Space Engineering Assignment 3 - Group 1

The University of Sydney

School of Aeronautical, Mechanical and Mechatronic Engineering

AERO2705

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1 Mission Overview

This mission is to design a Robotic Servicing Vehicle (RSV) and an Orbit Augmentation Platform (OAP) capable of one test slot and the servicing of 14 client spacecraft before relocating to an orbit graveyard. The RSV must be capable of attaching to the OAPs before relocating to the client spacecraft within the geostationary arc. The mission life is 7 years, and the RSV supplies 2x4K video streams at 24FPS of the client satellites to perform successful docking. The overall mission timeline is summarised in Figure 1.

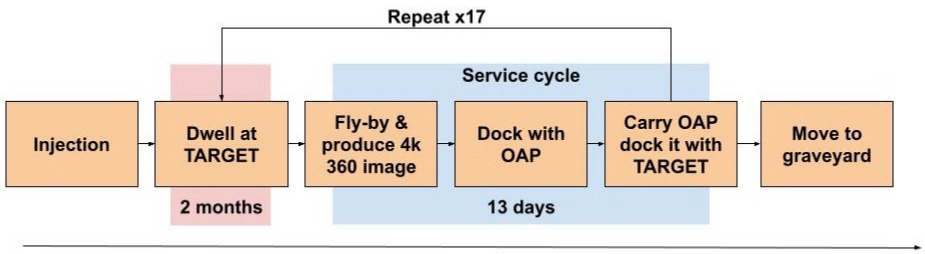


Figure 1: Overall Mission Timeline

Robotic servicing vehicles are still relatively new and similar missions are currently being undertaken by Northrop Grumman [6] and Defence Advanced Research Projects Agency [1]. This report looks at designing a competitive RSV and OAP mission plan.

2 RSV and OAP Design

The RSV and OAP were designed so that space could be maximised within the spacecrafts to ensure an efficient layout of the internal mechanisms. The RSV and the OAP were designed to compliment each other to ensure a seamless docking procedure and manoeuvres.

2.1 RSV Design

The RSV adapted a layout, similar to that of a cube-sat, which consists of a stacking method to store the main electrical components.These main components are housed in the data bus, in the order of their respective thermal requirements and stored at the front end of the RSV as it maintains a consistent centre of gravity throughout the system. Thus, the propulsion system is situated at the rear end of the RSV to counteract the weight present at the front end.

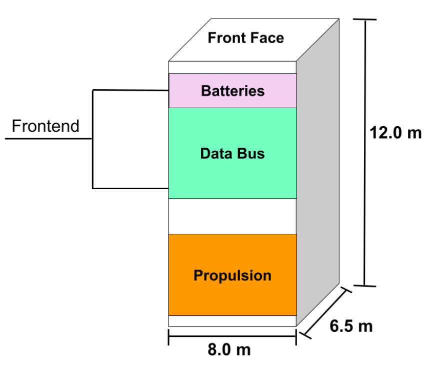


Figure 2: Overall RSV Layout and Design

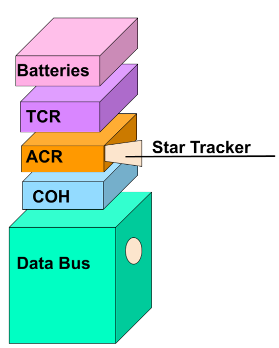


Figure 3: Design of RSV Data Bus

The exterior of the RSV composes of metals and coatings to prevent damages from the heat as well as potential debris the space craft could come in contact with. Aluminium, aluminium composites (aluminium-beryllium) and alloys (A6061) are used due to their lightweight, durability and resistance to degradation over the period of our mission.

The components housed atop the RSV include the communication dishes, antennas, cameras, sensors, solar panels and the robotic arm. The solar panel, antennas and the communication dishes are affixed on rotatable supports to ensure 360° of rotation. Additionally, the solar panels and the robotic arm are both retractable and expandable for when not in use and during deployment. After deployment of the RSV into orbit, the solar panels expand to their full width and remain in that position for the duration of the mission. The robotic arm, however, is controlled by linear actuators during the RSV-OAP rendezvous and can be retracted and extracted as per needed.

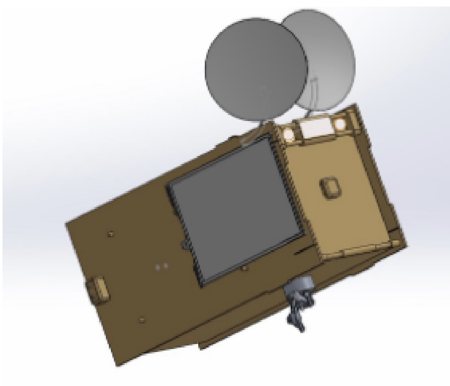


Figure 4: RSV Design with contracted arm and panels

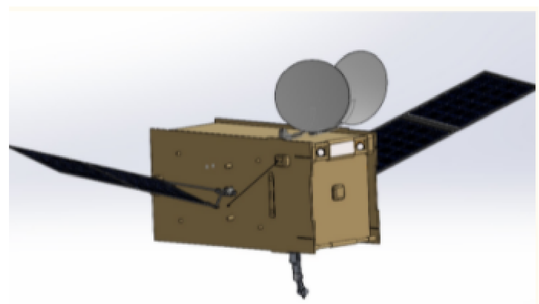


Figure 5: RSV Design with expanded arm and panels

2.2 OAP Design

The OAP is designed using the same general principles as the RSV. Since the OAP power requirements are lower than those of the RSV, the overall design is smaller, having dimensions of 2.5 m × 2.5 m × 3. In addition having the same capabilities for solar panel rotation as the RSV, the OAP has the satellite docking mechanism located at the front and attachment point for the RSV arm on top. As can be seen in Figure 6, the shape of the OAP is much the same as that of the RSV.

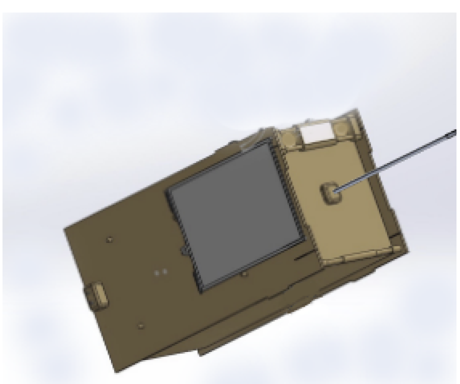


Figure 6: OAP design with contracted panels

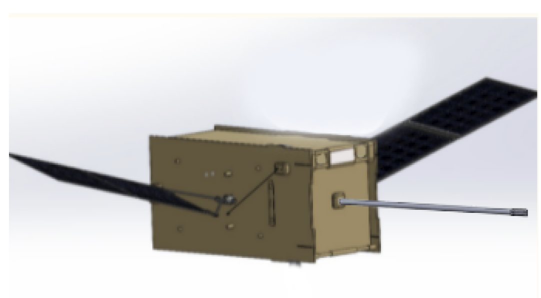


Figure 7: OAP design with expanded panels

3 System Level Requirements

3.1 Thermal

The temperature in space varies greatly based on the radiation of the sun, affecting the temperature sensitive components greatly. As a result, the thermal subsystem will be integral for the functionality of the spacecraft.

3.1.1 Thermal System Design

Given the temperature ratings specified for the integral components, we can identify the critical, important and non-critical components along with their operating temperature requirement. These ratings are determined by considering the placement of the space craft during solstices and eclipses.

Critical components include the most temperature sensitive components, that need to strictly adhere to the set operating temperatures. Important components refer to those that must remain at the operating temperature as their functionality is important during various stages in the cycles. Non-Critical components must remain within their operating temperatures, and a change in the temperature will not hinder their performance. This information is tabulated below in Table 1.

Table 1: Temperature requirement and ratings for the components

Components	Desired Operating Temperature(°C)	Operating Temperature(°C)	Rating
DC Current Sensor	0 to 70	45	Critical
4k Camera	-10 to 50	45	Critical
AC Current Sensor	-20 to 70	60	Important
Camera Actuators	-50 to 80	60	Important
Star Tracker	-20 to 60	60	Critical
Gyroscope	-40 to 70	60	Critical
CPU	-30 to 80	70	Critical
Data Bus	-30 to 85	70	Critical
Voltage Sensors	-40 to 85	70	Important
Batteries	-20 to 60	45	Critical
Thermal Sensor	-55 to 150	-55 to 150	Non-Critical
Pressure Sensor	-55 to 120	-55 to 150	Non-Critical
Position Sensor	-40 to 125	-40 to 150	Non-Critical

Given the temperature requirements for the integral subsystems, adequate thermal systems could then be implemented into the design stage. The stacking method proves to be compact and saves space, however, it also proves to be the most viable method to separate components into their various sub-systems, in accordance with their respective thermal components.

