CRANFIELD UNIVERSITY

William Easdown

Mission ORCA: Orbit Refinement for Collision Avoidance

School of Aerospace, Transport and Manufacturing
Astronautics and Space Engineering

MSc
Academic Year: 2019 - 2020

Supervisor: Dr Leonard Felicetti
March 2020

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This report is submitted in partial (30%) fulfilment of the requirements for the degree of Astronautics and Space Engineering

ABSTRACT

The likelihood of collisions between resident space objects (RSOs) is continuously increasing, driven by spacecraft fragmentations, collisions and the increasing use of 'mega-constellations'. This report details a space-based system proposed to address the need for space situational awareness to combat this issue. System requirements were defined based on literature review and the consideration of dependencies between spacecraft subsystems.

Trade-offs during the mission design included a pivot from space debris surveillance to space traffic management, with the final mission concept supplying end users with state vectors for objects at risk of collision with their spacecraft. This switch enabled the design of the constellation, which uses a constellation of 32 12 U CubeSats, providing coverage of SSO and wider LEO between altitudes of 800 and 1400 km. This altitude range was chosen because it was found after a literature review to be most at risk from collisions. A constellation model was created in STK to allow detailed analysis of coverage and ground station passes.

Spacecraft operational modes have been defined for the full mission lifetime, with payload operational modes being defined separately. These were included in a Concept of Operations (CONOPS) document used throughout the project.

Areas of future development will include risk analysis, validation of component selections, and detailed design of the payload hardware and software. These will all be completed before the system Critical Design Review.

Keywords:

Space debris surveillance, space traffic management, space situational awareness, resident space objects, constellation, CubeSat

Figure 0-1 - Mission ORCA logo

ACKNOWLEDGEMENTS

I would like to thank Dr Leonard Felicetti for his supervision during this project.

I would also like to thank my fellow ORCA team members for their hard work throughout the project and for the way they successfully adapted to the 2020 COVID-19 pandemic.

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LIST OF ABBREVIATIONS

ADCS	Attitude Determination & Control System
AI	Artificial Intelligence
AOCS	Attitude and Orbit Control System
ASAT	Anti-Satellite
AWS	Amazon Web Services
CAM	Collision Avoidance Manoeuvre
CCD	Charge-Coupled Device
CDR	Critical Design Review
CMG	Control Moment Gyroscopes
CMOS	Complementary Metal Oxide Semiconductor
CNES	Centre National d'Etudes Spatiales
CONOPS	Concept of Operations
COTS	Commercial Off The Shelf
COVID	Coronavirus Disease
DM	Direct Message
DoD	Department of Defense
EM	Engineering Model
EO	Earth Observation
EOL	End of Life
EPS	Electrical Power System
ESA	European Space Agency
FOV	Field of View
GDP	Group Design Project
IP	Intellectual Property
ISIS	Innovative Solutions In Space
ITU	International Telecommunications Union
km	Kilometre
krad	kilorad
LEO	Low Earth Orbit
LiDAR	Light Detection and Ranging
LV	Launch Vehicle

MP	Mega-Pixel
OBDH	On-Board Data Handling
OCDT	Open Concurrent Design Tool
ORCA	Orbit Refinement for Collision Avoidance
PDR	Preliminary Design Review
R&D	Research and Development
RAAN	Right Ascension of the Ascending Node
RADAR	Radio Detection and Ranging
RSO	Resident Space Object
SATCAT	Satellite Catalog
SDS	Space Debris Surveillance
SLACK	Searchable Log of All Conversation and Knowledge
SL-OMV	Small Launch Orbital Maneuvering Vehicle
SPENVIS	Space Environment Information System
SSA	Space Situational Awareness
SSN	Space Surveillance Network
SSO	Sun Synchronous Orbit
STK	Systems Tool Kit
STM	Space Traffic Management
TID	Total Ionising Dose
TLE	Two-Line Element
TRL	Technology Readiness Level
UHF	Ultra-High Frequency
UKSA	UK Space Agency
VHF	Very High Frequency
W	Watts
WP	Work Package

1 Introduction

This report discusses the author's experience as a systems engineer on the Mission ORCA: Orbit Refinement for Collision Avoidance Group Design Project (GDP). It describes the rationale behind the design of the mission, which involves using a constellation of 32 spacecraft for space traffic management, the need for which is based upon the increasing number of resident space objects (RSOs) and near collisions between them.

This report will detail the management of the ORCA project, steps taken to define the key mission requirements, the generation of the mission baseline and the process of designing the constellation. Mission operations will then be discussed, along with areas for future development.

1.1 Project Background

The first and most critical task for the members of the Systems WP was to define the trajectory of the project given related problems currently faced by the space sector. This project is based on the threat of collisions between resident space objects (RSOs). RSOs include active spacecraft, rocket bodies, and mission-related, breakup and anomalous debris.

The threat of collisions has been increasing, highlighted by several collisions and near collisions, examples of which are given in Table 1-1. The frequency of near collisions is high; from 1999 to 2018, the International Space Station conducted 25 debris collision avoidance manoeuvres [1]. As well as the number of near collisions, the total mass of RSOs is also increasing and now exceeds 7600 tons (see Figure 1-1). In 2018, debris RSOs accounted for almost two thirds of all RSOs in orbit (see Figure 1-2), meaning a large proportion of the space population is uncontrolled. Adding to the debris population, there has been an average of "approximately four fragmentations per year since 2001" [2, p. 13]. If the debris population were allowed to increase in an uncontrolled fashion, it could eventually lead to Kessler Syndrome, leaving whole orbits unusable [3].

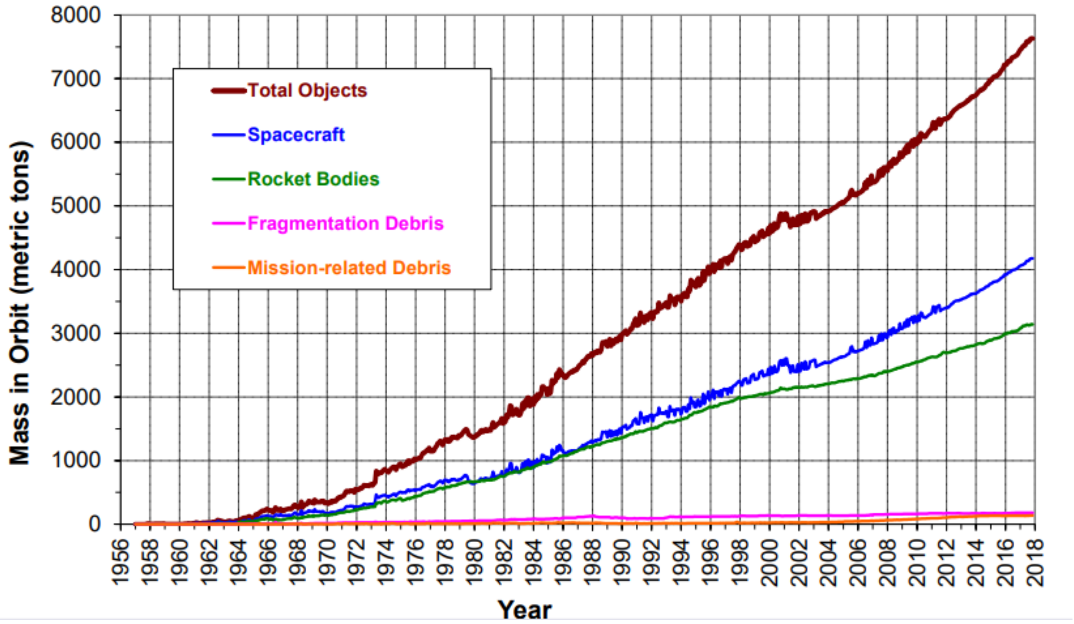


Figure 1-1 – Total mass of material in Earth orbit from 1956 to 2018 [1, p. 5]

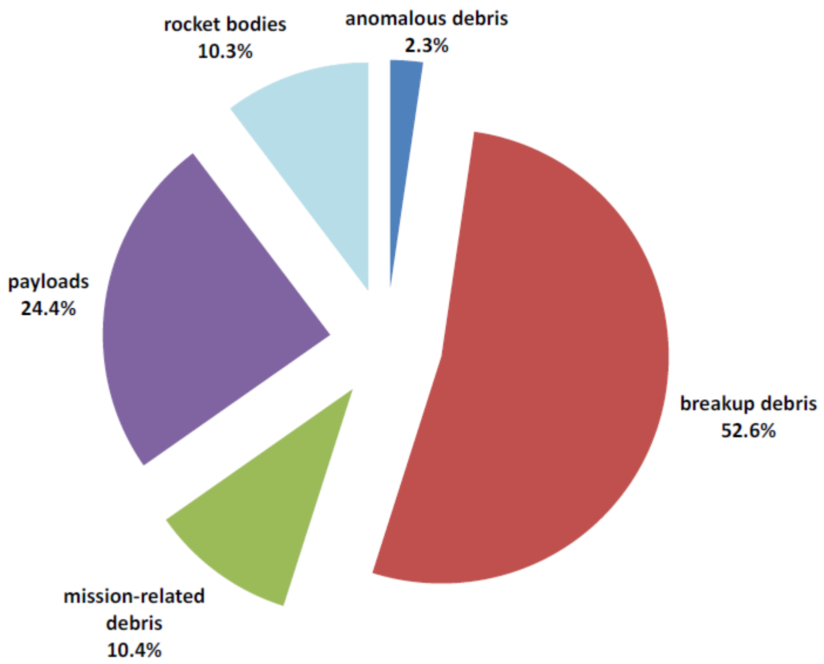


Figure 1-2 - Relative segments of the catalogued *in-orbit* Earth satellite population [2, p. 3]

Table 1-1 - Examples of RSO collisions and near collisions

Date	Event type	RSOs involved	Effects	References
September 13th, 1985	American anti-satellite (ASAT) missile test	Solwind P78-1 solar observatory	Destruction of satellite, generated 285 pieces of catalogued debris which all deorbited by 2008	[4] [5]
January 11th, 2007	Chinese ASAT missile test	Fēngyún FY-1C weather satellite	Destruction of satellite, over 3,000 trackable RSOs and an estimated 150,000 debris particles created	[6] [7] [8]
February 10th 2009	Accidental collision	Kosmos 2251, Iridium 33	Destruction of both satellites, creation of over 2000 large debris pieces	[9] [10]
March 27th, 2019	Indian ASAT missile test	Officially unspecified, reported to be Microsat-R Earth observation satellite	125 debris pieces catalogued six months after test, 50 still on orbit	[11] [12] [13] [14] [15]
September 2nd, 2019	Near collision	SpaceX Starlink44 communications satellite, ESA Aeolus weather satellite	Satellites did not collide; Aeolus performed a collision avoidance manoeuvre	[16]

January 29th, 2020	Near collision	IRAS, GGSE-4	Satellites did not collide; 18 m (+/- 47 m) separation at closest approach	[17] [18]

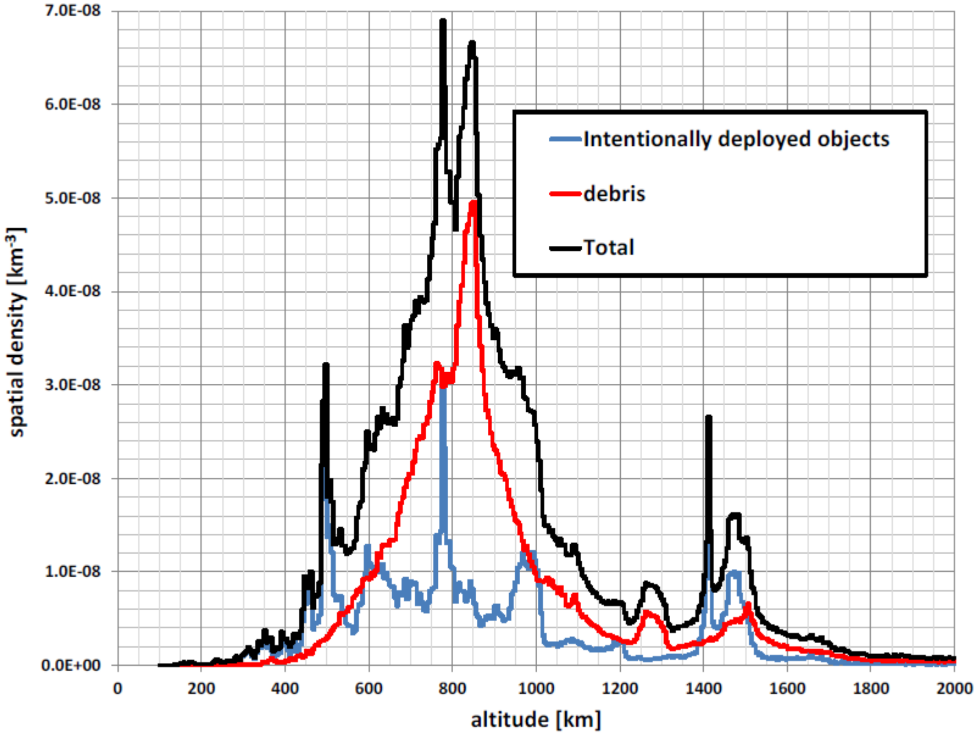


Figure 1-3 - The near Earth (up to 2000 km) altitude population [2, p. 6]

As of 2018, the highest spatial densities of RSOs in Low Earth Orbit (LEO) occur between around 750 km and 900 km, with the debris spatial density peak at around 850 km (see Figure 1-3). This altitude range is also dominated by high inclination Sun synchronous orbits (SSOs) that have “significantly higher collision rate as compared to those populated by lower inclination orbits” [2, p. 7]. These regions will therefore be the focus of this project. Further research into the space debris environment is detailed in Section 3.2 Debris environment research.

‘Mega-constellations’ now under development by a range of companies to increase the accessibility of internet connectivity are set to dramatically increase the number of RSOs. Examples of planned constellations are shown in Table 1-2.

The large number of spacecraft involved in mega-constellations increases the risk of a collision, especially if a mega-constellation satellite experiences a failure and becomes uncontrolled. For example, Amazon estimates that for an overall failure rate of 5% for their Kuiper satellites - a rate “similar to that of SpaceX’s first tranche of Starlink satellites” [19, p. 1] – there would be a 6.1% chance of a collision across the full constellation [20]. However, Amazon notes that any failures would be more likely to occur at the 350 km ‘check-out’ orbit, rather than in one of the operational orbits at 590, 610 or 630 km [20].

As part of their request to reduce the orbital altitude of some of their Starlink satellites from 1,150 km to 550 km, SpaceX performed an analysis of the time to de-orbit from 550 km. This determined that in a worst-case scenario, a failure of the spacecraft’s attitude determination & control system (ADCS) at solar minimum, the spacecraft would passively de-orbit from 550 km altitude in “approximately 4.5-5 years” [21, p. 40], or from 600 km in around 6.75 years. While far lower than the 25-years post-mission guideline [22], this is still a significant amount of time that a spacecraft would remain on-orbit without the ability to perform collision avoidance manoeuvres.

Maintaining space situational awareness (SSA), can be further complicated by the limitations of tracking systems. Debris RSOs in highly elliptical orbits can be more difficult to track with the United States' Space Surveillance Network (SSN) than other objects, due to the SSN's tracking limitations [2].

Table 1-2 – Examples of currently planned large satellite constellations

Manufacturer, Constellation name	Number of satellites planned	Orbital altitude (km)	Planned date of first operations	Current status	References
SpaceX, Starlink	Total up to 42,000	340, 550	2020	Around 12,000 satellites approved; application filed for 30,000 more	[23] [24] [25] [26] [27] [28]
Amazon, Kuiper	3,236	'Check-out': 350 Operations: 590, 610, 630	2021	Application filed to ITU	[20] [29] [30]
GalaxySpace, unnamed	Up to 1,000	500-1,000	Unknown	First satellite launched, rest of constellation under development	[31] [32]
OneWeb, unnamed	648	1,200	Uncertain	Uncertain after OneWeb filed for bankruptcy. 74 satellites already in orbit	[33] [34] [35]
KLEO, unnamed	300	1,100	Unknown	Under development	[36]
Telesat, Telesat LEO	117	Low Earth Orbit, exact altitude unknown	2022	Under development	[37] [38]

1.2 Brief

The initial project brief, set by Dr Leonard Felicetti [39], called for a space-based surveillance system for space debris surveillance (SDS) or space traffic management (STM). According to the brief, the advantages of a space-based system relative to a ground-based one are observation capability throughout a 24-hour period; improved accuracy; weather interdependency and no scattering, diffraction, aberration or turbulence effects from having to observe through the Earth's atmosphere. Other listed advantages include no geographical or political restrictions (contrasting with a ground-based system) and system scalability by increasing the number of spacecraft in the constellation.

The brief also described a series of suggested system requirements. These and the later changes made to them are discussed in Section 3.1 Requirements from original Brief.

Finally, the brief outlined a series of work packages (WPs) to be split between the team of 15 students. The allocation of these is described in Section 2.1 Team Structure.

1.3 Current State of the Art

The current state of the art systems in space traffic management are those provided by LeoLabs [40] and the United States' Space Surveillance Network (SSN). The commercial LeoLabs system, which used ground-based phased-array RADARs, can supply orbit state vectors as small as 0.25 U (10 x 10 x 2.5 cm) [41]. The US Department of Defense (DoD) SSN uses a selection of RADAR ground stations track objects as small as 5 cm diameter in low Earth orbit (LEO), or 1 metre across in geosynchronous orbit [42]. The SSN outputs two-line element (TLE) sets for each tracked object, with most of these data being publicly available in the Satellite Catalog (SATCAT) [43]. Both systems can be used by satellite operators to assess the risk of objects colliding with their spacecraft, although TLEs do not include orbital perturbations so will need to be propagated over time before an accurate probability of collision can be determined [41].

1.4 Business Case

As part of the Systems WP, a business case for the project needed to be understood, to ensure that the project was financially feasible. This business case will be refined as the project moves towards Critical Design Review (CDR).

The LeoTrack service offered by LeoLabs can be used as a cost baseline for the ORCA service [41]. The service offers 12-month subscriptions for $2,500 per month per satellite, with a rolling monthly subscription available for $4,000 per month per satellite for a minimum of three months. Operators of constellations can take advantage of volume discounts, although the level of these is unknown. ORCA will likely use a similar monthly subscription service to be adaptable to user requirements over time, rather than locking them into a contract.

The ORCA business plan envisages two user types. Primary users are satellite operators who need up to date, precise information about specific RSOs (either their own spacecraft or RSOs that risk collision with it). Secondary users include researchers and other academics who are not concerned about data being up to date but need a broad picture of all RSOs.

ORCA will regular search SATCAT’s list of potential collisions and will provide primary users with TLEs for object involved in relevant potential collisions. These TLEs will be updated regularly, giving operators good space situational awareness (SSA). The operator can then decide whether to carry out a collision avoidance manoeuvre (CAM), although this decision will lie solely with the operator to remove liability from ORCA.

2 Project Management

The ORCA project is expected to approximately follow the timeline shown in Figure 2-1.

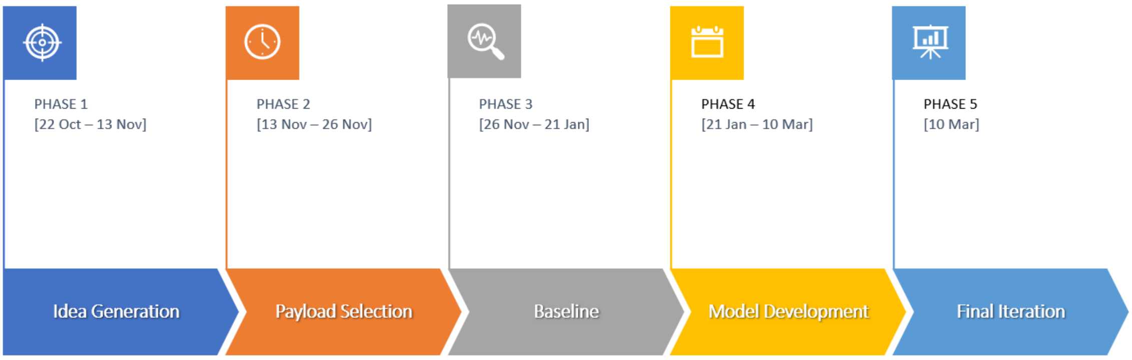


Figure 2-1 - Project timeline [44]

2.1 Team Structure

At the first group meeting on October 15th, 2019, the team of 15 was split into work packages (WPs). WP allocations were determined based on team members' experience and preferences to ensure the most effective team possible. These allocations were maintained throughout the project but were open to discussion if a team member had subsequently had a strong preference against their allocation.

The WPs themselves were also flexible, with team members frequently working across WP boundaries and using a system engineering mindset to help other team members. When WPs worked together in this way, a member of the systems team would oversee the work to ensure the correct factors were being considered and information transmitted accurately.

The overall WPs were systems, mission, payload, mechanical and electrical, with each of these being split into several sub-packages that would each be assigned to an individual team member. A team organogram is shown in Figure 2-2.

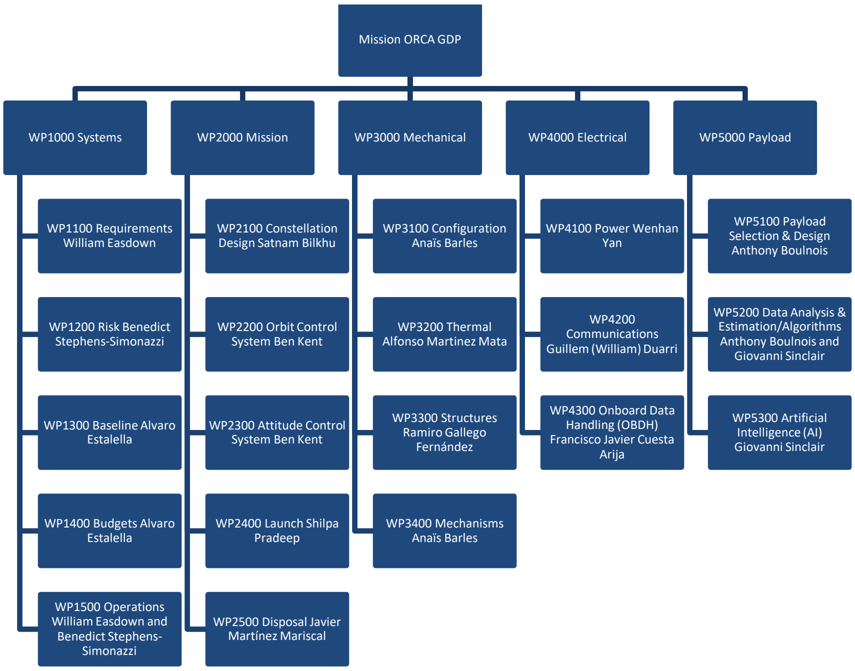


Figure 2-2 – Team organogram

2.2 Meetings

Throughout the project, meetings of the whole group (including supervisor) were held weekly, lasting around two hours. In each meeting, each team member would give an update on the status of their work, any issues faced and what their next steps would be. This was formalised in the latter half of the project using a Work Tracking PowerPoint template [45] that was used from that point onwards.

