Volcano eruption hazard	76.04%	82.29%	6.25%
Landslide triggered	83.33%	87.50%	4.17%
Agricolture	91.67%	93.75%	2.08%
Snow Cover	97.92%	98.96%	1.04%
Oilfield subsidence	91.67%	93.75%	2.08%
Deforestation Biomass	76.04%	82.29%	6.25%
Deforestation Tree height	76.04%	82.29%	6.25%
Cultural Heritage	41.67%	61.46%	19.79%
Ubranization control	41.67%	61.46%	19.79%

Table C.3: Variation of eccentricity and application availability

C.2 ADCS: Robert Arthur

This report describes the design process and final selection and sizing for the ADCS system of a proposed Laplace orbit geosynchronous SAR mission. The mission requirements and operational plane are used to define the control modes required. Environmental disturbance torques are assessed. Appropriate sensors and actuators are then selected.

C.2.1 ADCS Requirements

In addition to the mission requirements, specific requirements were derived for each control mode. Of these the most stringent requirements which drove the ADCS design are listed here.

1. Long lifetime in GEO of 25+ years.

2. Redundancy shall ensure that no single point failure shall end the mission.

3. Peak pointing accuracy of 0.07°, maintained for extended periods.

4. The attitude of the satellite shall be determined to an accuracy of 0.01°during operations.

5. Capability to safely recover from any foreseeable problems and prioritise the survival of the spacecraft.

6. Induce no damaging torques.

7. Control the spacecraft during the insertion burn and other orbit manoeuvres.

8. All modes shall demonstrate a phase margin of 30° and a gain margin of 6 dB.

The control modes relate closely to the operations schedule, beginning with separation from the launcher. They are illustrated in Figure C.2.

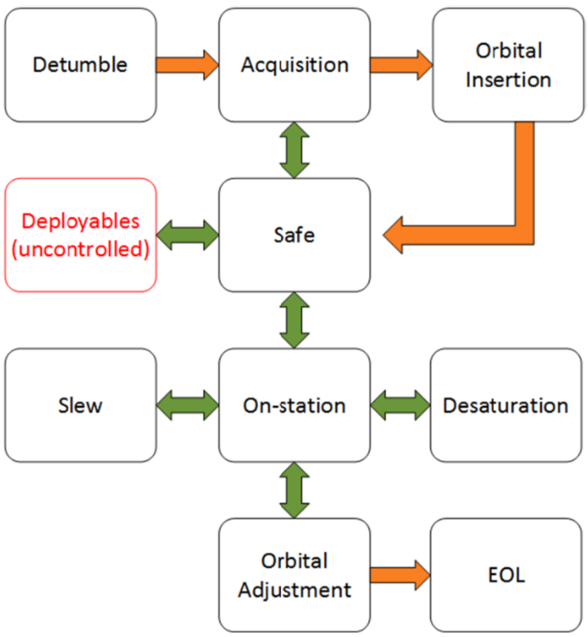

Figure C.2: GEO SAR control modes

C.2.2 Environmental Disturbances

The dominant disturbance torque was found to be solar radiation pressure, as a result of the large antennas necessary for the mission. Initial estimates found that SRP was capable of generating a cyclic torque that would necessitate a large storage of angular momentum. As a result of this analysis the satellite was configured so as to reduce the SRP torques, with a short focal length and the feed mounted on the front of the bus. A more detailed model was then produced to more accurately predict the angular momentum storage required. Results from this model are shown in Figure C.3.

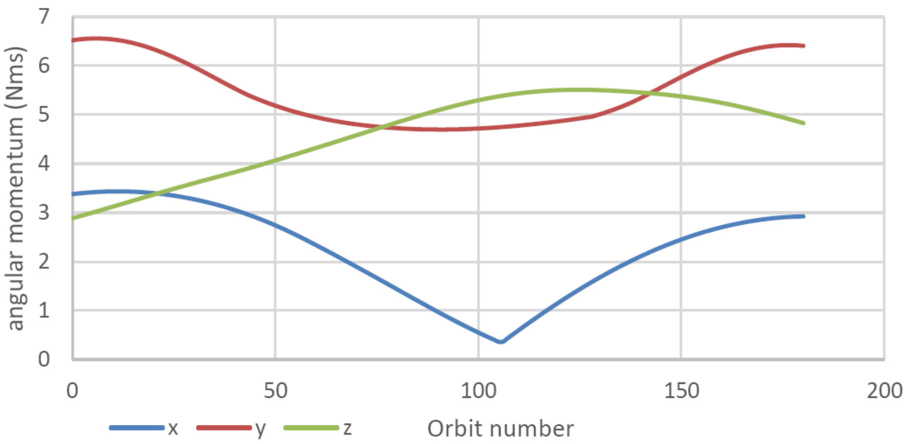

Figure C.3: Maximum angular momentum storage per orbit (6 months)

These results were then used to help size the attitude control actuators.

The effects of gravity gradient were more limited, amounting to 1.4 N m s over an orbit in the worst case. Magnetic torques were found to be negligible, being at least 2 orders of magnitude smaller than gravity gradient.

C.2.3 ADCS Hardware

A suite of sensors was selected in order to ensure continuous attitude determination to an appropriate level of accuracy throughout the mission. Using different types of sensors allows the mission to draw on the complementary strengths of each method and mitigate the weaknesses. The other driving requirement was the need for highly accurate determination during operations.

- 4 Fine Sun sensors
- 2 Coarse Sun sensors
- 2 Earth sensors
- 2 Star trackers
- Fibre optic IMU

In terms of actuators, the requirement for fine control and the dominance of cyclic torques made reaction wheels an obvious choice. Momentum wheels were also considered but once sized it was obvious that they would be far too large. Although the predicted angular momentum storage was low, the wheels were sized so that only a fraction of their maximum capacity would be used. This decision was taken in view of the long lifetime of the satellite, in order to reduce wear. 40 N m s reaction wheels were selected for this mission.

The thrusters would be used for attitude control in some modes of operation, but mainly for desaturation of the reaction wheels and orbital adjustment. Electric propulsion was discarded early due to concern over power supply later in the satellites life. 22 N thrusters from Airbus D&S were chosen as they make use of the same MMH/N2O4 propellants as the 500 N orbital insertion

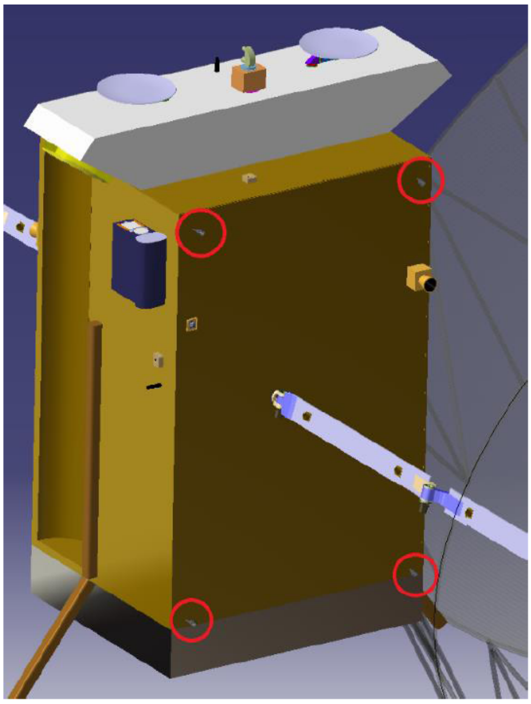

Figure C.4: Positions of thrusters (circled in red) on the satellite

thruster. This eliminates the need for a second propellant storage and supply system, and ensures that propellant can be used as efficiently as possible. Eight of these thrusters would be used, 4 on the North and 4 on the South face. These positions are circled in red in Figure C.4.

The thrusters are canted slightly. This enables the four thrusters on either face to produce torques about any body axis, and therefore provides redundancy.

For all hardware the primary concern during procurement should be quality control, as this will prove the key to achieving the full 25 year lifetime. If necessary, additional redundancy could be added.

C.2.4 Algorithms

Control would be kept as simple as possible, with the attitude being controlled via three independent SISO systems based on the error angle around each body axis. A proportional-derivative controller would be used for each system, with the proportional term used to reduce the steady state error to an acceptable level and the derivative term used to slow the response and reduce oscillations.

When in slew mode the angular momentum required for a manoeuvre could be reduced by initiating nutation. This greatly increased the time for the slew.

Possible reasons to use a more complex control system were to reduce vibrations in the antenna, or to introduce a degree of fault tolerance, where the satellite could diagnose and fix fault without loss of operational time or the involvement of the ground station. Given the long lifetime it may be worth the initial investment in a more complex system, which would pay for itself with savings from reduced demand on a ground station. A template for this sort of system could be found in communications satellites, where faults which impact operational time incur a substantial financial cost.

C.2.5 Additional Comments

Important issues for further consideration include verification of the SRP model for the antennas, and the structural dynamics of the antennas under the influence of various torques.

One technology that could potentially make the mission considerably easier is electronic beam steering. This would relax the pointing requirements, reducing the wear on the actuators and extending their life.

C.3 Opportunity Payload: Ezra Clarke

Opportunities for carrying "small" payloads which would provide additional services with minor impact on the mission were considered. Several candidates for an opportunity payload on the Laplace plane GEO SAR mission were suggested. These include:

- Electrodynamic tether
- Laser communications
- CubeSat
- Camera
- Radiation detector
- AIS (shipping information system)

The three opportunity payloads chosen after a trade-off considering parameters such as heritage, feasibility and risk are:

- Laser communication
  - The laser communication system shall provide communications between satellites, once the second satellite is fully deployed, with a minimum data rate of 1 Mbps.
  - The laser communication must achieve a pointing accuracy of 1 μrad.
  - The laser communication system must be able to relay information about the distance between the two satellites within a margin of 1 mm.
- Camera
  - The optical imager shall provide data on space debris of a cross sectional area of size of 0.3 m at GEO orbit.
  - The optical imager shall operate continuously during the sunlit part of the orbit.
- Radiation detector
  - Detect and study highly charged particles in the range of 10 Mev and above.
  - Radiometer shall operate continuously where data will be stored then transmitted back to the ground station.

The mission baseline includes provision for these payloads.
(The full technical report for this work package is not yet available.)

C.4 Operations, Electrical Power: Henrique Daitx

The Group Design Project (GDP) is part of the MSc in Astronautics and Space Engineering at Cranfield University. For the course year 2014–2015, one of the GDP subjects is a feasibility study on synthetic aperture radar (SAR) imaging from a geosynchronous orbit GeoSAR for short. The chosen orbit has parameters such that i) the daily repeating ground track is long enough to create a synthetic aperture and ii) the perturbations are minimized, reducing station-keeping requirements. One of the key requirements for the mission is that it has to cover a 50-year timespan. Although early on in the study it was determined such a lifetime wouldnt be possible with just one launch, making the spacecraft as durable as possible is quite important. Minimizing launches, further from being an obvious cost-saving measure by itself, helps in simplifying the operations plan.

In designing a long lifetime spacecraft, the power subsystem requires some careful consideration. Most satellites cant afford total redundancy because of the space, weight and expense of duplicating bulky items such as solar arrays and batteries. And its only logical that, if the power generation has to be well planned, so has the power consumption that is, the operations. This executive summary condenses information about the work done by the author for the feasibility study on these two fronts: the Power WP and the Operations WP.

C.4.1 Power

There are a few key figures for the mission which are most significant for the Power subsystem:

- The orbit, albeit not truly geostationary, is almost so. It shares most of GEOs significant traits, notably the eclipse seasons near the equinoxes, with one eclipse per sidereal day and a maximum eclipse duration of about 70 minutes. The insolation is roughly between 1300 W m^{-2} and 1400 W m^{-2}.

- The maximum power demand is about 6000 W, of which 75% is payload consumption. The payload consumption is highly variable with the specific imaging being done on any given moment. The rest of the load is spread among the other subsystems, and can also vary significantly with the spacecrafts activities (approximately between 750 W and 1.5 kW).

- These levels of power consumption should ideally be maintained for the 25 years of the spacecrafts planned lifetime.

- For power generation, a solar array composed of multi-junction cells was chosen. Multi-junction cells currently deliver the highest efficiency available, and have very good degradation properties. These characteristics help ensure the end-of-life power requirements will be met.

The chosen strategy to get the necessary battery lifetime was to use a reasonably large battery and reduce eclipse consumption as much as possible. This entailed dropping an early requirement that called for payload use during eclipse periods.

The final design for the Power subsystem uses a total solar array area of 30 m2. Spectrolabs NeXt triple junction solar cells are used. These cells offer a 29.5% BOL efficiency before accounting for packing factors (Spectrolab, 2015). The expected maximum BOL array power is 8400 W. Taking degradation into account, the minimum EOL array power after 25 years is 6200 W.

The energy storage device is a Li-ion battery composed by 56 Saft VES 180 cells in a 7-series-8-parallel configuration, offering a nominal 10.08 kWh capacity (Saft, 2015). Battery voltage is 25.2 V nominally, 28.4 V at 100% state of charge.

C.4.2 Operations

Operations plan

The chosen operations plan attains a 50-year mission lifetime using 3 spacecraft in total. It is illustrated in Figure C.5.

From this graph, some features are immediately apparent:

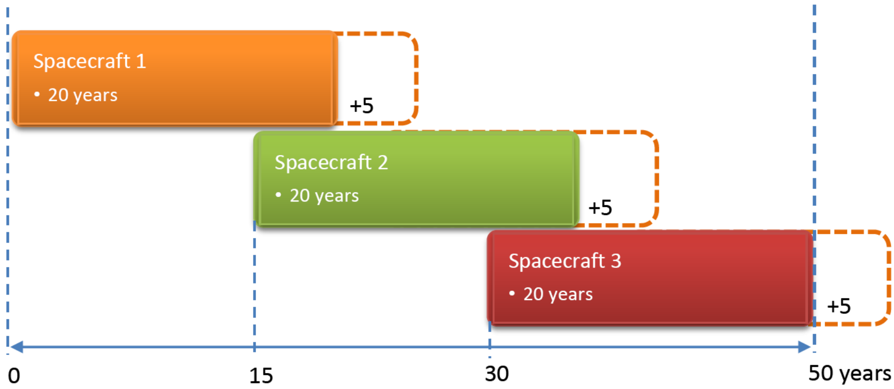

Figure C.5: Mission timeline

- Each S/C is launched separately, and the launches are staggered by 15 years.
- Although the design lifetime for the spacecraft is 25 years, the operational timespan for each is considered as only 20 years. This way, the overlap period may be as long as 10 years.
- There is the possibility of extending the mission by a further 5 years without any extra design effort.
- For a total lack of coverage to occur in the middle of the mission, a total loss of a spacecraft would have to take place at less than 60% of its design lifetime, which is highly unlikely.

Payload duty cycle, coverage, power

The imaging targets have to be selected, at any given time, in order to over a considered time horizon fulfil the user requirements. The sequence of selections is what is called, in the context of this feasibility study, the duty cycle. For a given set of user requirements, there are many duty cycles which are able to fulfil it. Nevertheless, only a few can actually be done, given the coverage area of the satellite as a function of time which is determined by the orbital parameters (cumulative coverage over a sidereal day is illustrated in Figure C.6). Of those, even fewer will be the most convenient in terms of power consumption. Its also important to remember that, in the case of an emergency, everything might change and the required duty cycle could be (and probably will be) one that was never considered before.