A mixture of passive and active thermal systems were used. The temperature sensitive ACR components are included in one stack, with a heater. The Batteries sit at the forefront of the RSV to power the payload systems, with variable heat pipes and MLI's as it creates the right flow of heat. While, the TCR stack uses another heater to control the temperature and ensures the temperature is consistent through the mission cycle. However, the most important thermal component used in the satellite is the flat plate radiator to ensure a flow of the excess heat that will be injected into space as the components are constantly running and being powered to ensure the mission proceeds seamlessly.

Table 2: Thermal components and their locations for the subsystems

Thermal Component	Subsystem	Location	Temperature(°C)
	External Structure	All faces of the RSV	-
MLI- Passive	Batteries	In the Data Bus	-
	Cameras	Front end of the RSV	45
Flat plate radiator - Active	Impacts all subsystems	Side faces of RSV	-
Variable Conductance Heat Pipe - Passive	Payload	Front end of the RSV	45
Heater- Active	ACR	Front end of the RSV	60
Heater- Active	TCR	Front end of the RSV	70
Cyrogenic cooler- Active	Batteries and CDH	Front end of the RSV	-40 to 150

For the OAP, the RSV's thermal system design was adopted as the components are the same throughout both systems, designed to withstand a similar amount of solar radiation throughout its life cycle.

3.2 Command and Data Handling

The command and data handling subsystem consists of the CPU, data bus, encoders and decoders and micro-controllers.

Considering the radiation hardened CPUs available at the moment the RAD750 was chosen for both the RSV and the OAP. Although there are more powerful CPUs available, the RAD750 meets the requirements of the mission and is able to process the 2x4K video streams as well as other command required by the spacecraft. The data bus chosen was the MIL-STD-1553b. Similarly to the CPU selection this data bus is sufficient for this mission and it is unnecessary to use anything more powerful. The specifications of these components can be found in Appendix A.

Sensors will also be important in the CDH subsystem in providing feedback to micro-controllers before this feedback is decoded and processed by the data bus and CPU so that decisions on spacecraft operation can be made. Voltage and current sensors will be required at each of the electronic components to check for fluctuations. Thermal and pressure sensors will also be required within the spacecraft and at each electronic component to check that they are at the correct operating temperature. Angular position sensors will also be placed on the reaction wheels and cameras to ensure that they are angled correctly.

The docking process will use TriDAR sensors to provide high precision imaging of the client satellite and help in collision avoidance. TriDAR combines LIDAR and triangulation technology to allow for precision both at close proximity and over long distances [7]. This is advantageous for autonomous rendezvous and docking, obstacle avoidance and inspection.

Without this subsystem the satellite will not be able to function and complete the mission, as a result there will be adequate redundancy and a backup will be carried for each of these components.

3.3 Telemetry, Control and Ranging Subsystems

The satellite operates within the Ku FSS frequency band. To receive the command signal, the satellite requires an omnidirectional antenna, the output of which is passed through a Berkeley Packet Filter to reduce incoming packets (by allowing only certain types of traffic), then denoised with a low-noise amplifier and downconverted (using band pass filters) to the output frequency. Then, it is passed through a demultiplexer, so we now have access to individual channels. This information is dealt with by the CPU.

In addition to returning the ranging signal, telemetry data from sensors needs to be passed through a linearised travelling wave tube amplifier, which amplifies the signal, and all channels are combined into one signal with a multiplexer. This signal is sent to the telemetry (transmission) omnidirectional antenna.

The same process is used for the video transmission, but this uses QPSK, a different modulation type allowing for more data in a given bandwidth. Given the much larger bandwidth, the video antenna needs to be high-gain, and uses a reflector. As gain is directional, we use a rotational actuator to point the reflector at the ground station to ensure the signal is received.

3.4 Power

The power subsystem consists of solar panels, batteries, discharge and charge controllers and voltage regulators. When choosing these components consideration was primarily given to the power requirements of the RSV and OAP and the lifetime of the components, but mass, efficiency and operating temperatures of all of the components were also considered. The solar cells and batteries chosen for both the OAP and RSV were triple junction gallium arsenide solar cells [3] and the 60Ah Space Cell (lithium ion cell) [2].

Before sizing and determining how many of each of these cells was required the power budget for the OAP and RSV needed to be calculated (**Part 2.4.1**). This budget considers four separate cases; the June solstice, when power supplied by the solar arrays will be at a minimum; the transient case; during an eclipse (maximum of 70 minutes; and the case of moon shadowing (maximum of two hours) to ensure that the RSV and OAP have enough power to meet their loads in all scenarios.

When sizing the solar arrays, the angle of tilt of the arrays to the sun, distance to sun and a worst case degradation 2%/year due to solar radiation was considered. The output power of the arrays could then be plotted over the total mission time in days based on an inputted array area (Figures 5 and 6) to find the required surface area of the solar panels for the RSV and for the OAP.

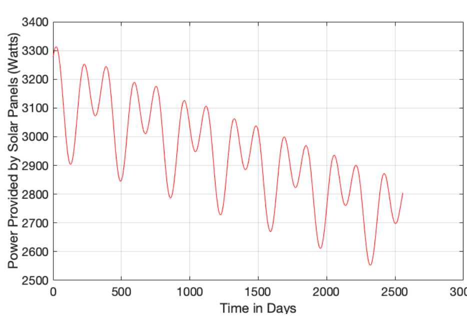


Figure 8: OAP Solar Array Output over the Mission

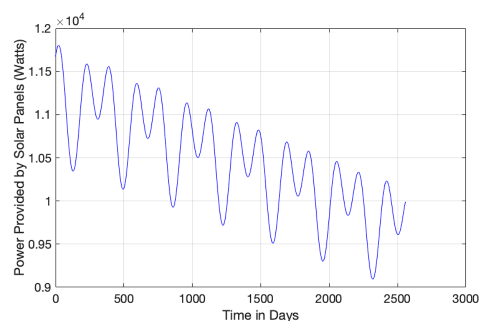


Figure 9: RSV Solar Array Output over the Mission

Consideration also had to be put into choosing the number of battery cells required for the mission. The 60Ah Space Cells have a 10 year lifespan at a 40% maximum depth of discharge, so there had to be sufficient cells to ensure that the batteries would not discharge past this point. By calculating the supply of each battery cell it was found that 10 cells would be required on the OAP and 24 cells on the RSV.

3.4.1 Power Budget of the OAP and RSV

Table 3: OAP Power Budget

Subsystem	June Solstice (W)	Transient Case (W)	Eclipse (W)	Moon Shadowing (W)
Payload	30	30	30	30
TCR	30	30	30	30
CDH	30	30	30	30
ADCS	11	11	11	11
Ion Thrusters	0	1520	0	0
Thermal	200	200	200	200
Reaction Wheels	0	120	0	0
Battery Charge	500	0	0	0
Total OAP Load	792	1941	292	292
Solar Array Capability	2550	2550	-	-
Shadowing Loss	-300	-300	-	-
4x Solar Array String Failures	-200	-200	-	-
Total Solar Power Capability	2050	2050	-	-
Solar Array Power Margin	1258	119	-	-
Margin (%)	466	5.7	-	-
Battery Capability	-	-	2560	2560
Battery Discharge Harness Loss	-	-	-400	-400
2x Battery Cell Failures	-	-	-512	-512
Total Battery Capability	-	-	1648	1648
Battery Power Margin	-	-	1356	1356
Battery Depth of Discharge (%)	-	-	20.7	35.4
Battery Max Discharge (%)	-	-	40	40
Max Recharge Duration (mins)	-	-	64	110

Table 4: RSV Power Budget

Subsystem	June Solstice (W)	Transient Case (W)	Eclipse (W)	Moon Shadowing (W)
Payload	150	150	120	120
TCR	330	330	330	330
CDH	40	40	40	40
ADCS	14	14	14	14
Ion Thrusters	0	6900	0	0
Thermal	400	400	400	400
Reaction Wheels	0	120	0	0
Battery Charge	600	0	0	0
Total RSV Load	1534	7954	892	892
Solar Array Capability	9100	9100	-	-
Shadowing Loss	-300	-300	-	-
2x Solar Array String Failures	-190	-190	-	-
Total Solar Power Capability	8610	8610	-	-
Solar Array Power Margin	7076	656	-	-
Margin (%)	460	8.2	-	-
Battery Capability	-	-	6144	6144
Battery Discharge Harness Loss	-	-	-400	-400
2x Battery Cell Failures	-	-	-512	-512
Total Battery Capability	-	-	5232	5232
Battery Power Margin	-	-	4340	4340
Battery Depth of Discharge (%)	-	-	19.9	34.1
Battery Max Discharge (%)	-	-	40	40
Max Recharge Duration (hours)	-	-	3.5	5.9

3.5 Actuary Control

The actuary control system (ADCS) for the OAP and RSV consists of the sun tracker, star tracker, gyroscope, reaction wheels, thrusters and actuators.