By discussions involving all team members, situational awareness across the whole team was maintained. This allowed team members to autonomously gain an appreciation of potential impacts from other systems or WPs, although the systems team always drew the team’s attention to issues of particular note, determined qualitatively by severity or breadth of impact.

To avoid overrunning all hands meetings, if an issue arose that was of high severity but did not affect the whole team, a separate meeting would be organised to discuss it. For example, when concerns were expressed about the data rate of the on-board data handling (OBDH) subsystem, a meeting was organised involving just the OBDH, payload and communications team members.

For some all hands meetings to discuss trade-offs between options for critical hardware, the group was split into smaller groups. These would each have around two to four people, with everyone remaining in the same room. Each small group would then be assigned a topic to discuss and would find advantages and disadvantages. After an adequate amount of time, the small groups would then re-unite to discuss their findings. It was in this manner that an initial trade-off for payload type was carried out; more detail on this trade-off can be found in Section 4.1 Payload and bus trade-off. Using this method, trade-offs could be performed rapidly, considering expertise and experience from members across the whole team, rather than just from a particular WP.

Meetings were also held by WPs between the all hands meetings. These would involve the team members from the given WP, with a member of the systems team also attending if able, to provide oversight of decisions.

Non-all hands meetings would typically last around an hour, but no fixed time limit was set to enable problems to be worked to a satisfactory conclusion.

The later stages of the project were impacted by the 2020 COVID-19 pandemic. Social distancing necessitated that meetings be held online – Microsoft Teams' video calling was used for this. Meetings otherwise proceeded as normal.

2.3 Tools

2.3.1 Slack

The productivity tool Slack was used for communication among the team. This enabled splitting of conversation into several 'channels', with each channel covering a separate topic and team members able to opt in or out of channels as desired. A list of the channels used is shown in Figure 2-3.

By separating discussion in this way, team members could narrow their focus onto only the work areas that were relevant to them, avoiding distraction from other areas in which they may have less expertise.

Slack also features tags, which send notifications to certain people. For example, if a team member wanted to attract the attention of this author, they could use @Will in their message. This would send a specially highlighted notification. Another available tag, @channel, could be used to get the attention of anyone in the channel in which it was used.

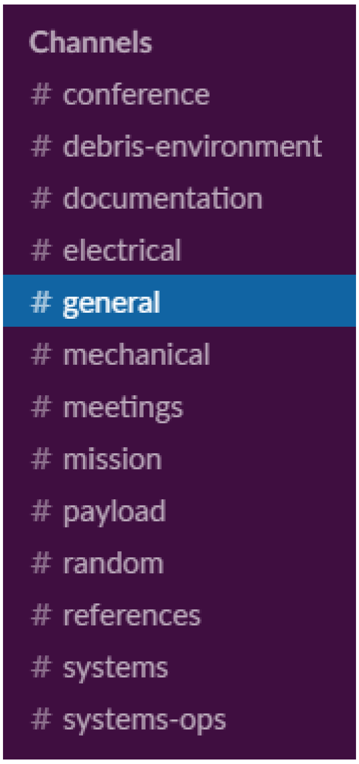


Figure 2-3 – Screenshot of channel list within Slack

Polling functionality within Slack, initiated using the /poll command, allowed team members to quickly choose from a list of given options, so decisions such as meeting times could be made quickly. An example of this is shown in Figure 2-4.

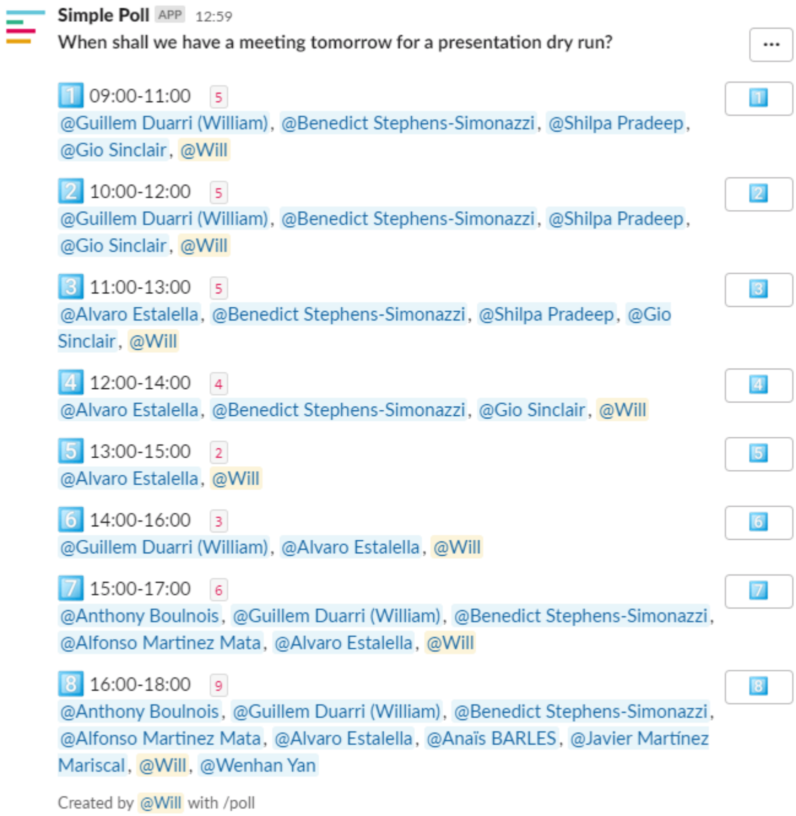


Figure 2-4 – Screenshot of a poll within Slack

Slack also supports direct messages (DMs). This allows sending of messages to a specific team member or small group of team members, useful when an issue relates only to a single team member. These were rarely used however, as the systems team were keen for work and decisions to be transparent to the whole team where possible.

2.3.2 Trello

Work management tool Trello was setup in February 2020 to clarify outstanding tasks and to which team members they were assigned. A 'list' was setup for each WP, with a set of cards in each list showing that WP's tasks. This is shown in Figure 2-5. Gantt chart functionality was also provided through a Google Chrome plugin called Elegantt [46].

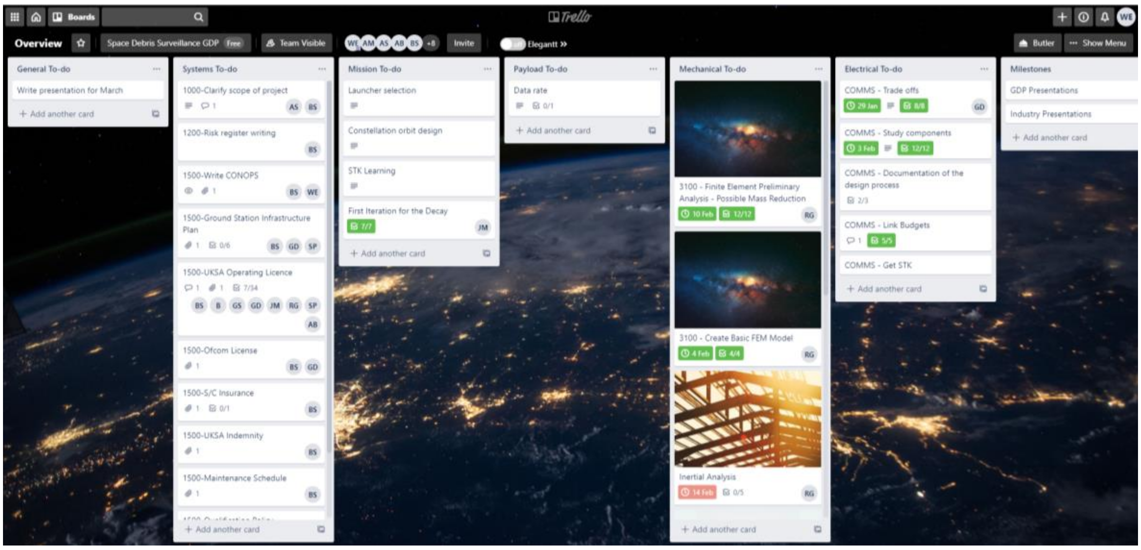


Figure 2-5 – Screenshot of Trello dashboard

While Trello could have been a useful tool for the ORCA team, the systems team lacked sufficient experience to guide the rest of the team in optimal use of the tool. This meant that systems lacked understanding that would have enabled them to use Trello to its full potential. It was also deployed too late in the project, by which time other tools and methods had already been successfully adopted. Ultimately, while Trello could have been a useful tool if used earlier in the project and to its full potential, it proved unnecessary for the ORCA team.

2.4 Documentation

As well as tools for communication and project management, another selection of tools was used for project documentation. These are described in the following sections.

2.4.1 OneDrive

All project documentation was stored on a shared OneDrive folder. This use of cloud storage enabled the whole team to access files anywhere they had an internet connection. This also eliminated the need for team members to store versions of files locally then merge them into a master copy manually. By using Microsoft Office's collaboration functionality, multiple users could edit the same document simultaneously, allowing rapid generation of project documentation.

Backups of the OneDrive folder were taken regularly by downloading the whole folder and storing the copy locally. This meant that should OneDrive become inaccessible, the team could still work using information from the local files, thus mitigating the risk to the project.

Occasionally, errors in OneDrive caused previously written content to be deleted. In this case, OneDrive version history functionality was used to fully restore the earlier version of the document, which was then edited to add any new content.

To make documents within the OneDrive easy to find, a logical file structure was setup, with separate folders for different WPs and other themed folders as required, as shown in Figure 2-6. This avoided document loss among a mess of other documents.

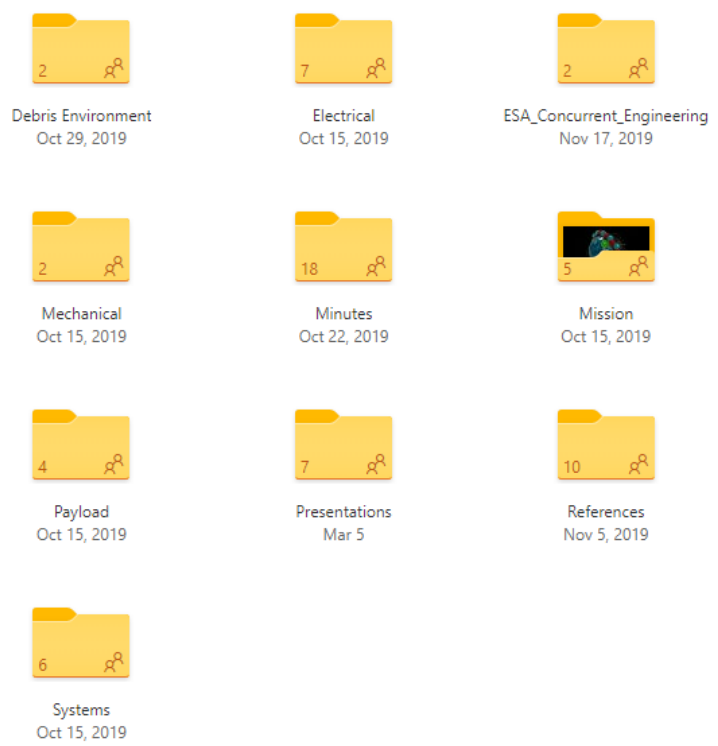


Figure 2-6 – Screenshot of OneDrive top level folder structure

2.4.2 File naming convention

It was important to ensure that all document names followed the same convention, to make it easier to find them, identify their owner and date of creation. To enable this, a File Naming Scheme was written [47]. This established the following convention for file names:

yymmdd creation date_File-name_version number_Team_Creator-acronym

For example, for a requirements document to be used by the systems team:

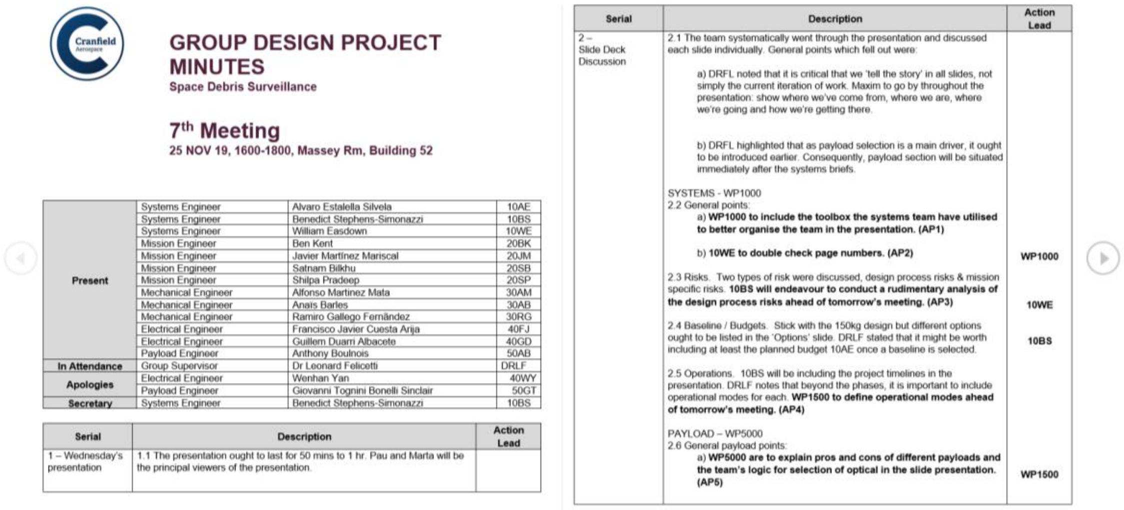
191027_Requirements-Specification_v1.0_Systems_10WE

The naming scheme made use of a list of acronyms [48] that were setup to give each team member a unique and easily identifiable code number to be used throughout the project. These codes were used, for example, to identify team members in meeting minutes (see Section 2.4.3 Minutes).

The file naming scheme also clarified the use of version numbers, with minor modifications causing a change of the number after the decimal point e.g. v1.10 to v1.11, and major changes necessitating a new integer e.g. v1.11 to v2.0.

2.4.3 Minutes

At each weekly all hands meeting, minutes were written to document the topics discussed, decisions made and actions going forward, with the minutes disseminated to the team afterwards. An example of part of a minutes document is shown in Figure 2-7. Minutes were also taken where possible at smaller meetings. Each minutes document included an attendance log and a list of action points (APs), with a person or group assigned to each.



GROUP DESIGN PROJECT MINUTES
Space Debris Surveillance

7th Meeting
25 NOV 19, 1600-1800, Massey Rm, Building 52

Present	Systems Engineer	Alvaro Estalella Silvela	10AE
	Systems Engineer	Benedict Stephens-Simonazzi	10BS
	Systems Engineer	William Easdown	10WE
	Mission Engineer	Ben Kent	20BK
	Mission Engineer	Javier Martínez Mariscal	20JM
	Mission Engineer	Satnam Bilkhu	20SB
	Mission Engineer	Shilpa Pradeep	20SP
	Mechanical Engineer	Alfonso Martinez Mata	30AM
	Mechanical Engineer	Anaïs Barles	30AB
	Mechanical Engineer	Ramiro Gallego Fernãndez	30RG
	Electrical Engineer	Francisco Javier Cuesta Arija	40FJ
	Electrical Engineer	Guillem Duarri Albacete	40GD
	Payload Engineer	Anthony Boulnois	50AB
In Attendance	Group Supervisor	Dr Leonard Felicetti	DRLF
Apologies	Electrical Engineer	Wenhan Yan	40WY
	Payload Engineer	Giovanni Tognini Bonelli Sinclair	50GT
Secretary	Systems Engineer	Benedict Stephens-Simonazzi	10BS

Serial	Description	Action Lead
1 – Wednesday's presentation	1.1 The presentation ought to last for 50 mins to 1 hr. Pau and Marta will be the principal viewers of the presentation.	

Serial	Description	Action Lead
2 – Slide Deck Discussion	2.1 The team systematically went through the presentation and discussed each slide individually. General points which fell out were: a) DRFL noted that it is critical that we 'tell the story' in all slides, not simply the current iteration of work. Maxim to go by throughout the presentation: show where we've come from, where we are, where we're going and how we're getting there. b) DRFL highlighted that as payload selection is a main driver, it ought to be introduced earlier. Consequently, payload section will be situated immediately after the systems briefs.	
	SYSTEMS - WP1000 2.2 General points: a) **WP1000 to include the toolbox the systems team have utilised to better organise the team in the presentation. (AP1)** b) **10WE to double check page numbers. (AP2)**	**WP1000**
	2.3 Risks. Two types of risk were discussed, design process risks & mission specific risks. **10BS will endeavour to conduct a rudimentary analysis of the design process risks ahead of tomorrow's meeting. (AP3)**	**10WE**
	2.4 Baseline / Budgets. Stick with the 150kg design but different options ought to be listed in the 'Options' slide. DRLF stated that it might be worth including at least the planned budget 10AE once a baseline is selected.	**10BS**
	2.5 Operations. 10BS will be including the project timelines in the presentation. DRLF notes that beyond the phases, it is important to include operational modes for each. **WP1500 to define operational modes ahead of tomorrow's meeting. (AP4)**	
	PAYLOAD – WP5000 2.6 General payload points: a) **WP5000 are to explain pros and cons of different payloads and the team's logic for selection of optical in the slide presentation. (AP5)**	**WP1500**

Figure 2-7 - Example meeting minutes [49, p. 1]

2.4.4 Master Central Control spreadsheet

To enable use of concurrent engineering practices, a spreadsheet named Master Central Control [50] was setup. This contained a sheet for each work package, with team members holding responsibility for their WP's sheet. In the sheet, users could setup key data and equations and interlink these both within the sheet and between sheets. In this way, subsystem dependencies were built into the spreadsheet model, allowing the team to use concurrent engineering practices during mission development.

The spreadsheet gave the systems team oversight over the project in a single location while also allowing any team member to check the status of their or another work package. As such, it became a useful tool for the whole team to monitor the status of the mission design. An example of a sheet in the Master Central Control spreadsheet is shown in Figure 2-8.

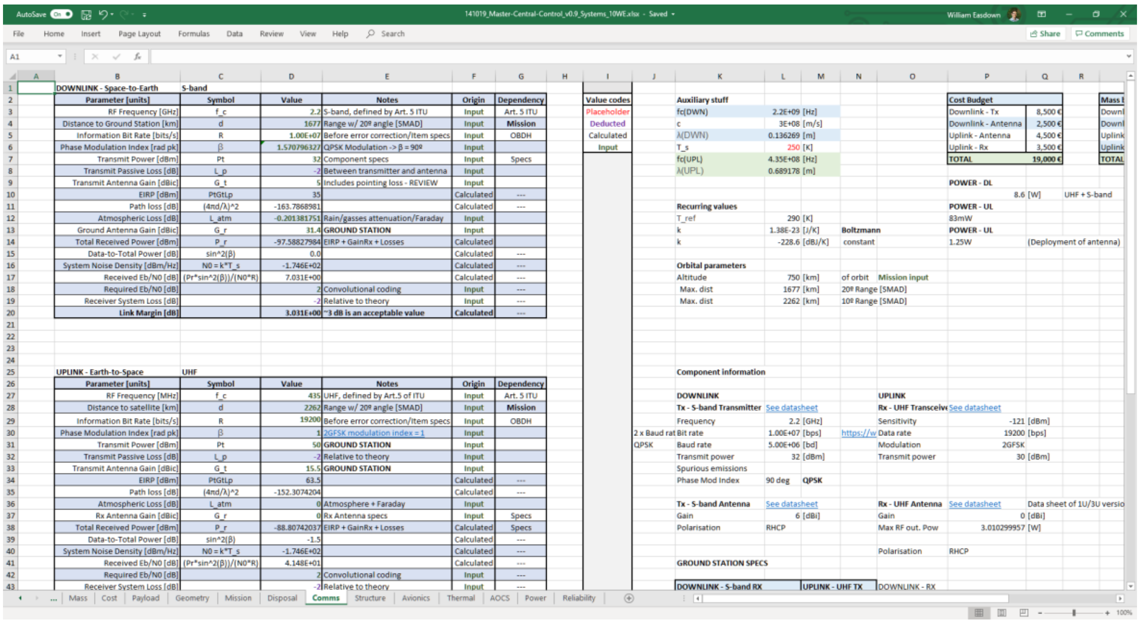


Figure 2-8 - Screenshot of a sheet within the Master Central Control spreadsheet [50, p. 11]

While any member of the team could edit any sheet, team members were encouraged to only edit their own sheets and to seek approval from the systems team before making major changes. This ensured continuous oversight of the project to avoid major design changes not being communicated between WPs.

2.4.5 European Space Agency (ESA) Open Concurrent Design Tool (OCDT)

In January 2019, the team trialled using the European Space Agency's Open Concurrent Design Tool (OCDT) to help track system data. The tool takes the form of an add-in for Microsoft Excel called ConCORDE. The add-in is shown in the Excel toolbar in Figure 2-9.

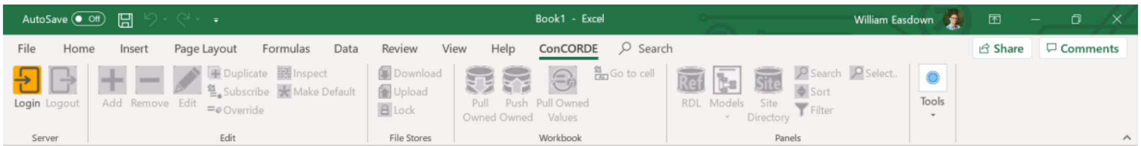


Figure 2-9 - Microsoft Excel toolbar showing the ESA OCDT ConCORDE add-in

While trialling the tool, technical issues around login details were encountered, which delayed the rollout of the tool to the whole team. These were resolved, but once the tool was working, the systems team determined that the amount of time required to fully integrate it into the Master Central Control spreadsheet (see Section 2.4.4 Master Central Control spreadsheet) was disproportionate to the benefit gained by using the tool. This was furthered by the tool being investigated fourth months into the seven-month project, by which point the team had good awareness of the key variables that needed to be tracked throughout the design process. These could be adequately managed without the OCDT tool, with the systems team instead tracking changes in Master Central Control, at meetings and through Slack (see Section 2.3.1 Slack).