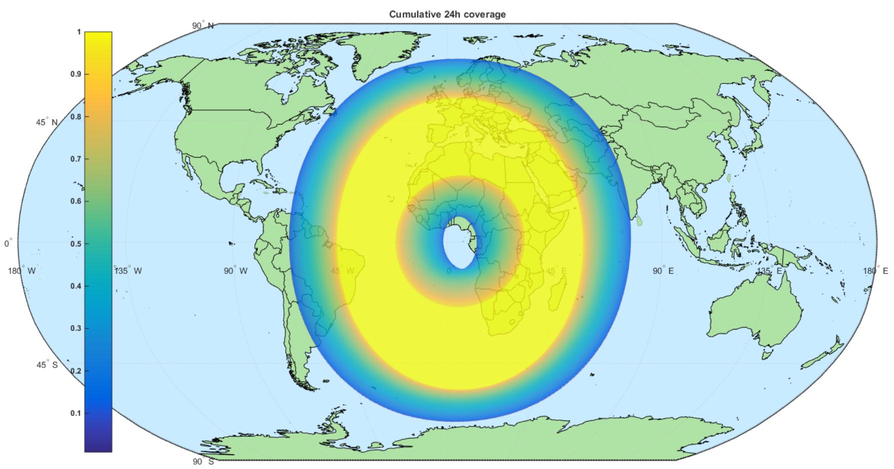

Figure C.6: Cumulative coverage over a sidereal day

It is the task of the operations ground segment to take all these constraints into account and come up with a satisfactory duty cycle. This optimization task is too complex for an analytical approach; it has to be done numerically. Even then, doing it manually would not be cost-effective; a specialized tool a computer program is required for this to work satisfactorily. A small proof-of-concept software was developed which shows the feasibility of this approach.

C.4.3 References

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Saft, Space Products Brochure. Accessed at http://www.saftbatteries.com/battery-search/ves-vl-batteries-satellites on 03/05/2015

C.5 Thermal: Daniel Erkel

C.5.1 Main Aims of the Laplace GEO SAR Feasibility Study

The Laplace Plane Geosynchronous Synthetic Aperture (GEO SAR) Feasibility Study is currently running at the Cranfield Space Research Centre with the goal to investigate the possibility of creating an ESA mission with a SAR satellite operating on a Laplace geosynchronous orbit. The satellite would perform SAR and InSAR imaging over its 50 year mission duration for various users.

C.5.2 The GEO SAR Thermal Control Subsystem (TCS) Feasibility Study - Analysis and Design Process Summary

As part of this project one of the studies, presented in this document, investigated the problem of designing a thermal control subsystem for the GEO SAR satellites and provided solutions to complete this.

The purpose of the thermal control subsystem (TCS) is to regulate the temperature of every element in the spacecraft ensuring constant functionality and operability. The task of the corresponding work package was therefore to establish thermal requirements, analyse the spacecrafts thermal environment prior to, and during its mission, and design the thermal control subsystem meeting all system needs.

The different components of the spacecraft operate and survive in a fixed range of temperatures, which may vary from component to component. In the meantime, the spacecraft, depending on its mission, can encounter extreme variations in temperature.

The GEOSAR mission's greatest challenges from the TCS designs aspect are the 20–25 years spacecraft lifetime, the SAR payloads large heat dissipation (approximately 3900 W), the shadowing of the reflectors, and the thermo-elastic distortion of the large deployable mesh antennas, with each having an effective aperture of 13 m (actual diameter is 16 m). While designs employed in geosynchronous missions have good heritage, few spacecraft have reached 25 years, with one example being the Inmarsat-2 mission, which was used as a benchmark designing the GEOSAR TCS.

During the design process, the greatest emphasis was placed on the interfaces with the configuration, payload, and power subsystems. However, thermal control is linked to nearly all other work packages and requires a complete understanding of the spacecraft system and the mission architecture.

The design process started with investigating the elements of the mission which have a key influence on the TCS, analysing the environments the spacecraft is exposed to determining the greatest challenges of geosynchronous TCS design and the mission itself, and conducting a literature review to find previously used solutions for similar problems. This was then followed by defining the derived requirements for the spacecraft.

The primary inputs from the baseline were then used to analyse a hot and a cold case, performing preliminary calculations. Creating the hot and cold cases (worst case scenarios) also included the preliminary selection of passive thermal control methods which can offer the reliability and longevity required by the mission.

The baseline analysis using simple heat balance calculations was followed by modelling in ESATAN, a tool for thermal radiative analysis based on the lumped-parameter approach.

This simple ESATAN model, created without the antennas and the solar panels and using a total area heat load as boundary condition for internal heat dissipation, was then validated using the heat balance equations and a Matlab code based on theoretical models for the calculating sun beta angle and the solar heat flux on the spacecraft.

The initial ESATAN model was then improved using further data from the other subsystems and on the configuration, adding first the solar panels and the reflectors and later modelling all of the equipment. Results from the thermal analysis was then used to iterate towards creating the final spacecraft configuration and adjusting the requirements of the other subsystems.

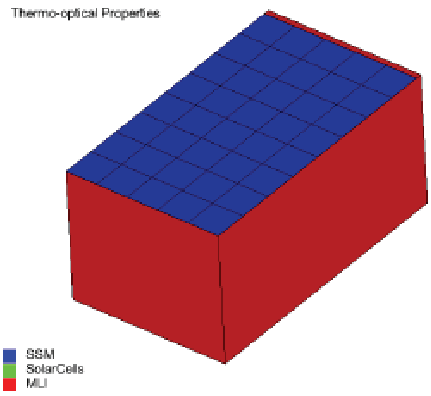

Figure C.7: First basic model created in ESATAN for the spacecraft bus with the thermo-optical shown with different colours

C.5.3 Final TCS Design

1. The front of the solar panel: solar cells. Rear: Chemglaze Z306 (high emittance, low degradation black paint) manufactured by Lord Corporation Europe
2. The solar panels are thermally decoupled from the spacecraft bus.
3. Mesh antennas require no thermal control.
4. All panels of the bus, with the exception of the radiators, are covered with a 20 layer MLI blanket with conductive ITO outer layer from RAL Space
5. OSR CMX galss radiators with ITO from Qioptiq
6. The reflectors are thermally isolated from the bus and the booms they are installed on are covered with MLI
7. The different components of the secondary payload are thermally isolated from the their environments and the remainder of the spacecraft
8. The back of the comms. antenna reflectors is covered with MLI

1. Most of the inside of the spacecraft is covered with Chemglaze Z306
2. Approximately 120 Ruag patch heaters are used on the various internal equipment
3. The internal part of the structural cylinder is covered with 10 layer MLI
4. All propellant tanks, including the two bi-propellant ones and helium tank, are covered with 20 layer MLI.
5. High dissipating equipment split into two groups with approximately equal heat dissipation values, placed on the OSR radiators using thermal doublers and heat pipes in the case of the TWTAs
6. Internal panels are also covered with Chemglaze Z306
7. Temperature of the propellant tanks and the piping is controlled with line heaters
8. The various electronic equipment are scattered around the different parts of the spacecraft to ensure sufficient distance from and between the high dissipating equipment. The batteries with the narrowest operating temperature range are located near the Zenith side of the spacecraft and are isolated from the other equipment.

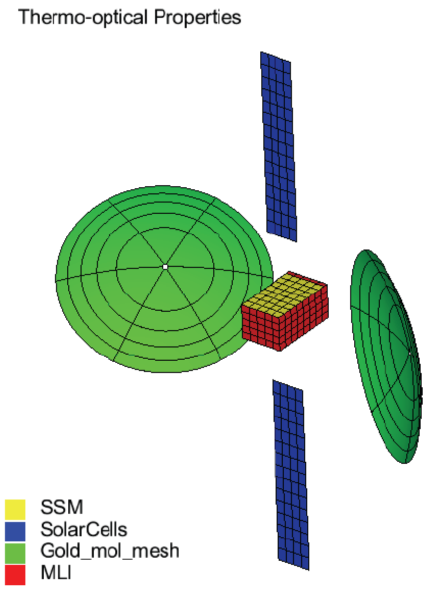

Figure C.8: Second model created in ESATAN for the spacecraft with the thermooptical shown with different colours

C.5.4 Conclusions and Further Work

The study achieved its main goal and showed that it is possible to create a TCS for the GEO SAR missions requirements with European Technology. Further work is required on various parts of the study, such as: modelling contact conductances within the spacecraft; creating a higher fidelity model; modelling the launch and pre-launch phases; addressing the thermo-elastic problems of the antenna.

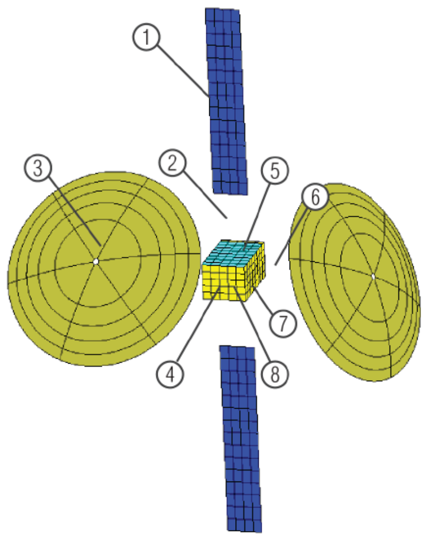

Figure C.9: External view of the satellite with numbers indicating the different external elements of the TCS

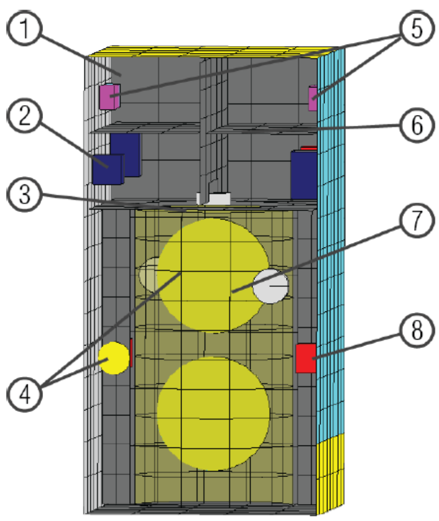

Figure C.10: Internal view of the satellite bus with numbers indicating the different components of the TCS

C.6 Collision Risk Analysis and End of Mission, Debris: Dorien Gegout

The aim of this study is to perform a collision risk analysis and to design the end of mission of the Laplace orbit GeoSAR mission. The collision risk was therefore evaluated, a graveyard orbit was chosen for the satellites and the passivation (removal of any kind of internal energy in order to avoid explosions) of the spacecraft was designed.

C.6.1 Collision risk analysis

Three categories of debris have been identified. The first category corresponds to small pieces of debris (a<1 cm where a is the length of the object along its major axis), which do not represent a threat for the satellite. Category 2 includes medium pieces of debris (1cm <a<10 cm), which represent a threat for the satellite. However if a collision occurs, the mission of the satellite may continue. Finally, category 3 contains big pieces of debris (10 cm<a). A collision must not occur during the mission otherwise the mission is over. Using DRAMA, a tool developed by ESA, the collision probability during the mission with these pieces of debris was determined.

The acceptable probability of collision was defined by the probability of collision of a 15 years lifetime geostationary satellite. Therefore the probability of collision was identified to be as follow:

Probability / Category	$p < 7.15 \times 10^{-5}$	$7.15 \times 10^{-5} < p < 5.73 \times 10^{-4}$	$p > 5.73 \times 10^{-4}$
1	acceptable	acceptable	acceptable
2	acceptable	acceptable	non-acceptable
3	acceptable	non-acceptable	non-acceptable

Table C.4: Collision risk categories for 30 year lifetime

The probability of collision for each satellite (30 years lifetime) was calculated :

- 4.99×10^{-4} for a collision with a category 2 piece debris

- 2.85×10^{-5} for a collision with a category 3 piece debris

Hence, the probability of collision during the mission should be enough to ensure a successful mission.

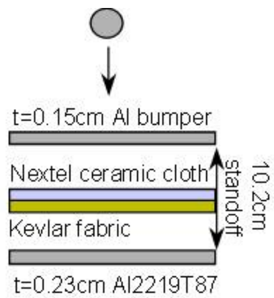

Figure C.11: Whipple shield construction

Pieces of debris from category 1 can however be a threat if a collision occurs close from the propellant tanks. Indeed even a small impact can cause an explosion. That is why it was decided to use Whipple shield around the propellant tanks in order to reduce this risk of explosion. An example of Wipple shield is shown in the opposite figure.

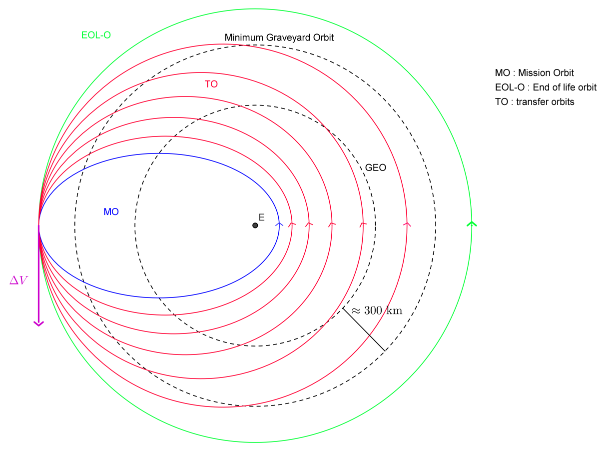

Figure C.12: Transfer to graveyard orbit

C.6.2 End of life considerations

Transfer to the graveyard orbit

At the end of mission, IADC regulations demands a minimum increase of the semi-major axis by 301.008 km and a eccentricity lower than 0.003 for the minimum graveyard orbit.

However the satellite has an eccentric orbit (e=0.089). That means the apogee of the orbit will be higher than the minimum graveyard orbit defined by the IADC regulation.

Five different options were considered and a trade of analysis was made to select the best one. The most important parameters for the trade-off analysis were to meet the IADC regulations, to provide a low risk of collision in the end of life orbit, and to have a low cost in term of ΔV for the transfer. Two other parameters were also considered for this trade-off analysis.

The idea of the option chosen is to reach the circular orbit at the apogee of the mission orbit at the end of life. The ΔV corresponding to such a manoeuvre is $\Delta V = 134.152$ m.s^{-1}.

Only the thrusters for station keeping can be used to reach this orbit. However, it appeared that the thrusters do not have an enough efficient thrust to reach the orbit in one manoeuvre. Therefore the transfer was split into seven manoeuvres to reach the final orbit as it is shown in the figure above.

Passivation of the propulsion system

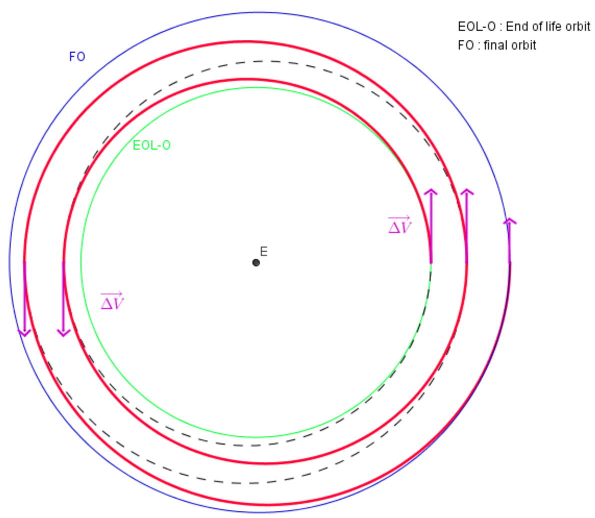

Figure C.13: Orbit raising manoeuvres

Propellant tanks Once the graveyard orbit is reached, the propellant tanks must be emptied. As it is difficult to exactly know how much propellant there is in the tanks at the end of life of the satellite, the spacecraft will do small manoeuvres to slowly rise the orbit and to keep the eccentricity lower than 0.003 until the propellant tanks are fully emptied. The opposite figure represents the design chosen for the passivation of the propellant tanks.