Many factors can cause disturbance in attitude including solar radiation, solar frequency, gravity gradient and magnetic. Because of these disturbances actuary control components are integral for the spacecraft to determine its location and rotate the spacecraft in the correct orientations. Mass, lifetime and power requirements of these systems were considered when selecting each component. See the full specifications in Appendix B.

Three reaction wheels will be used to ensure that the spacecraft can rotate in every direction. Three sun trackers will be used, as each only has a 120 degree field of vision and actuators will be used for the RSV docking arm and OAP probe and to move and angle the cameras, torch and solar panels.

As integral components for the mission, to have adequate redundancy, a backup will be carried for each of these.

3.6 Propulsion and Station Keeping

3.6.1 RSV Propulsion

The RSV propulsion system consists of a liquid apogee engine and eight ion thrusters, which alongside the three reaction wheels allows for six degrees of freedom.

The liquid apogee engine (LAE) will be used for large Δv manoeuvres, in particular the manoeuvres from injection orbit to GEO and GEO to graveyard orbit. Further, to minimise the satellite downtime during servicing, the LAE will also be used for rendezvous manoeuvres 7 and 8 (see Section 5). The selected LAE is Northrop Grumman's TR-312-100MN, which has been tested and approved for commercial geostationary satellite applications. The LAE runs on N_2O_4 and MMH propellants, which will be pressurised to a density of 1370 kg/m^3, providing a specific impulse of 325s and 502N of thrust.

For more precise manoeuvres with small Δv requirements such as those for close approach and station-keeping, the eight ion thrusters will be used. The selected ion thrusters are RIT 10 Evo thrusters, using xenon pressurised to 1500 kg/m^3. These each have a specific impulse of 3000s and a thrust of 25mN. The configuration of ion thrusters can be seen in Appendix D.

3.6.2 OAP Propulsion

The OAP's propulsion system is able to provide enough Δv for one year of stationkeeping prior to docking, stationkeeping for five years once docked to the customer satellite, and final relocation to graveyard orbit with the customer satellite. Eight RIT 10 Evo thrusters were also selected for OAP propulsion. The pressurisation level and configuration are the same as those used for the RSV.

3.6.3 Propulsion Budgets

The method used to determine the total amount of propellant and the size of the fuel storage and feed systems required for the RSV and OAP propulsion systems is outline in Appendix C. Using these final masses to

Table 5: RSV Propulsion Budget

	Manoeuvre	**Δv (m/s)**	**Δm (kg)**
LAE	Injection to GEO	2453.6	3686.3
	Servicing	227	143.3
	GEO to graveyard	12.4	7.64
Xenon	Servicing	7.44	4.59
	Stationkeeping	282	177
Total		**2982**	**4202**

Table 6: OAP Propulsion Budget

Manoeuvre	**Δv (m/s)**	**Δm (kg)**
Undocked stationkeeping	55	3.22
Docked stationkeeping	275	16.1
Deorbit	12.4	0.725
Total	**342**	**20**

3.7 Imaging

3.7.1 Sensor and Microcontroller

The satellite operators require two 4k, 24fps streams. Meeting these requirements is the Space Micro Spacecam 4K. It is a 12-bit bayer sensor, meaning there are 12 bits of data per pixel, and a Bayer colour filter is used. As this camera is designed for LEO, we have accounted for an extra 50% volume and mass to account for further radiation hardening. The sensor is multispectral, with bands in the visible range. While a hyperspectral camera may be useful for examining defects, a rad-hardened hyperspectral system with sufficient resolution is not currently commercially available. As the maximum relative velocity is only 0.013 m/s, motion blur will not pose an issue.

3.7.2 Optics

The defined system requirements need our camera to capture up to 5mm detail on a 16.2m client satellite. We image the client during our flyby orbit, which uses 3 relative orbits around the client with 45° inclination difference to reconstruct a 3D image for inspection, and has $r_{apoapsis} = 2r_{periapsis}$. With a resolution of 4096x3072 pixels, it is impossible to achieve 5mm detail at apoapsis and fit the whole satellite in frame at periapsis. As such, we need a variable zoom lens and/or pointing actuators. As a moving part is unavoidable, we will design the optics for a 2.5mm pixel size (the Nyquist sampling rate for 5mm defects). The required Field of View will be between 4.4-8.5° (see fig. 19 in appendix E).

For the 25mm diagonal sensor size and given FoV, we find the required focal length of the variable zoom lens will be 165mm - 331mm. The zoom will be calculated to keep the client a constant size in frame, facilitating the 3D image reconstruction. This lens will be custom manufactured by Asphericon, using rad hardened glass with very low aberration and roughness. The aperture will be adjustable to account for high and low light situations; a maximum aperture of f/2.2 will require a 15cm lens diameter at maximum zoom. As we will know the distance to the subject in advance, focus will be manually calculated on the ground and sent in command, as auto-focus will have poor performance in low-light situations. We also require a small linear actuator to provide 6.3 ° of pointing left and right, to inspect the solar panels in full detail. As we must use moving parts, full redundancy has been accounted for.

The 2nd camera is designed for the docking procedures, and includes the same sensors and aperture/focus capabilities, but has a fixed 80° field of view, which comes stock with the Spacecam 4K.

3.7.3 Illumination

Parts of our flyby orbit will be shadowed, and we will not be able to accurately assess the client without a method of illumination. We use one high-powered LED, providing 4292 lumens, and approximately 1000 lux on the body of the client, which is an equivalent light level to a medical theatre. Due to the inverse square law, we can't afford the power required to illuminate the whole satellite (including solar panels) at this level, but we can adjust the reflector to provide lighting to the solar panels at a lower intensity.

3.8 Payload Subsystem

3.8.1 RSV

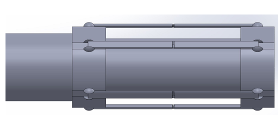
The KRAKEN® robotic arm has been selected for the RSV. The specifications for the KRAKEN® robotic arm is shown below:

Table 7: KRAKEN® robotic arm specifications [8]

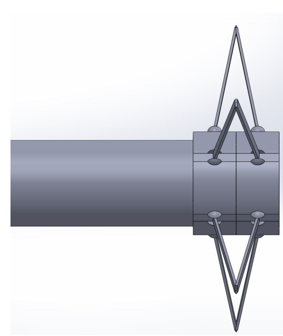
Mass (kg)	5
Joint Rotation (deg)	±345
Reach (m)	1
Payload Capacity (N)	4.91
Joint control	EtherCAT Protocol
Joint Torque (Nm)	60
Storage Dimensions (cm)	19x27x36

3.8.2 OAP

The OAP's probe will be manipulated using two independently controlled concentric rods. These consist of a 325mm outer rod (10mm diameter) and a 305mm inner rod (8.5mm diameter). Both rods are manufactured from carbon fibre and epoxy resin so that they have sufficient tensile strength to pull the OAP and the satellite together during docking (carbon fibre tensile strength is 2500MPa [4]). The rods are independently controlled to allow the head of the probe to expand once inside the combustion chamber of the satellite's LAE during the docking procedure. A CAD model showing the expansion of the head of the probe is shown below:



(a) Head during insertion



(b) Head expanded

Figure 10: Docking probe mechanism

3.9 Overall Mass Budget of RSV and OAP

The mass of the OAP can be broken up into dry mass and total mass as seen in Table 8. The below table includes redundancy. This can also be done for the RSV (see Table 9). As it has a greater payload and greater fuel and power requirements it will have a greater dry and total mass.