2.4.6 Concept of Operations (CONOPS) document

To drive and formalise the design of the mission operations in a way used across industry [51] [52], the author created and structured a concept of operations (CONOPS) document. This included key mission elements such as a detailed description of the constellation, operational modes and operations guidelines. The CONOPS was referred to throughout the mission design process. The current version of it as of the publication of this report can be found in full in Appendix E Concept of Operations (CONOPS).

3 Requirements Definition

The requirements for the overall mission and for individual subsystems were the first part of the project to be tackled due to the project's total dependence upon them. The requirement definition process needed to reflect the top-level goals of the mission as well as the requirements that these goals placed on the various mission aspects such as spacecraft subsystems or constellation design.

Requirements were split into sections: top level (TLR), operational (OPS), payload (PAY) mission (MIS), mechanical (MEC) and electrical (ELE). Each requirement was given a code, showing which section it was associated with and having a unique number for it to be identified by, for example TLR-0110 for a top-level requirement. References between requirements could be used, enabling key requirements to drive other requirements throughout the system.

3.1 Requirements from original Brief

Several requirements were stated in the original project brief [39], based on the proposed space debris surveillance mission. These were as follows:

[R-1] Payload shall detect small objects in LEO ≥ 1cm (preferably less)

[R-2] Full coverage of LEO region with selected payloads

[R-3] Detection of threat of collisions with more than 28.5 hours' notice (as per ISS)

[R-4] The platform shall use commercial off the shelf (COTS) technology (possibly CubeSat technology) to reduce cost

[R-5] Mission duration: at least five years

Once the mission focus pivoted from space debris surveillance to space traffic management (see Section 4.2 Mission Type Trade-Off), the requirements were changed as follows, where the requirements relating to the codes can be found in full in the System Requirements [53] and Appendix C Requirements:

[R-1] was replaced by TLR-0010, setting the minimum trackable RSO size as 300 cubic centimetres volume

[R-2] was modified slightly to state that global coverage of the 800 to 1400 km altitude band should be achieved

[R-3] was replaced by TLR-0040, which states that within six hours of a potential collision becoming known, relevant end users should receive state vectors for objects involved

[R-4] has been kept and formalised as TLR-0110

[R-5] has been kept and formalised as TLR-0060

In this way, a direct connection was made between the original GDP brief and the requirements used moving forward, with this connection being maintained even after a significant change to the mission type.

3.2 Debris environment research

In order to define the top-level requirements, research was necessary to enable characterisation of the space debris environment. The author and another team member, Guillem Duarri, wrote a document outlining the current state of the debris environment [54]. This included information on the number of resident space objects (RSOs) of different sizes, their typical orbits and the level of threat to spacecraft posed by different sizes of RSO.

In this document, the authors concluded that “the highest concentration of space debris is found in the 800 km – 1000 km [55] and 1200 km – 1400 km height layers of LEO [56]”. The document also revealed the range of sizes of debris objects, with 34,000 objects larger than 10 cm and an estimated 128 million objects between 1 cm and 1 mm in size [57]. These became key factors moving forward, as they drove the requirements for payload sensor range and spacecraft altitude. These requirements are discussed further in Section 3.3 System Requirements spreadsheet and shown in full in Appendix C Requirements.

3.3 System Requirements spreadsheet

A spreadsheet called System Requirements [53] was setup to contain the requirements. Like the Master Central Control spreadsheet (see Section 2.4.4 Master Central Control spreadsheet), this had a sheet for each work package so that requirements could be kept separate for ease of tracking. The date of creation of each requirement was shown in the spreadsheet, along with a person responsible for that requirement, so team members knew who to contact if they had an issue regarding it. A full list of the requirements from this spreadsheet can be found in Appendix C Requirements.

4 Baseline Generation

4.1 Payload and bus trade-off

When setting a baseline for the ORCA mission, the biggest driver was the payload requirements. A literature review carried out by the members of the payload WP determined that the payload would require approximately six CubeSat units, or 6 U, of volume within the ORCA bus [58]. However, the author's experience working on the CubeSat camera (CCAM) at RAL Space [59] meant this 3U optical payload was also considered. Given the volume required for the spacecraft's other sub-systems, the systems team estimated, partially by using volume breakdown estimates in Space Mission Engineering: The New SMAD [60], that an overall spacecraft size of at least 12 U would be needed. 12 U buses were investigated, as well as larger sizes that could enable a larger and more accurate payload. The SSTL-150 150 kg bus was found and considered as an alternative, with its extensive space heritage being an advantage [61] [62].

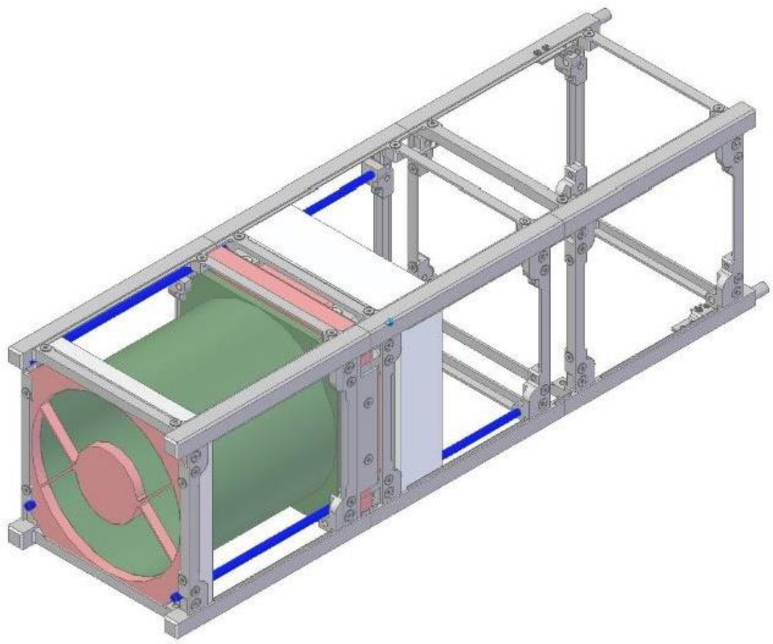


Figure 4-1 - RAL Space's CCAM imaging system uses a 3U optical design [59]

Power was the other major consideration when choosing a payload type for the baseline. Three payload types were considered: optical, RADAR and LiDAR. The team's research revealed that RADAR and LiDAR payloads would use significantly more power than an optical system [63]. This meant that a RADAR or LiDAR payload's power requirement would drive moving to a bigger spacecraft bus with a solar array large enough to supply the payload and other systems. However, other limitations of the sensors compared to optical systems were also contributing factors in their deselection.

In this way, the volume and power budgets became the drivers for the lower limit of bus size. The upper limit was set by the availability of launch slots, with slots for smaller spacecraft tending to be more commonplace due to the availability of ride sharing. Use of a smaller bus would also allow use of standardised dispensers [64] and launch on upper stage vehicles such as the Moog Small Launch Orbital Maneuvering Vehicle [65]. This is discussed in greater detail in Section 7.1 CubeSat Deployer Trade-Off.

In November 2019, after discussion of payload types but before the final selection, the author set a baseline for the system. This was as follows:

- *12 U spacecraft (up to 24kg, 23 x 24 x 36cm)*
- *1500 km altitude sun synchronous orbits (SSO)*
- *Minimum equipment technology readiness level (TRL) of 6*
- *Payload scanning range of 400 km*

This enabled the team to be iterating from a common design. The 12 U bus was chosen to give sufficient volume for a 6 U payload, with the 1500 km altitude orbit chosen because of the plan at that time to look down onto the RSOs being tracked. This was subsequently revised once it was decided that the payload would be of the optical type, as the author knew from his experience on CCAM that backlighting of payloads by the Earth could present a major challenge. The 400 km payload range was set as it was believed at the time to be near the upper limit, although after further analysis by the payload team this was later extended to 1000 km.

This baseline was iterated in January 2020. More work by the payload team confirmed its 6 U volume requirement, making a large bus unnecessary in terms of volume. An optical payload type was also chosen at this time, negating the need for large solar arrays. The SSTL-150 bus would also have been more costly than a smaller bus. Hence it was at this point that SSTL-150 was ruled out and a 12 U bus selected.

4.2 Mission Type Trade-Off

For the ORCA system to be viable, it had to at least match the current state of the art. When the system was being designed for space debris surveillance, its main competitor would have been New Zealand-based company LeoLabs (see Section 1.3 Current State of the Art), who offer tracking of satellites as small as 0.25 U [41] and have previously tracked satellites in potential collisions [17]. This would therefore act as the state of the art.

Other factors that the system could be designed to beat are temporal resolution or tracking accuracy. However, LeoLabs offers satellite state vector updates as frequently as one to three times per day, depending on the satellite orbit [41]. During the near miss between IRAS and GGSE-4 in January 2020, LeoLabs gave an uncertainty of +/- 47 m in its closest approach estimate of 18 m [17]. Orbit determination uncertainty varies depending on several factors: ground location knowledge accuracy, the angle swept by the measured satellite between observations and rounding errors during computation [66]. Despite these, LeoLabs' level of accuracy is currently world-leading for a publicly available system [67]. This combination of factors made beating the state of the art a significant challenge.

Furthermore, to meet requirement TLR-0020 and provide global coverage [68] (also see Appendix C.1 Top Level Requirements), the constellation would need a very large number of spacecraft. This is because the small debris size would necessitate a narrow field of view (FOV) on the payload camera, hence, to cover the full sphere around the Earth a very large number of satellites would be required.

The field of view could be widened, but for a fixed resolution, this would make objects appear smaller in the frame, limiting the minimum resolvable object size. Sensor resolution has a maximum of around 85 mega-pixels (MP) for commercially available space-rated complementary metal-oxide-semiconductor (CMOS) or charge-coupled device (CCD) sensors, although many sensors are around 4 MP [69] [70] [71]. Other sensors with space heritage include COTS parts such as the CMV4000 from AMS [72], which will be flown on NASA’s Mars 2020 Perseverance rover [73] and has been proposed to use the SpaceFibre protocol [74]. The CMV4000 has also been tested by the French space agency, CNES, to determine its resistance to radiation [75]. This and its COTS nature would make it a strong contender for selection as ORCA’s payload sensor. Future work will need to be undertaken to further evaluate this sensor against the Cheetah C4020 camera module that has been baselined (see Appendix B.3.1 Payload Selection & Design).

Field of view is determined by a camera’s optics and is independent of its resolution [76]. Due to this combination of factors, a trade-off had to be performed between the sensor resolution, amount of coverage and the number of satellites required for global coverage. As a result of this, in January 2020, the systems team decided that the mission focus should switch from space debris surveillance to space traffic management.

This pivot relaxed the resolution requirement by allowing ORCA to primarily target larger RSOs. The mission would now focus on existing objects in RSO catalogues, such as that managed by the US’ Space Surveillance Network (SSN), rather than trying to find small unknown objects and add them to the catalogue. To compete with the LeoLabs system, the systems team decided that ORCA’s focus should be on precise orbit determination and high temporal resolution. Rather than a service simply for tracking RSOs, this would make ORCA a valuable tool for spacecraft operators to stay informed of potential collisions with their spacecraft.

The global coverage requirement was relaxed by having the constellation not observe around the whole planet continuously, but instead tracking a few objects at a time, with the option to choose objects based on end user requirements. The systems team termed this ‘discrete global coverage’. By using discrete global coverage, the constellation could again use a small field of view to target RSOs of interest. This meant that the resolution and number of spacecraft could also be reduced to reasonable levels, and the system became feasible.

4.3 Current Baseline

As this report is approximately equivalent to a Preliminary Design Review (PDR), the following baseline represents a good approximation to the final spacecraft design but may change as more analyses are carried out and the project approaches Critical Design Review (CDR). Areas of future work that may affect the design going forward are discussed in Section 8 Areas for Future Development.

4.3.1 Constellation Design

For details of the baseline constellation design, see Section 5.4 Final Constellation Design. Work to define the constellation was carried out in collaboration with the mission WP team, particularly Satnam Bilkhu.

4.3.2 Ground Segment

For details of the ground segment baseline, see Section 6.3 Ground Segment. Work to define the constellation was carried out in collaboration with Guillem Duarri from the communications WP.

4.3.3 Individual ORCA Spacecraft

Details of the current spacecraft baseline can be found throughout Appendix B Common Appendix. The baseline was generated in collaboration with the whole ORCA team.

5 Constellation Design

The author and mission WP team member Satnam Bilkhu worked on the constellation design, selecting orbits and the number of spacecraft to be used for the ORCA constellation.

5.1 Constellation Requirements

A key requirement for the constellation was for its orbits to be between 700 km and 800 km altitude (see requirements MIS-0010 and MIS-0020 in Appendix C.4 Mission Requirements), due to the high density debris field between 800 km and 1400 km altitude (see Section 1.1 Project Background and Section 3.2 Debris environment research). The constellation was also required to provide global coverage (see requirement MIS-0030 in Appendix C.4 Mission Requirements) to enable full space situational awareness. The interpretation of global coverage used initially, combined with the payload field of view and resolution limitations, resulted in an early version of the constellation design requiring an excessive number of satellites. This is discussed in more detail in Section 4.2 Mission Type Trade-Off.

While not formally defined, the constellation could not use a constellation with an excessive number of spacecraft that would take a long time to launch. Thus, the time from first launch to a fully operational constellation, and reliance on the ability to launch many spacecraft on a single launch vehicle (LV) could be reduced.

5.2 Constellation Design Process

The textbooks Space Mission Engineering: The New SMAD [77] and Spacecraft Systems Engineering [78] were used to gain an understanding of current industry best practice for constellation design.

The first step in the constellation design process was to gain a good understanding of the payload range and limitations. This was because the viewable areas from each spacecraft would need to overlap to meet the global coverage requirement. After discussion with the payload WP, it was agreed that the payload would provide sufficient resolution with debris at a maximum range of 1000 km. This figure was then used to drive the constellation design, particularly its altitude. The key payload limitation was not pointing it towards the Sun, to avoid permanently blinding the sensor, with the payload team determining that the payload boresight should not be pointed within 15° of the solar vector (see requirement PAY-0110 in Appendix C.3 Payload Requirements).

The inclination of the constellation's orbital planes was driven by a trade-off between launch site latitude and the inclination of the RSOs to be observed. As described in Section 1.1 Project Background and Section 3.2 Debris environment research, there are a large number of RSOs in Sun synchronous orbits (SSO) that require tracking. The tracking of these became an important driver for the shape of the constellation. The author also noticed that by positioning the constellation's spacecraft at high inclination to primarily track RSOs in SSO, they could also make use of the rotation of low inclination RSOs, which would be in approximately orthogonal orbits. The orbits of the near equatorial RSOs would mean they would frequently pass through the field of view of the constellation's chain of high inclination satellites. Thus, the constellation could supply global high temporal resolution data, similar to how Earth observation (EO) satellites use polar orbits to observe the spinning Earth beneath them. To enable this orbit selection, a launch site supporting SSO was required. The Sutherland spaceport currently under construction in Scotland [79] was chosen, as this will provide native support for small LVs such as the Firefly Alpha [80], which was selected to launch the ORCA constellation [81]. This UK-based site will also enable easy transport of the ORCA satellites to the launch site by road if they are built in the UK.

Thus, it was decided that the ORCA constellation would have a chain of spacecraft in Sun synchronous orbit, with their 1,000 km payload ranges overlapping to improve performance. An added benefit of SSO was its continuous illumination from the Sun, which simplified the design of the electrical and thermal sub-systems as it mostly eliminated time-varying effects. However, eclipses still needed to be considered [82] [83]. An altitude of 750 km was selected, to be in the middle of the required altitude range – for SSO, this gives an inclination of 98.39°. By considering the 1000 km payload range around the circumference of the 750 km orbit, it was found that 14 satellites would give full coverage with an amount of overlap for margin. It was decided that to expand the range of inclinations that could be scanned continuously, the constellation would use two Sun synchronous orbits, with their right ascensions of the ascending node (RAANs) spaced by 7.5° and at 52° and 59.5°. 14 satellites were always used in each plane for a minimum total of 28 operational spacecraft.

The constellation's provision for spare satellites also needed to be considered, to minimise the interruption of service in the event of a satellite failure. It was found that the launch vehicle could launch 16 satellites per launch by using a pair of stacked Moog Small Launch Orbital Maneuvering Vehicles (SL-OMVs) [81] [65]. The SL-OMV is described in Section 7.1 CubeSat Deployer Trade-Off. This meant that for two launches, 32 satellites could be launched, giving an excess capacity of four spacecraft over the constellation minimum requirement. This capacity was therefore chosen to be used to launch four operational spare satellites, bringing the constellation total to the maximum two launch capacity of 32. The four spare spacecraft would be deployed into the operational orbits and be used as fully operational, in-orbit active spares. In this way, they would negate the need for the SL-OMV to perform additional manoeuvres to place the spacecraft in a different plane, and would allow the spares to quickly enter service when required, performing simple phasing manoeuvres if necessary to fill any gaps in coverage. This use of spares in the same orbit as operational spacecraft has previously been demonstrated on constellations such as Galileo [84], the Global Positioning System (GPS) [85] and OneWeb's internet constellation [86]. The two spares in each plane were positioned opposite each other in the orbit to enable the quickest possible filling of any coverage gaps. To improve constellation coverage, the spares in each plane were placed such that the spares in one orbit were spaced 90° further round the orbit than in the other plane.

The ground segment also needed to be considered during the constellation design process. It was decided early on that to avoid having to rent another facility, the ground station would be installed on the Cranfield University campus. This could also make use of existing ultra-high frequency (UHF) equipment, although an S-band system would also need to be installed to fully support the ORCA constellation. More details concerning the ground segment can be found in Section 6.3 Ground Segment.

5.3 STK Model

A constellation model was created in AGI Systems Tool Kit (STK) [87], using the version provided by Cranfield University. This model was refined as understanding of payload and launch limitations grew. The satellites were shown using a generic satellite 3D model, with the 1000 km payload range represented by 1000 km radius hemispheres with their centres at each satellite. The two orbital planes were established, with 14 operational and two active spare satellites in each. 1000 km orbits were also setup – one equatorial and one in a 99.48° SSO- for two simulated RSOs. These were then used with STK's access report tool to find how often the RSOs would pass within range of the ORCA payloads. The tool also allowed team members to analyse ground station passes, displaying the pass length and showing all passes within a specified date range. The STK model is shown in Figure 5-1.

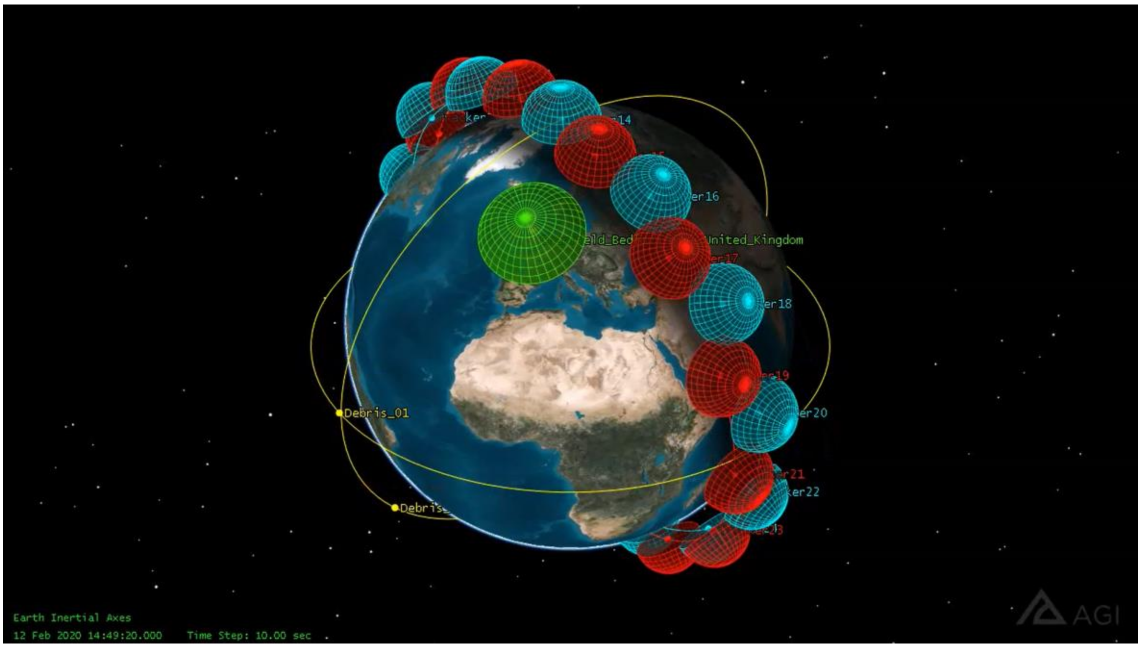


Figure 5-1 - Screenshot of the STK model of the final ORCA constellation (shown without in-orbit spares)

5.3.1 Model Limitations

Due to the limitations of the free version of STK, the payload range could only be modelled as a hemisphere, rather than a full sphere. This mostly had little effect but did mean that in some cases an RSO detection opportunity was not picked up by STK's access report.

To account for the Sun exclusion zone required by the payload to avoid sensor blinding (see Section 5.2 Constellation Design Process), a Sun-pointing notch was added to the payload hemisphere. However, when the STK model was worked on from home using the free version STK due to the 2020 COVID-19 pandemic, the free version did not allow this exclusion zone to be shown in the model.

5.4 Final Constellation Design

The final constellation uses two orbital planes, with 14 operational satellites and two active spares per plane. The two planes have inclinations of 98.39° and

RAANs of 52° and 59.5°. Launch into these planes will be from the Sutherland spaceport via a Firefly Alpha launch vehicle and Moog Small Launch Orbital Maneuvering Vehicle deployer. Two SL-OMVs will be stacked one on top of the other in each Firefly Alpha launch, with each SL-OMV carrying eight ORCA CubeSats to orbit [81].

6 Mission Operations

While not a spacecraft subsystem, the mission operations form a core part of the overall mission design. This section discusses the work carried out for the operations work sub-package, which fell under the systems work package.

To define operations aspects of the mission, in a way that would keep them separate from other systems, a Concept of Operations (CONOPS) document was created [88]. This use of a standard industry document type ensured that best documentation practices were followed, and that operations information was detailed clearly.