Pressurized system The helium tank will be separated from the propulsion system just after the transfer in GEO. The helium tank will immediately be depressurized. To perform this, two valves will be opened. The gas will be expelled tanks to two piped placed in opposite directions to avoid any torque or force. The remaining pressured gas in the propellant tanks will be emptied with the propellant. (Indeed there is no diaphragm between the propellant and the gas)

Passivation of the power system

Disconnecting solar panels from batteries To meet this requirement, it was decided with the power subsystem to use pyrotechnics circuit breakers. To reduce risk of failure, it was also decided to place these pyrotechnics circuit breakers at the exit of solar panels.

Discharging the batteries As the batteries are not supplied once they are disconnected from the solar panels, they will automatically discharge. Indeed they will still supply some systems like the communication subsystem until full discharge.

Stop reaction wheels

The reaction wheels must be stopped at the end of mission. It was shown that they will automatically stop due to friction. The time needed for the wheels to stop was estimated as 1 600 seconds.

C.7 Constellation: Ilji Jang

In this Laplace Orbit GeoSAR Feasibility Study Group Design Project, three constellation concepts are suggested for the mission requirements. Three satellites constellation designs are examined to see the reconfiguration for the reliability and the maintenance for its feasibility. In this report, feasibility study for the three recommended constellation designs are investigated numerically. For the global coverage, three satellites are deployed separately to each position. When the Satellite1 which is located above the Europe is failed, reconfiguration need to account for failure of the mission. This reconfiguration method can make the mission possible to continue its mission for the Europe region.

Table C.5: Orbit element for the global coverage (Epoch Time = 1 Jan 2020 00:00:001)

Orbit element	Satellite1 (Europe)	Satellite2 (USA)	Satellite3 (Asia)
e	0.089	0.089	0.089
i / °	7.5	7.5	7.5
ω / °	270	270	270
Ω / °	0	0	0
θ / °	190	100.5	290

Table C.6: Orbit element for double monostatic mission

Orbit element	Satellite1 (Main)	Satellite2
a, km	42164	42164
e	0.089	0.089
i, deg	7.5	7.5
Ω, deg	0	102
ω, deg	270	270
θ, deg	190	100
M, rad	-2.9340	1.5682

C.7.1 Orbit reconfiguration

Orbit reconfiguration method for failure of the Satellite1 is examined to move the Satellite3 to the position of the Satellite1. Hohmann transfer method is more efficient than Bi-elliptic transfer method through ratio of Final/Initial orbit. However compared with Hohmann transfer, orbital phasing method also shows less ΔV budget and it only take account for two tangential burn compared with four tangential burn for phase change by Hohmann transfer. Therefore orbital phasing method is selected as reconfiguration method. To decide SMA and eccentricity of phasing orbit, cost effectiveness study is done by different weighting ΔV and Δt . As the importance of ΔV is increased, Δt is not affecting significantly. However weighting on ΔV is small, the effect of Δt became significant.

Finally, the cost-effective semi-major axis and eccentricity for phasing orbit are decided to $a =$ 30815, $e = 0.2465$. The minimum ΔV is 840.5 m s^{-1}and Δt is 1.264 days by this selected phasing orbit to reconfigure the constellation. This selected phasing orbit suggested that this orbit is the cost-effective phasing orbit in terms of ΔV and Δt in the most cases of the different weighting to ΔV .

C.7.2 Maintenance

Based on the Draper Semi analytic Satellite Theory (DSST) which is based on Hills equation, along-track separation is examined for global coverage, double monostatic and bi-static mission to maintain formation keeping for further study to quantify perturbations. Orbit maintenance is needed to consider for formation keeping propagation. Based on DSST theory, each constellation mission orbit were propagated to see its propagation and to decide proper maintenance plan.

To find the elements for the bi-static mission for 77 km distance between two satellites, Newton Raphson method was used to find the corresponding mean anomaly of the two satellites. The propagation graph shows it has ±6 km variation.

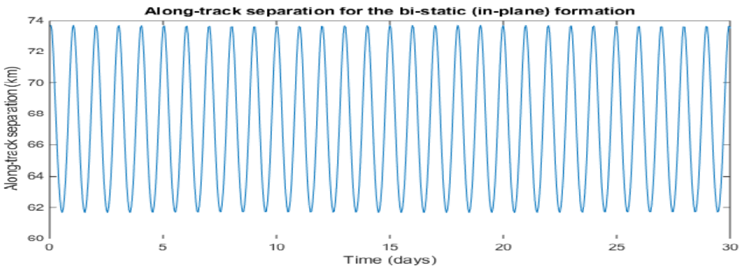

Figure C.14: Along-track separation for the bi-static (in-plane) formation

The reason for the perturbations are caused from the earth oblateness, atmospheric drag, and tesseral resonance and those effects should be accounted for in the further studies quantitatively. In bi-static mission the propagation shows that 68 km distance caused 62–74 km variation. To keep the distance between two satellites in 77 km, 68 km is new initial distance for two satellites to account for the perturbations.

Table C.7: Bi-static mission orbit elements

Orbit elements	Satellite1 (Main)	Satellite2
a, km	42164	42164
e	0.089	0.089
i, deg	7.5	7.5
Ω, deg	0	0
ω, deg	270	270
θ, deg	0.11	0
M, rad	0.0016	0

C.7.3 References

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C.8 Baseline, Cost Engineering: Oscar Jennings

This Laplace Orbit GeoSAR Feasibility Study, was a 5 month study conducted by 18 students at Cranfield University in 2014/15. The role of the baseline and cost engineer was to produce, monitor and update the budgets of the design. This report details a design chosen by the students to create a geosynchronous synthetic aperture radar imaging satellite, and the design choices, and their justification, for every subsystem and mission operation decided on by members of the group.

The final mission baseline is detailed below:

C.8.1 Mission Design

1. Satellite will launch for Kourou on the Ariane 5 ECA with a launch mass of 6000kg.
2. The satellite will use a bipropellant AKM to raise to a geosynchronous orbit with an inclination of 7.5° and an eccentricity of 0.089. This will take 7 burns, and 2.15 days
3. The satellite will begin nominal operations and remain in space for an estimated 20 years.
4. A second satellite will launch 15 years after the initial launch, and carry out a similar launch operation. This satellite will then work in a constellation with the first to provide a higher performance.
5. When the performance of the first satellite drops below a point, it will carry out an end of life burn to raise its perigee, and passivate its propellant system. This manoeuver costs a ΔV of 134.2 m s^{-1}
6. The second satellite will continue GeoSAR operations on its own.
7. 30 years after the first launch, a third satellite will undergo a similar launch. And join the second in a constellation.
8. Once the performance of the second satellite drops past a point, then it will carry out an end of life burn to raise its perigee, and passivate its propellant system.
9. The third satellite will continue GeoSAR operations until the end of its life where it will carry out an end of life burn to raise its perigee, and passivate its propellant system.

C.8.2 Satellite Design

The satellite consists of 2 large mesh antennas with an effective diameter of 13m. These antennas will image the earth with L-band and X-band radar to form a Synthetic Aperture Radar (SAR). The system will be powered by 30 m^2 of solar arrays that will provide 6.2 kW end of life power output. The satellite has a dry mass of 2153 kg

The antennas are fed by 36 feed horns, 6 for L-band, 30 for X-Band

The propulsion system is a monomethylhydrazine and nitrogen tetroxide bipropellant system. There is one 500 N apogee kick motor for orbit raising, as well as eight 22 N thrusters for station keeping manoeuvres all using a total of 3087 kg liquid bipropellant. The tanks are kept pressurized by liquid helium.

There are 4 reaction wheels in a pyramid structure, and a total of 13 attitude sensors.

The On Board Data Handling is carried out by 2 buses linked to 3 CPUs, and 4 FPGAs. This system is complemented by an identical system separated by optical insulators. The whole system has 2TB of storage and is protected by 6mm thick aluminium shielding.

The data transmission to earth is handled by two 0.75 m antennas pointing at ground stations in Kiruna and Maspalomas which can maintain a downlink speed of 300 Mbps. Telemetry, Telecomand and Control orders are received by 4 conical low gain antennas.

The bus carries 3 secondary payloads: and optical sensor, a laser communications unit, and a radiation detector.

Thermal control of the satellite involves 2 radiators to disperse heat, as well as 120 small patch heaters to keep systems operating during eclipse periods, or launch. The propellant tanks, boom arms, and satellite are all covered by Multi-Layered Insulation blankets

The satellite has a secondary payload consisting of a radiation detector, an optical imager, and laser communications unit.

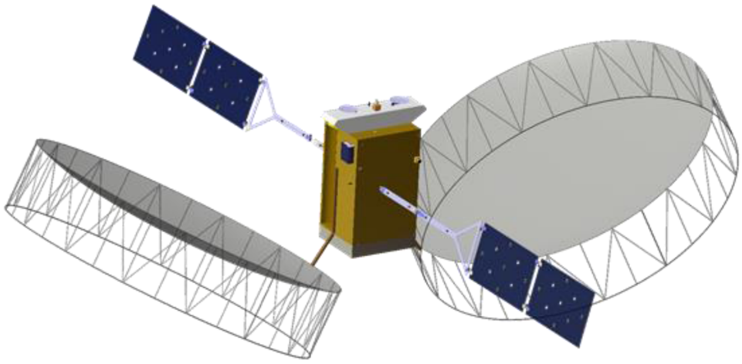


Figure C.15: Deployed GeoSAR Satellite

C.8.3 Satellite Budgets

Table C.8: ΔV Budget

Manoeuvre	ΔV (m s^{-1})
Orbit Insertion	1461.3
Eccentricity Correction	2.8 (yr^{-1})
Inclination Correction	8.76 (yr^{-1})
Longitudinal Correction	1.6 (yr^{-1})
End Of Life Manoeuvres	134.2
Total Lifetime Budget	1924.5

Table C.9: Mass and Power Budgets

Subsystem	Mass (kg)	% Dry Mass	Power (W)	Power (%)
PAYLOAD	627	29.10%	4464	72%
STRUCTURE	524	30%	0	0%
THERMAL	90	4.20%	650	10.10%
POWER	493	22.90%	236	3.80%
TT&C	61	2.80%	198	3.20%
OBDH	130	6.00%	360	5.80%
ADCS	70	3.30%	130	2.10%
PROPULSION	158	7.30%	74	1.20%
TOTAL (DRY MASS)	2153		6112	98.20%
PROPELLANT	3087			
Power Generation			6200	

It is hoped that the information in this report, and the reports of the entire group, will provide useful information for the future planning of geosynchronous synthetic aperture radar imaging missions.

Table C.10: Cost budget

Item	Cost (€ M)
Satellite Development	144
Satellite Construction	240 (per Satellite)
Launch	125 (per Launch)
Operations	12.5 (per Year)
Mission total	1.86B(for 50 years)

C.9 Payload Data: Pranav Keskar

C.9.1 PDGS Layout

PDGS Layout can be represented in a nutshell as shown above comprising two Core Ground Stations and two Processing and Archiving Centres providing the required redundancy. All the other units like Data Management Centre, Mission Performance Centre assume their usual roles. The choice of locations PACs is proximal to the CGS to minimise the internal data shipping time

C.9.2 Data Processing

Data processing requirements of the GeoSAR mission are inherently demanding owing not only to the large volumes but also the different processing requirements of the vast variety of data generated by a wide spectrum of user applications.

Data Processing Concept could be summed up as follows:

- The RAW quick look data for NWP that needs to be delivered within 20 min from the capture to be transmitted directly from the core ground station to the Meteorological agencies assimilating the data into NWP code.
- The other NRT data can be split into two categories
- High bit rate data requiring a delivery within 3 hours.
- Low bit rate data that needs a delivery on the same day but not necessarily within 3 hours from capture of image.

Core ground stations to process the data of first category saving the internal transmission time and cost. Second category data to be processed in the PACs

All the on demand and offline data to be handed over straightaway to the PACs after downlinking at the CGS and processed and delivered from there.

All the payloads raw data received, shall be processed to Level 1B as that is the minimum requirement of all the applications considered in both X as well as the L band.

All raw mission data to be temporarily stored in rolling archives at the Core Ground Stations and later be stored for long term in archiving facility at the Processing and Archiving centres

C.9.3 Data Archiving

The major challenges in PDGS were presented by the targeted operational lifetime of the satellite generating unprecedentedly huge volumes of data, imposing very demanding storage requirements. This load can be significantly reduced by processing this data on board. But the cumulative cost for processing data on board over the entire satellite lifetime would threat the economic viability of the mission. Besides, On board data handling system is the most susceptible to failures and degradation over such an extended length of life time.

This necessitates the on board processing to be kept limited to minimal level of requirement. Thus it also implies that all the level 0 mission data should be processed and stored on ground. The archiving capacity required to meet this challenge has been calculated as follows:

Limiting case imaging data rates in both L-band and X band were considered, and then the data storage requirement for a month was calculated considering the limiting case monthly duty cycle load. This is extrapolated to estimate the total storage requirement over the satellite lifetime of 25 years.

The following Solutions were proposed to meet this requirement

- Archiving only the level 0 data
- Adherence to ESAs Long Term Data Preservation Guidelines
- Provision of a redundant Processing and Archiving Centre

Total Archiving Capacity					
Payload Type		Most Challenging Data rates Mbps	With 20% Margin applied Mbps	Most Demanding duty hours/month	Total Archiving Capacity (25 operational yrs) TB
Radar	L Band	9	10.8	61	74.115
	X Band	175	210	109	2575.125
Secondary payload	Imager	21	25.2	744	2109.24
	Radiation detector	0.0003	0.00036	744	0.03
				Total Capacity (PB)	5

Table C.11: Data Archiving Capacity Required

C.9.4 Data Transmission rate

The most stringent demand for data transmission which also drives the data delivery link design is imposed by Numerical weather prediction (NWP) application. It requires RAW quick look images to be delivered within 20 min. from capture in order to assimilate them in NWP code to be converted to water vapour and hydrometeor components.

Data Rates for Raw quick look data delivery to NWP	
L Band Data rate (Mbps)	12
Integration time for NWP(2 spots)	240
Image data (Mb)	2880
Time available for downlinking (min)	13
Time available for delivery (min)	5
Downlinking data rate(Mbps)	3.69
Delivery data rate(Mbps)	9.60
Downlinking with 20% Margin	4
Delivery with 20% Margin	12

Table C.12: Most Demanding Data Rate for NWP

The maximum time for image integration in L band (2 min.), imaging frequency (15 min.) is considered along with Considering limiting case L band imaging data rates the downlinking and delivery data rate requirements have been calculated.The imaging data rates calculated by the OBDH work package already had a built in margin thus adding 20% margin on top of that ascertains a conservative figure leaving enough room for flexibility.

C.9.5 Dissemination Link Architecture

The ultimate success of any mission is decided by its ability to deliver the data to the targeted user meeting the requirements of timeliness and quality.