Table 8: OAP Mass Budget

Component	Mass (kg)
Structure	321
Ion Thrusters	80
Fuel Storage and Feed System	10
Solar Array 1	193
Solar Array 2	193
Batteries (10 cells)	16
Capture Mechanism (Probe)	17.3
CPU	1.1
TC&R and CDH Electronics	28
ADCS	70
Thermal	2
Total Dry Mass	931.4
Xenon	20
Total Mass	951.4

Table 9: RSV Mass Budget

Component	Mass (kg)
Structure	1600
Liquid Apogee Engine	6
Ion Thrusters	80
Fuel Storage and Feed System	30
Solar Array 1	398
Solar Array 2	398
Batteries (24 cells)	38.4
CPU	1.1
Telemetry/Command Antenna	2
TC&R and CDH Electronics	29.4
ADCS	70
Thermal	5
Payload (Arm and Cameras)	7.5
Total Dry Mass	2665.4
Xenon	21
LAE Propellant	4181
Total Mass	6867.4

4 Command/Telemetry Link Budget

The link budget is found by considering sources of gain and loss for each signal, calculating the signal-to-noise ratio per bit, and comparing to the minimum E_b/N_0 required to achieve a bit error rate of 10^{-7} for BPSK (binary phase shift-keying) for telemetry and control and QPSK (quadrature phase shift-keying) for video downlink. Notable assumptions made for computing the link budget include a) The satellite is at 152°E, 0°N for distance calculations; b) Doppler shift does not need to be accounted for, as there is no or very limited relative motion between the RSV and ground station; c) The client satellite is turned off during servicing, removing the possibility of signal interference and d) We require 99.9% availability, and values for atmospheric attenuation have been determined as such. We can also reduce the minimum acceptable E_b/N_0 by using a Hamming algorithm to correct bit errors.

4.1 Signal Polarisation

As we can see from figure 11, it is imperative the polarisation of signal and receiver is aligned properly - if this is not the case, both the telemetry and video link may not be closed. To avoid the possibility of issues stemming from this, we will chose a circular polarisation for our telemetry, control and video links - this does not require the receiving antenna to be calibrated to the specific (and changing with location) polarisation angle, but instead either be calibrated to right or left hand polarisation. We will select one of these based on other nearby signals, choosing the option that will have less possible interference. From table 15, we find that by eliminating attenuation due to polarisation orthogonality, we have a comfortably closed link with acceptable margin.

4.2 Link Calculations

Table 10: Link Budget

Value	Command (Up)	Telemetry (Down)	Video
Reflector Diameter (m)	13	N/A	1.5
Frequency (GHz)	14	12.252	12.252
Band Size (MHz)	0.001	0.016	45
Power (dB)	28.75	14.77	24.77
Gain (dB)	63.38	6	43.47
Output Loss (dB)	1	1	1
EIRP (dB)	91.13	19.77	67.24
Free Space Loss (dB)	162.07	162.07	162.07
Isotropic Conversion Loss (dB)	44.46	43.40	43.30
Rain Loss (dB)	10.13	10.13	10.13
Polarisation Loss (dB)	0	0	0
Implementation Loss (dB)	1.5	1.5	1.5
Total Loss (dB)	216.65	211.69	211.69
Link Performance (dB)	98.08	74.68	122.15
Signal/Noise (dB)	66.58	31.12	41.10
Required Signal/Noise (dB)	12	12	16
Margin (dB)	**42.58**	**19.12**	**25.10**

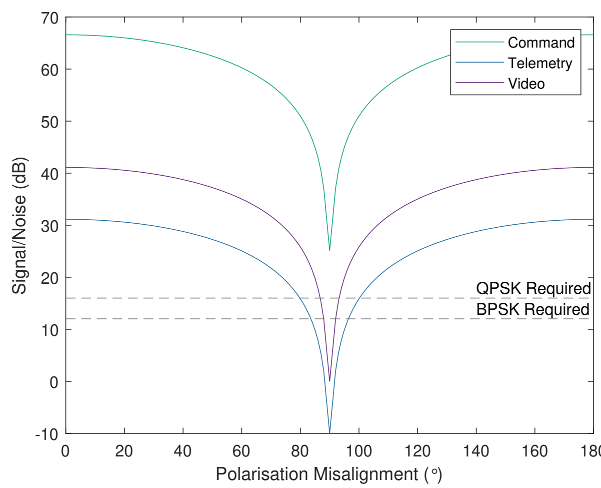


Figure 11: Signal/Noise vs. Polarisation Angle

5 Manoeuvres, rendezvous & service cycle design

5.1 Injection

The RSV is given to be in a 250 × 14880 km, 27° inclined injection orbit. Assuming a launch from Cape Canaveral, and this orbit has an argument of perigee $\omega = 0°$, the most fuel efficient burn to geostationary is a Hohmann elliptical transfer starting from injection orbit perigee (250 km), as the transfer energy ratio $\alpha = 3.02 < 11.94$. This is shown below:

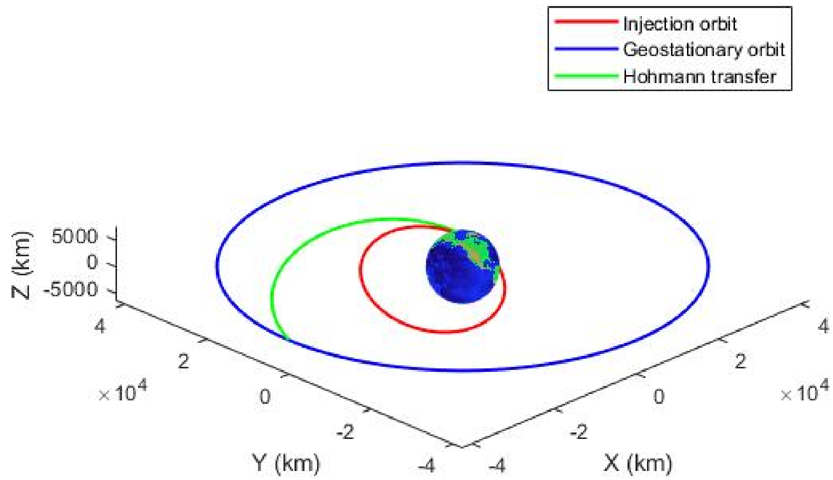


Figure 12: Injection trajectory for given orbit w/ $\omega = 0°$

The manoeuvre consists of two burns, the first being a perigee raise of the injection orbit, allowing the RSV move through the Hohmann ellipse pictured above, followed by a plane change combined with an orbital speed change burn at the Hohmann apogee to geostationary. After this, to account for the inability to predict launch time accurately, the RSV will launch with the fuel required for a 180° phase to its first service dwell site. The phase Δv was calculated in MATLAB with a given time input. This phase time was set to one month, to conserve fuel if a full 180° phase is required, and allow time for systems to be checked before the design mission is started. Not the dwell site lies $-0.5°$ behind the target spacecraft. The Δv requirements were calculated with the functions in Appendix F, assuming impulsive manoeuvres and are summarised below:

Table 11: Injection manoeuvre strategy

Injection Manoeuvre 1 - Move to GEO

Burn	Type	Location	Δv (m/s)	Time (hrs)
1.1	Hohmann perigee raise	Injection perigee	619.56	+0
1.2	GEO plane & speed change	Hohmann apogee	1800.30	+5.3
Totals			2419.86	5.3

Injection Manoeuvre 2 - Phase to dwell site

Burn	Type	Location	Δv (m/s)	Time (hrs)
2.1	Phasing	GEO	16.802	+0
2.2	Phasing	Dwell site	-16.802	+730.0
Totals			33.604	730

It is to be noted that if $\omega \neq 0°$ for the injection orbit, the plane change and apogee Hohmann burn cannot be combined in one burn, and the satellite must coast in an inclined geosynchronous orbit until it reaches its ascending or descending node, requiring more fuel, hence why $\omega = 0°$ was selected for the injection orbit.

5.2 Service cycle: fly-by & rendezvous

Post-injection, the RSV lies at its first dwell site, a geostationary orbit $-0.5°$ behind the target spacecraft. During the mission timeline, the spacecraft is designed to perform 17 services (factoring for two redundancies/dead target spacecrafts). The most efficient way to manoeuvre the spacecraft through each service was to design a service cycle, a set of 8 manoeuvres the RSV completes 17 times throughout its mission. The primary consideration of this design was the use of three fly-by orbits which the spacecraft moves through, each producing a different relative motion between the target and the RSV (or the chaser), optimising 4K imagery of the target for damage assessment. The service cycle strategy is summarised in Figure 13 below:

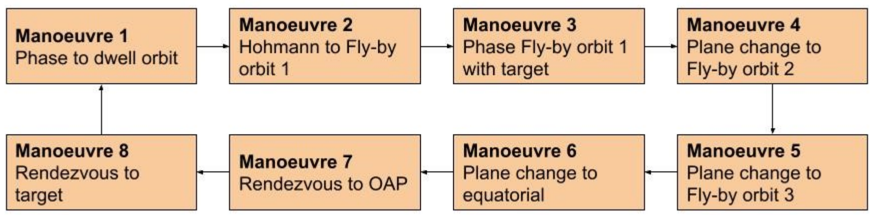


Figure 13: RSV service cycle

In summary, the RSV starts each cycle with a 2-month dwell time to allow for systems and conditions checks (such as for solar storms, upcoming moon eclipses), before executing Manoeuvre 2 and starting the fly-by and rendezvous part of the service which lasts 14 days. It is important to specify the differing orbits the RSV moves through in one cycle. For the rest of this section, the target spacecraft is moving on Orbit A, the dwell orbit is Orbit B, fly-by orbits 1, 2 and 3 are Orbits FB1, FB2 and FB3 respectively, and the OAP is moving on Orbit C, $+0.5°$ from the Target in geostationary. Also, for each service, it is assumed the target lies over Sydney, at 152° longitude, and hence ω_A is set to 152° for the Target. For reference, these orbits are briefly summarised in Table 12 below:

Table 12: Orbits the RSV moves through during service life

Orbit	Name	h (km^2/s)	e ($\times 10^{-6}$)	i (°)	ω (°)	Ω (°)
A	Target	129640.444584	0.000	0.000000	152.0	-
B	Dwell	129640.444584	0.000	0.000000	151.5	-
FB1	Fly-by 1	129640.444584	1.185	0.000000	152.0	-
FB2	Fly-by 2	129640.444585	1.185	0.000136	0.0	332.0
FB3	Fly-by 3	129640.444585	1.185	0.000136	0.0	152.0
C	OAP	129640.444584	0.000	0.000000	152.5	-

Orbits A, B and C are geostationary orbits each out of phase by 0.5° of each other. In order to select a desirable fly-by trajectory, the fly-by orbits FB1, FB2 and FB3's state vectors were first defined in an LVLH frame centred on the Target, orbit A. By defining the RSV's motion in this frame in terms of the target's mean motion, $n = 2\pi/\text{day}$, the RSV will move along an ellipse relative to the target spacecraft. So for orbit FB1, the co-planar fly-by, with $\omega = 152°$, $\theta = 0°$ (perigee):

$$\delta \mathbf{r}_{FB1} = [-0.050, 0, 0] \text{ km} \quad \delta \mathbf{v}_{FB1} = [0, -2n\delta \mathbf{r}_{FB1}(1), 0] \text{ km/s} \tag{1}$$

And for optimising the amount of surface area that can be imaged, a *relative* incline of ±45° is required. By defining FB2 and FB3 with the same position but velocity in both the y and z axes in terms of the target's mean motion, this is achieved:

$$\delta \mathbf{r}_{FB2} = [-0.050, 0, 0] \text{ km} \quad \delta \mathbf{v}_{FB2} = [0, -2n\delta \mathbf{r}_{FB2}(1), -2n\delta \mathbf{r}_{FB2}(1)] \text{ km/s} \tag{2}$$

$$\delta \mathbf{r}_{FB3} = [-0.050, 0, 0] \text{ km} \quad \delta \mathbf{v}_{FB3} = [0, -2n\delta \mathbf{r}_{FB3}(1), 2n\delta \mathbf{r}_{FB3}(1)] \text{ km/s} \tag{3}$$

It must be noted that the x-component of each $\delta \mathbf{r}$ state vector is proportionate to the perigee of the RSV's $e = \sqrt{3}/2$ ellipse about the target. As this is chosen as 50m, the relative motion induced between the RSV and the target during fly-bys are each 50×100 m ellipses, as shown in Figure 15, which is an optimal distance to minimise chance of collision and maximise imaging. The rotational operator $\mathbf{Q}_{xX}$ (see Appendix F) to convert LVLH to GCEF frame vectors was then used to convert these state vectors to GCEF state vectors relative to the target, and then this relative motion was converted to absolute motion:

$$(\mathbf{r}_{rel})_X = \mathbf{Q}_{xX}\delta \mathbf{r} \quad (\mathbf{v}_{rel})_X = \mathbf{Q}_{xX}\delta \mathbf{v} \tag{4}$$

By the equations of relative motion, to absolute GCEF frame, using the state vectors of orbit A (at $\theta = 0$), and the precession of orbit A's reference frame, the state vectors were converted to absolute motion:

$$\mathbf{r}_X = (\mathbf{r}_{rel})_X + (\mathbf{r}_A)_X \quad \mathbf{v}_X = (\mathbf{v}_{rel})_X + \mathbf{\Omega} \times (\mathbf{r}_{rel})_X \tag{5}$$

This method was used to convert eqs (1), (2) and (3) to GCEF, and then converted to classical orbital elements. To summarise, these elements were placed in Table 12 above, and were used to calculate the required Δv's for each manoeuvre using the formulae in Appendix F. Below in section 5.4 pictures each fly-by's absolute motion around Earth and their relative motion with the target as a result of setting these initial bounds.

Now, with the 6 orbits the RSV moves through during a service, the cycle will be detailed in full below. This is split into Phasing & dwell (manoeuvre 1), Fly-by (manoeuvres 2-6) and Rendezvous (manoeuvres 7 & 8). The sections below detail these stages of the service cycle, with the required Δv's and elapsed times calculated with the formulae in Appendix F, and are published in Table 14. All values in this table are calculated using the MATLAB script *Master-Calculator.m* in Appendix F.

5.3 Phasing & Dwell: manoeuvre 1

After completion of a service (ending at manoeuvre 8), the RSV must be relocated to the dwell orbit, Orbit B. This is a week-long phasing manoeuvre, moving the RSV back from A to B. Note, for a geostationary orbit, the required phasing orbit period is given as a function of geostationary angular velocity, ω_{geo}, phase difference, $\Delta\Lambda$, and a specified number of phaser rotations n:

$$T_{phase} = \frac{2\pi n + \Delta\Lambda}{n\omega_{geo}} \tag{6}$$

Eq. (6) implies that the greater the selected n, the closer T_{phase} is to T_{geo}, meaning the less change in energy and hence Δv is required to phase the orbit. This is utilised in mission design, as n is conservatively chosen as 7 for this phasing manoeuvre to minimise fuel consumption, however if there need be the phasing orbit can be set to a longer elapsed time to conserve fuel if the upcoming service is scheduled further ahead than the one just completed. At Orbit B, the required dwell is predicted at 2 months, allowing the the go-no go criteria outlined in section 6 table 11 to be green-lit. The required Δv's and elapsed time are specified below:

Table 13: Manoeuvre 1 strategy: phase & dwell requirements

Burn	Type	Location	Δv (m/s)	T+
1.1	Phase	Orbit A apogee	0.203310	+0 d 0 hrs
1.2	Phase	Orbit B apogee	-0.203310	6 d 25.52 hrs
NET TOTAL, 1 cycle + dwell			0.40662	66 d 25.52 hrs
NET TOTAL, 17 cycles + 17 dwells			6.91254	1 yr 268 d 15.84 hrs

5.4 Fly-by: manoeuvres 2-6

Manoeuvre 2: Transfer to fly-by 1

Following a green-light from ground control, the spacecraft will move from Orbit B via. a Hohmann transfer to Orbit FB1 (see Table 14). Burn 1.1 is at perigee so the following phasing and plane changing between fly-bys are at apogee to increase fuel efficiency.

Manoeuvre 3: Phase fly-by 1 with target

At the end of burn 2.2, the RSV is at apogee of Orbit FB1. The RSV is out of phase by 0.5° with the target, and hence must be phased. To conserve fuel, this phase is set with repeat factor $n = 7$ (using eq. (6)), taking a week to complete (see Table 14). This phasing implies that after Burn 3.2 is completed, finishing manoeuvre 3, the spacecraft's first fly-by trajectory, one rotation through orbit FB1, can begin, starting at apogee of orbit FB1 (or, with reference to the target, the LVLH vector $\delta\mathbf{r} = [0.05, 0, 0]$ km).