6.1 Operations Requirements

The operations requirements, which are also listed in Appendix C.2 Operational Requirements, were as follows:

OPS-0010: The system shall be capable of storing telemetry for 1 missed orbit in the event that communications breakdown

OPS-0020: The system shall be capable of delivering two orbits worth of data to the ground in one pass

OPS-0030: The system shall incorporate a ground operations centre(s) capable of co-ordinating and planning spacecraft operations

OPS-0040: The system shall operate with a valid UKSA operating license

OPS-0050: The system shall be insured to cover any indemnity & third-party liability costs that may arise

OPS-0060: The system shall operate with a valid Ofcom radio operating licence

OPS-0070: The system shall inform the relevant people / organisations of the orbit and orbit uncertainty of resident space objects (RSOs) with enough notice as defined in TLR-0030.

From a systems engineering perspective, these requirements mostly impacted the data budget and hence the design of the OBDH, communications and payload subsystems. Licensing and insurance, while not a distinct subsystem, also need to be managed carefully. Benedict Stephens-Simonazzi on the systems team worked on these [44].

6.2 Operational Modes

A key step in defining the mission operations was the definition of the space segment operational modes. These describe various system states that the spacecraft switches between throughout different phases of the mission or if a certain trigger occurs, such as a computer failure. They were defined by carefully stepping through the mission timeline (see Section 6.3

Mission Timeline) and considering what actions the spacecraft would need to take at each phase. Failure states were also borne in mind during creation of the modes, with the SAFE mode acting as the ultimate fallback. Operational modes were split into spacecraft and payload modes. The operational modes were defined in the CONOPS [88] and can be found in full in Appendix E.5 Operational Modes.

The operational modes had a significant interlink with the design of the spacecraft's attitude and orbit control system (AOCS). For example, early in the project the systems team, while defining an early version of the operational modes, decided that while imaging an RSO, the ORCA spacecraft should slew to keep the RSO at a constant position within the frame. However, Ben Kent, who led the design of the AOCS, reported that the slew rate required for this type of operation was impractical, as it would require the use of control moment gyroscopes (CMGs) that would have too much volume and mass and have too high a power requirement to be suitable for use in the 12 U bus. This is also discussed in Appendix B.3.12 AOCS.

Reaction wheels were then chosen, which necessitated a change to the RSO imaging operations. The author decided that rather than taking several images with a relatively stationary target against a moving stellar background, a similar isolation effect could be achieved by fixing the camera's field of view relative to the background and having the RSO move through the frame. This meant that the AOCS had a requirement to accurately hold a desired attitude and the tighter the payload field of view, the stricter the attitude holding requirement would be. However, the AOCS WP deemed this much more feasible than a high slew rate requirement. The operational modes were then updated, with the ACQUIRE mode describing the new pointing strategy.

6.3 Ground Segment

Working with communications team member Guillem Duarri, the Cranfield University ground station was selected as ORCA's primary ground station due to the team's access to staff who are familiar with its systems. Its relatively high latitude of 52° North also means it has frequent passes from the ORCA satellites. The author worked with mission team member Satnam Bilkhu to develop an STK model of the mission (discussed in Section 5.3 STK Model), through which accurate measurements of the length and frequency of ground station passes could be gathered. STK's access report tool revealed that the average pass duration was five minutes for the S-band downlink, with these occurring approximately every 12 hours [89]. This may not meet the top level requirement for temporal resolution (see TLR-0050 in Appendix C.1 Top Level Requirements) and will be investigated further (see Section 8.8 Requirement validation and system verification).

The ground station selection had a significant impact upon the requirements for the communications and onboard data handling (OBDH) subsystems. The average pass length correlates with the maximum downlink speed required on the spacecraft's S-band transmitter, which is used to downlink data from the payload (see Appendix B.3.10 Communications). A shorter average pass length requires a transmitter on the spacecraft that can support a higher data rate, and hence likely uses more power from the electrical power system (EPS). OBDH is impacted by pass interval because for a higher interval between passes and a given system data production rate, more data must be stored onboard before being downlinked. To add margin into the design, this must also take account of the potential for missed passes due to a failure on the ground or onboard the satellite. Given the size of memory modules available for spacecraft computers, this meant that once the pass interval was known, a maximum payload data production rate could also be calculated. In this way, ground station selection had a large impact that rippled across various spacecraft subsystems.

One disadvantage of the Cranfield ground station is that while it has very high frequency (VHF) and ultra-high frequency (UHF) equipment already installed, an S-band will need to be added to support ORCA's payload data downlink. Purchasing of this equipment will be included in the ORCA cost budget. To avoid unnecessary design effort, a system will likely be bought off the shelf, such as that supplied by Innovative Solutions in Space (ISIS) [90].

To give the ORCA system redundancy in case of a failure at the Cranfield station, or to increase coverage, a backup system is needed. Options from Amazon Web Services (AWS) [91] or the ESA GENSO network [92] were studied. Both networks were found to be feasible backups, with the AWS option potentially being more accessible due to its commercial nature. Both options have global networks, meaning the time to regain communication with an ORCA spacecraft in SAFE mode would be shorter than if only the Cranfield ground station were used.

6.4 Mission Timeline

The mission timeline was developed in collaboration with systems team member Benedict Stephens-Simonazzi and Satnam Bilkhu. The current mission timeline is shown in Figure 6-1; this will be refined as the project moves forward.

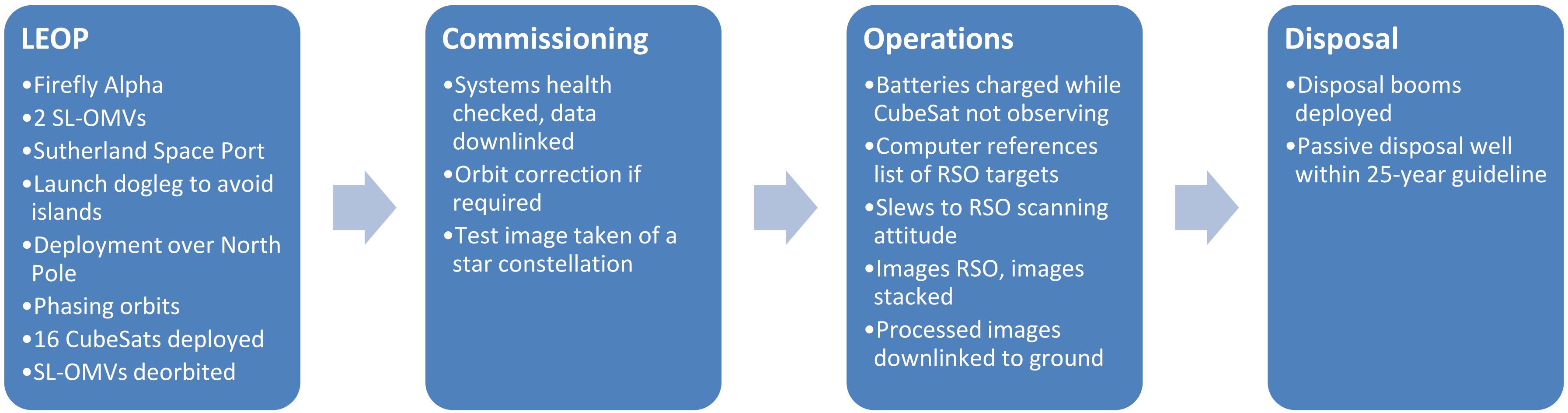


Figure 6-1 - ORCA mission timeline

7 Other Contributions

7.1 CubeSat Deployer Trade-Off

The author worked with mission team member Shilpa Pradeep [81] to select a deployer for the ORCA CubeSats. This vehicle was used to carry the CubeSats atop the Firefly Alpha launch vehicle and to position the satellites in their correct orbital positions using a series of deployment and phasing manoeuvres. After initial separation of the pair of stacked deployers from the LV (see Section 5.2 Constellation Design Process), it needed to be able to use its own propulsion system to move to the deployment point for the first CubeSat. The first satellite would then be jettisoned, before the deployer would use its propulsion system again to enter a phasing orbit and relocate to the second deployment position. This sequence would be repeated until all eight CubeSats on the deployer had been released.

To meet this mission outline, the author defined the following requirements for the deployer:

- Can carry no less than eight 12 U CubeSats
- Can be stacked on top of another copy of itself
- Two stacked deployers can fit within the payload fairing envelope of a Firefly Alpha LV
- Has sufficient Δv to manoeuvre to and between the eight deployment points

During the deployer trade-off, two options were considered: the Small Launch Orbital Maneuvering Vehicle (SL-OMV) from Moog, Inc. [65] and the SHERPA family of vehicles (SHERPA 400/1000/2200) from Spaceflight, Inc [93]. The properties of these for the purposes of the trade-off are shown in Table 7-1. These show that the SHERPA's capabilities are limited relative to the SL-OMV. Due to its lack of stacking capability, use of the SHERPA would also require the number of Firefly launches to be doubled, in turn doubling that element of the overall project cost. It was therefore decided that better value and capabilities would be achieved by using the SL-OMV.

Table 7-1 - SL-OMV and SHERPA properties

	SL-OMV	**SHERPA 400/1000/2200**
12 U CubeSat carrying capacity	8, max. 25 kg each	6, max. 20kg each
Stackable?	Yes	No
Stacked deployers fit within Firefly envelope?	Yes	N/A
Sufficient Δv for orbital manoeuvres	Yes - preliminary	Yes
References	[65]	[81] [93]

Pricing for use of the SL-OMV is currently unknown, but this will be researched further as the design approaches CDR. Also, close to the publication of this report, the author discovered a similar vehicle called the Orbital Transfer Vehicle (OTV), which is built by Firefly Aerospace [94]. This will need to be evaluated against the SL-OMV as the fact that it is built by the same manufacturer as the launch vehicle may lead to improved integration processes and reduced time to launch.

8 Areas for Future Development

8.1 Detailed payload hardware and software design

While the Cheetah C4020 camera module has been selected for the payload (see Appendix B.3.1 Payload Selection & Design), more work needs to be done to define and refine the payload software. This will include the algorithms used to stack frames and remove background stars. The software to determine an RSO's state vector from a final image also needs to be designed. Furthermore, due to radiation (see Section 8.3 Radiation analysis), the Cheetah C4020 module may need to be swapped for a more radiation hardened device. This decision will be taken once radiation analysis has been completed. The payload team will make a recommendation on whether the module should be changed, with the systems team maintaining oversight to ensure compatibility with other subsystems.

8.2 Completion of CONOPS

The CONOPS document (see Section 2.4.6 Concept of Operations (CONOPS) document and Appendix E Concept of Operations (CONOPS)) needs a significant amount of work before it is ready for publication. The spacecraft and payload operational modes are currently well defined, but areas of future work in the CONOPS include a full mission timeline with all required orbital manoeuvres, more details of ground station provisioning including backups, contingency plans for eventualities such as a spacecraft entering SAFE mode, and detailed Δv calculations for the entire mission. The ground station analysis will be of particular importance to ensure that the system meets the top level requirement for data temporal resolution (TLR-0050, see Appendix C.1 Top Level Requirements). This should be worked on as a priority and will need to be completed before CDR.

8.3 Radiation analysis

While the current spacecraft design considers volume, mass and thermal properties among others, it does not currently take into account the effects of radiation. To gain an preliminary estimate of the total ionising dose (TID) that the spacecraft would experience during its five-year lifetime, the author generated a model of the spacecraft and its orbit in radiation modelling software SPENVIS [95]. This used the SHIELDOSE-2 model, assumed the shielding was a finite aluminium slab to model the sides of the CubeSat bus and used silicon as the target material to represent the onboard electronics. The results of this are shown in Figure 8-1. The ORCA bus uses aluminium walls with a thickness of 1.5 mm – this can be considered the thickness of the shielding in the SPENVIS model. Using the results table also generated by SPENVIS (see Appendix F SPENVIS raw results), the TID for 1.5 mm of shielding is 4.338 krad, over a five-year mission.

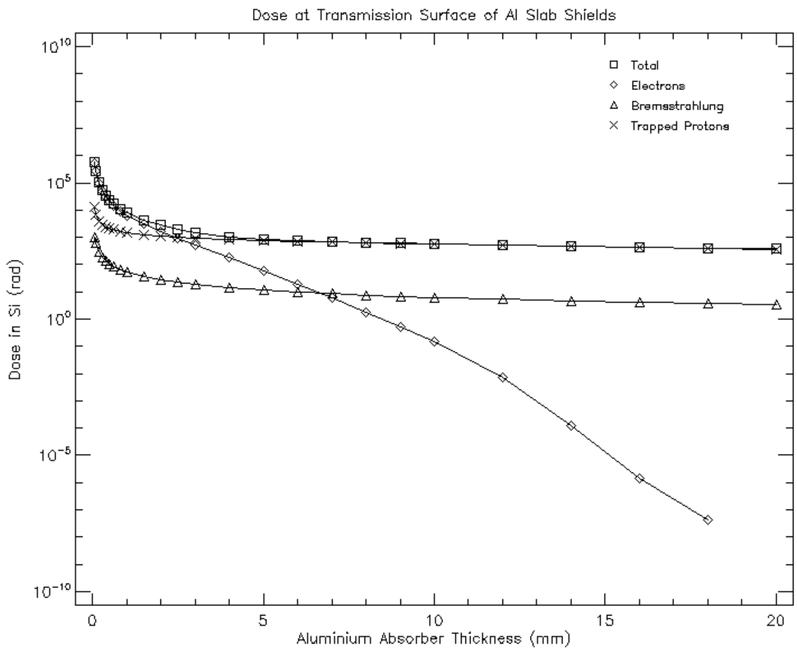


Figure 8-1 – Model generated in SPENVIS [95] of total ionising dose (TID) in silicon over five-year mission lifetime

The effect that this level of radiation will have on the spacecraft components as currently designed is unclear and requires further analysis. Space rated components are often tested for significantly higher levels of radiation, such as 42 krad for the HAS2 CMOS imaging sensor [96, p. 6]. However, COTS components will not have been radiation tested and will either need qualifying for the expected TID or will need shielding to be applied around them to reduce the TID. Due to this, radiation will be a significant factor in whether ORCA can make use of low-cost COTS components and will require extensive further analysis so that final component selections can be made prior to CDR.

8.4 Detailed spacecraft failure modelling

While a risk model has been generated by systems team member Benedict Stephens-Simonazzi [44], it is not sufficiently detailed to give an accurate estimate of the probability of a premature critical spacecraft failure. This is to be expected at this phase of the project, but more detailed analysis will need to be carried out on this. The results of the analysis, which will require detailed estimation of failure rates for each subsystem, will inform the systems team of the likelihood of a spare satellite being called into service to replace a malfunctioning satellite in the constellation. However, this will not affect the number of spare spacecraft provisioned – as described in Section 5.2 Constellation Design Process, it would be excessively expensive to launch more than four spares. Instead, the results will enable the systems team to quantify any long-term degradation of service caused by spacecraft failures. If no more than four satellites fail, full service can be maintained, but more failures than this will lead to gaps in constellation coverage.

8.5 Deployment vehicle evaluation

As discussed in Section 7.1 CubeSat Deployer Trade-Off, the SHERPA and SL-OMV deployers were initially discovered, but Firefly Aerospace’s Orbital Transfer Vehicle (OTV) was also discovered shortly before publication of this report. A trade-off will now need to be carried out between the existing SL-OMV solution and the OTV. Both seem to offer very similar capabilities, but the using a launch vehicle and deployer may have advantages, as it could potentially reduce the time needed to integrate the ORCA satellites onto the deployer and the deployer onto the LV. Other potential advantages include the removal of any negotiation required between Firefly Aerospace (who make the LV) and Moog (who make the SL-OMV) over intellectual property (IP). Firefly Aerospace may also be willing to negotiate a cost reduction for use of their LV and deployer together. All these factors will be considered in the trade-off, which should take place in a timely manner so that the deployer selection can be finalised before CDR.

8.6 Star tracking mounting accuracy impact analysis

Mission requirement MIS-0090 calls for the AOCS to have a pointing accuracy of at least 0.1 degrees (see Appendix C.4 Mission Requirements). However, this may prove very challenging to meet due to the limit on the best accuracy that can be achieved when mounting the star tracker, which is used for attitude determination, to the spacecraft bus. Even if the star tracker is good enough that a perfectly mounted unit could achieve a pointing accuracy well within the limits of the requirement, the mounting inaccuracy means that AOCS may not be able to meet the requirement. Therefore, the mounting accuracy needs to be quantified and any impact of it assessed. As with other key design issues, this needs to be addressed before CDR.

8.7 Full project costing

A detailed cost budget for ORCA remains to be calculated. While a maximum cost has been specified in the top level requirements (see Appendix C.1 Top Level Requirements), the current limit of £250m is likely a gross overestimate. The cost budget will include the design and manufacturer of the 32 ORCA satellites and any engineering models (EMs), launch (including the Firefly Alpha LVs and the SL-OMVs) and operations costs (including licensing).

8.8 Requirement validation and system verification

While the requirements set out in Section 3 Requirements Definition and Appendix C Requirements have not currently been formally validated due to time constraints, a formal verification process such as a System Requirements Review (SRR) will be held as soon as possible to ensure the requirements match up with the system needs.

The designed ORCA system will also be formally verified against these requirements, with any requirements that are not met requiring a waiver. This will ensure that before CDR, the system will be capable of achieving the goals defined by the requirements.

9 Conclusion

This report has detailed the decisions taken during the design process so far of the ORCA constellation. It has shown how the system is based upon the need for greater space traffic management and space situational awareness, allowing spacecraft operators to make informed decisions on whether to carry out collision avoidance manoeuvres.

The project management steps taken, such as tools used and how the team was split into WPs, have been detailed.

The requirements were shown along with how they fed into the system baseline, which was generated by considering several trade-offs within and between subsystems. The author performed specialist work on constellation design and mission operations, which has been described, showing how the constellation design was driven by the design requirements and the operational modes were influenced by subsystem interlinks.

A range of areas for future development have been outlined, giving the ORCA team a strong foundation for its work as the project moves towards the Critical Design Review.

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APPENDICES

These appendices comprise of:

Appendix A, an Executive Summary of this report.

Appendix B, a Common Appendix written jointly by all 15 members of the group and including key budgets, figures and other results.

Appendix C, a complete list of the system requirements.

Appendix D, a debris environment research document created near the start of the project and which drove the mission concept.

Appendix E, a Concept of Operations (CONOPS), created to document the operations aspect of the project. This will be used through CDR and flight phases.

Appendix F, raw data generated by the SPENVIS radiation analysis tool.

Appendix A Executive Summary: Mission ORCA: Orbit Refinement for Collision Avoidance

A.1 Introduction

The number of resident space objects is continuously increasing, with many objects being generated by spacecraft collisions or fragmentations. The risk of collisions is also increasing, with highly populated and high inclination orbits such as Sun synchronous orbit (SSO) being particularly under threat. This Group Design Project, proposed by Dr Leonard Felicetti, sought to design a space-based solution to improve space traffic management, enabling satellite operators to avoid collisions.

A.2 Project Management

The author and wider systems team implemented a range of tools and documentation items to aid the ORCA team's work throughout the project. These included a Slack workspace for communication and meeting planning, a OneDrive folder for easy access to shared documents, and a Master Central Control spreadsheet for tracking of details of the spacecraft subsystems' designs.

A.3 Requirements

The requirements were derived from a detailed study of the space population, including its current state and its trends over time. Comparing this to the current state of the art provided by LeoLabs and the Space Surveillance Network revealed a market niche for a CubeSat constellation to provide high temporal and spatial resolution data to satellite operators regarding objects involved in potential collisions with their spacecraft.

Once this mission concept had been selected, the formal system requirements were defined by considering the resolutions needed and the type of payload needed to supply this. Detailed requirements were generated at the top level and for each spacecraft subsystem and considered impacts subsystems would have on each other. A full list of requirements can be found in Appendix C Requirements.

A.4 Baseline generation

A literature review was carried out, primarily by the members of the payload WP, into the types of payload that could be used for the ORCA mission, as this would drive the wider system baseline. Three main payload types were considered: RADAR, LiDAR and optical. A subsequent trade-off revealed excessive volume, mass and power use by RADAR and LiDAR payloads, leading the team to select an optical payload.

The specific payload was then designed. The final design used a Cheetah C4020 COTS camera module with a custom Ritchey-Chrétien Cassegrain telescope design. The payload takes 6 U of volume at launch, with the telescope deploying through the end of the CubeSat bus for operations.

After the payload selection, the rest of the WPs could work to design their subsystems around the payload. This included considerations such as the amount of electrical power and data rate needed for the payload, with data rate having a particularly wide-reaching impact across WPs.

A.5 Constellation design and operations

The constellation design was challenged by the initial selection of a space debris surveillance mission for ORCA but switching this to space traffic management with a focus on specific RSOs made the constellation feasible.

The operations baseline has been defined as launching the ORCA CubeSats from the Sutherland spaceport using Firefly Alpha LVs and Small Launch Orbital Maneuvering Vehicle (SL-OMV) deployers. The ground segment will primarily be based at Cranfield University, with systems such as that supplied by Amazon Web Services acting as a backup.

A constellation model was developed in STK alongside mission WP member Satnam Bilkhu, enabling analysis of ground station passes and coverage of RSOs by the constellation. It was determined that two Sun synchronous orbit planes with RAANs of 52° and 59.5° would give sufficient coverage of SSO and the wider LEO region in the densest areas that are most at risk of experience collisions. A minimum of 28 operational satellites will be used to complete the system coverage.

Spare satellite provision has been considered, with an extra four satellites being launched. These will make use of the capacity of the SL-OMVs and LV, providing in-orbit active spare capability. This will ensure that should a spacecraft fail, it can be rapidly replaced with minimal loss of coverage.

Spacecraft operational modes have been defined to cover the full mission lifetime from launch through to disposal, with contingencies also being accounted for. Operational modes have also been defined for the payload, with them driven by a change of the mission concept from a slewing detection strategy to maintaining a constant spacecraft attitude while the target RSO passes overhead.

A.6 Future Work

Areas for future development as the project progresses include detailed payload hardware and software design, operations design and risk management, including the impacts of radiation, spacecraft failures and sensor mounting inaccuracy. Further work will also be needed to assess the Orbital Transfer Vehicle, which has appeared as a competitor to the SL-OMV. These will all be addressed as the project moves towards the Critical Design Review (CDR).

Appendix B Common Appendix

This appendix summarises the overall ORCA mission, including requirements, baseline hardware selections and sub-system analyses. It was written by all ORCA team members.

B.1 Work Breakdown Structure

The wider ORCA team of 15 was split into five main work packages (WPs): systems, mission, mechanical, electrical and payload. Individual team members were then assigned specific sub-WPs, as shown in Figure B-1, although they were free to work across these if desired.