- Delivery via Standard Land Lines: Ground based high speed lines (GEANT and NERN) operated by Hiseen for low-rate fast-delivery product are generated by the ESA ground stations and re-distributed to nationally nominated user centres.

- Delivery via the Broadband Data Dissemination Network (BDDN): when high rate FD products from Kiruna and Maspalomas ground stations are transmitted to nominated receiving stations by means of a data relaying satellite telecommunication channel like Artemis service

The concept can be summarised as follows,

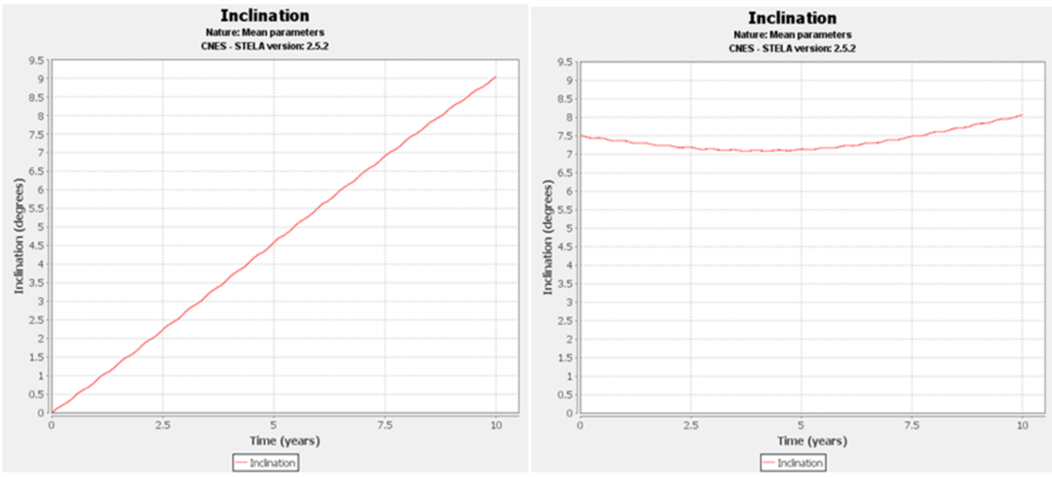
- Delivery of on demand user data upon online orders or requests from users that can access the data through a subscription (Majority of commercial users)
- Product retrieval via internet using ESAs high speed internet back bone (GEANT / NREN)
- Product distribution via satellite in a future scenario possible; could be added as a redundancy
- Electronic shipment and internal circulation of data amongst the CGSs ,PACs ,PDMC and MPC using high speed intranet (GEANT / NREN based)
- The data rate bandwidths available on the GEANT / NREN high speed links at Core Ground Stations at Maspalomas (RedIRIS,40 Mbps) and Kiruna (SUNET,100 Mbps) and at Processing and Archiving Centre at Farnbnorough (Ukerna,100 Mbps) are more than sufficient to cater for the delivery data rate requirements of all the user applications

C.10 Orbits: Eric Li

The GeoSAR is a concept consisting of placing a spacecraft in geosynchronous orbit that would be capable to perform SAR imaging. It hasn't been done before, yet, it increasingly interests space agencies such as the European Space Agency. This report is an attempt to study the feasibility in term of orbit design of a GeoSAR mission with a lifetime superior to 50 years.

C.10.1 Laplace plane orbit advantageous properties

The Laplace plane in this report is defined as the mean plane that is perpendicular to the axis around which a satellite orbit pole precesses. It was discovered by Pierre-Simon Laplace as he observed Jupiters satellites. The main advantage of this plane is that it experiences low inclination drift as shown in Figure C.16. The corresponding North-South station keeping DeltaV budget is usually about 5 to 6 times smaller than the DeltaV required for geostationary orbit.



(a) Geostationary orbit (b) Geosynchronous Laplace plane

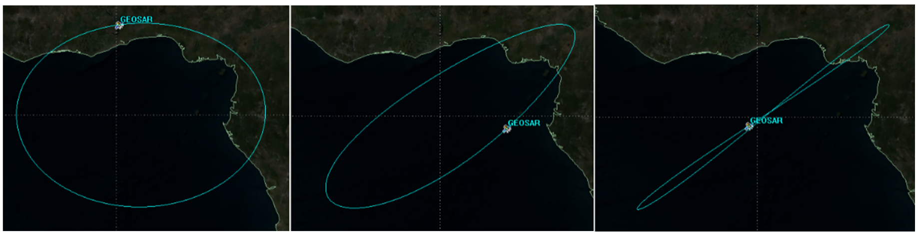
Figure C.16: Inclination drift comparison

Hence, the Laplace plane orbit an ideal candidate for the design of a mission with a long lifetime such as GeoSAR.

C.10.2 Orbit design

The eccentricity of the orbit is a key parameter as it will impact almost all of the aspect of the global design such as the power consumption, the integration time, the ground-track or orbital drifts. Hence, collaboration with the concerned work packages was of absolute necessity. The most noticeable impact of the eccentricity will be to allow better imaging performances but at the cost of a higher station keeping cost. The argument of perigee will determine the shape of the ground track. The shapes will vary between an ellipse and inclined analemmas as shown in Figure C.17.

As mentioned above, collaboration with other work packages is of utmost importance for this study. The final orbit design proves it as the design was obtained after several iterations from a circular shaped orbit-track. As the report will demonstrate, an analemma shaped orbit track is a better design. Unfortunately, by the time I had enough understanding of SAR and InSAR imaging to realise that an analemma shaped orbit track could potentially be better, it was wisest to continue by slightly modifying the circular shaped orbit rather than using a totally new design since each iteration required lot of work from every work packages. Just as the launch vehicle designs are ameliorated versions of existing designs. That could have been avoided if interactions within the work packages were stronger at the beginning of the project.



(a) Circular (b) Inclined ellipse (c) Inclined analemma

Figure C.17: Different attainable ground-track shapes

C.10.3 Station keeping

The report will estimate the DeltaV budget required for station keeping for the final orbit design and the analemma orbit design. According to (Ahmad, 2014), station keeping will consist of keeping the eccentricity, inclination and longitudinal position drift within dead bands. However, (Ahmad, 2014) did not consider the drift of the Argument Of Perigee (AOP) and without correction, the ground-track would change from an ellipse to an inclined analemma within the first decade. I realised this toward the end of the project. Thus, there was no time to include the DeltaV budget required to control the AOP drift in the final design. However this report evaluates the added DeltaV by using a single burn method or a two burn strategy inspired by Gauss's Perturbation Equations for Osculating Elements.

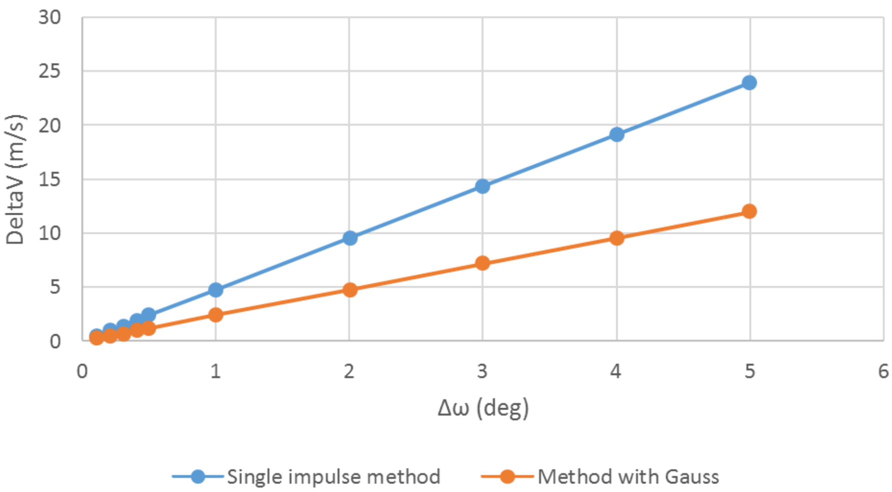


Figure C.18: Apse line change DeltaV cost

As the increase in DeltaV budget impacts the overall design and might render it more expensive than the ESA sentinel missions, ways to reduce the overall DeltaV budget will be suggested.

C.10.4 Analemma orbit

As mentioned above, an analemma shaped orbit-track could be for more efficient than an elliptic one. For example, one of advantages of this orbit is that it uses more efficiently the relative orbital speed as shown in Figure C.19.

The report will attempt to demonstrate quantitatively that such an orbit will give better result for a smaller station keeping cost.

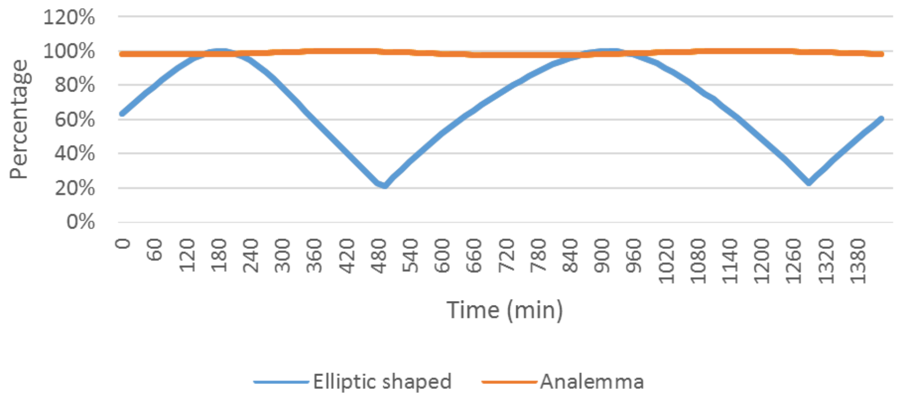


Figure C.19: Percentage of the total relative velocity used comparison during 24 hr

C.10.5 Conclusion

GeoSAR is an ambitious project that could greatly influence Earth observation. The orbit work package proved to be challenging as it required to take a multitude of parameters into account. Even though the orbit design only has two important variables, each of them has enormous impact on the overall design. Thus, interaction with almost all the other work packages was mandatory.

The Station keeping DeltaV budget might pose a threat to the feasibility of the mission. However, the design still present many area that can be optimized and many solutions exist to make the 50 years long GeoSAR mission.

C.11 Software Architecture, OBDH: Moreno Peroni

The basic idea developing the software architecture has been to create a link between the user and the SAR data. This link is depicted in the following

UML Use-Case diagram in Figure C.20. It is possible to understand the functionalities and features of the software from different points of view. Note the rule of the scheduler, that contains the mission duty cycle. An important aspect is to ensure an access to mobile devices as well, integrating other resources like satellite maps, GNSS (e.g. Galileo), and so supply a global product to the final user, for instance in an emergency theatre.

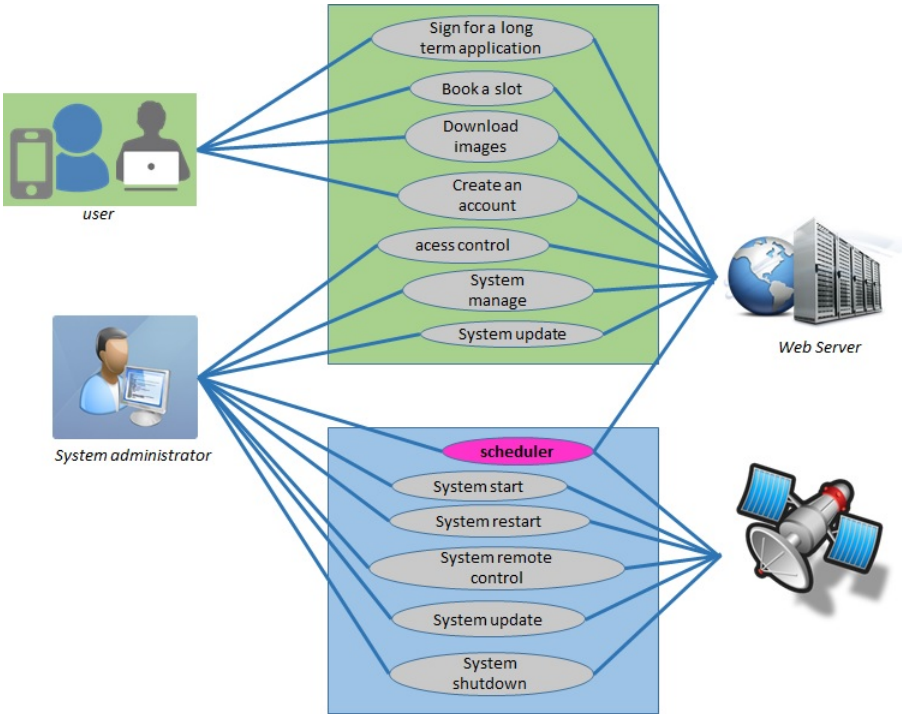


Figure C.20: UML Use-Case diagram for the GeoSAR mission.

In the on board software a key point is to provide "Failure Detection Isolation and Recovery" functionality, both at top level and in each mode.

Following a modular approach, picture (C.21) presents the on board software layered architecture, as an expansion of the basic concept of "Hardware, Software, Documentation".

Note that the scheduler is not the "*task manager*" but the database that contains payload scheduled applications, which must be updated after the interaction with the user, who is able to book a slot or a service.

Looking at past missions lines of code, throughput and RAM memory have been estimated. The reference language is C with a 32-bit processor. Results in Table (C.13).

Parameter	Estimating formula	Estimate
SLOC	$92000 \times 4 + \text{margin}$	**400000 *SLOC***
Throughput	$2 \times 10.5\, MIPS$	**21 *MIPS***
RAM memory	$4 + (4.8 \times 1.3_{EDAC})\, Mbyte + margin$	**16 *Mbyte***

Table C.13: Final estimates for the On Board Software.

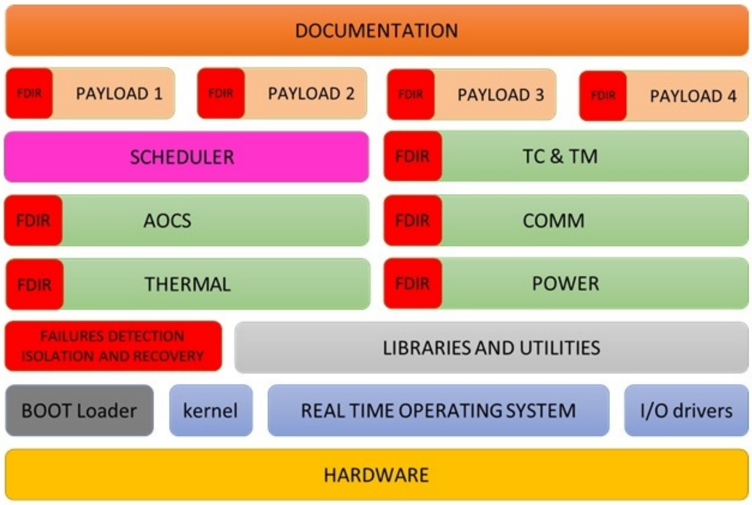


Figure C.21: Software layered architecture.