Manoeuvres 4, 5 & 6: Plane changing between fly-bys 1, 2 & 3

At the end of burn 3.2, the RSV is in phase along orbit FB1, which is depicted below, alongside the target, in absolute motion and relative motion:

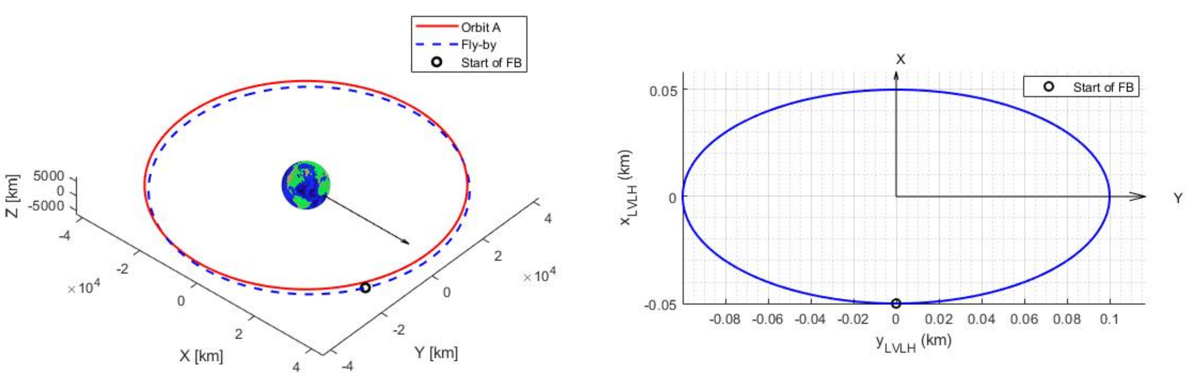


Figure 14: Orbit FB1 exaggerated absolute motion (left) and true motion relative to target (right)

By turning the 4K stream on at the start point labelled above, and letting the RSV move along this orbit FB1 for one period, the ellipse on the right (above) is moved around, with the target at the centre, hence producing an image of the target spacecraft to be assessed before docking. After one revolution of orbit

FB1 is completed, manoeuvre 4 is executed, changing the plane by 0.000136° to orbit FB2, where the RSV moves around a second revolution, producing an inclined relative motion, followed by manoeuvre 5, changing the plane by 0.000272° allowing one the RSV to move through one complete revolution of orbit FB3. In summary, the entirety of the fly-by imaging takes 3 days (one day per FB orbit), requiring the Δv's given in Table 14 and producing the relative motion as shown below, plotted by using the Runge Kutta method to solve the LVLH two-body problem in Appendix F:

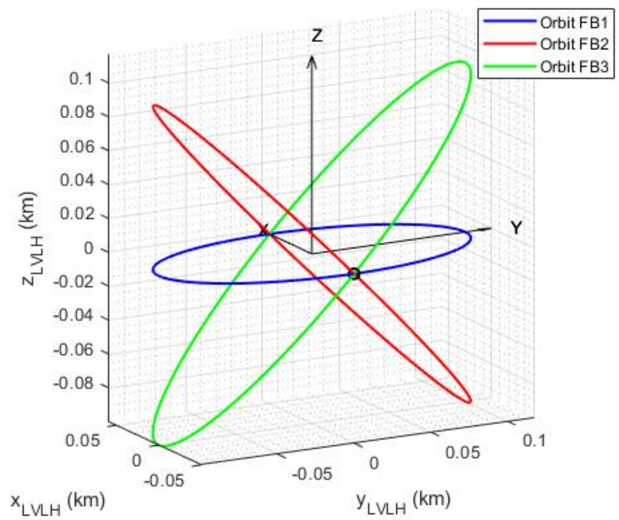


Figure 15: Complete relative fly-by motion from FB1, FB2 and FB3

As shown in Figure 15, the RSV successfully completes 3 'orbits' around the target, each at 45° to one another, maximising the imaged surface area, allowing a 360° 4K photograph of the spacecraft to be produced, which can be analysed to give a green-light for manoeuvres 7 and 8, the rendezvous. At the end of fly-by 3, manoeuvre 6 is performed, a plane change back to an equatorial geostationary orbit (see Table 14). The spacecraft then sits in orbit FB1 again for one more day awaiting the green light to rendezvous with the OAP.

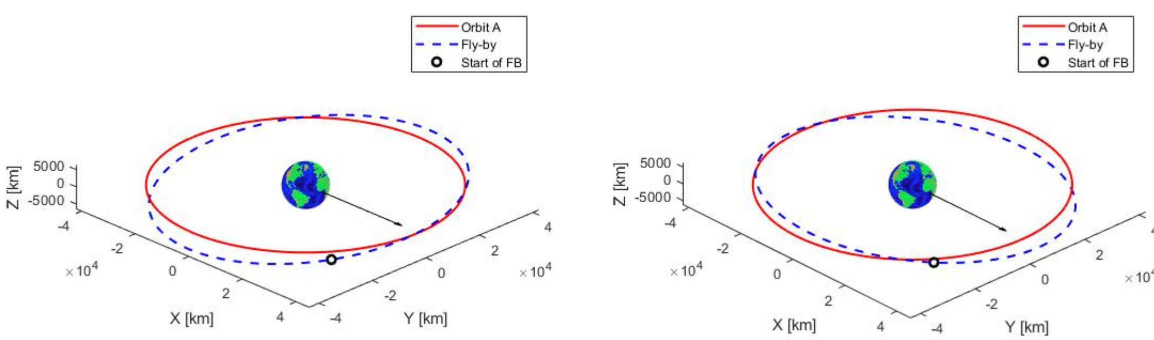


Figure 16: Exaggerated absolute motion of Orbit FB2 (left) and FB3 (right)

5.5 Rendezvous: manoeuvres 7 & 8

If, when inspected, the target is deemed acceptable to dock with, manoeuvre 7, a two-impulse rendezvous, is executed at apogee of FB1. The required Δv was computed using a two-impulse Δv in LVLH frame centred on the OAP in orbit C. The MATLAB algorithm *delta-V-rendezvous.m* written takes flight time as the input, so in practice this travel time between orbit FB1 and C can be shortened if need be at the cost of fuel, and the altered Δv required can be found using this MATLAB script. The function assumes the target orbit is an ideal geostationary orbit, using the Clohessy Wiltshere frame matrices for trajectory propagation to produce the Figures below (see Appendix F). A flight time of 12 hours was selected to minimise fuel usage, but also using as little time as possible between the inspection and docking to minimise risk. For manoeuvres 7 and 8, the Δv's are calculated using $\delta \mathbf{v}_0$ in LVLH and the boundary condition $\delta \mathbf{v}_f = 0$ in LVLH, so using the method in Appendix F, each burn for M7 and M8 in LVLH vector form is found:

$$\Delta \mathbf{v}_{0)7} = [6.742, 0.219, 0] \text{ m/s} \quad \Delta \mathbf{v}_{f)7} = [6.743, 0.0148, 0] \text{ m/s} \tag{7}$$

$$\Delta \mathbf{v}_{0)8} = [-6.605, 0.191, 0] \text{ m/s} \quad \Delta \mathbf{v}_{f)8} = [-6.605, 0.0435, 0] \text{ m/s} \tag{8}$$

Result (7) and (8)'s magnitudes of Δv are tabulated in 14. This produces the motion of the RSV relative to the OAP below (M7), and relative to the target (M8), over a flight time of 12 hours:

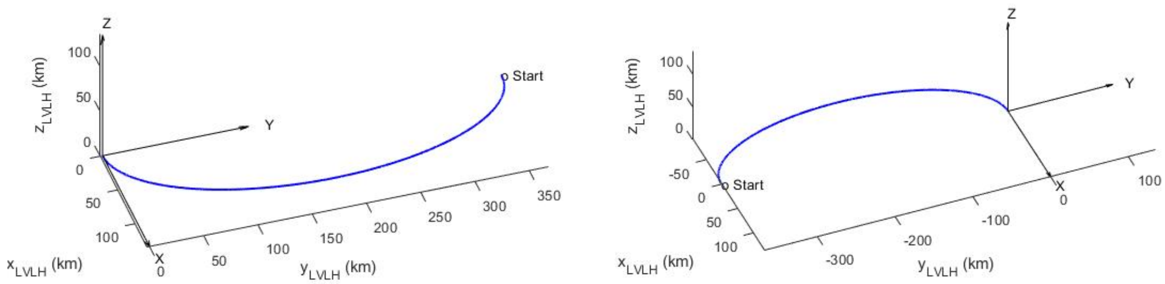


Figure 17: 12hr rendezvous of RSV and OAP (manoeuvre 7, left), and RSV and Target (manoeuvre 8, right)

Note that manoeuvre 8 is executed after a 6 hour docking time at the OAP, which moves the RSV back to the target to dock the OAP. As mentioned before, the specifications of each manoeuvre is listed in Table 14. After completion of 17 of these cycles, the spacecraft must use a Hohmann transfer to a graveyard orbit 350 km above geostationary. This manoeuvre would need an additional $\Delta v = 12.386$ m/s, and is accounted for in the fuel mass budget.