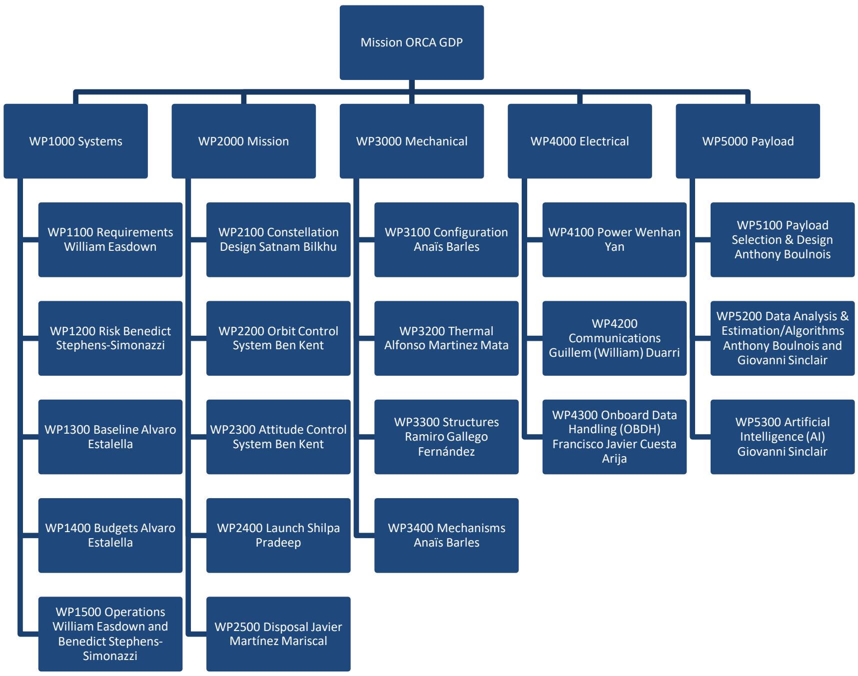

Figure B-1 - Work Breakdown Structure

B.2 Objectives & Requirements

With the amount of material in Earth orbit and the risk of collision constantly increasing, a space traffic management (STM) system is required to track objects and assess the risk of collisions between them. The system should give sufficient warning of a potential collision to allow satellite operators to perform collision avoidance manoeuvres if necessary. This is achieved through high spatial and temporal resolution.

The following top-level requirements were used:

- The system shall be capable of tracking objects as small as 300 cubic centimetres volume
- The system shall provide global coverage of the LEO region (from 800 km to 1400 km altitude)
- The system shall analyse all potential collisions with a likelihood greater than 1 in 10,000 within 12 hours
- The system shall notify primary users of state vectors for RSOs involved in potential collision within six hours of the potential collision becoming known
- The system shall provide collision object state vectors as frequently as every 6 hours if required by operators
- The system shall operate for a minimum of five years, measured from beginning of operational service to End of Life (EOL)
- First launch for the final constellation shall occur before 2028
- The mass of an individual spacecraft shall not exceed the limit for a 12 U CubeSat that is specified in the CubeSat specification

B.3 Baseline

This section gathers the main aspects of all the work packages. Decisions were taken in order to guarantee a viable, feasible and useful mission. In order to achieve such objectives as a student team, technology readiness and simplicity were key drivers in many decisions. COTS elements have been used where possible.

The mission consists of 32 12 U CubeSats, of which 4 are active spares, in two 750 km Sun-Synchronous orbital planes, with direct communication with the ground segment. This baseline focuses mainly on the demonstration satellite, which will serve as proof of concept, while its architecture will be applicable to all the satellites that will form the future fully operational constellation.

Preliminary cost and mass budget analyses have been performed. The manufacturing of each satellite will have a cost in the order of £250,000, with the mass budget complying with the 12U CubeSat regulations. Details of the mass budget are shown in Table B-1.

Table B-1 - Spacecraft mass budget

Part	Total %	Mass (kg)	Margin (%)	Margin (kg)	Total (kg)
Wet mass (top down)	-	19.05	5	1	20
Dry mass (top down)	-	18.73	5	0.94	19.65
Payload	28.0	4.89	10	0.49	5.38
Structure	20.6	3.60	10	0.36	3.96
Thermal	1.9	0.33	5	0.02	0.35
Power	31.3	5.47	4	0.22	5.69
TT&C	2.8	0.49	5	0.02	0.52
OBDH	2.3	0.40	10	0.04	0.44
AOCS	11.0	1.93	5	0.10	2.02
Other	2.0	0.34	10	0.03	0.38
Total	**100**	**17.45**	**7.38 (mean)**	**1.28**	**18.73**

B.3.1 Payload Selection & Design

The payload subsystem has been designed to accomplish the mission objectives: debris detection and trajectory prediction to enable space traffic management. A passive optical system has been chosen from a trade-off analysis between existing technologies, considering that the system should include a demonstration mission run by a university if possible. The key parameters during the analysis were cost, power consumption, volume and weight. The selected system is an optical design that uses a telescope paired with a camera sensor.

The main drivers to design the telescope were field of view (FOV), amount of light gathered, range of detection, complexity, mass and volume. Both FOV and light gathered are significant by respectively defining the area covered and the minimum debris size to be detected.

Table B-2 - Telescope parameters

Aperture	150 mm
Focal length	350 mm
F-number	2.3
Dimensions (stowed)	14x20x7 cm
Mass	2.2 kg

The telescope is a two-mirror Ritchey-Chrétien Cassegrain design. It is deployable and is fitted with a piezoelectric alignment mechanism. The main telescope specifications are detailed in Table B-2.

The final selection for the payload camera is the Cheetah C4020, which meets the required accuracy and FOV. After performing a trade-off analysis, a camera using a complementary metal-oxide-semiconductor (CMOS) sensor was chosen mainly because of the low cost and low power consumption compared to a charge coupled device (CCD) sensor. Camera specifications are shown in Table B-3.

Table B-3 - Camera sensor properties

Pixel size	3.45 mm
Image size	4112x2176 pixels
Sensor size	14.1x7.5 mm
Resolution	2.03” /pixel
Power consumption	2.4 W
Mass	92 g

The overall payload subsystem performance is presented in Table B-4.

Table B-4 - Payload performance

FOV	2.37°x1.23°
Accuracy	984 m at 1000 km
Smallest detectable RSO size (solar phase angle: 160°)	5 cm at 1000 km

B.3.2 Detection strategy selection and implementation

The chosen detection strategy is the ‘stellar background subtraction method’, which was selected following a trade-off shown in Table B-5. It involves matching each frame output from the camera to a specific image containing an identical stellar background. The two images are then aligned and subtracted, revealing any visible debris. This method requires a highly detailed dataset of stars covering the entire celestial sphere. The subtraction of the stellar background scored the highest during the trade-off because it is a method which yields accurate results thanks to the minimal image processing required. The background noise generated will be substantial due to possible misalignment when the subtraction of the images is performed, but the consequences can be mitigated by implementing AI algorithms to 'spot' the debris in the frames (see Appendix B.3.3 Data analysis and AI decision making). Global Motion Estimation yields high accuracy in detecting the debris; however, it uses complex algorithms which increase the processing power requirements in return for small performance increments and hence is not suitable for our mission. Motion history images do not meet the accuracy requirements specified by our mission.

Table B-5 - Trade-off analysis for detection strategy

Trade-off parameters					
	Accuracy	Background noise	Required processing power	In-situ implementation	Total
Weighting	3	2	1	4	
Background subtraction method	3	1	5	5	36
Global Motion Estimation	4	3	1	1	23
Motion history images	1	2	1	1	12

B.3.3 Data analysis and AI decision making

The data analysis described in this section is to be performed by the on-board computer of the CubeSat and its primary goal is to reduce the quantity of data generated by the camera while maintaining the crucial information about the debris. The output of the detection (see Section B.3.2 Detection strategy selection and implementation) will be large packets of raw images from the same detection event. The first step will be to normalise the data and convert all images to greyscale to eliminate inverted colours and minimise noise. Subsequently, all the frames from the same packet will be superimposed into one image; in the case that debris is present, it will then form a trail. This image will be fed to the AI neural network which will then assign a label to it:

- 0 (No debris detected)
- 1 (Debris detected)

The neural network will also assign a percentage accuracy and percentage loss (error) linked its classification. Images which are classified as class 1 with an accuracy percentage above 70% will be passed to OBDH for downlink, and all other images will be deleted.

B.3.4 Data generated

Limits to the quantity of data produced by the CMOS detector:

- Frame rate of the camera: 25 fps (arbitrary)
- Max number of frames taken: 20 (arbitrary)

By programming the CMOS sensor to meet the imposed limitations, and taking into consideration the image processing described in (Sections B.3.2 Detection strategy selection and implementation and 0

), the following are the expected data transfers from the payload sub-system (excluding house-keeping data):

- 268 MB from CMOS to Nano-mind FPGA (20% margin)
- 13 MB from OBDH to comms (image processing + 2:1 compression ratio)

B.3.5 Spacecraft configuration

Figure B-2 to Figure B-4 present the final design configuration of the ORCA satellite. The first step of the configuration design process was to obtain the final design of the structure after several iterations with the structure work package. Then, in order to obtain the final configuration, iterations were made with all the other subsystems to convert their requirement inputs into design outputs.

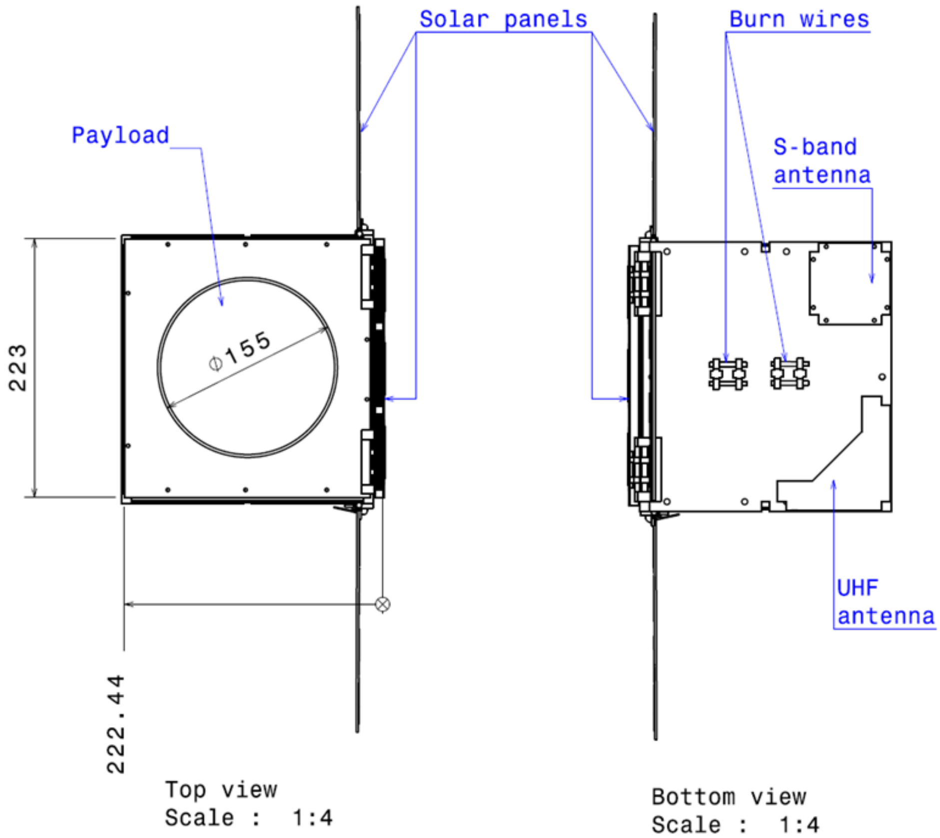

Figure B-2 - ORCA spacecraft top and bottom views

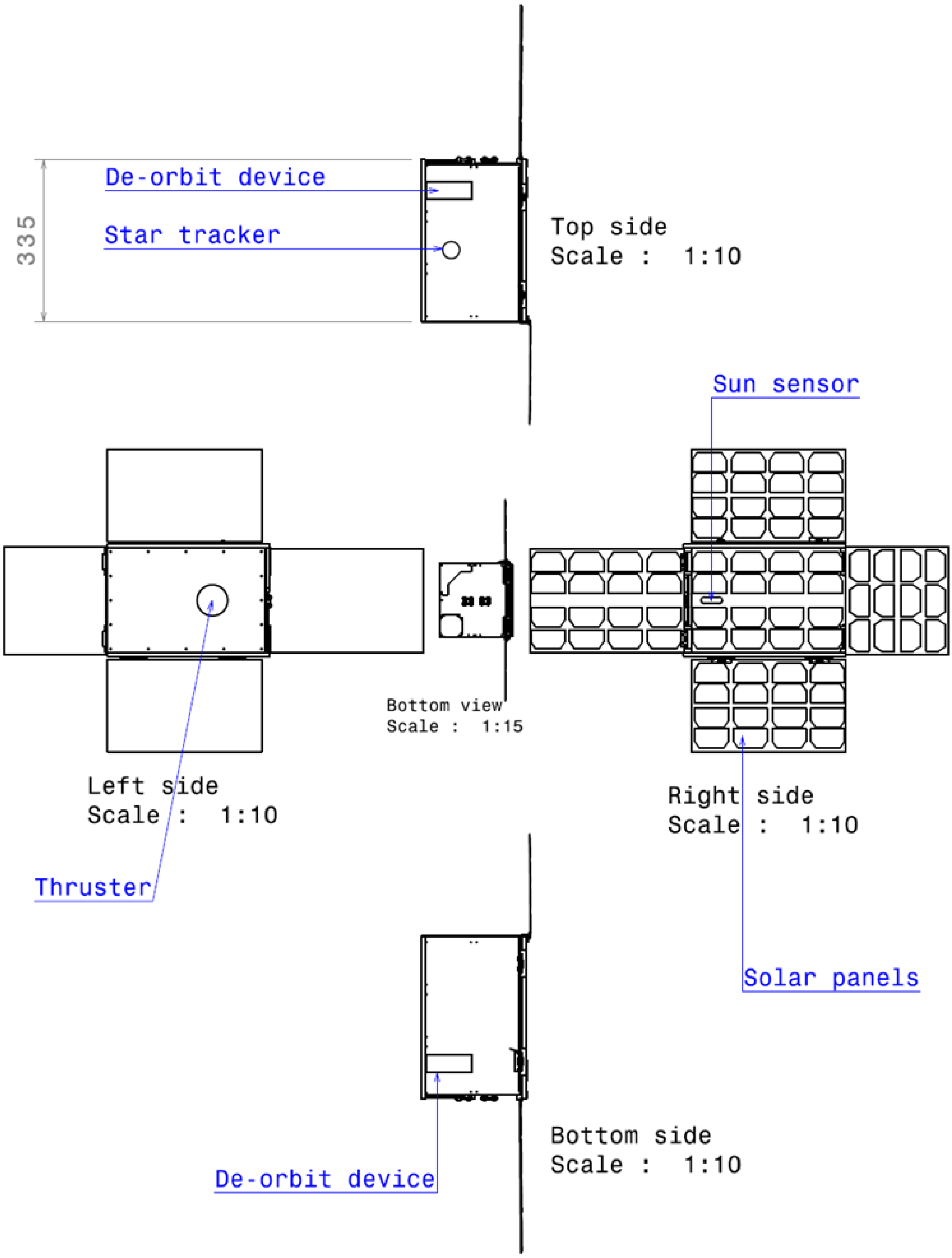

Figure B-3 - ORCA spacecraft side views

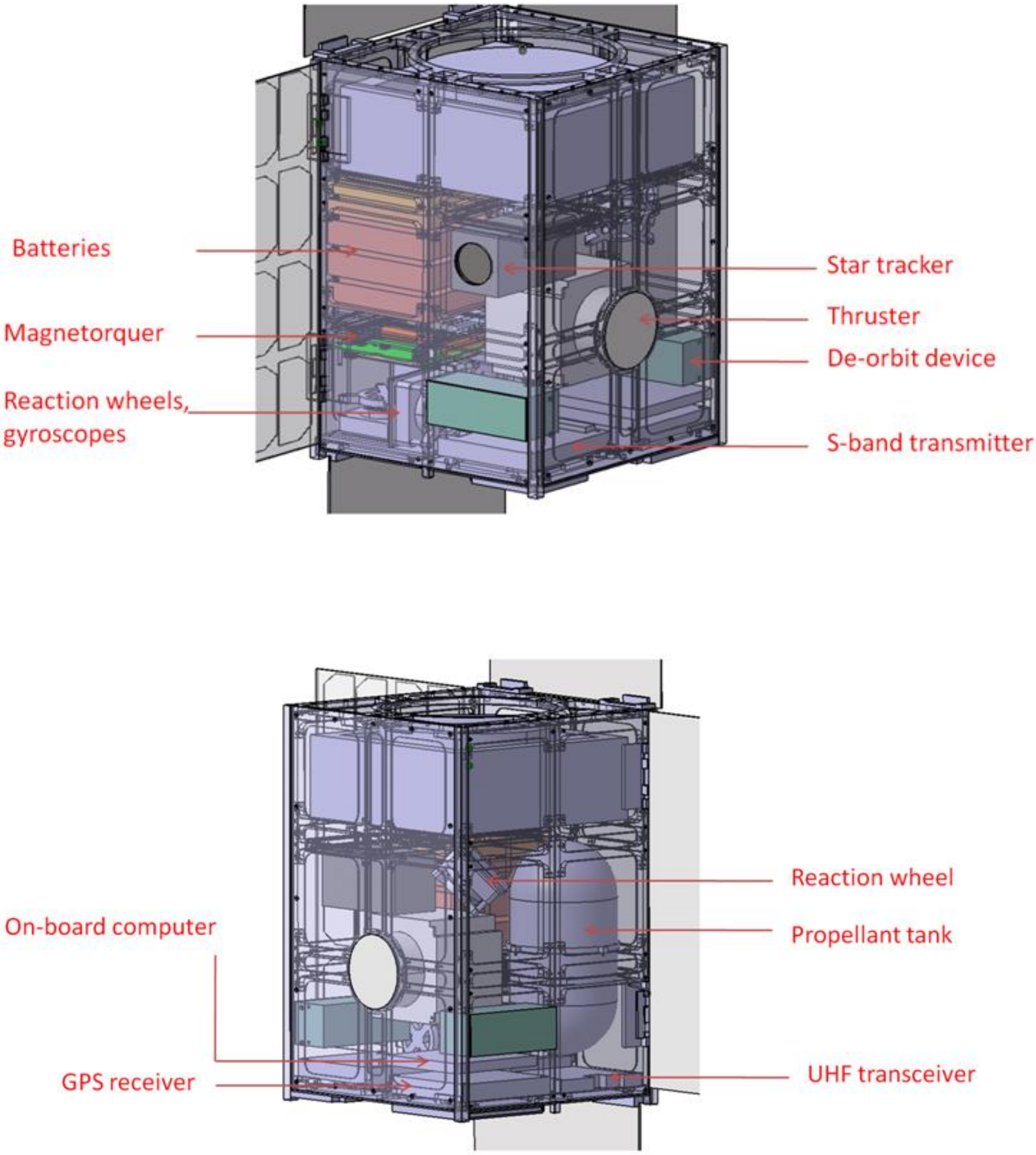

Figure B-4 - ORCA spacecraft internal configuration

B.3.6 Thermal

The thermal subsystem design is centred around the requirement to constantly maintain equipment temperatures within their limits during the mission lifespan.

Thermal design for the ORCA satellite is driven by two main factors: the low mass and power budgets available for the subsystem, and the limited TRL rating of 12 U CubeSat COTS thermal solutions.

The thermal analysis for the final model was carried out using ESATAN software and analytically validated with MATLAB. The model can be seen in Figure B-5.

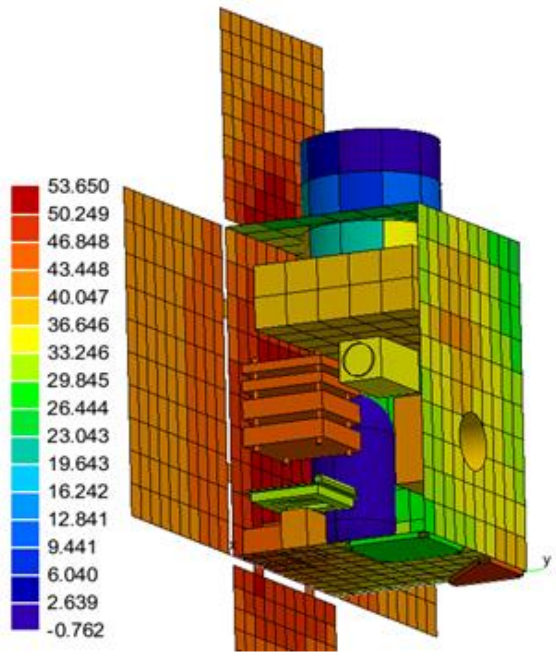

Figure B-5 - Final thermal model

The final design primarily uses passive control elements, as well as two small heating units.

The surface coatings used are aluminium Kapton for the majority of the CubeSat's exterior and optical solar reflector (OSR) to form a heat dissipating surface that acts as a radiator. Thermal copper straps are used to conductively connect the heat dissipating equipment with this radiative area. Finally, as mentioned, two simple electrical heaters were added for the GPS receiver and battery unit.

With these solutions the temperature requirement was verified for all worst-case scenarios studied.

B.3.7 Structures

The structural design was constrained by two factors: the possibility of employing COTS structures and the hazardous launch environment survival requirement. Launch phase induces in the satellite a combination of quasi-static and frequency dependent vibration loads which must be withstood by the structure to create a safe environment for the payload.

After performing a trade-off among existing COTS 12 U structures, the 12 U structure from Innovative Solutions in Space shown in Figure B-6 was selected. This uses Aluminium 7075 T6.

Figure B-6 - Selected ISIS 12 U COTS structure

This structure was modified to accommodate internal equipment by adding two apertures, one on the top face of the structure to ensure payload deployment and another on one the side for the thruster nozzle exhaust.

A finite element model was created in Patran-Nastran software to simulate the selected structure. Quasi-static inertial and random vibration analyses were performed to ensure the structure’s ability to withstand the stresses and displacements caused these loads. A frequency analysis was also performed to ensure the resonance phenomena was not present during launch phase. Post-processing of the final results and requirements fulfilment will validate the structure and the launch vehicle selected for the mission.

B.3.8 Mechanisms

To give a sufficiently large area to power the spacecraft, the deployable solar panels are required for the ORCA satellites. Two mechanisms are used: an actuation mechanism and an initial release mechanism.