C.12 On Board Data Handling

OBDH system will comprehend two central units and a mass memory storage, in addition to a smaller buffer memory inside each unit. Internal design of the unit is in Figure (C.22). Units

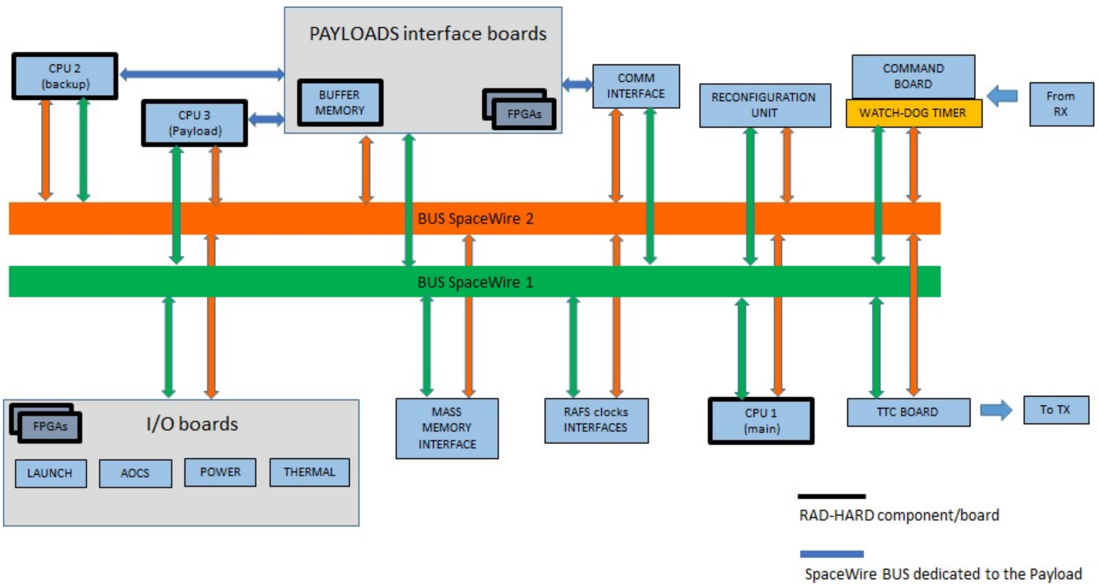


Figure C.22: OBDH unit architecture.

are separated by an optical insulator unit that allows connections with subsystems and between primary and secondary units. The optical unit avoids direct electrical connection within units and subsystems.

The selected bus is SpaceWire, because it can easily handle 300 Mbps, and using SpaceWire routers and interfaces is possible to create fully redundancy and more patterns between components.

The mass memory can be used if the buffer memory fails, or to store data if there are problems in the downlink channel, or at ground station level. The 72 *Gbit* buffer memory can store 3 minutes of satellite work, while the 2 *Tbit* storage can contain one hour of continuous work.

Rubidium Atomic Frequency Standard (RAFS) is the implemented clock to synchronise the

	Max Data Rate considering multispot in L (Mbps)	Adding EDAC 30% (Mbps)	Max Data Rate considering multispot in X (Mbps)	Adding EDAC 30% (Mbps)	Max Data Rate considering multispot in L and X (Mbps)	Adding EDAC 30% (Mbps)
L band	9	12	0	0	2	3
X band	0	0	175	228	140	182
IMAGER	21	27	21	27	21	27
RAD DETEC	0,0003	0,0004	0,0003	0,0004	0,0003	0,0004
	TOTAL	39		255		212

Table C.14: Mission estimated data rates.

satellite. To achieve full redundancy and a graceful degradation the twin unit design has been

	Component name
PROCESSOR	GR712RC Dual-Core LEON3FT SPARC V8
FPGA	XILINX VIRTEX-5QV
BUFFER MEMORY	AEROFLEX UT8SDMQ64M48
STORAGE RECORDER	AIRBUS D&S SS Recorder
RAFS Clock	EXCELITAS Space-Qualified RAFS

Table C.15: Components.

adopted, Figure (C.23).

Because one of the main problem for space electronics is the radiation environment, units are enclosed inside pods. Several materials have been considered: metals, composites and plastics. Aluminium has a good density, although it cannot stop all different types of radiation (e.g. GCRs). It has been used for years in space applications and with last measurements made by Indian GSAT-2 Bhat, Upadhyaya and Kulkarni, 2005, 6 mm thickness can ensure a good shielding.

In closing, mass and power budgets are in Table (C.16).

OBDH	Mass (Kg)
Components, wires, cables, supports for **two units**	120
Mass Memory (SS Storage)	10
Shielding boxes 6mm Al thickness	35
TOTAL	165

(a) Mass budget

OBDH	Power (Watt)
System with one unit ON	270
1/3 secondary unit ON	90
TOTAL	360

(b) Power budget

Table C.16: Mass and power budgets.

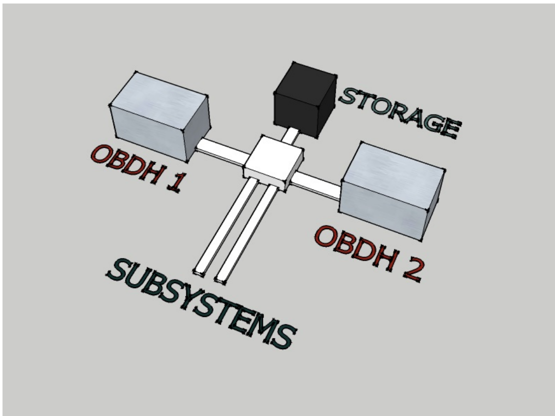


Figure C.23: Basic OBDH concept design.

C.13 Mechanisms, Payload Mechanical: Jesus Quirce-Garcia

This section summarizes the activities and results of the following two related work packages carried out by the author as part of the GeoSAR Group Design Project for the MSc in Astronautics and Space Engineering in the academic year 201415 at Cranfield University:

- 3400 Mechanisms
- 5400 Payload Mechanical

Both work packages revolve around the mechanisms needs of the overall system while the second focuses on the specific needs of the payload.

During the first period of the project the activity focused on the evaluation of all sort of mechanisms related aspects relevant for the feasibility of the mission. During the second one, the scope of the activities has been narrowed to those mechanisms that are not covered explicitly by other subsystems. Four assemblies performing major functions within the system were identified within this scope:

- Payload large antenna reflector and deployment mechanisms
- Solar arrays
- Secondary payload camera rotation mechanism
- Data download antenna re-pointing mechanism

Special effort has been devoted to the solar arrays, performing a conceptual design intended to cover a wide range of aspects of mechanisms technology.

C.13.1 End to End Issues

For feasibility purposes, the major challenge to the mission and systems is its very long duration of 50 years. Technologies on board GEO communication satellites can potentially cover the needs of our mission, but in almost every case they have been qualified and marketed for lifetimes of just 15 years.

Mechanism Considered	Risk or Opportunity
Solar array pointing drives	Number of cycles?
Antenna reflector	No European technology. Fatigue issues in the long term?
Antenna reflector deployment and holding boom	Fatigue issues in the long term?
Active thermal control mechanisms	Number of cycles? Long term degradation of thermal fluids?
Spinning and/or reaction wheels or momentum control gyros	Number of cycles?
“High performance” or magnetic bearings	Potentially less wear and tear. No clear in flight heritage.
Power storage momentum wheels	Combined power and momentum management subsystem.
“Solar flaps”	Potential reduction of thrusters usage for orbit management and attitude control.

Table C.17: Design options and potential concerns for GEO SAR mechanisms

This fact called for the identification of architectural and technology solutions beyond the usual envelope. Not only mechanisms seemed strained by lifetime, but other subsystems like power management or attitude control were as well affected so an effort was made to consider unconventional mechanisms solutions.

C.13.2 Antenna Reflector and Boom Assemblies

The most relevant aspects to consider for the selection of a large antenna reflector are mechanical reliability, long term stability and surface accuracy. Another relevant aspect is the ability to accurately verify performance under 1 g conditions. With these general criteria, plus the common ones of heritage and suitability for our application, a trade off was performed, choosing AstroMesh by Northrop Grumman as reference solution.

Figure C.24: Example Astromesh antenna

Reflector Model	AM-1
f/D	0.4
Aperture Diameter	13 m
Mechanical Diameter	16 m
Number of Bays	42
Batten Length	3.39 m
Longeron Length	1.2 m
Stowed Height	4.59 m
Stowed Diameter	1.175 m
Feed to Reflector Base	5.295 m
Mass, including Boom	115 kg

Table C.18: Astromesh antenna specification

It is worth noting that an aperture diameter of 13 m is not an issue with the current state of the art, but AstroMesh presents the best option in terms of surface accuracy control and testability on ground. Regarding robustness in the long term, NASA has launched a mission that will spin a smaller reflector for several years using this technology. Once chosen, the reflector was dimensioned according to the needs of the mission, deployed and stowed mechanical envelopes and off-axis configuration.

C.13.3 Solar Array Assemblies

The feasibility of the solar array has been covered in a less conventional manner. The consolidation of operations and products led to the conclusion that the power needs of the system were modest compared with GEO communications missions. Accordingly most technologies and solutions are actually oversized for our purposes.

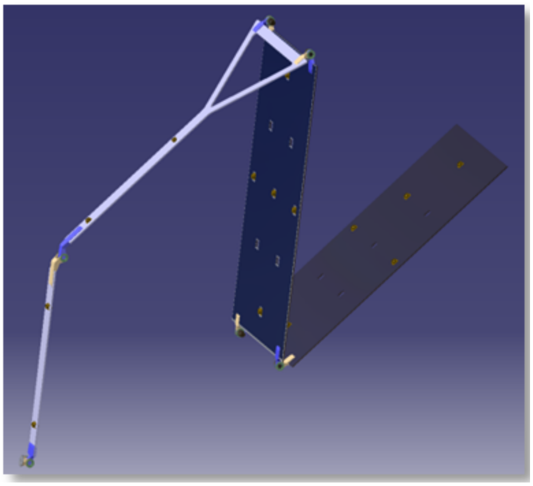


Figure C.25: Example solar array

Parameter	Value
Mass	257 kg
Lowest Natural Frequency (Bending)	0,298 Hz
Motorization Budgets	0,158 (> 0)
	298% (> 175%)
	408% (> 175%)
Power budget (Peak)	7,615 W
Power budget (Nominal)	5,941 W

Table C.19: Solar array mechanical specification

This was taken as an opportunity to design an ad hoc solution using an innovative architecture based on joints actuated by electrical motors, all based on European technologies, by RUAG. To achieve a sensible baseline several aspects had to be considered in some detail: mass, vibration modes, hold and release devices, motorization budgets, operations, folded and deployed configurations. Overall, it serves as an exercise of mechanism conceptual design, demonstrating as well feasibility.

C.13.4 Other Mechanisms

The activity covered the identification of the best options to implement a rotating optical camera and the pointing of the data downlink antenna.

The debris scanning secondary payload requires the rotation of a telescope at low speed. A well established design solution was chosen, featuring a hollow shaft brushless motor, slip rings for data and power transmission, a clamp band based hold and release mechanism and optical resolvers to achieve the required accuracies. The European company VTT manufactures a compliant device, scalable for our needs.

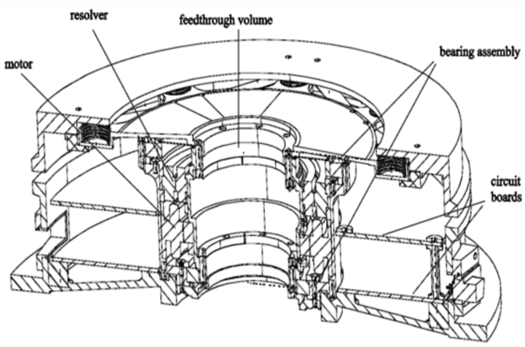
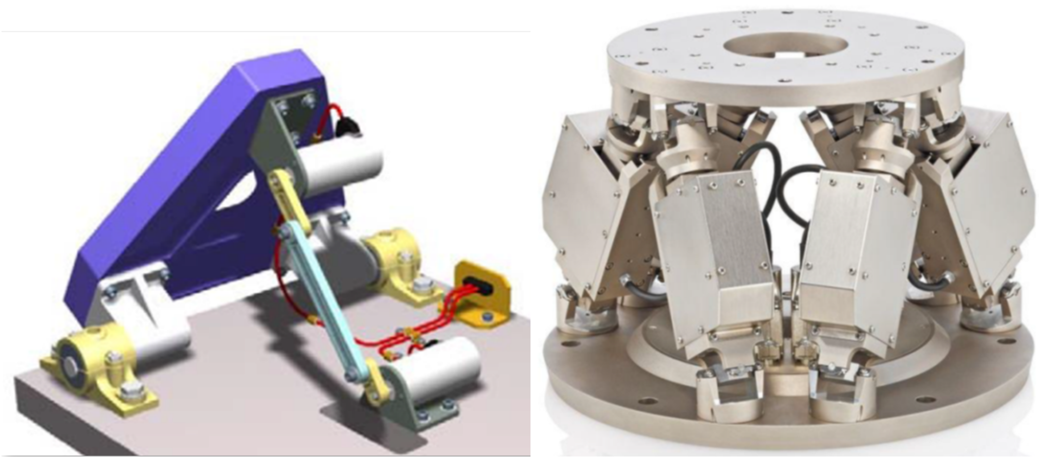


Figure C.26: Camera rotation bearing

Despite the wide 3 dB lobe of the data download dishes, two scenarios require re-pointing to ensure optimal communication. The first involves re-pointing the secondary dish to the primary station, while the second implies free re-pointing of both antennas in specific acquisition scenarios. Several alternatives have been identified, being the most promising a deceptively simple mechanically multi-stable device developed by NASA and a compact hexapod. New implementations are needed, but feasibility is not compromised.

C.13.5 Conclusions

In the scope of these Work Packages a number of mechanisms have been evaluated, dimensioned or designed to cover the need of the chosen mission and system baseline.



(a) NASA multi-stable device (b) Compact hexapod

Figure C.27: Candidate payload steering mechanisms

The choice of architecture and selection of technologies has been somehow conservative. There is a wealth of GEO technology with long heritage. Continuously operating mechanisms are restricted to the solar array drives, and the performances required in terms of power and number of rotations are reduced compared with other missions. The chosen devices can be considered derated for this application. The effect of a failure in the secondary payload is not critical for the mission. The number of cycles of antenna re-pointing is an open issue, but it cannot be considered a case of continuous operation. The actual, but rather fuzzy, incognita is the combined effect of radiation, vacuum, thermal cycling and time on the most critical devices. Loss of surface accuracy in the reflector due material fatigue seems a remote possibility. Overall, feasibility of the GeoSAR mission is not compromised by mechanisms.

C.14 Configuration, Structure: Carlos Roldan Rueda

This appendix provides a summary of the configuration and the structural subsystem development process of the GeoSAR. It describes the requirements which both subsystems should meet as well as constraints imposed by other work packages (AOCS, Payload Performance, Thermal, Secondary Payload and Mechanisms).

C.14.1 Introduction

GeoSAR is a word formed by two acronyms, Geo which means Geosynchronous Orbit and SAR which means Synthetic Aperture Radar. A satellite in GEO orbit is subject to a continuous view over continental regions, either at day or at night and during all kind of weather. The technology used is sensitive to the atmosphere and the land surface, which provides new significant imaging capabilities.

The applications cover a wide range of areas, in order to directly improve life on Earth as well as being a complement to LEO satellites. Some applications which can be considered are the detection of snow mass, volcano eruption hazard, earthquake response, agriculture, hydrology, methane gas, flooding and glacier motion.