5.6 Service Manoeuvre strategy

Table 14: Service manoeuvre strategy (excluding phase & dwell time)

Manoeuvre 2

Burn	Type	Location	Δv (m/s)	T+
2.1	Hohmann	Orbit B perigee	0.000912	+0d 0.00 hrs
2.1	Hohmann	Orbit FB1 apogee	-0.000912	+0 d 11.97 hrs

Manoeuvre 3

Burn	Type	Location	Δv (m/s)	T+
3.1	Phase	Orbit FB1 apogee	0.203311	+11.97 hrs
3.1	Phase	Orbit FB1 apogee	-0.203311	+7 d 11.54 hrs

Manoeuvre 4

Burn	Type	Location	Δv (m/s)	T+
4.1	Plane change	Orbit FB1 apogee	0.007292	+8 d 11.54 hrs

Manoeuvre 5

Burn	Type	Location	Δv (m/s)	T+
5.1	Plane change	Orbit FB2 apogee	0.014584	+9 d 11.54 hrs

Manoeuvre 6

Burn	Type	Location	Δv (m/s)	T+
6.1	Plane change	Orbit FB3 apogee	0.007292	+10 d 11.54 hrs

Manoeuvre 7

Burn	Type	Location	Δv (m/s)	T+
7.1	Two-impulse rendezvous	Orbit FB1 apogee	6.7460	+11 d 11.54 hrs
7.2	Two-impulse rendezvous	OAP	6.7429	+11 d 23.54 hrs

Manoeuvre 8

Burn	Type	Location	Δv (m/s)	T+
8.1	Two-impulse rendezvous	OAP	6.6081	+12 d 5.54 hrs
8.2	Two-impulse rendezvous	Target	6.6050	+12 d 17.54 hrs

NET TOTAL, 1 cycle	13.791662	12 d 23.54 hrs
NET TOTAL, 17 cycles	234.458254	220 d 16.21 hrs

5.7 Docking

The OAP will dock with the client satellite via the satellite's liquid apogee engine (LAE). The RSV will position the OAP approximately 0.3m behind the target satellite and extend a 325mm long probe into the satellite's LAE. The head of the probe will then expand so that it can grab the satellite by the combustion chamber. The probe can then retract and pull the OAP and the satellite together until the two are a single unit.

The docking procedure is as follows:

Table 15: Docking procedure

Step	Description	Time
1	Rendezvous with the OAP (Manoeuvre 7).	+12.00 hrs
2	Position the satellite one meter directly above the OAP, and orthogonally such that the OAP's solar array is not shaded by the RSV's solar array.	+1 hrs
3	Using the robotic arm of the RSV, grab the OAP with the engine of the OAP pointing toward the RSV.	+5 hrs
4	Rendezvous with the client satellite (Manoeuvre 8).	+12.00 hrs
5	Position the OAP 0.3m from the satellite, using the cameras and TriDAR so that the docking probe is in line with the satellite's LAE.	+5.67 hrs
6	Extend the outer rod to insert the docking probe into the combustion chamber of the LAE.	+3.00 min
7	Retract the inner rod to expand the head of the docking probe.	+3.00 min
8	Retract the outer rod to pull the RSV+OAP and the satellite together.	+3.00 min
9	Once the OAP and the client satellite are flush, release the OAP from the arm of the RSV.	+11.00 min

6 Collision Avoidance and Fault Protection Routines

The RSV and OAP systems are designed to have appropriate fault protection routines to ensure the system stays functional. These are outlined in Table 16 below.

In the case of severe failures - for example if no directions from ground are received over a 24 hour period - the RSV and OAP are also equipped with a safe mode.

If safe mode is triggered, all systems will switch to their redundant units, all non-essential systems and payload will be switched off and the spacecraft will rotate itself to have its solar panels facing the sun at all times. The spacecraft will remain in safe mode until the source of failure has been resolved and it is rebooted by command.

Table 16: Procedures for Fault Protection and Correction

Data	Source	Expected Value	Contingency
Ongoing Checks			
Position and attitude	Sun tracker, star tracker, gyroscope, ranging	All agree	Re-calibrate position with reaction wheels and thrusters (mono and ion)
Space Debris	Tridar, command	No expected debris within 50km [5]	Plan debris avoidance manoeuvre
Temperature	Temperature sensors	10-40 °C	Too cold: more power to thermal control, engage redundant components if error due to defect. Too hot: reduce clock rate of device.
Voltage	Voltage sensors	Rated for component	Adjust voltage. If 0 and no signal from component, engage redundancy measure. Capacitors are used for surge protection.
Current	Current sensors	Below max component rating	Reduce current/power to component, enter safe mode to avoid damage. Diodes used to direct current flow.
Link	Command/telemetry	Clear signal	Point ground station antenna, change band to avoid interference, engage redundancy measures
Battery	State of Charge (SOC) sensor	Charge $\geq$ 40 %	Load shed, if SOC $\leq$ 50% turn off thermal, if SOC $\leq$ 30% turn off payload
CPU	Watchdog Timer	No timeout	CPU and hardware into safe mode, CPU reboot
Close Approach			
Rain Rate	Command	$\leq$ 22 mm/h	Delay flyby/approach due to broken link
Video	Command feedback	Clear, usable 4k video feed	Adjust camera zoom, aperture or focus. Direct signal with antenna pointer. Engage redundant camera or downlink systems if necessary.
Client/OAP Condition	Command	No visible defects, communicating properly	Terminate docking if damage deemed significant
Docking			
Strain	Strain gauges	Very low	Terminate docking
Alignment	Camera, tridar	Aligned properly $\pm$ 1cm	Attempt to adjust position, terminate launch and re-attempt
Probe position	Angular position sensors	Fully extended when docked	Retract probe, inspect LAE for damage, adjust OAP position

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Appendix A: CDH Components

Description	Component	(b x w x h)mm	Mass (g)	Peak Power (W)	Operating Temperature C°	Price (USD)
CPU	RAD750	130x130x0	549	10	-30-85	200000
Data Bus	MIL-STD-1553b	-	17	1.1	-30-85	40
Seial Interface	R232	-	-	-	-	-

Appendix B: ADCS Components

Description	Component	(b x w x h)mm	Mass (g)	Peak Power (W)	Operating Temperature C°	Price (USD)
Star Tracker	Hydra CP	146.5x146.5x283	1400	5	-20-60	2000
Sun Tracker	Hydra CP	34.9x34.9x10.4	76.5	0	-40-93	-
Gyroscope	Arietis	-	2700	8.5	-40-70	-
Reaction Wheels	RSI 45 Reaction Wheels	-	9200	40	-	528000
Camera Linear Actuator	Model-N	247.5x267x0	2625	60	-50-80	-

Appendix C: Determination of structural and propellant mass

In order to calculate the amount of propellant needed for both the RSV and the OAP, it is necessary to know their respective masses; however, these are both dependent on the structural mass, which is in turn dependent on the volume of fuel that is needed for the mission. Evidently, this is not able to be solved directly.

There are a few ways to tackle this issue. A simple solution would be to overestimate the structural mass needed to house the fuel and simply go forward using this value. The approach used in this report aims aims to use kinematic constraints in order to reduce the complexity of the problem to a simple converging calculation, which can be solved numerically to a desired tolerance.

The algorithm follows the following steps:

1. First, constants are defined and an estimate of the structural mass is made.
2. The total dry mass of the structure to be moved, m_f is calculated. For the RSV, this is simply the mass of the components and the mass of the structure. For the OAP, it also includes the mass of the attached satellite.
3. The mass of propellant used to move the structure over a given Δv, m_{prop}, is calculated using a rearranged form of the Tsiolkovsky rocket equation:

   $$m_{\text{prop}} = m_f \Big(e^{\frac{\Delta v}{I_{sp}\, g_0}} - 1 \Big)$$

   An additional 10% is added on to account for inaccuracies in the estimation.
4. If the calculated propellant mass is the same as the previously calculated propellant mass to a required degree of precision, the required values have been found. If not, the previous propellant mass is set as the current propellant mass.
5. The volume of propellant required is calculated from the mass and density.
6. The structural volume for housing the propellant is set as the volume of the propellant plus an additional 10% to account for thickness and other components.

7. A single dimensional variable for the structure is calculated from the volume from predefined dimensional ratios.

8. The surface area of the entire structure (both propellant and component housing) is calculated from the determined constant.

9. The structural mass is calculated from a predefined relationship between surface area and mass.

10. Steps 2 to 10 are repeated until Step 4 is satisfied. Then, the relevant values for propellant mass, structural mass, and the dimensional variable is output to the console.