A trade-off analysis was performed; the chosen actuation mechanism is a torsion spring in a hinge. Two torsion springs are used per hinge and two hinges are used per solar panel. These hinges are shown in Figure B-7. Calculation of the torques in a '1g' environment led to the determination of the spring design.

The solar panels are released thanks to two nichrome burn wire release mechanisms. To melt the nichrome wire that holds the solar panels in their stowed position, two individual loops are used with a 4 Amp current flowing through them.

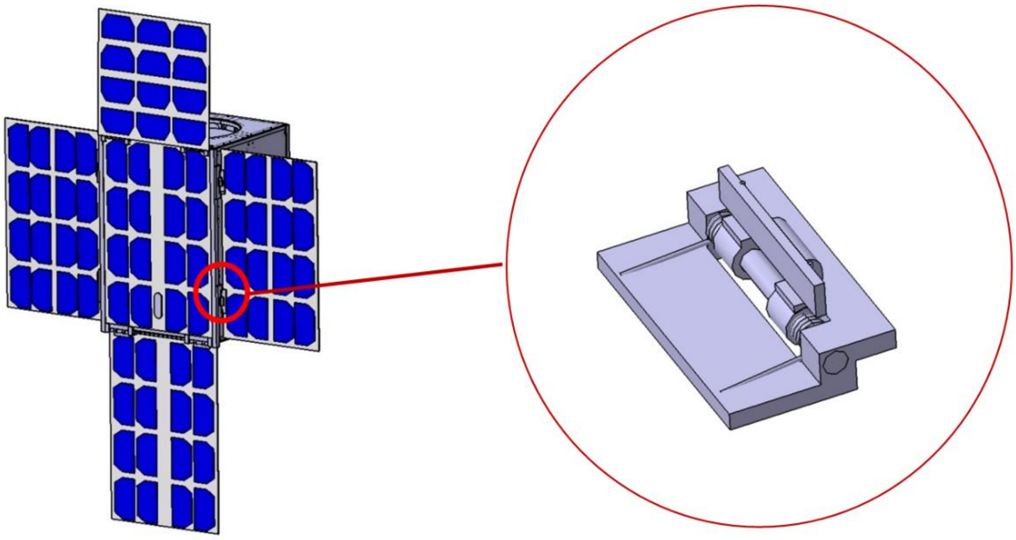

Figure B-7 - Deployed solar panels and hinge

B.3.9 Power

The power subsystem's main duty is supporting all components, which need to be supplied with power both when the spacecraft is illuminated and in eclipse. The subsystem is required to function throughout the entire mission duration with high reliability. A block diagram of the power subsystem is shown in Figure B-8.

According to the constellation design, the 12 U CubeSat will operate in a 750 km SSO. The power subsystem includes three main components: solar arrays, a micro control system (MCU), and a battery.

The solar array is of a gallium arsenide (GaAs) design to make use of this chemistry’s high efficiency. The MCU should both be suitable for power input and output (general performance and specific port). It uses maximum power point tracking (MPPT) technology in order to maintain a high transfer of power from the solar array. A lithium ion battery was selected due to its energy density of over 100Wh/kg.

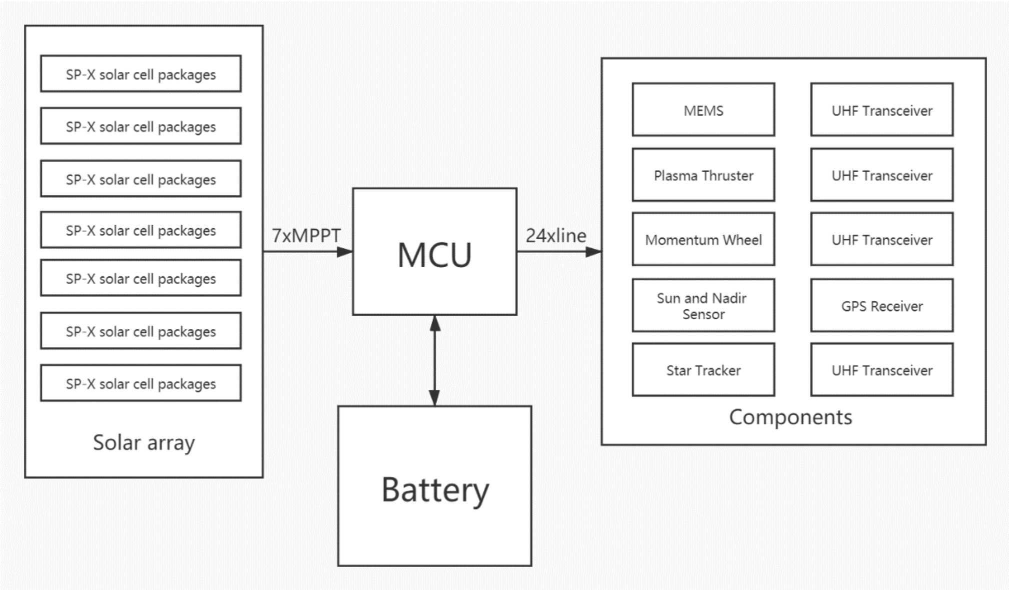

Figure B-8 - Power system block diagram

The solar arrays use SP-X cells from SPACEQUEST. The total power output is over 46W, with an efficiency of up to 29.5%. The MCU and battery unit is the Modular Electrical Power System from ISIS and includes a 135Wh battery, 8xMPPT input, and 24-line output.

B.3.10 Communications

The communications WP designed the links to transfer data between the CubeSat and ground stations. There are two sources of data: payload data and TT&C (housekeeping) data. A block diagram of the communications subsystem is shown in Figure B-9.

In order to maximise the available data rate, a dedicated S-band downlink is used for the payload data, while TT&C data uses a separate UHF link (for both uplink and downlink). The UHF link transmits continually as a beacon for tracking purposes and is used in SAFE mode as well (see Section B.3.16 Operations).

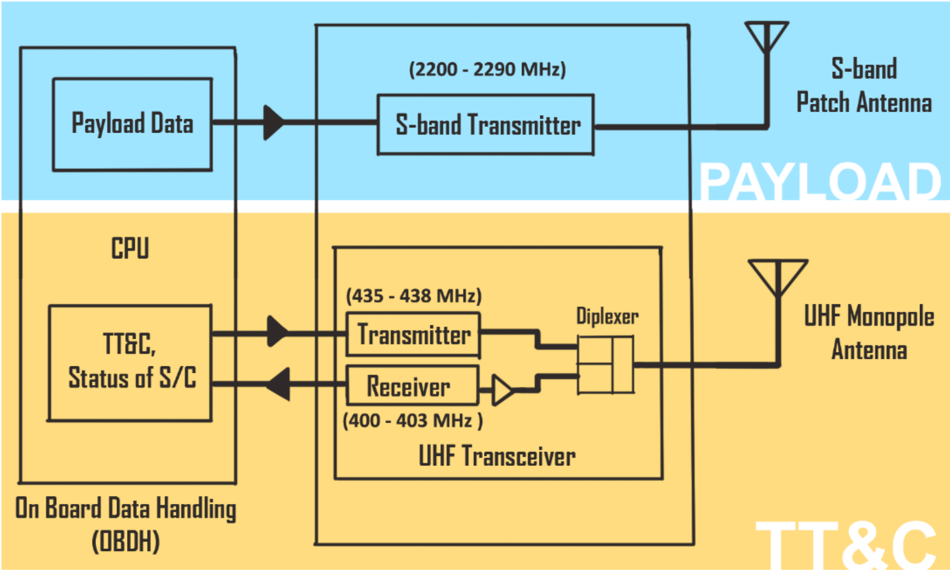

Figure B-9 - Design of the space segment communications subsystem

The S-band link (2200 – 2290 MHz) uses the GOMspace ANT2000 patch antenna and an Endurosat S-band transmitter. According to the link budget calculations, in the worst-case scenario during the contact with ground stations the link margin is 5 dB.

The UHF link uses an ISIS UHF antenna system for 6 U/12 U CubeSats and the Endurosat UHF transceiver II that handles both the uplink (400-403 MHz) and the downlink (435-438 MHz). According to the link budget calculations, during their contact with the main ground station the link margin for the uplink is 33 dB and 106 dB for the downlink.

The main ground station selected for the communications links is the Cranfield University ground station. Whilst at present it only features VHF/UHF capabilities, an upgrade to S-band capabilities is suggested, and is deemed feasible before the mission's launch. For link budget purposes, the ground station was modelled after the ISIS VHF/UHF/S-band full ground station kit.

An STK simulation of the communications system design corroborated the calculations by giving link margins that were found to be within the same order of magnitude, confirming the results obtained. It also confirms that the S-band downlink can transmit up to 375 MB of payload data to the ground station per pass.

B.3.11 OBDH

The on-board data handling (OBDH) subsystem processes the payload data and sends them to the antenna, as well as executing telecommands, managing other devices and housekeeping. The data flow is shown in Table B-6.

Table B-6 - Data flow

	Max Data Rate		Frequency
Camera	2237 Mb/s	280 MB/s	0.8 sec each 30 min
S-band Transmitter	10 Mb/s	1.25 MB/s	some min per day
UHF Transceiver	19.2 kb/s	2.4 kB/s	
M.E. Power System	13 kb/s	1.625 kB/s	continuously
GPS			
Star Tracker			
Sun Sensor			
Gyroscope			
Reaction Wheel			
Magnetorquer			
Thruster			

The OBDH solution is a hybrid architecture, a combination between centralised and bus, with a central computer, the NanoMind Z700. For the connections, the NanoDock SDR motherboard is needed, which manages all the different buses required by the electronic devices. A block diagram of the OBDH subsystem is shown in Figure B-10.

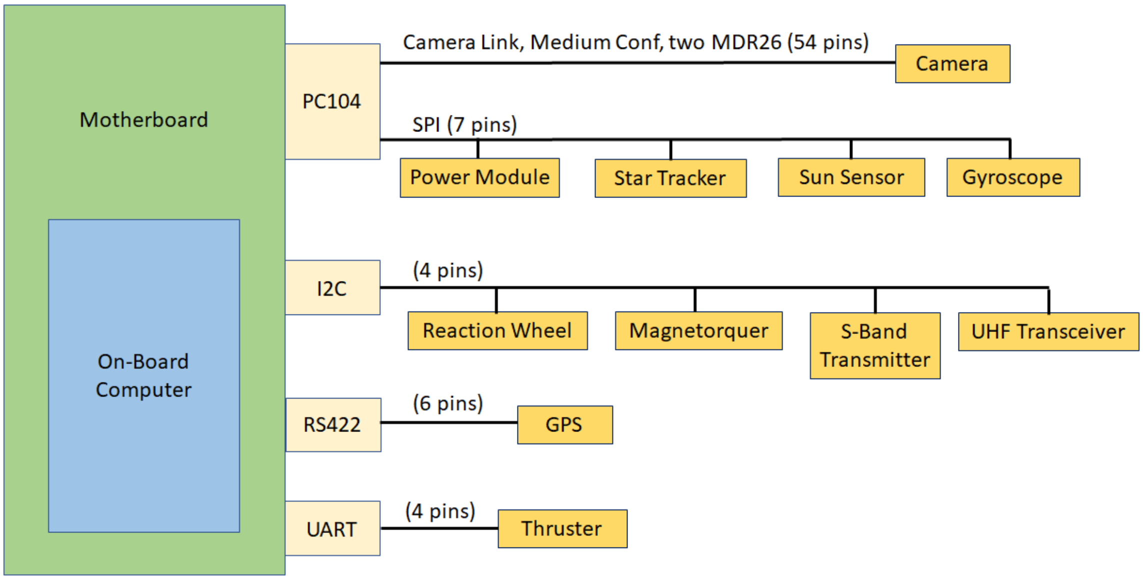

Figure B-10 - OBDH configuration

The on-board computer (OBC) has an FPGA and 800 MHz processors to perform the debris orbit determination and carry out operational modes (see Section B.3.16 Operations). The OBC can store up to 32 GB, consumes 2.3 W, and is provided with an aluminium case to act as radiation shielding. The image sensor is connected directly to the OBC, handling a data rate of 280 MB/s when it is taking images. After processing, the result and the housekeeping data are sent to the S-band transmitter to accomplish the mission.

B.3.12 AOCS

The Attitude and Orbit Control System (AOCS) is responsible for determining and controlling the orientation (attitude), and the orbit (station keeping) in which the satellite is placed. In the ORCA mission, the AOCS is also the system which allows the satellites to rotate in order to track space debris.

Due to the high determination and control accuracy requirements, it was clear that the attitude control system would have to be a three-axis system. For attitude determination, it was necessary to employ both inertial sensors and reference sensors. Two MEMS gyroscopes were chosen to provide 3-axis inertial reference with sufficient redundancy. For reference sensors, the four arcsecond requirement mandated the use of a star tracker. The KU Leuven Star-Tracker can provide two-arcsecond accuracy (one sigma), in a lightweight and low power configuration. This alone would have been sufficient to meet the requirements, but two sun sensors were added to allow precise calculation of the solar vector, helpful to ensure the solar panels are receiving the optimal solar radiation. The 0.1° control accuracy required the use of either reaction wheels or control moment gyroscopes (CMGs).

As the mission was initially designed with the expectation of a high slew rate, CMGs were first analysed. Unfortunately, it was soon apparent that the mass requirements of the CMGs far exceeded the entire subsystem budget. Therefore, reaction wheels were the only possible solution for this mission.

The presence of reaction wheels mandated a method of desaturation. The two potential methods were magnetorquers or small thrusters. Magnetorquers were chosen due to the mass constraints. In addition, the magnetorquers include a detumbling algorithm, useful for stabilising the CubeSat after separation.

Of all the existing propulsions systems, the vast majority were immediately ruled out. Hydrazine-based monopropellants do not satisfy CubeSat regulations, and were therefore discarded immediately. The high I_{sp} electric propulsion options, such as Hall thrusters, were also immediately ruled out; they drastically exceeded the entire subsystem power budget. Cold gas thrusters were analysed, but the low I_{sp} values required a large propellant mass. This mass exceeded the entire subsystem mass budget and was therefore discarded. This left the choice of electrospray, pulsed plasma thrusters (PPT) and green propellants. Green propellants had mass and power values which would have utilised the majority of the respective subsystem budgets, severely constraining the remainder of the AOCS system. The remaining two were entered into a trade-off study, with the PPT being the final selection.

B.3.13 Orbit and Constellation Design

The Orbit and Constellation Design WP concentrates on the orbit type and the arrangement of satellites best suited for our mission. The ORCA mission consists of two Sun-synchronous orbit (SSO) planes, at an altitude of 750 km, an inclination of 98.39° and initial Right Ascension of the Ascending Nodes (RAANs) of 52° and 59.5°. Having a payload range of 1000 km and with the chosen orbit planes, the satellites will have some overlap in the coverage, meaning if one satellite cannot gather enough data on an RSO, another satellite can gather further information on it, which can then be collated. Being in a near polar orbit means that the whole space debris field in LEO is covered, whether the RSO is in equatorial or polar orbit, and at some point, all RSOs will be detected by the satellites. From average relative velocities, a given piece of debris is detected once every 50 minutes by a given satellite. With the distribution of RSOs in LEO, this means all satellites should be constantly detecting and gathering data on space debris. In order for the data to be off-loaded, the satellites in this orbit pass the Cranfield ground station twice in a 24-hour period. The number of ground station passes can increase depending on the number of ground stations used.

The ORCA mission has a system of 28 operational satellites and 4 active spare satellites. There are 14 operational satellites, with 2 active spare satellites in each orbit plane of RAAN's 52° (as shown by the Red line in Figure B-11) with a Local Time of the Descending Node of 06:00:00.000 (HMS) and 59.5° (as shown by the Blue line in Figure B-11) with a Local Time of the Descending Node of 06:30:00.000 (HMS). The active spare satellites are placed at opposite sides of their orbit planes (a difference of 180°) and have a difference of 90° between the active spare satellites between the two orbit planes. For example, if the orbit plane with RAAN of 52° has its active spare satellites at the North and South poles, then the orbit plane with a RAAN of 59.5° will have its active spare satellites at either side of the Equator.

The active spares are ready to go into the operational orbit at any time, but rather than just having hardware in orbit and not used whilst the operational lifetime decreases, these satellites will gather extra data. When the active spares are required to replace a non-functional operational satellite, a phasing manoeuvre will be used in order to fill the gap in the constellation.

The orbit planes cross each other at the poles (as shown in Figure B-11) and due to the time intervals between the satellites, there is no risk of any collisions. The yellow lines seen in the model are representative of RSOs, with one in equatorial orbit and the other in a polar orbit. This allowed a quick analysis of the two main RSO orbit planes, and whether the satellite constellation could track the RSO field.

There are 32 satellites in total, with 16 deployed in one operational orbit (14 operational and 2 active spares) from two deployers (SL-OMVs) in a single launch. The next launch will launch the same but into the other operational orbit.

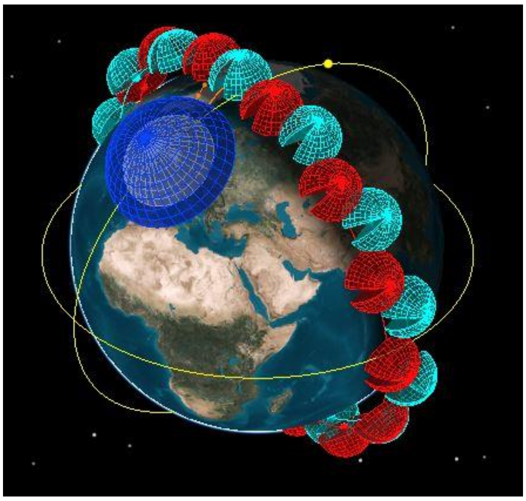

Figure B-11 - AGI STK Model Orbital Planes

B.3.14 Launch

The launch subsystem aimed at choosing a suitable launcher and deployment system to insert the ORCA satellites into the desired orbit.

The main criterion considered during launcher selection is budget. After trading this off with other important criteria such as TRL and secondary payload capability, the final launcher selected for the mission is Firefly Alpha. Firefly Alpha is a two-stage expendable launch vehicle with a TRL 8. It is shown in Figure B-12.

Figure B-12 - Firefly Alpha Launch Vehicle

The next step was to decide the deployer vehicle for putting CubeSats into orbit. From the two possible options, SHERPA and SL-OMV, the latter was selected since the former imposed mass limitations and less accommodation capability. The SL-OMV, shown in Figure B-13, can deploy eight 12 U CubeSats into orbit provided that the CubeSats are not inside a container. This eliminates the need for Quad pack CubeSat deployer and instead a Mark II Motorized Light band will be used for the deployment.

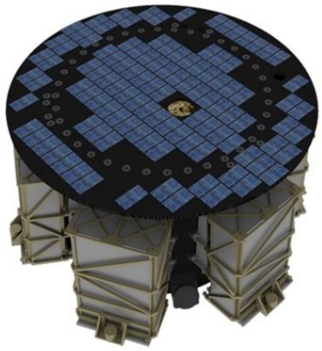

Figure B-13 - SL-OMV deployer

Finally, for the ease of launch into the SSO, the Sutherland spaceport, which provides inclinations of approximately 85-100°, is chosen as the launch site for the mission.

B.3.15 Disposal

The disposal subsystem focuses on providing a solution which complies with the ESA requirements on end of life (EOL) de-orbit, avoiding the rejection of the mission.

The passivation specifications have been accomplished with two separate studies. Firstly, the satellites will have all internal sources of energy (batteries, propellant and reaction wheels) drained. To ensure high system reliability levels and to diminish the risk of generating space debris, the housekeeping data will be analysed periodically via a Kaplan-Meier analysis. If the reliability of the satellite or the disposal system drops below 90%, and the results are confirmed on the ground, an immediate disposal is performed.

The orbital decay model is run in DRAMA and STELA simulators for the demonstrator satellite at the EOL (2026). The 12U CubeSat with the solar panels deployed has an orbital lifetime greater than 25 years at that altitude. Acceleration with passive de-orbit devices is preferred for its simplicity, high TRL and low cost. In Figure B-14 the altitude decay over the years within the regulated lifetime limit is obtained after the deployment with a mean area (tumbling), providing a mean area (tumbling mode) of 1.26 m^2.

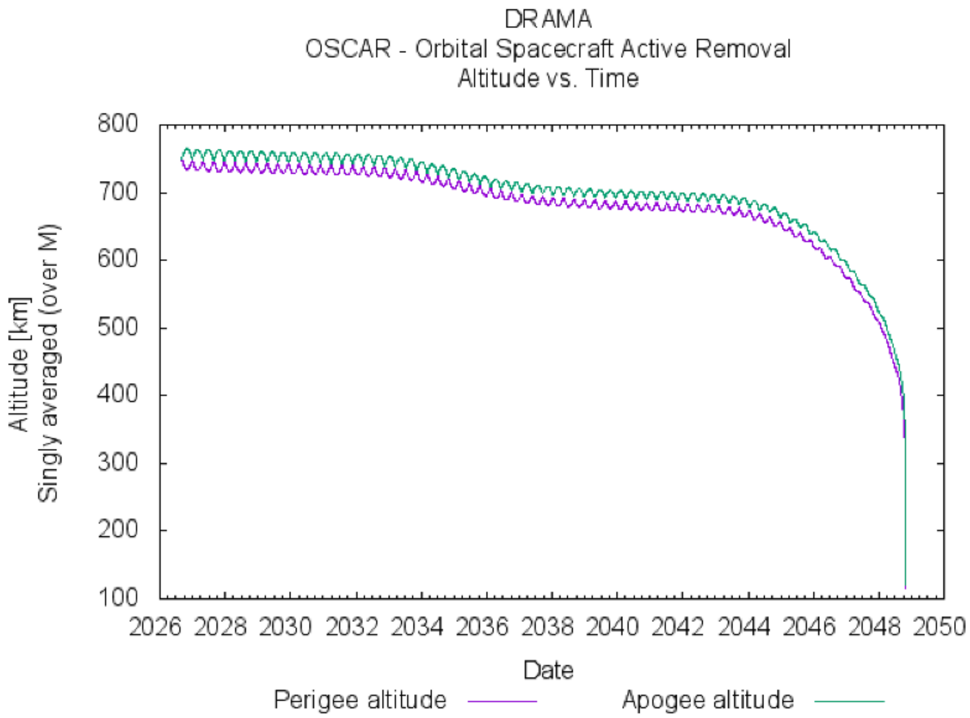

Figure B-14 - DRAMA decay analysis for the minimum area required with the latest Solar Activity prediction by ESA

From the COTS solutions available, the RODEO (Roll-Out DE-Orbiting) device from CTD (Composite Technology Development, Inc) is chosen. As calculated with STELA Mean Area Tool, two de-orbit devices will each deploy a boom of at least 5m and a cross section (sails) of 0.15 m in width and depth. Figure B-15 shows the final configuration after the deployment.