C.14.2 Requirements

The derived requirements from the top level ones, which the Configuration and Structure subsystems need to meet are:

- Lifetime of the satellite shall be 20 years
- Equipment accommodation shall be in order to avoid interference between each other during launching or in orbit
- Minimise perturbations and shadows from solar panels and SAR antennas
- The configuration shall accommodate all payloads, taking into account fields of view from SAR antennas, communication antennas, solar arrays, AOCS sensors, secondary payload, etc.
- Carry a high amount of fuel in order to reach final orbit, orbital manoeuvres and provide enough fuel to the end of life propulsion
- If it is possible, the structure should be made by European technology
- The satellite structure shall be constructed of materials that ensure structural integrity whilst optimising mass and cost

Likewise, there are many constraints from other work packages, which have been developed during the project:

- The satellite must withstand the maximum launch loads (accelerations)
- Natural modes of vibration higher than the limits stated by the launch vehicle, in longitudinal and lateral axes
- Centre of gravity at launch must be within limits imposed by the launcher
- Satellite must fit into the fairing and if it is possible, we would use the SYLDA 5 module to give the chance of a dual launch, decreasing the launch cost of our satellite
- Axes of inertia shall be as close as possible to spacecraft axes, giving the moments of inertia greater than the products of inertia, producing a diagonal matrix of inertia
- In orbit, the centre of pressure (large SAR antennas) shall be as close as possible to the centre of mass so that solar flux perturbations are avoided
- Safely deploy of SAR antennas in orbit while maintaining the satellite, in stowed configuration, within limits of the fairing volume.

C.14.3 Configuration

After careful consideration of the requirements and constraints, as well as trade-offs between different body shape possibilities and locations for the payload, the next figure shows the final configuration with the components labelled. The platform selected is Alphabus (European technology).

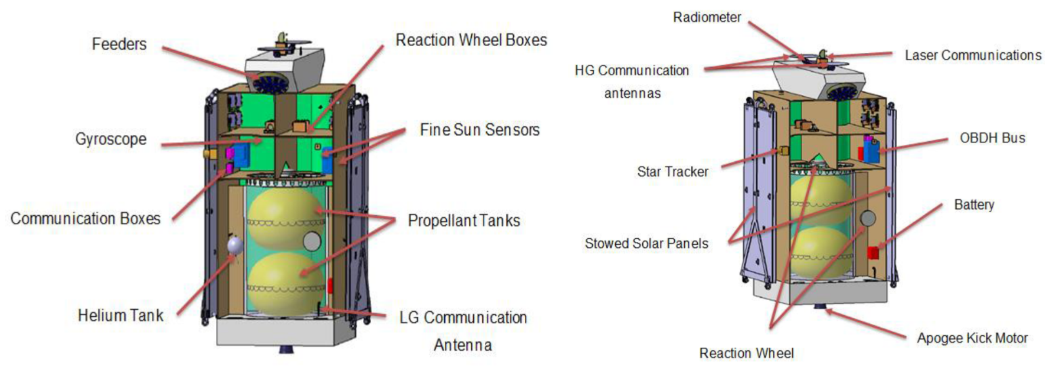


Figure C.28: Final body configuration with components labelled

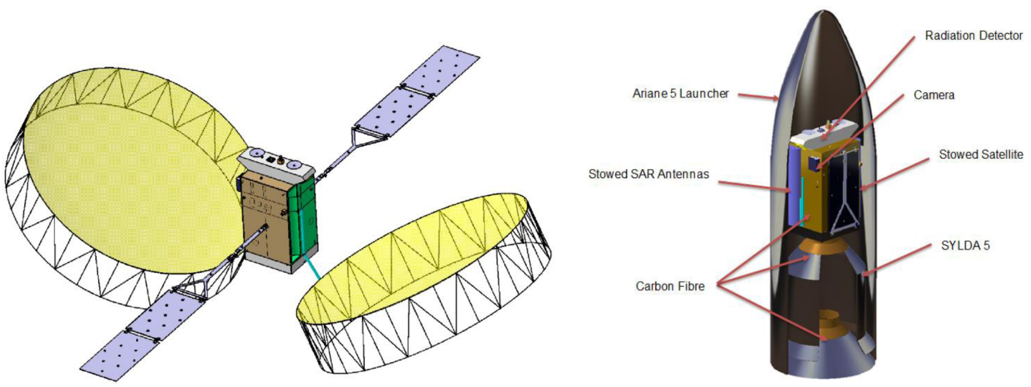


Figure C.29: Deployed Satellite Representation and within the fairing

C.14.4 Structure

In order to size the structure and ensure that it withstands the worst load cases of the entire mission, hand calculations and simulations using Nastran/Patran have been performed. Also, a first trade-off between different materials usually applied to spacecraft structures has been done.

	Material	Thickness (μm) Core/Skin
Cylinder	CFRP	0.00202
Panel E/W (2,45x4,3)	Sandwich CFRP/Al	0.03/0.0002
Panel N/S (2,8x4,3)	Sandwich CFRP/Al	0.04/0.0002
Deck (2,8x2,45)	Sandwich CFRP/Al	0.025/0.0002

Table C.20: Final materials used for the primary structure and their thicknesses

The final figures show simulations performed using Nastran/Patran with the same materials and thickness calculated in the previous step. The left image shows the first normal mode of vibration of 34 Hz, on the middle the axial stresses produced by launch acceleration and on the left, the lateral stresses at launch:

	Total Mass (kg)	Total Mass with Margin (kg)	Quantity
Cylinder	95,76	114,91	1
Panel E/W (2,45x4,3)	30,6	36,72	2
Panel N/S (2,8x4,3)	23,1	27,72	2
Deck (2,8x2,45)	13,7	16,44	3
Total mass		293,11	

Table C.21: Total mass of the structure

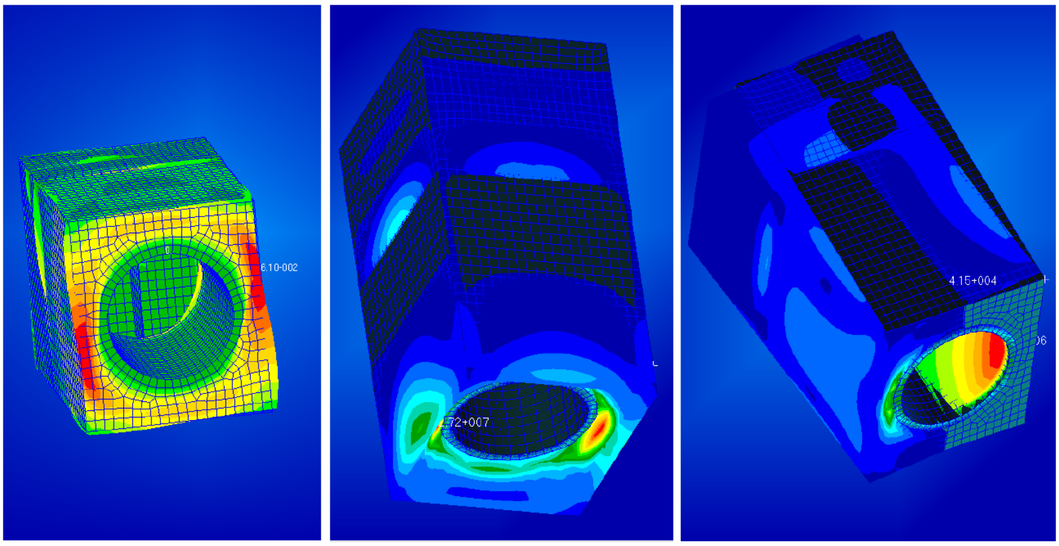


Figure C.30: Nastran/Patran simulation results

C.15 Launch: Eddie Ross

The mission objective was to produce a feasibility study of a long life GeoSAR mission to propose to ESA. The GeoSAR satellite of mass no less than 3100 kg shall be inserted into a Laplace orbit of inclination 7.5°.

The purpose of this study was to select the most suitable launch vehicle (LV) for the mission and determine a method of inserting the GeoSAR satellite into its operational orbit.

C.15.1 Orbit Raising Selection

Upon inspection of the LVs available, it was clear that no LV would be able to insert the GeoSAR satellite into the Laplace orbit, inclined at 7.5°. Therefore a method of inserting the satellite into its operational orbit was required.

Traditional methods, performing manoeuvres at apogee and perigee can change the semi-major axis and eccentricity to what is required. Chemical propellant motors such as solid propellant, liquid monopropellant and liquid bipropellant are used for these manoeuvres and were investigated in this study.

Chemical engines require a large percentage of mass as fuel. Electric propellant motors have a much higher performance in this regard and use a significantly smaller percentage of their mass on fuel. Electric propellant motors however have much lower magnitudes of thrust and hence take significantly longer to achieve a given ΔV than chemical motors. The balance between the time taken to provide the ΔV and the required fuel mass was the main influence of the trade-off analysis on orbit raising options.

Table C.22 shows the trade-off performed between the different orbit raising options considered in this study. The trade-off highlighted a liquid bipropellant propulsion system to be the most suitable system for inserting the GeoSAR satellite into GEO from a GTO. Despite saving a large amount of mass if electrical propellant was used, it was found that the orbit raising period would be too long to meet the requirement of the satellite being in operation in 2020.

A number of liquid bipropellant motors were researched and the European Apogee Motor (EAM) was selected to provide the apogee and perigee manoeuvres in order to raise the orbit. The EAM has a thrust of magnitude 500 N, Isp = 325 s and a qualified accumulated burn life of 8.5 hours or 135 burn cycles.

C.15.2 Launch Vehicle Selection

The mission requirements highlighted that the LV shall be European, if possible; hence mainly European LVs were considered, with other non-European options investigated which may reduce cost or improve performance.

In selecting the LV for the GeoSAR mission, the key drivers were mass capability inserted into GTO (and GEO), cost and reliability, with other parameters based on the individual LV performances used to determine the most suitable LV.

Table C.23 shows the options considered and the weightings of each of the parameters used in the trade-off analysis. GTO load (i.e. mass inserted to GTO) was the highest weighted parameter as the main responsibility of the LV is to insert sufficient mass into orbit. If it could not do this, it should not be a feasible selection to launch the GeoSAR satellite. Cost was the other main parameter, as the mission requirements state that the mission shall be cost effective in relation to the ESA sentinel mission which launched on the low cost Soyuz LV.

From this trade-off analysis, the Ariane 5 ECA LV was selected for the GeoSAR mission launch. It has the capability of launching 10.05 tonnes into a GTO with an inclination of 6°, semi-major axis of 24474.5 km and eccentricity of 0.7297. Ariane 5 is a very reliable, European LV, typically used for dual GEO satellite launches; hence its heritage and performance makes it an ideal LV for this mission.

C.15.3 Launch and Budgets

A dual launch is to be used and the GeoSAR satellite shall be attached to the Ariane 5 ECA on top of the SYLADA 5 type E adapter; hence it shall be the upper satellite in the dual launch and

Table C.22: Trade-off analysis of orbit-raising options

	Weight	Solid	Mono-prop	**Bi-prop**	eSpiral	eApogee boost	MEO mix
Raising period	*7*	9	7	**8**	2	1	5
Mass budget	*9*	4	3	**5**	8	9	7
Fuel reusability	*1*	0	8	**8**	8	8	8
Power budget	*1*	7	7	**7**	4	4	4
Cost	2	7	5	**5**	5	5	4
Total:	19	120	101	**126**	108	110	118
	%	60.0	50.5	**63.0**	54.0	55.0	59.0

first inserted into the GTO.

The launch into GTO shall commence in late December 2019. A total of 7 burn manoeuvres is required to insert the satellite into the operational orbit from the GTO. This orbit raising campaign will take approximately 2.15 days (51.46 hours), to apply a total $\Delta V = 1461.3209$ m s^{-1}. Allowing the GeoSAR satellite to be in operation in early 2020.

To achieve the baseline mass inserted into GEO (with around 10% margin), a launch mass of 6000 kg is required. The breakdown of masses and volumes used in the orbit raising is shown in Table C.24.

Table C.23: Launch vehicle trade-off analysis

	Weight	Ariane 5 ES	**Ariane 5 ECA**	Ariane 6	Angara A5	Proton Me	Zenit 3SL	Falcon 9	Falcon Heavy
GTO load	*10*	8	**10**	10	7	6	6	5	10
Cost	*7*	3	**2**	4	5	4	7	6	5
Reliability	*5*	10	**9.8**	0	5	9	8.9	10	0
Injection inclination	*4*	8	**8**	8	3	3	6	3	3
Availability	*2*	7	**7**	7	0	1	4	4	4
Injection accuracy	*3*	7	**7**	7	5	4	6	7	7
GEO load	*3*	0	**0**	0	5	4	0	2	2
Total:	*34*	218	**229.9**	195	172	171	203.4	189	182
	%	64	**67.6**	57	50.6	50	59.8	55.6	53.5

Table C.24: Mass and volume budgets of the orbit raising

Item	Mass / kg
Launch Mass	6000
Propellant Mass Required	2206.027
Propellant Mass (+20% margin)	2647.233
Mass into GEO	3352.767
MMH Mass	998.956
MMH Volume (m^3)	0.9459
N2O4 Mass	1648.277
N2O4 Volume (m^3)	0.9539
EAM Mass	5

C.16 Communications: Adrian Segura

This report is organized in different parts where, based on the requirements of the mission and inputs from other WPs, the decisions that will implement the final design will be justified. It can be summarized in:

- Ground Stations. Given the available GSs that are part of ESA Network a trade-off will be carried out to determine those that will be necessary during Launch and Early Orbit Phase (LEOP) and the ones that will perform best during nominal operations.

- Link Budget. Once the GSs have been chosen the link budget can be calculated as the path of the link from SC to GS can be defined. This will provide a knowledge of the atmospheric and free space losses affecting the signal thus allowing a calculation of the power required to reach the GS within requirements. A trade-off is needed when designing the X-Band antenna in order to achieve a balance between gain, beamwidth and mass.

- Design. Finally, based in the previous points, the configuration of the communications system onboard the spacecraft will be developed. This will include the equipment needed, the block diagram showing the connections between the different devices onboard and possible commercial products that could be suitable for the mission.

C.16.1 High-gain antenna

Figures C.31 and C.32 contained in this section include information about the performance and characteristics of the high-gain antenna considered as a function of its diameter.

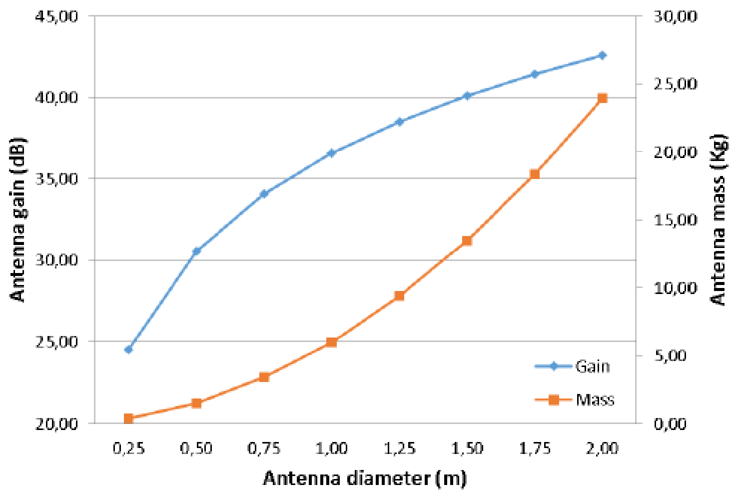


Figure C.31: Antenna gain vs mass

C.16.2 Antenna location

Figure C.33 shows the position of the S-Band and X-Band antennas in the SC. There are 4 S-Band antennas although there is one of them that is not visible in the figure. The radiation patterns of the 4 antennas are combined to surround the SC. The X-Band antennas are placed on the face of the SC that is pointing to Earth.