First, the mass of the components (see Section 3.9 for both the RSV and OAP must be calculated). This gives the following values:

$$m_{\text{RSV components}} = 1042.4 \text{ kg} \qquad m_{\text{OAP components}} = 613.4 \text{ kg}$$

Then, the total Δv for each spacecraft must be determined. While the OAP uses only ion propulsion, making the calculation easy, the RSV uses both LAE and ion propulsion, meaning that the Δv must be split into two values. Using the values seen in Section **??** and making note of the specifications in Section 3.6 provides the following values:

$$\Delta v_{\text{RSV ion}} = 289.36\text{m/s} \qquad \Delta v_{\text{RSV LAE}} = 2693.48\text{m/s}$$
$$\Delta v_{\text{OAP}} = 338\text{m/s}$$

As seen in Section 3.6 the following values can be used for ISP and propellant density:

$$I_{\text{sp ion}} = 3000 \text{ s} \qquad I_{\text{sp LAE}} = 325 \text{ s}$$
$$\rho_{\text{ion fuel}} = 1500\text{kg/m}^3 \qquad \rho_{\text{LAE fuel}} = 1370\text{kg/m}^3$$

Finally, the dimensions for the RSV and OAP can be defined. The RSV cross sectional area was defined to be $5 \times 6.5 \text{ m}^2$, as seen in Section 2 to allow space for components. This means that for a length L, the total volume of propellant housing is given by

$$V = 5 \times 6.5 \times L = 32.5L \text{ m}^3$$

Now, a length of 4 m is allowed for component housing, giving a total surface area of

$$SA = 2 \times (5 \times 6.5) + 2 \times (5 \times (L + 4)) + 2 \times (6.5 \times (L + 4)) = 157 + 23L \text{ m}^2$$

The OAP dimensions are less constrained. Hence, a cross section area of $(L + 2) \times (L + 2)$ was selected. For a length L, the volume of propellant housing is given by

$$V = (L + 2)^2 \times L = L^3 + 4L^2 + 4L \text{ m}^3$$

Now, a length of 3 m is allowed for component housing, giving a total surface area of

$$SA = 2 \times (L + 2)^2 + 4 \times ((L + 2) \times (L + 3)) = 6L^2 + 28L + 32 \text{ m}^2$$

Then, as defined in the design specifications, every square metre of surface area is equivalent to 10 kg of mass. Using all of the above values and constraints produces the structural and propellant mass values seen in Section 3.9 and the dimensions seen in 2.

Appendix D: Ion Thruster Configuration

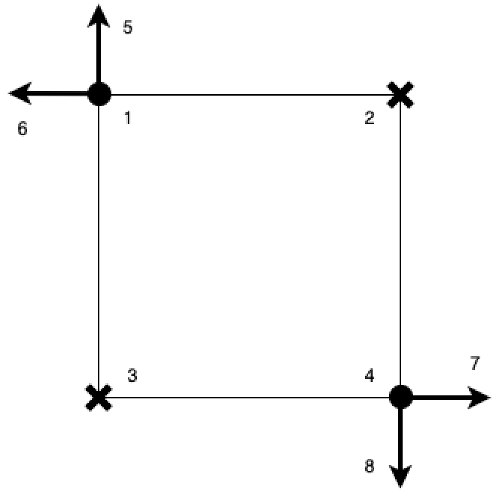
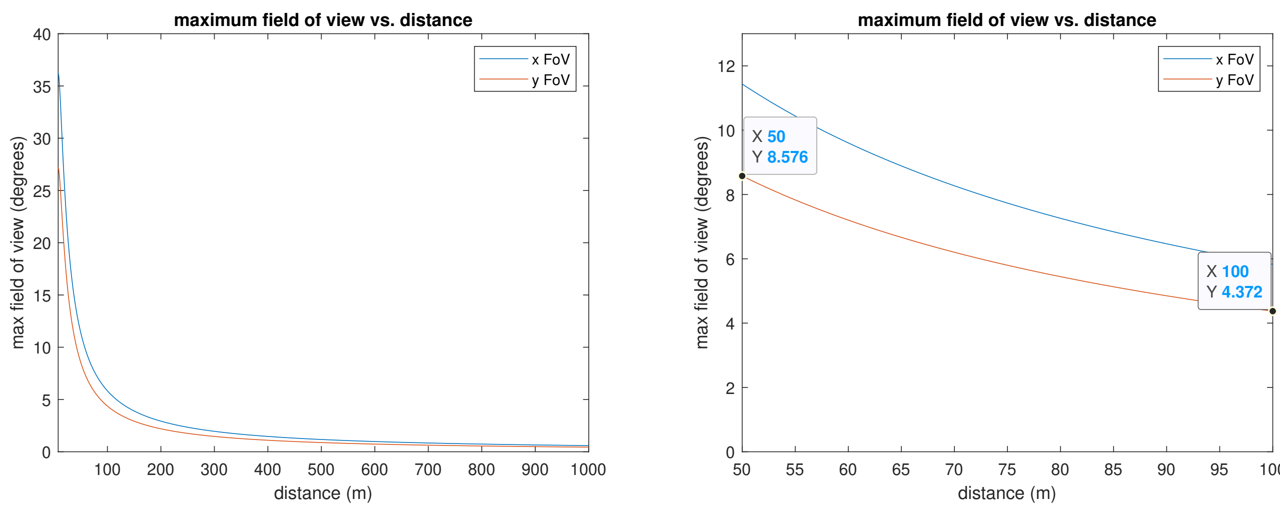


Figure 18: Ion thruster configuration for RSV and OAP providing six degrees of freedom

Appendix E: Field of View vs Distance from Satellite



(a) Required FoV 0-1000m **(b)** Right ascension normalised by Earth's orientation to sun

Figure 19: Required FoV 50-100m

Appendix F: Manoeuvres: Useful formulae & methodology

For Δv's found in GCEF frame:

Plane change on common apse:

$$\Delta v = \sqrt{{v_1}^2 + {v_2}^2 - 2 v_1 v_2 \cos(\Delta i)} \tag{9}$$

Geostationary phasing:

$$T_{phase} = \frac{2\pi n + \Delta\Lambda}{n\omega_{geo}} \tag{10}$$

LVLH and GCEF rotational operators:
Defining the reference frame at the Target:

$$\hat{i} = \frac{\mathbf{r}_A}{r_A} \quad \hat{k} = \frac{\mathbf{h}_A}{h_A} \quad j = \hat{k} \times \hat{i} \tag{11}$$

GCEF to LVLH DCM:

$$\mathbf{Q}_{Xx} = \begin{bmatrix} \hat{i} \\ \hat{j} \\ \hat{k} \end{bmatrix} \tag{12}$$

LVLH to GCEF DCM:

$$\mathbf{Q}_{xX} = [\mathbf{Q}_{Xx}]^{\intercal} \tag{13}$$

TWO-BODY PROBLEM in LVLH FRAME for ODE45 solving:

$$\delta\ddot{x} = \left(\frac{2\mu}{R^3} + \frac{h^2}{R^4}\right)\delta x - \frac{2(\mathbf{V}\cdot\mathbf{R})h}{R^4}\delta y + 2\frac{h}{R^2}\delta\dot{y} \tag{14}$$

$$\delta\ddot{y} = -\left(\frac{\mu}{R^3} - \frac{h^2}{R^4}\right)\delta y + \frac{2(\mathbf{V}\cdot\mathbf{R})h}{R^4}\delta y - 2\frac{h}{R^2}\delta\ddot{x} \tag{15}$$

$$\delta\ddot{z} = -\frac{\mu}{R^3}\delta z \tag{16}$$

CLOHESSY-WILTSHIRE equations used in calculating manoeuvres 7 & 8:
Clohessy-Wiltshire operators:

$$[\mathbf{\Phi}_{rr}(t)] = \begin{bmatrix} 4-3\cos nt & 0 & 0 \\ 6(\sin nt - nt) & 1 & 0 \\ 0 & 0 & \cos nt \end{bmatrix} \quad [\mathbf{\Phi}_{rv}(t)] = \frac{1}{n}\begin{bmatrix} \sin nt & 2(1-\cos nt) & 0 \\ 2(\cos nt - 1) & (4\sin nt - 3nt) & 0 \\ 0 & 0 & \sin nt \end{bmatrix}$$

$$[\mathbf{\Phi}_{vr}(t)] = \begin{bmatrix} 3n\sin nt & 0 & 0 \\ 6n(\cos nt - 1) & 0 & 0 \\ 0 & 0 & -n\sin nt \end{bmatrix} \quad [\mathbf{\Phi}_{vv}(t)] = \begin{bmatrix} \cos nt & 2\sin nt & 0 \\ -2\sin nt & 4\cos nt - 3 & 0 \\ 0 & 0 & \cos nt \end{bmatrix}$$

Clohessy-Wiltshire solution for relative motion to a target in circular orbit (for Δv rendezvous calculations and rendezvous motion plotting):

$$\{\delta\mathbf{r}(t)\} = [\mathbf{\Phi}_{rr}(t)]\{\delta\mathbf{r}_0\} + [\mathbf{\Phi}_{rv}(t)]\delta\{\mathbf{v}_0\} \tag{17}$$

$$\{\delta\mathbf{v}(t)\} = [\mathbf{\Phi}_{vr}(t)]\{\delta\mathbf{r}_0\} + [\mathbf{\Phi}_{vv}(t)]\delta\{\mathbf{v}_0\} \tag{18}$$

MATLAB functions
Run *Master-Calculator.m*:
`https://github.com/Oscar-Ansted/AERO2705-Assignment-3`