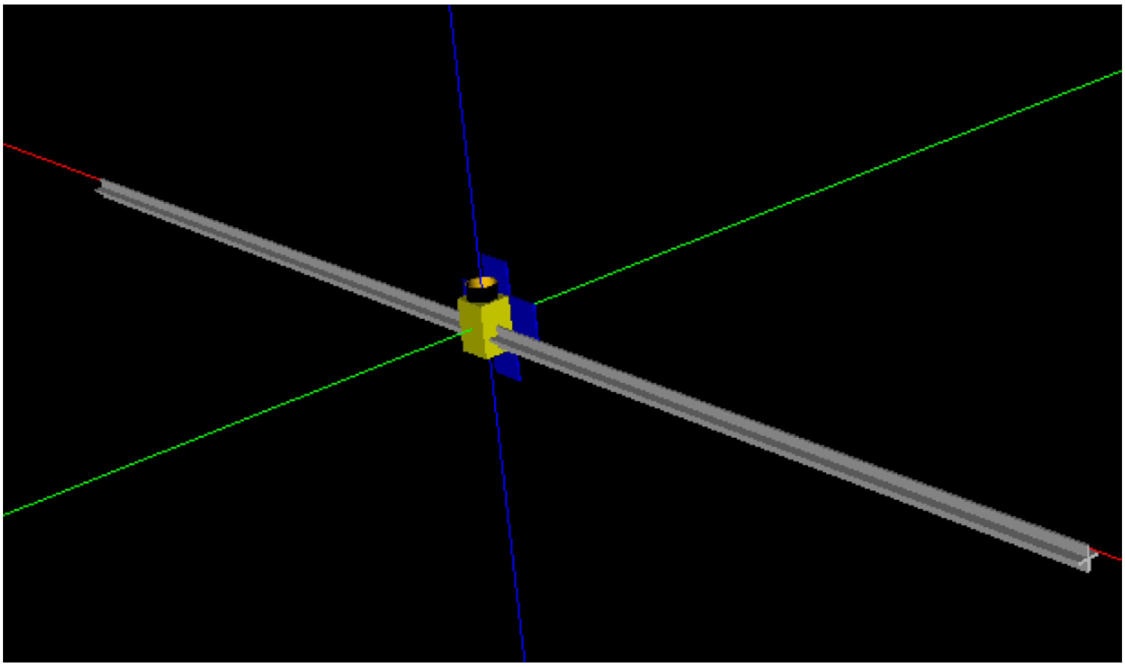

Figure B-15 - Satellite view after RODEO deployment with STELA Mean Area Tool

An uncontrolled re-entry is performed as the COTS for CubeSats components follow the design-for-demise strategy to avoid active re-entries. It is either way simulated (and the survivability) in DRAMA to ensure the safety of this EOL manoeuvre.

B.3.16 Operations

Operations of the ORCA constellation will be controlled from a Mission Operations Centre based at the Cranfield University campus. The university will also act as the main system ground station using its S-band and UHF equipment. Computers at the ground station will be used for any further analysis of the payload data, before the data are disseminated to end users. The Amazon Web Services network of ground stations is provisioned as the backup for ORCA.

Operational modes are defined for the full range of mission states, including LEOP (OFF, DETUMBLE, SUN ACQUISTION, ORBIT INSERTION and COMMISSION modes), operations (SLEW, ACQUIRE, TRANSMIT/RECEIVE) and a SAFE mode to be used in case of emergency. Separate operational modes have also been defined for the payload (OFF, STANDBY, EXPOSE, READOUT, STORE).

After launch on the Firefly Alpha LV and SL-OMV deployer, the ORCA satellites are injected into their SSO using a series of deployments and phasing manoeuvres, equally spacing the satellites around that launch's orbital plane. Once deployed, the satellites carry out initial commissioning and payload checks, the results of which are downlinked to the ground station before initial operational commands are received and enacted. During standard operations, the satellite stores a list of target RSOs. For each one, it slews to face an attitude such that the RSO will pass through the payload field of view, before taking several pictures of the RSO (see Section B.3.2 Detection strategy selection and implementation). It then processes the picture (see Appendix B.3.3 Data analysis and AI decision making) and stores the resulting data in onboard memory, before slewing back to a battery charging attitude. This process is repeated for following RSOs, with the collected data being downlinked to the ground station during the next pass. The constellation geometry means RSOs can be scanned at least every roughly 45 minutes.

The ORCA project roadmap is shown in Figure B-16.

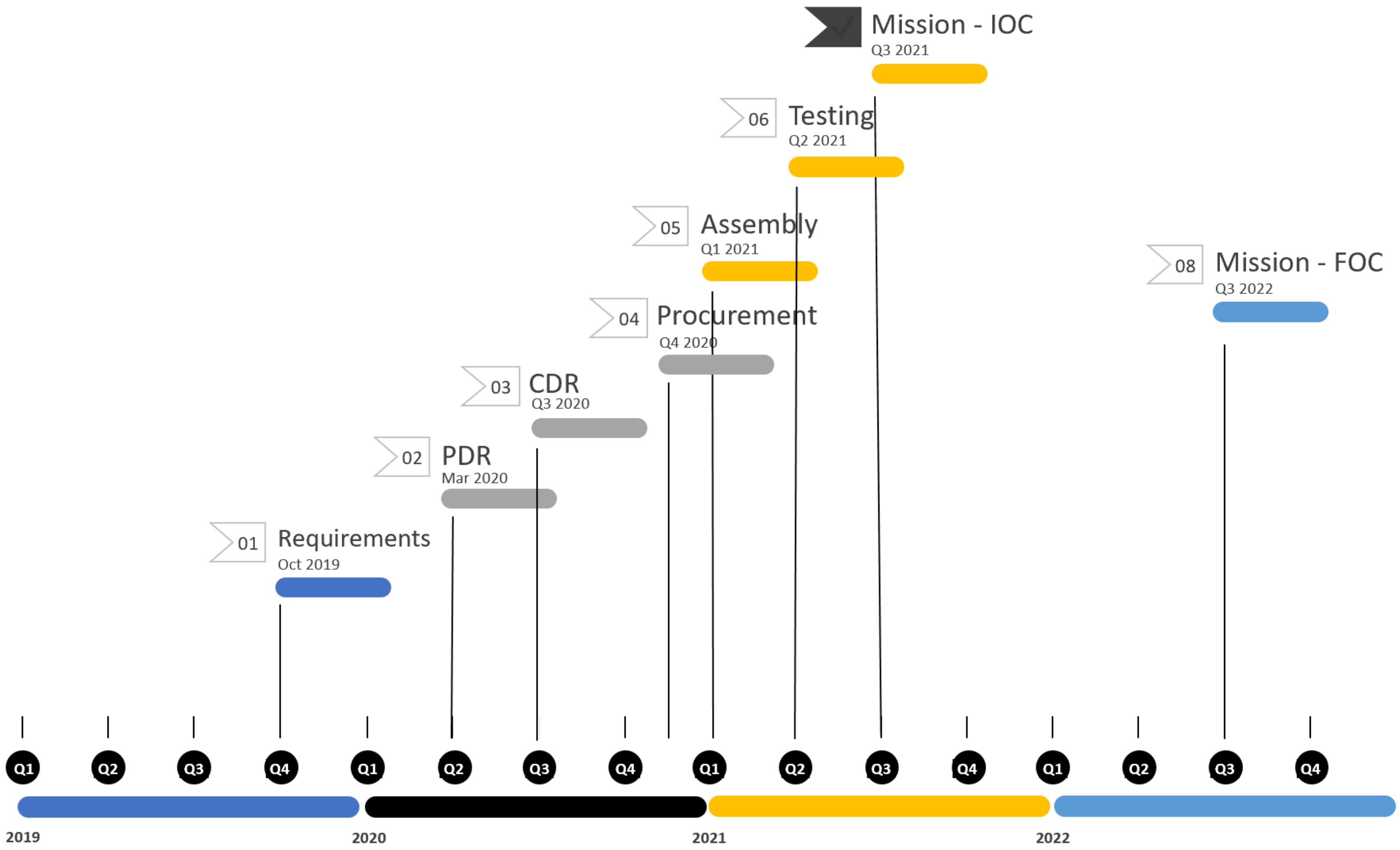

Figure B-16 - ORCA project roadmap

Appendix C Requirements

The following requirements are from the System Requirements spreadsheet [53].

C.1 Top Level Requirements

TLR-0010 Minimum RSO size

The system shall be capable of tracking objects as small as 300 cubic centimetres volume.

TLR-0020 Coverage

The system shall provide global coverage of the LEO region (from 800 km to 1400 km altitude).

TLR-0030 Response time

The system shall analyse all potential collisions with a likelihood greater than 1 in 10,000 within 12 hours.

TLR-0040 Collision notice

The system shall notify primary users of state vectors for RSOs involved in potential collision within six hours of the potential collision becoming known.

TLR-0050 Temporal resolution

The system shall provide collision object state vectors as frequently as every 6 hours if required by operators.

TLR-0060 Lifetime

The system shall operate for a minimum of five years, measured from beginning of operational service to End of Life.

TLR-0070 Earliest launch

First launch for the final constellation shall occur before 2028.

TLR-0080

The mass of an individual spacecraft shall not exceed the limit for a 12 U CubeSat that is specified in the CubeSat specification.

TLR-0100 Cost limit

The overall system shall cost no more than £250m.

TLR-0110 COTS components

Commercial Off The Shelf (COTS) components shall be used where possible.

C.2 Operational Requirements

OPS-0010 Telemetry storage

The system shall be capable of storing telemetry for one missed orbit in the event that communications breakdown.

OPS-0020 Data delivery

The system shall be capable of delivering two orbits worth of data to the ground in one pass.

OPS-0030 Operations centre

The system shall incorporate a ground operations centre(s) capable of co-ordinating and planning spacecraft operations.

OPS-0040 Operating license

The system shall operate with a valid UK Space Agency (UKSA) operating license.

OPS-0050 Insurance

The system shall be insured to cover any indemnity and third party liability costs that may arise.

OPS-0060 Radio license

The system shall operate with a valid Ofcom radio operating license.

OPS-0070 Constellation design

The system shall inform the relevant people / organisations of the orbit and orbit uncertainty of resident space objects (RSOs) with enough notice as defined in TLR-0030.

C.3 Payload Requirements

PAY-0010 Minimum detectable size

The system shall track objects as small as 125 cubic centimetres at 1000 km.

PAY-0020 Debris description

The system shall determine the size of tracked objects.

PAY-0060 Deep learning

The artificial neural network shall be able to train itself with a dataset of the order of 1E3.

PAY-0070 Dataset

The payload subsystem shall use on-board artificial intelligence to detect trails of debris on the frames taken.

PAY-0080 Tracking

The system shall identify the RSO trajectory.

PAY-0090 Obstruction

The field of view shall not be obstructed by other instruments.

PAY-0100 False positives

The subsystem shall have a false positive detection rate of no more than one in 50.

PAY-0110 Sun exclusion

The payload shall not be pointed within 15 degrees of the Sun vector.

PAY-0120 Accuracy

The payload shall have an accuracy of under 1 km.

PAY-0130 Range

The payload shall observe debris in the range of 50-1000 km.

PAY-0140 System size

The payload volume shall be no more than a 6 U CubeSat platform (12x24x36 cm, 10,368 cubic centimetres).

PAY-0150 Field of view

The payload shall have a field of view of 2 degrees.

C.4 Mission Requirements

MIS-0010 Orbit minimum altitude

The space segment's orbit shall remain above 735 km from the Earth's surface during operations at its periapsis.

MIS-0020 Orbit maximum altitude

The space segment's orbit shall remain below 765 km from the Earth's surface during operations at its apoapsis.

MIS-0030 Constellation design

The system shall have a sufficient number of satellites for the payload to have global coverage of the LEO region (TLR-0020) with a margin of no less than 10%.

MIS-0040 Orbit control

The AOCS subsystem shall keep the satellite on its defined orbit within tolerance (as defined in MIS-0010 and MIS-0020) for the entire operational life (as defined in TLR-0060).

MIS-0050 AOCS Δv

The AOCS subsystem shall have a capacity of 90 m/s delta-v to maintain orbit in compliance with MIS-0060.

MIS-0060 AOCS peak power

The AOCS Subsystem shall draw no more than 10 W of peak power.

MIS-0070 AOCS mass

The AOCS subsystem shall use no more than 11% of the total system mass, as defined in TLR-0080.

MIS-0080 Slew rate

The AOCS system shall have a maximum slew rate of at least 0.05 deg/s.

MIS-0090 Pointing accuracy

The AOCS system shall have a pointing accuracy of at least 0.1 deg.

MIS-0100 Attitude determine accuracy

The AOCS shall have a determination accuracy of at least 4 arcseconds.

MIS-0110 Licensing

The AOCS system shall contain no prohibited components (as defined in Calpoly General Requirements for CubeSats).

MIS-0115 Maximum launch mass

Total launch mass shall be no less than 700 kg, including all spacecraft being launched.

MIS-0120 Initial orbit

Launched spacecraft shall be deployed into initial orbits as defined in the CONOPS.

MIS-0130 Vehicle structure

Vehicle structural mass shall be no more than 10% of the total.

MIS-0140 Launcher and deployer TRL

TRL of the launch vehicle and deployment systems shall be no less than 8.

MIS-0150 Disposal guidelines

At EOL, spacecraft shall follow the ESA Requirements on EOL de-orbit.

MIS-0160 Time to deorbit

In an uncontrolled de-orbit the S/C shall be moved into a reduced lifetime orbit (ISO24113,6.3.3). The calculus of the life should include a 5% margin below 25 years (ISO27852:2016).

MIS-0170 Reliability of the system

An immediate disposal of the spacecraft shall take place if its reliability drops below 90% (ISO24113:2019).

MIS-0180 Reliability of disposal

The Post Mission Disposal system requires a success rate of at least 90% (ISO24113:2019).

MIS-0190 Passivation

During disposal the S/C shall be depleted from all sources of stored energy in a controlled sequence (ISO24113:2019,6.2.2.3).

MIS-200 Re-entry

During the re-entry the casualty risk to ground population shall be below 10^-4 (ISO27875).

C.5 Mechanical Requirements

MEC-0010 Quasi-static launch load

The system shall resist axial quasi-static launch loads of up to 7.7 g and lateral loads up to 2.4 g.

MEC-0020 Resonance

The system shall have natural frequencies higher than 25 Hz, as driven by the launcher natural frequencies.

MEC-0030 Materials (structures)

The system's structure shall be a COTS structure according to TRL-0130.

MEC-0040 Random loads

The system shall resit random loads during launching without reaching the maximum stress material and with a security margin greater than zero.

MEC-0050 Centre of gravity

The system shall have centre of gravity that fits within the static unbalance limit of the launcher.

MEC-0060 Fairing volume

The system shall fit within the envelope of the launch vehicle's fairing.

MEC-0070 Operating temperatures

The system shall maintain the electronic and mechanical equipment under their operational temperature ranges.

MEC-0080 Heat dissipation

The system shall be able to dissipate the heat from inside the satellite produced by batteries/electronics/rocket motors.

MEC-0090 Space environment

The system shall protect the payload from the radiative heating from the Sun, albedo and Earth.

MEC-0100 Structural mass

Structure mass shall not be over 4.22 Kg.

MEC-0110 Mechanisms

The system shall ensure the correct deployment of solar panels, antenna and telescope.

C.6 Electrical Requirements

ELE-0010 OBDH volume

The onboard data handling hardware shall have a volume no larger than 100x100x10mm.

ELE-0020 OBDH mass

The electrical subsystem shall have a mass of no more than 7 kg.

ELE-0030 OBDH power and voltage

Consumption, TBD (around 1 W and 3.3V)

ELE-0040 Hardware operational temperature

Temperature, TBD (max 65C min -25C)

ELE-0050 Hardware radiation tolerance

The spacecraft shall remain operative after a total ionising dose of 6 krad over its lifetime.

ELE-0060 Data compression

The spacecraft shall optimise data handling (if possible).

ELE-0070 Hardware lifetime

The spacecraft shall be operative a minimum time, TBD.

ELE-0080 Battery size

Battery shall provide enough power when CubeSat in shadow in whole lifetime.

ELE-0090 Solar array size

Choose GaAs multi-junction array, size depends on power requirements, based on the worst situation.

ELE-0100 Solar array tracking system

Maybe need a tracking system, it depends on the orbit and attitude, which can influence the solar array performance.

ELE-0110 Battery temperature

The battery temperature shall be maintained between 60°C and -20°C.

ELE-0120 Solar array temperature

The solar array temperature shall be maintained between 125°C and -40°C.

ELE-0130 Ports and voltage

Electrical appliances quantity and voltage requirement.

ELE-0140 Radio frequency

Antenna(s) shall operate within the UHF and S-band spectrums allocated by the ITU.

ELE-0150 Antenna power

Antenna(s) shall have 8.6 W of power available whenever it is expected to transmit/receive data.

ELE-0160 Antenna size

Selected antenna assembly shall be compatible with CubeSat structures.

ELE-0170 Acceptable data error rate

The data error rate shall be no more than 10^-7 during 10% of any month.

ELE-0190 Receiving commands

To understand commands from ground or other satellites.

ELE-0200 Executing commands

To be able to execute external commands.

ELE-0210 Telemetry and health monitoring

The spacecraft shall include temperature, charge and voltage sensors for health monitoring.

ELE-0220 Connection with subsystems

Connection between the computer and the subsystems, especially PL.

ELE-0230 Data storage capacity

The OBDH subsystem shall include sufficient data storage to store all payload subsystem output data if a single ground station downlink is missed.

ELE-0240 Data rate handling

How much data the S/C shall process per second, TBD (around 1Mbps?).

Appendix D Debris Environment Analysis

The following document was created to summarise research carried out by this author and Guillem Duarri into the space debris and RSO environment.

Debris Environment Analysis

D.1 Abstract

This document seeks to make a brief, qualitative overview of the situation concerning resident space objects (RSOs), including space debris, orbiting Earth (specifically, artificially created objects in space). This will provide understanding of RSOs' size, how they affect the orbits where they can be found, and how they can endanger space missions. Implications for the Space Debris Surveillance project will also be discussed.

D.2 Resident Space Object (RSO) size

According to ESA's information up to January 2019 (European Space Agency, 2019), out of the 8950 orbital satellite launched since the beginning of the space age, about 5000 remain on orbit, with about 1950 still functional. These old satellites, remains of their rocket stages, as well as disintegrating and colliding parts accumulated about **129 million pieces of debris,** of which:

Only **34 000** objects are larger than **10 cm**.

900 000 objects are between **1 cm** and **10 cm**.

128 000 000 objects are between **1 cm** and **1 mm**.

Only about 22 300 objects are regularly tracked by current space surveillance networks [citation needed]. Hence, the need to track more and smaller RSOs and debris pieces is evident.

D.3 RSO orbits

LEO: Low Earth Orbit is defined as an orbit between 200 km and 2000 km over the Earth [citation needed]. The main advantages of this orbit are the fact LEOs require the least amount of energy per satellite placement, and low latency when transmitting and receiving data due to the short distance to the ground. The main downsides are the limited field of view at each pass (Anon., n.d.) and its large amount of atmospheric drag (Space Weather Prediction Center, NOAA, n.d.). LEO has been used since the beginning of the space race, and is now the region containing the largest amount of space debris [citation needed], bringing concerns for a domino effect called the *Kessler syndrome*, named after Donald J. Kessler who first proposed it (Kessler & Cour-Palais, 1978): a cycle of uncontrolled space debris collisions that generate even more space debris until the point where a screen of space debris would make it impossible for any spaceship to leave Earth. However, while it is critical to develop countermeasures to clean and mitigate space debris, the critical point of a Kessler cascade might not be as close as initially estimates (Drmola & Hubik, 2018). The urgency of taking measures against this phenomenon should not be denied though; the increase in space debris would still happen even if no further launches were made (Zhang, et al., 2019).

Popular orbits are polar LEO orbits (Portillo, et al., 2018), which, given the Earth's rotation, allows satellites to get a different vertical swath of the planet with every revolution, and allow them to fly over every point of the Earth twice every 24 hours [citation needed – is that second part accurate for all polar orbits?]. If a simultaneous observation of different parts of the Earth is necessary, a LEO satellite constellation can be used (Roberts, 2017).

MEO: Medium Earth Orbit is defined as an orbit between 2000 and GEO orbit, at 36.786 km over the Earth. This orbit is used mostly by navigation satellites, such as the GPS, GLONASS and Galileo. While these are satellite constellations, the space debris problem is still not that important in this region.

MEOs are interesting because they have a greater view of the Earth than LEOs, and shorter transmission times than satellites at GEO, but the high radiation from the Van Allen belts can damage electronic systems. A common orbit is a highly elliptical orbit (high eccentricity) that goes between MEO and HEO (Roberts, 2017).

GEO: Geostationary Earth Orbit is located at 35,786 km over the Earth. Many telecommunication satellites are located at this orbit. Because they are located at the same plane, altitude and speed collisions are less likely than in LEO, and at a lower speed, however, as this region of interest becomes congested, this probability increases, and by 2017 it was found to be up to 4 orders of magnitude greater than the first estimates (Stephens, 2017). On top of that, in its current state, observation of debris smaller than 10 cm is inaccurate with the state-of-the-art equipment on Earth (Henry, 2017).

Beyond GEO is considered High Earth Orbit, which is neither in the scope of the project nor in the risk of space debris related congestion. Figure D-1 provides a good perspective of the scale of these orbits.