C.16.3 Link budget

Figures C.34 and C.35 show the power required for transmitting data to Maspalomas and Kiruna as a function of the data rate with a 3dB link margin.

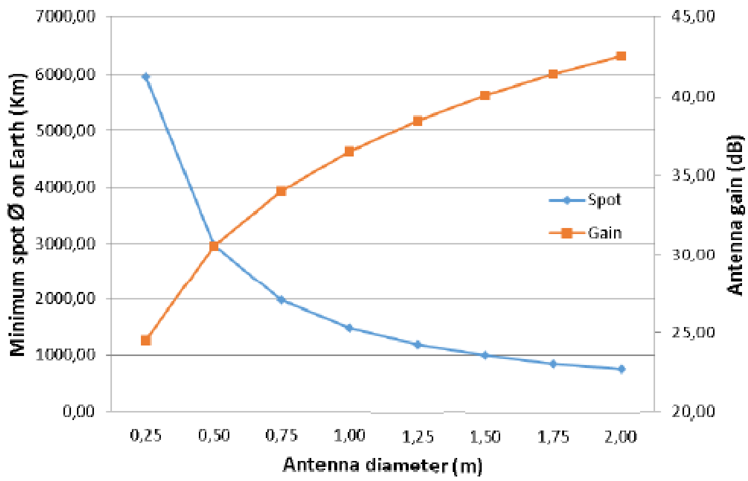


Figure C.32: Spot on Earth vs gain

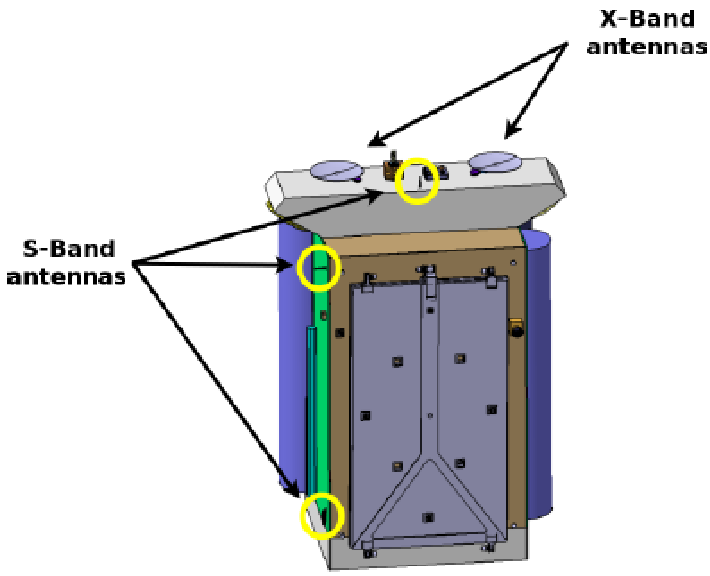


Figure C.33: Antenna location on SC

C.16.4 Link margins

Figures C.36 and C.37 show a comparison of the power required for transmitting data to Maspalomas and Kiruna with different link margins as a function of the data rate.

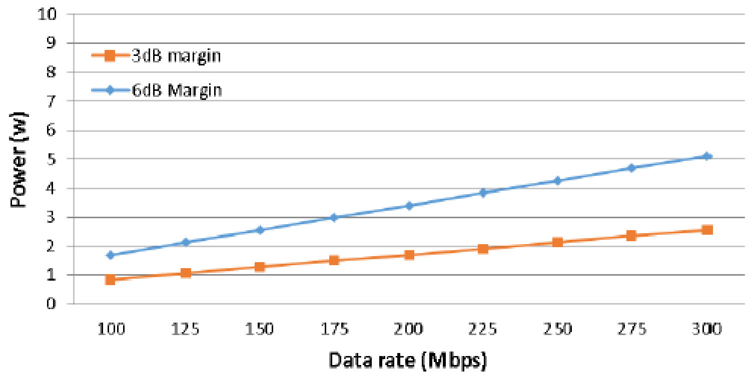


Figure C.34: Power required for Maspalomas GS

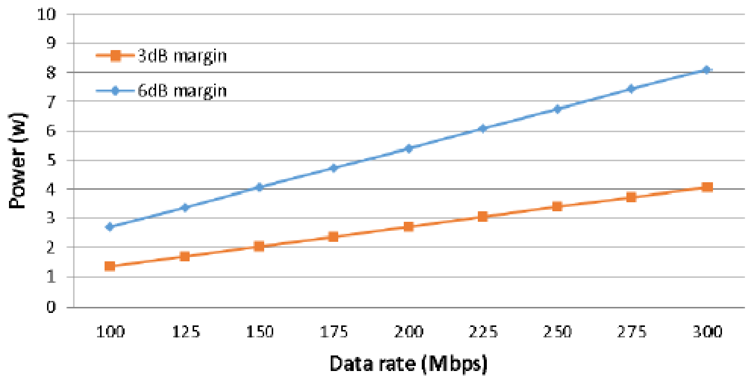


Figure C.35: Power required for Kiruna GS

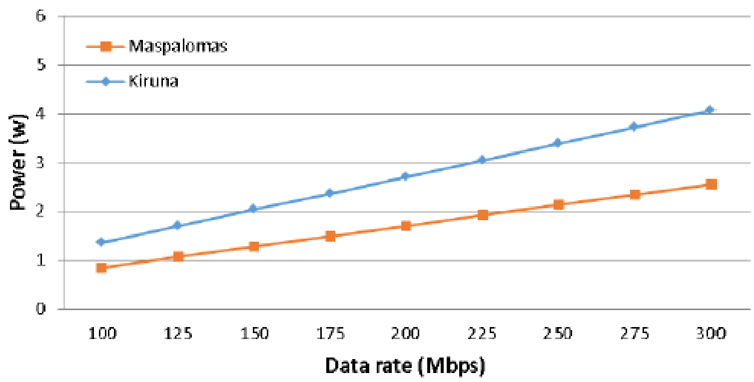


Figure C.36: Power required for 3 dB margin

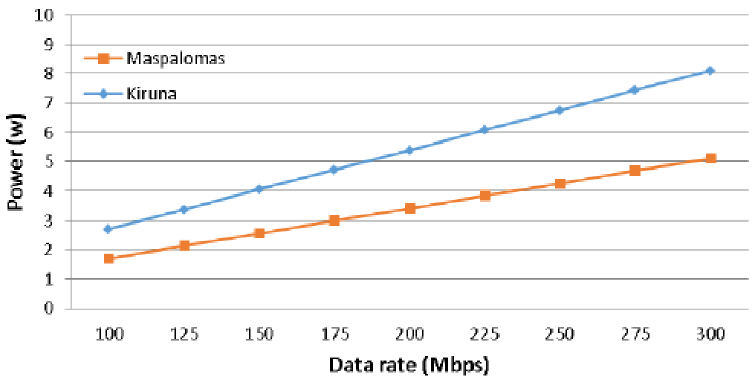
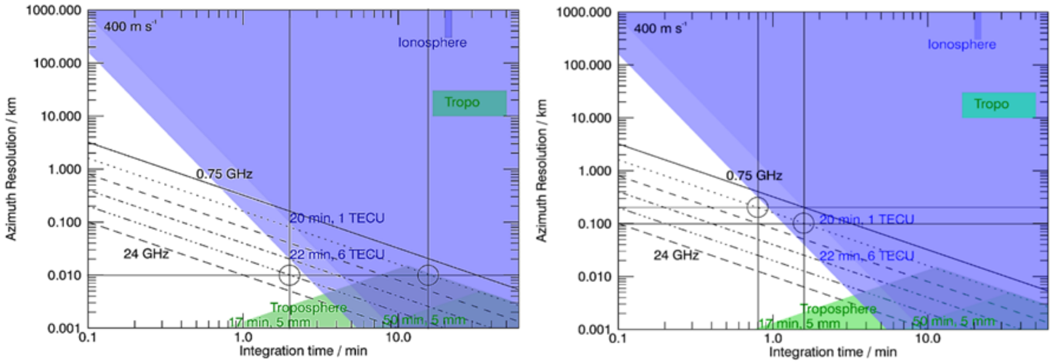


Figure C.37: Power required for 6 dB margin



(a) X-band

(b) L-band

Figure C.38: Resolution / integration time trade-offs for the Laplace plane GEO SAR

C.17 Payload Performance: Chris Slattery

C.17.1 Introduction

In determining the parameters for the Synthetic Aperture Radar (SAR) the first task was to identify and eliminate the main challenges which will affect the performance of our satellite. One challenge is the effect that the changing atmosphere has on the radar signal as it travels through it.

The choice of frequency bands was made through a trade-off between the resolution and coverage. L-band provides broad coverage at a course resolution and X-band provides fine resolution with approximately 10% of the L-band spot diameter.

C.17.2 Integration time limits

Changes in the atmosphere cause a change in the refractive index which in turn has an impact on the path length that the radar signal travels. If the atmosphere changes during the image integration period (while the synthetic aperture is being formed) then the phase information will become erroneous making it difficult to determine range and producing incoherent images. Because of this the integration times are limited in each band. As L-band is affected more significantly by the ionosphere it has a shorter integration time. X-band is affected more by the troposphere which changes more slowly so a longer integration time is permitted.

Here we see how the integration time will be limited in both bands in order to avoid having to correct for atmospheric effects:

- The image integration time shall not exceed 180 s (3 min) in X-band to avoid Tropospheric interference.

- The image integration time shall not exceed 120 s (2 min) in L-band to avoid Ionosphere interference.

This requirement can be relaxed if Atmospheric Phase Correction is used however for the preliminary design this restriction has been imposed for simplicity. If it is possible to image in this way from GEO then the processing of images will be both quicker and simpler.

C.17.3 Antenna Sizing and SNR

When considering the sizing of the on-board reflector antennas the signal to noise ratio has been over-estimated in order to give margin to the system.

The antenna size can be determined for a SAR system independently of the wavelength being used as long as the limits to the integration time have been identified.

Cluster #	Application	Spatial Resolution	Temporal Resolution (hrs)	No. Looks	No. Spots	Usable Velocity (m/s)	Tint (s)	Power (W)
1	APS	1000	0.25	3	2	337.2	22.3	271.2
	Hydrology	1000	1	3	2			
	NWP	1000	0.25	3	2			
2	Snow Mass	200	2	3	1	433.4	70.9	882.9
	Snow Cover	200	24	3	1			
3	Agriculture	100	24	3	1	542.1	98.8	1705.2
	CH4	100	36	3	1			
4	Volcano	200	3	1	1	431.7	70.2	832.2
5	Earthquake Response	100	6	1	1	541.0	99.4	1739.6

Table C.25: L-band power requirements averaged over the usable orbit

This is done using the equation:

$$A^2 = \frac{16\pi r^2 S F_n k T_s}{P_t F_t \sigma^0 \cos\theta} v^2 t_{int} \tag{C.2}$$

Plotting this allows for the antenna diameter to be read off. As both bands require an antenna 13 m in diameter, the parameters were optimised for to allow them to be equal (at 13 m) for symmetry, making other areas of the system design easier.

C.17.4 Power requirements

The power requirements for each application have been determined through the use of a rearrangement of equation C.2 relating the spatial resolution and wavelength. This gives the power requirements for the different applications considered. Table C.25 is an example table.

As more information about our orbit became available a more detailed analysis into the power requirements was produced; how it changes over the course of orbit and also for the different areas on the ground in which the applications would be located. This information has not been included here as it would take up too much room but more details can be found in the Laplace Orbit GeoSAR Feasibility Study: Payload Performance work package report (2015).

C.17.5 Antenna Configuration

Having identified that the feeds for the reflectors would be an array of horns to allow for phase controlled beam steering, and having sized the horns (Laplace Orbit GeoSAR Feasibility Study: Payload Performance full report) an offset design was produced. The result of this was a 1 m offset antenna with focal length of 5.2 m.

Including this offset keeps the aperture size as 13 m but increases the physical diameter of the antenna to 16.03 m a result which would be factored into the rest of the design in a further iteration.

C.17.6 Conclusion

The key result of this study is that it appears the Laplace 7.5° orbit is suitable for providing data fulfilling the requirements of the majority of the applications considered while not correcting atmospheric perturbations. The design proposed has a high power requirement and large antennas for imaging. Multi-looking would be implemented in order to reduce speckle noise and improve SNR.

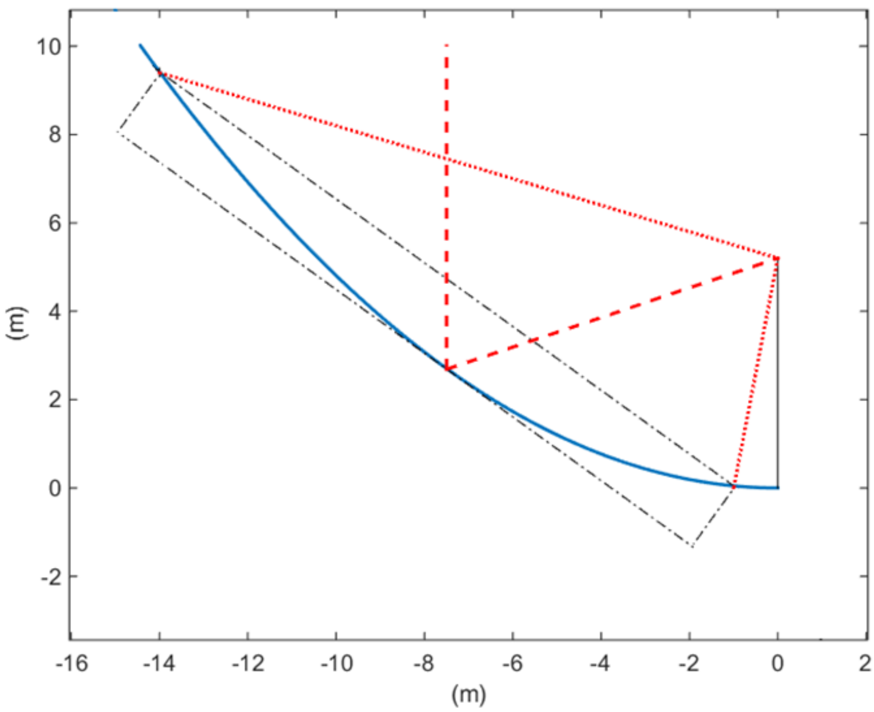


Figure C.39: Reflector geometry suggested to require small offset of antenna feed

C.18 Payload Constellation: Andrea Testore

This report is intended as a summary of the work related to the Group Design Project focused on a feasibility study for a Geosynchronous SAR mission. It collects the considerations about three main mission aspects: Users Requirements, Payload Operations and Constellation study.

C.18.1 Users Requirements

Starting from previous SAR missions the most relevant applications that this system can provide have been identified. Thanks to its Geosynchronous orbit and its long lifetime expectation, this mission can ensure not only a large amount of scientific and commercial data but also valuable aid in case of natural disaster. The applications identified have been grouped considering their arguments. The list of the main functions of these applications with their users requirements is shown below.