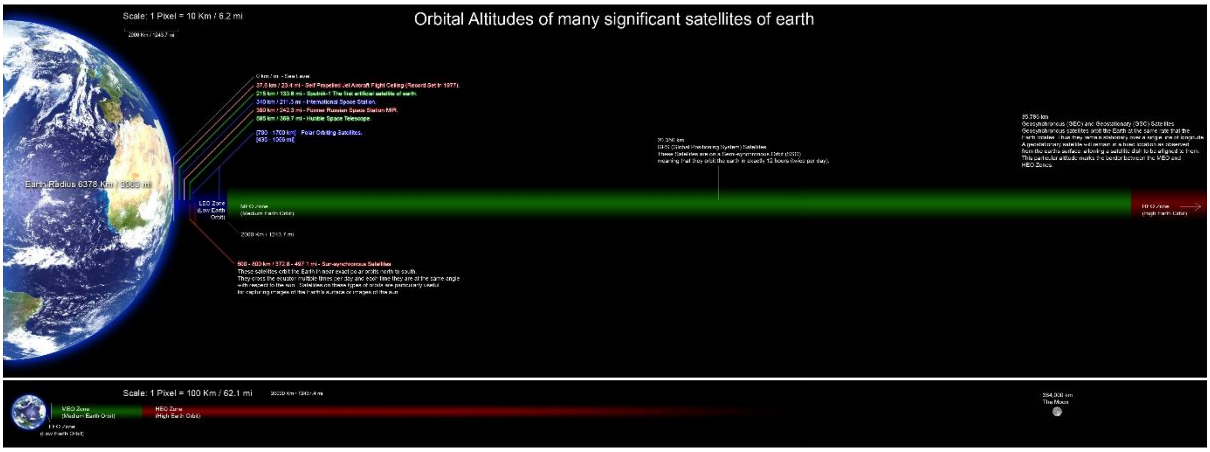

Figure D-1 - Diagrams showing a range of orbital altitude classifications (Rrakanishu, 2008)

D.4 Location: Density of space debris in LEO

According to some models, the highest concentration of space debris is found in the 800 km – 1000 km (European Space Agency, n.d.) and 1200 km – 1400 km height layers of LEO (Zhang, et al., 2019).

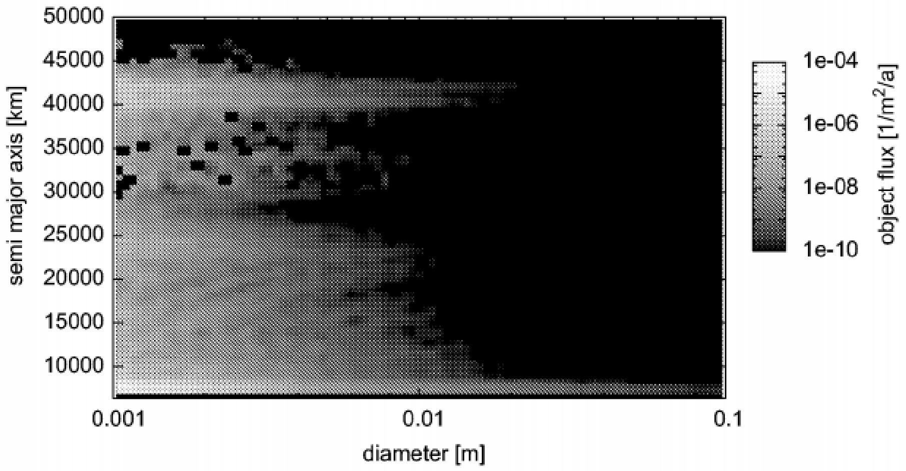

Figure 1. Debris object flux per year vs. semi major axis and diameter. Object flux cut at 1e-10 [$1/m^2/a$].

D.5 Threat to space missions

The high speed at which space debris fly makes them carry a great amount of energy, if we make some rough estimates:

A .50 BMG bullet used in anti-materiel rifles carries **17491 J**

1 cubic centimetre of aluminium at LEO carries **81190 J**

Should such a fragment impact a space object with its full kinetic energy, the impact would have approximately 4.6 times the energy of an armour piercing shot.

However, in the event of a head on collision at LEO (about 16 km/s), double the speed implies four times more kinetic energy:

1 cubic centimetre of aluminium at LEO carries **341632 J**

One of the main effects of such a high velocity impact is “spalling”, where the object does not penetrate the protective layer, but the protective layer breaks on the inside, propagating particles at high speed (Anon., n.d.).

Given the current state of space debris observation technologies, only larger, catalogued space debris can be located in order to perform evasive manoeuvres, smaller debris have to be dealt with through passive protection, such as Whipple shields (Layered protection). When it comes to simulating these impacts for study, given that physical tests of these cases are complicated and expensive, rely on complex codes called “hydrocodes” (or hydrodynamic codes) (European Space Agency, n.d.).

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Appendix E Concept of Operations (CONOPS)

Concept of Operations

Version number	**Date created**	**Description**	**Modified by**
v0	18/11/2019	Work began	10BS
v0.1	28/01/2020	Structure and first details added	10BS // 10WE

Document objective

A rough description of activities from CubeSat deployment to end of life.

List of Acronyms

LEOP Launch and Early Orbit Phase

LV Launch Vehicle

List of Figures

List of Tables

E.1 Mission Objective

To design a constellation of small satellites that allows for detection and tracking of space resident objects for prompt and global identification of collision threats and traffic management in LEO regions.

-> fallout: improvement of SSA, principally in LEO

E.1.1 Mission Justification

From NASA 2009b on 6 Nov 2009 21 crew members were awoken by flight controllers as debris was passing nearby (wasting precious resources)

Earlier, on 12 Mar 2009, ISS astronauts have to shelter in a docked Soyuz S/C as a precaution for the same reason (also from NASA 2009b)

Flight rules (Foster et al 2001)

[Understanding how the ISS manoeuvres - https://www.nap.edu/read/5532/chapter/8#48]

E.2 Timelines

E.2.1 Project timeline

- Design

- Procurement

- Assembly

- Launch & Mission

E.2.2 Mission timeline

- Launch

- Separation and Manoeuvre

- Deployment of equipment

- Initial start-up

- Operation

- Disposal / Renewal

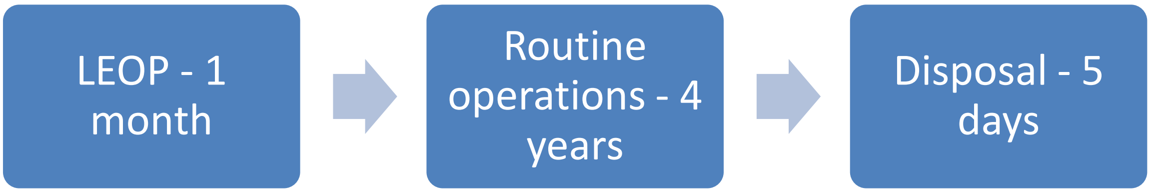


Figure E-1 - Overall satellite mission phases

Launch and Early Orbit Phase (LEOP)

LEOP is concerned with injecting the satellite into its required orbit and confirming that all systems are working nominally before operations

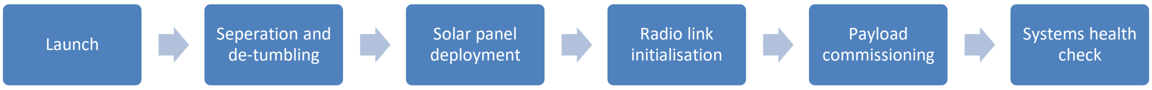


Figure E-2 - LEOP timeline

Launch

1) Launch into target orbit on one of a selection of LVs

Separation and de-tumbling

1) Spacecraft separates from LV payload adapter/P-POD
2) Reaction wheels used to de-tumble spacecraft

Solar panel deployment

1) Start generating power

Radio link initialisation

1) Turns on radios and downlink initial status
2) Confirm sufficient SNR received at spacecraft

Payload commissioning

1) Slew spacecraft to known target constellation
2) Open telescope sun shield (if fitted)
3) Take first image
4) Onboard image quality assessment
5) Downlink image for further analysis
6) Proceed with LEOP unless interrupted by telecommand from ground

Systems health check

1) Measure battery and solar array voltages
2)

Nice way to represent the sequence of events (taken from another GDP) …

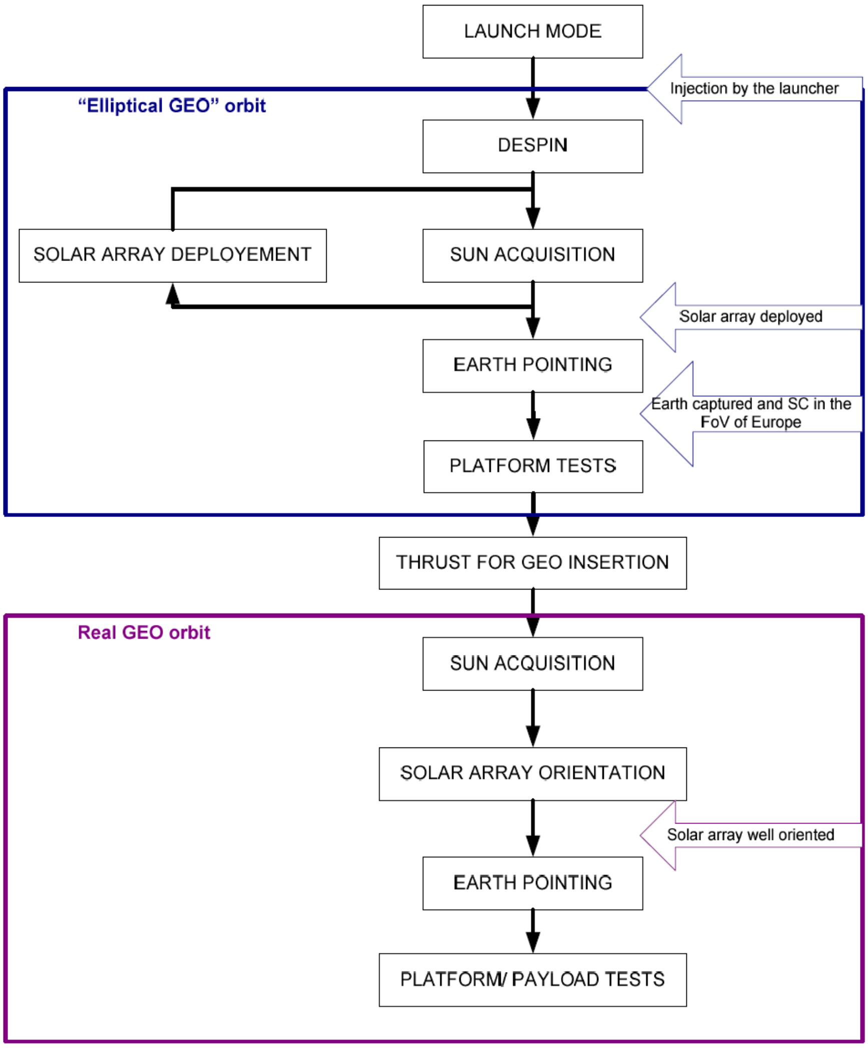


Figure 25: Event sequence before Operational phase

Routine Operations

Disposal

E.3 Constellation Design

E.3.1 Operational Satellites

The operational satellites consist of 2 orbital planes, both of the same altitude of 750 km, but with a difference in Right Ascension of Ascending Node (RAAN). One of the orbit planes is at a RAAN of 52° and the other has a RAAN of 59.5°. The difference in the RAAN between the two planes allows for a wider coverage of the debris field, and the potential for a 'hand-over' of debris tracking where if a satellite from one plane receives a limited amount of data, then another satellite in the other plane can pick it up. The inclination of these planes is 98.38°. There are 14 satellites in each plane, making 28 in total in the operational satellite constellation. There are an additional 4 satellites as spares.

E.3.2 Active Spare Satellites

There are 4 active spare satellites which will be at the same altitude as the operational satellites of 750 km, but with a RAAN of 55.75°. With the Active Spare satellites, the total number of satellites required in orbit for the mission are 32, which can be done with 2 launches.

E.4 Ground Segment Design

E.4.1 Data Processing

S/C Information Output => Useful information to the customer

Compare against…

Qs:

Does Cranfield have a data processing centre we could use?

E.4.2 Communications System

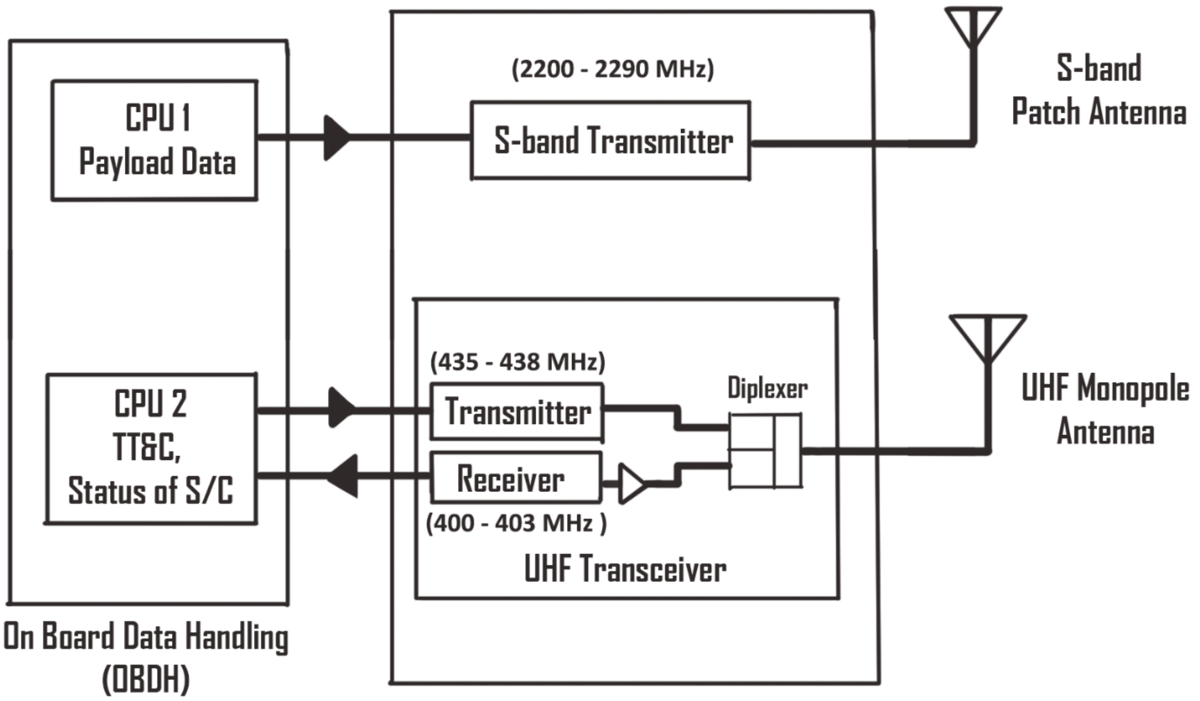
Spacecraft

The spacecraft communications subsystem uses bandwidths within the following frequency bands, as indicated by Article 5 of ITU for “Earth exploration-satellite services”.

The system is actually made of two separate systems operating at the same time. A UHF half-duplex communication between the satellite and the ground station to exchange satellite status information and commands, and a higher frequency S-band downlink that can deal with the increased data rate demands of the payload.

The S-band transmitter features a 32GB hard drive that can store data from the payload, given that the CPU connected to it may a different data rate. It can also hold data from a previous cycle, in the event of a missed pass over the ground station.

The UHF communications link will also be used in emergency mode for the satellite.



Ground segment

The ground segment of the communications link is mainly composed of ground UHF and S-band antennas with tracking capabilities. The main ground station would be Cranfield ground station, but the coverage of the communications link can be greatly improved by using the ESTRACK network of antennas.

The volume of data that can be transmitted to the payload increases proportionally to the linking time.

E.5 Operational Modes

The operational modes in Figure E-1 on the following page are defined for the space segment, with those in Figure E-2 being applicable solely to the camera payload.

Table E-1 - Spacecraft operational modes

Mode name	Mission Phase	Description	Spacecraft Attitude
OFF	LEOP	• Used during launch only • Entire spacecraft is kept deactivated by microswitches in the CubeSat deployer	N/A
DETUMBLE	LEOP	• Stop the tumbling triggered upon spacecraft separation from the LV. • Reduce all angular rates to approximately zero so SISO controllers can be used.	Undetermined initially, stabilises to best charging attitude
SUN Acquisition	LEOP/Operational	• Attain and maintain the best attitude for battery charging. • The solar panels are then deployed to start charging the batteries.	Any initially, solar panels towards Sun at end

		• Rotate around one axis then, if Sun not yet found, halt then rotate around an orthogonal axis to find the Sun in the Sun sensor's initial blind spots. • Data processed from previous debris pass once Sun pointing to make use of high power availability	
ORBIT INSERTION	LEOP	• Corrects any errors in the insertion orbit caused by inaccuracies in the LV	As required for orbit correction
COMMISSION	LEOP	• Systems health check – sensors read, and data downlinked • First image taken by camera then downlinked	Towards reference constellation for first image
SLEW	All	• AOCS orients spacecraft to the desired attitude	As desired
safe	Any, as required	• Used during emergency situations • Uses SUN ACQUISITION mode to maintain fully charged batteries • No downlinking of payload data • All unessential systems turned off	Solar panels towards Sun

		• Health and GPS data transmitted intermittently on half-duplex UHF system • UHF receiver active to receive rescue commands from ground • Transmit/receive duty cycle about 25%	
acquire	Operational	• Special case of SLEW mode • Camera in its STANDBY mode (distinct from the overall spacecraft STANDBY mode) ready for imaging of the target RSO	Towards target RSO
Transmit/receive	Operational	• S-band transmitter used to downlink payload data • UHF transmitter used (half duplex) to downlink health data • UHF receiver used to uplink commands for future operations	S-band and UHF antenna nadir pointing

Table E-2 - Payload operational modes

Mode name	Mission Phase	RSO Scanning Phase	Description
OFF	LEOP	N/A	• Camera is completely disabled by removal of all supply voltages • Used while spacecraft in HIBERNATE mode
STANDBY	Operational	During spacecraft SLEW mode	• Necessary voltages are supplied to camera • No images are taken, or other data transferred
EXPOSE	Operational	Taking picture	• Exposure of sensor for image taking
READOUT	Operational	Once image exposed	• Use FPGA/frame grabber to assemble image from sensor pixel exposure values
STORE	Operational	Once image generated	• Send image to on-board computer via data bus for storage

E.6 Key Constraints

- [R-1] Payload shall detect small objects in LEO ≥ 1cm (preferably less)

- [R-2] Full coverage of LEO region with selected payloads

- [R-3] Detection of treat of collisions with more than 28.5 hours' notice (as per ISS)

- R-4] The platform shall use commercial off-the-shelf technology (possibly CubeSat technology) to reduce cost

- [R-5] Mission Duration: at least 5 years

E.7 Operations Guidelines

- prevent contamination of outer space and adverse changes in the environment of the Earth (OSA Licence – ref licensing plan)
- avoid interference in the space activities of others (OSA Licence – ref licensing plan)

E.8 On-Orbit Servicing & Upgradability

E.9 Test & Simulation

Currently covered in Risk document

E.10 Workforce Requirement

Possible personnel bins:

- Flight Operations
    - RTFO - Real-Time Flight Operations (real-time monitoring & C2)
- Mission Analysis (can be combined or separated)
    - SO - Spacecraft Operations (general support)
    - FD - Flight Dynamics (plans for the future)
    - PO - Payload Operations (payload instruments)
- Data Processing & Management
- Launch Operations (assembly principally as most will be covered by LV organisation)
- Operations management & quality assurance
- Operations infrastructure & support
- Software development & maintenance

Have to consider extra WR at certain phases of the mission (Launch, commissioning etc.)

Need to compute WR

Appendix F SPENVIS raw results

The follow results were generated in the SPENVIS tool, as described in Section 8.3 Radiation analysis.

Table F-1 - SPENVIS radiation results

Total mission dose (rad)								
Al absorber thickness			Total	Trapped electrons	Brems-strahlung	Trapped protons	Tr. electrons+ Bremsstrahlung	Tr. el.+Bremss. +Tr. protons
(mm)	(mils)	(g cm^{-2})						
0.050	1.968	0.014	5.571E+05	5.435E+05	1.040E+03	1.252E+04	5.446E+05	5.571E+05
0.100	3.937	0.027	2.736E+05	2.664E+05	6.035E+02	6.581E+03	2.670E+05	2.736E+05
0.200	7.874	0.054	1.039E+05	9.986E+04	2.984E+02	3.771E+03	1.002E+05	1.039E+05
0.300	11.811	0.081	5.323E+04	5.017E+04	1.884E+02	2.864E+03	5.036E+04	5.323E+04
0.400	15.748	0.108	3.247E+04	2.994E+04	1.355E+02	2.399E+03	3.007E+04	3.247E+04
0.500	19.685	0.135	2.238E+04	2.016E+04	1.070E+02	2.109E+03	2.027E+04	2.238E+04

0.600	23.622	0.162	1.679E+04	1.480E+04	8.887E+01	1.905E+03	1.489E+04	1.679E+04
0.800	31.496	0.216	1.098E+04	9.267E+03	6.720E+01	1.641E+03	9.334E+03	1.098E+04
1.000	39.370	0.270	7.938E+03	6.406E+03	5.429E+01	1.478E+03	6.460E+03	7.938E+03
1.500	59.055	0.405	4.338E+03	3.060E+03	3.657E+01	1.242E+03	3.096E+03	4.338E+03
2.000	78.740	0.540	2.768E+03	1.635E+03	2.753E+01	1.106E+03	1.662E+03	2.768E+03
2.500	98.425	0.675	1.952E+03	9.144E+02	2.223E+01	1.015E+03	9.366E+02	1.952E+03
3.000	118.110	0.810	1.491E+03	5.244E+02	1.875E+01	9.474E+02	5.431E+02	1.491E+03
4.000	157.480	1.080	1.042E+03	1.775E+02	1.431E+01	8.502E+02	1.918E+02	1.042E+03
5.000	196.850	1.350	8.497E+02	5.907E+01	1.164E+01	7.789E+02	7.071E+01	8.497E+02
6.000	236.220	1.620	7.515E+02	1.911E+01	9.868E+00	7.226E+02	2.898E+01	7.515E+02
7.000	275.590	1.890	6.893E+02	6.025E+00	8.602E+00	6.747E+02	1.463E+01	6.893E+02

8.000	314.960	2.160	6.440E+02	1.821E+00	7.663E+00	6.345E+02	9.484E+00	6.440E+02
9.000	354.330	2.430	6.065E+02	5.265E-01	6.934E+00	5.990E+02	7.461E+00	6.065E+02
10.000	393.700	2.700	5.742E+02	1.437E-01	6.345E+00	5.677E+02	6.488E+00	5.742E+02
12.000	472.440	3.240	5.202E+02	7.117E-03	5.444E+00	5.148E+02	5.451E+00	5.202E+02
14.000	551.180	3.780	4.752E+02	1.184E-04	4.777E+00	4.704E+02	4.777E+00	4.752E+02
16.000	629.920	4.320	4.378E+02	1.420E-06	4.250E+00	4.335E+02	4.250E+00	4.378E+02
18.000	708.660	4.860	4.051E+02	4.129E-08	3.821E+00	4.013E+02	3.821E+00	4.051E+02
20.000	787.400	5.400	3.766E+02	0.000E+00	3.463E+00	3.731E+02	3.463E+00	3.766E+02