- Civil Defence: thanks to its unprecedented reactivity the GeoSAR mission is particularly useful in disaster aftermath monitoring. Moreover thanks to its InSAR capability GeoSAR can provide data related to the soil deformation due to earthquakes and volcano eruptions, helping to forecast natural disaster.
- Hydrology: Using backscatter and InSAR modes it is possible to monitor the amount of water stored within the snow mass in mountain areas. This function can be helpful to forecast flooding risk and agricultural productivity.
- Human Activities: Being focused only on a desired area, GeoSAR mission can provide frequent images of many human activities such as crude oil drilling, soil moisture monitoring for agriculture, and cultural heritage monitoring.
- Weather: thanks to the SAR capability to produce measurement of the water vapour in the low atmosphere layer GeoSAR can provide images needed to implement Numerical Weather Prediction models and to correct other SAR images affected by Atmospheric phase shift.
- Climate Change: With its long lifetime GeoSAR mission can be particularly useful in monitoring the climate change. Methane release in wetland and deforestation can be monitored with high accuracy.

An example table showing the Applications Requirement relative to civil defence applications is reported in Table C.26.

Table C.26: Application requirements relative to civil defence applications

Civil Defence	Wavelength	Mode	δa	Time Resolution	Delivery
Earthquake	0.3 m (L)	InSAR	100 m	6 hr	6 hr
Volcano	0.3 m (L)	Phase Coherence	200 m	3 hr	2 hr
Flooding	0.03 m (X)	Intensity Mapping	30 m	2 hr	/
Earthquake	0.03 m (X)	Phase Coherence	10 m	12 hr	1 hr
Volcano	0.03 m (X)	Phase Coherence	20 m	3 hr	2 hr
Landslide	0.03 m (X)	Phase Coherence	30 m	6 hr	12hr

C.18.2 Payload Operations

Due to its relation with the satellite velocity, the imaging capability of the system is not constant along the day. This phenomenon is due to the time needed to produce the SAR images and the atmospheric shift. The integration time is strictly related to the V_{perp} (perpendicular component of the relative velocity between the satellite and the target). Thus comparing the V_{perp} trend and the minimum velocity required to produce image without atmospheric disturbance it has been possible to identify the imaging availability along the orbit. Table C.28 reports the orbit availability for imaging process with optimal and suggested requirements for the application mostly affected by velocity issues. Proposed requirements have coarser azimuthal resolution in order to reduce the integration time required.

Table C.27: Users Requirements listed with their GeoSAR requirements codes

Ref.	Requirements
SR.02	The system shall provide spatial resolution down to 10 m (UR)
SR.03	The system shall provide imaging capability within 12 hours after a natural disaster (UR)
SR.04	The system shall provide images of at least the European Union (UR)
SR.05	The system shall satisfy all the applications listed in the WP5100 report (UR)
SR.06	The data produced shall be delivered to the customers within scheduled times (UR)
PSuR.01	The system shall ensure imaging capability also in shadowed conditions
PSuR.02	The system shall provide 60 m*hr imaging in X-Band
PSuR.03	The system shall provide 400 m*hr imaging in L-Band
PSuR.04	The system shall ensure flexibility of use in order to be able to provide disaster management aid when required

Table C.28: Orbit percentage available for imaging process for some applications considering customer requirements and proposed requirements and the increment in overall availability

	Application Req.	Suggested Req.	Increment
Earthquake response	35,42%	71,88%	36,46%
Volcano eruption hazard	71,88%	71,88%	0,00%
Agricolture	35,42%	71,88%	36,46%
...	...	...	...

C.18.3 Constellation

Throughout the last chapter three different constellation designs have been analysed in order to understand which of them can provide better advantages to the mission. Each of them has been suggested because of their capability to solve a specific GeoSAR mission issue. Since an handover operation between three S/C has been implemented in order to ensure the lifetime expectation, slightly overlapping the satellites operative life it is possible to ensure cooperation between satellites studied in this report.

Double Monostatic

This design is characterised by a second satellite, placed on a different orbit. This orbit has been studied in order to have a similar ground track but different V_{perp} trend. With this configuration, one of the two spacecraft will always have a higher performance V_{perp} value than the other. The overall imaging capability of the system is significantly increased.

Moreover while with a single satellite design in order to produce a InSAR measurement the satellite has to complete orbit ($\simeq$ 24 hrs) with this design it is possible to produce a InSAR image every 6 hours.

The requirements identified for this constellation design concern the orbital parameters of the second satellite and in Table C.29. As side effect, this configuration has a higher station keeping cost for the satellite that is placed in this orbit, it provokes a higher fuel mass in the satellite.

Bistatic

This design has been studied in order to extend the first generation satellite lifetime. The second generation satellite will gradually substitute the old satellite performing the most demanding func-

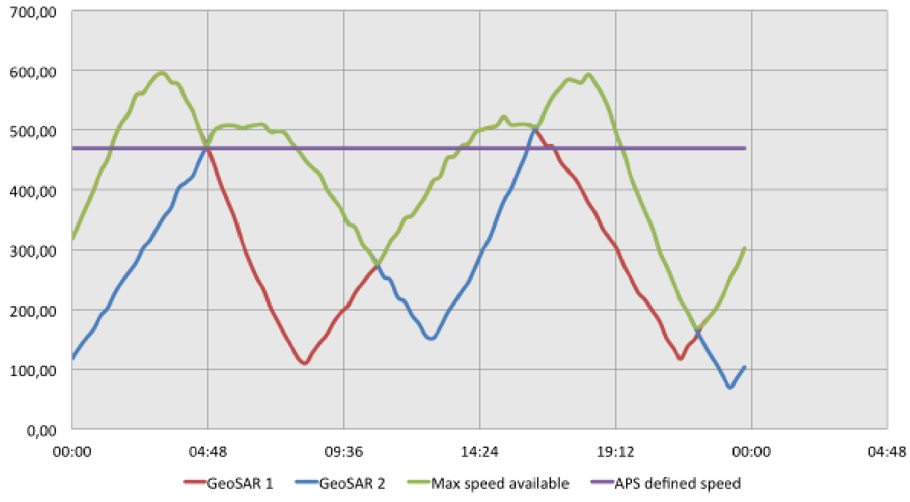


Figure C.40: GeoSAR 1 and 2 velocity trend (m s^{-1}) during the day (hr).

Table C.29: Orbital parameters for Double Monostatic constellation design

SMA	e	i	RAAN	AOP
42164	0.089	7.5	102	90

tions. Near to the EOL of the first satellite will be used just as a passive receiver. Figure C.41 shows the possible evolution of the functions handover between GeoSAR 1 and 2.

To implement this design the system shall satisfy the following requirements:

- Perpendicular baseline: the distance between the two satellites has to be less than 77 km.
- Baseline knowledge: the system has to know the inter-satellite baseline with precision up to 3 mm.
- Time precision: the two satellites have to be synchronised with a precision down to nano-seconds.

Multiple coverage

Increasing the number of satellites placed in the same orbit focused on different locations can increase the amount of data provided. China is the largest food producer of the world, a system such as GeoSAR can provide a huge amount of information. North America, has a quite high percentage of the world gas and oil deposit. Function such as oilfield subsidence can be useful in this area. Finally Civil Defence functions are very important all around the world.

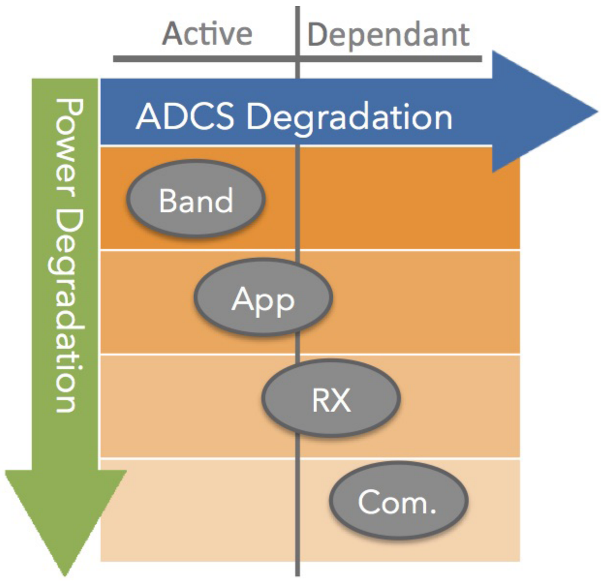


Figure C.41: Functions handover for Bistatic design.

C.19 Requirements and Risk: Alex Trolley

C.19.1 Introduction

This report focuses on elements of systems engineering required for the system level design of the GeoSAR mission, namely requirements and risk. This investigation was conducted at Cranfield University and was designed as a feasibility study. The GeoSAR concept discussed focuses on two main areas of interest; the implementation of synthetic aperture radar from GEO and the use of a 7.5° Laplace plane orbit to reduce station keeping

C.19.2 Requirements

Requirements identification and analysis followed the following format. Figure C.42 displays graphically the requirements derivation process. The mission objective was selected from which a set of mission requirements was derived. Subsequent system and subsystem requirements were also derived. System requirements were categorised into functional requirements, operational requirements and constraints. Subsystem requirements were divided into parent requirements, which were categorised as before, and derived requirements which refer to individual components or modes.

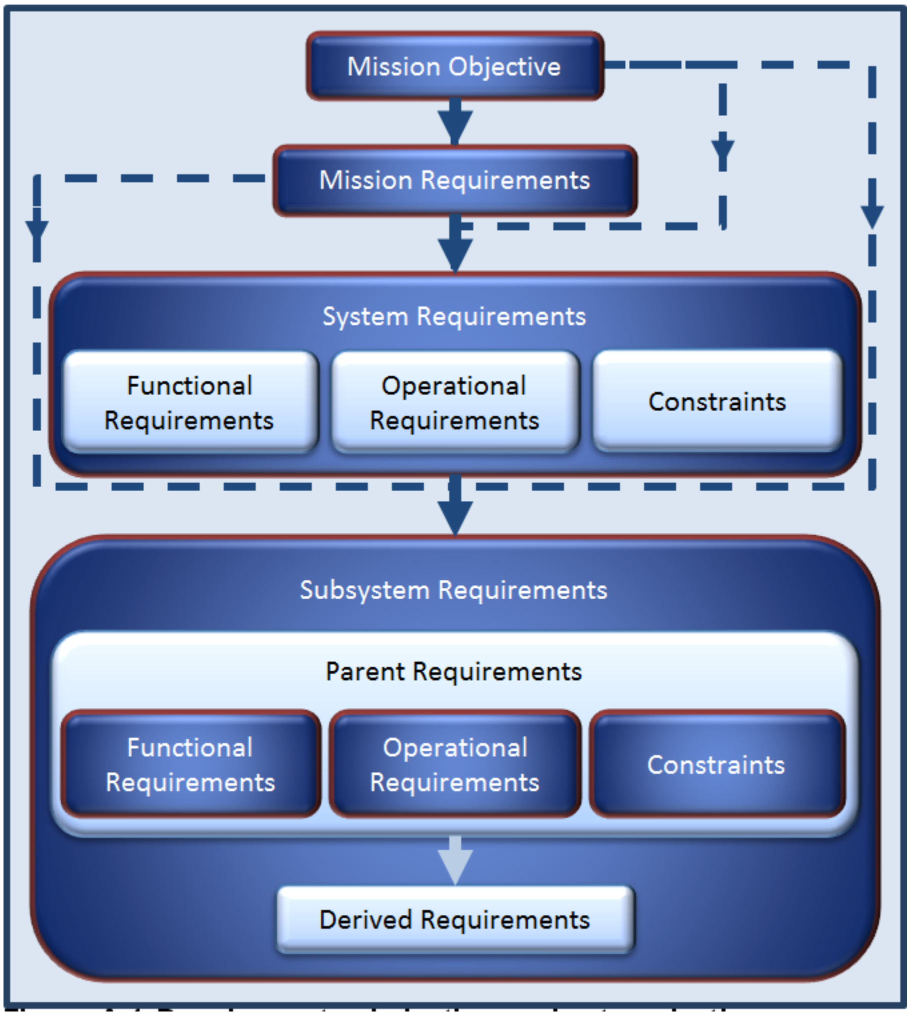


Figure C.42: Requirements derivation and categorisation process

The mission objective was as follows:

MO.01. The mission objective is to produce a feasibility study outlining the system design of a long life GeoSAR mission to propose to ESA.

The mission requirements were derived accordingly

- MR.01. The mission shall be in operation for users no later than 31/12/2020.
- MR.02. The mission shall, in a safe and responsible manner, last for 50 years.
- MR.03. The mission shall implement SAR and InSAR from geosynchronous orbit.
- MR.04. The mission shall be cost effective with respect to the ESA Sentinel missions.

A set of system level requirements was subsequently derived along with individual subsystem system requirements details of which are outlined in the full report.

C.19.3 Risk

Due to the fact that the investigation was conducted as a feasibility study, the approach to risk analysis was a relatively top level one, attempting to identify the highest areas of risk to the mission. For further investigations a bottom-up approach will be necessary including the construction of a FMECA.

Analysis of SpaceTrak database highlighted common failures/anomalies to GEO EO satellites, categorised according to the subsystem to which they occurred and classified according to the severity of their effects.

Risk was calculated as the multiplication of severity of effects and frequency of occurrence. The highest area of risk (49.8% of total calculated risk) to EO GEO missions was Payload Instrument/Amplifier/On board data/Computer anomalies.

Classification	Class I	Class II	Class III-LR	Class IV
No. of occurrences	3	24	7	37

Table C.30: Frequency of different classifications of Payload Instrument/ Amplifier/ On board data/ Computer anomalies.

The sources of risk were investigated in particular long term effects of the space environment on spacecraft. This included considering effects of the following; vacuum environment, collision with space debris, radiation of varying energy and intensity.

A risk register was constructed which listed the main areas of risk remaining to the mission. These risks were classified according to the subsystem to which they would be likely to occur, cause and both local and global effects. The risk level was calculated for each and suggested resultant action. Figure A 3 shows an excerpt from the risk register highlighting the highest level risks.

#	Source	Effect	Class	Resultant Action
3	SPE	Displacement damage and ionisation	V. HI	Shielding & Redundancy
6	Collision with debris 1cm - 10cm (natural or man-made)	Subsystem damage	HIGH	Shielding & Redundancy
9	Rapid SAR ant. pointing	Damage to extrusions/ reduced image cap.	HIGH	Use electric/ mechanical pointing

Figure C.43: Risk register excerpt

C.19.4 Conclusion

While requirements identification is crucial during the early stages of the investigation, risk analysis plays an increasingly important role as the design develops and often feeds back into initial requirements. Therefore this report is paramount for identifying the key drivers in this feasibility study.

It is evident that the requirement for a 50 year mission is a highly influencing factor on the design of the mission. Certain design choices, such as the Ariane 5 as the choice of launch vehicle, are not likely to change were certain individual requirements to change. However the 50 year lifetime remains an important issue and should be the further investigated in subsequent investigations.

C.19.5 Key References

J. Wertz, D. Everett, and J. Puschell (Eds.), Space Mission Engineering; The New SMAD. HawThorne, California: Microcosm Press.

Seradata. (2015, February 16). Retrieved from SpaceTrak: `http://www.seradata.com/`

C. Schrijver, and G. Siscoe (Eds.), Heliophysics, Space Storms and Radiation: Causes and Effects (pp. 381-399). Cambridge University Press.