MACE 31521 CONCEPTUAL AEROSPACE SYSTEM DESIGN

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Group Number:			
Role	**Full Name**	**Responsibility**	**Pages**
Chief Engineer	Jakub Zemek	Mission and Overall System	
Project Manager	Samir Motwani	Project Management	
Mission Analysis Engineer	Samir Motwani	Mission Analysis	
Mechanical Engineer	Bartek Haško	Mechanical Subsystem	
Electrical Engineer	Martim Baptista	Electrical Subsystem	
Payload Engineer	Liqi Zhang	Payload	
Operations Engineer	Sam Fok	Operations	

Table 1: Individual Tasks

Table of contents

List of figures

List of tables

Nomenclature

AI	Artificial intelligence
ADCS	Attitude determination and control
COTS	Commercial of the shelf
GEO	Geostationary orbit
Isp	Specific impulse
LEO	Low Earth orbit
T	Thrust

1 Final concept summary

1.1 Introduction

The Czechmate is a group of students acting as a start-up company that aims to produce a feasible conceptual design for a Space Debris Removal (SDR) mission. This aerospace system must provide the product and/or services to enter and utilise the business opportunity in this emerging market sector. [1] This report investigates the user needs and derives system requirements. Based on these requirements, every subsystem engineer performs a trade-off analysis to find the most suitable solution. Additionally, a feasibility study is conducted for every subsystem and the overall design to ensure that not only does the system design comply with requirements but also that it is realistic and financially viable.
Since the Czechmate start-up aims to enter the market with its design, it is first necessary to investigate the need which creates demand from the customers.
The need comes from a problem which is high and growing amount of space debris around the Earth. Figure 1 shows that the number of space objects has been increasing since the start of the space era and, according to Figure 2, this trend is likely to remain in the future, especially due to the advent of so-called mega-constellations consisting of thousands of satellites.

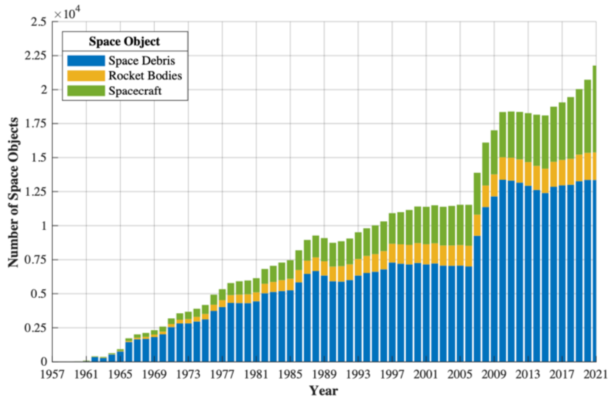


Figure 1: Number of space objects vs time [1]

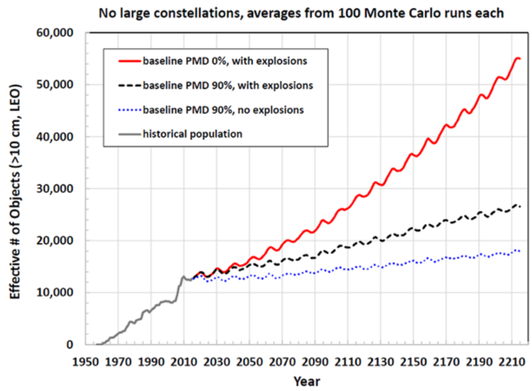


Figure 2: Future population of space objects >10 cm

The size of space objects can range from rocket paint fragments that are millimetres large to dead satellites such as Envisat that has dimensions of several meters and weighs more than 8 tons. [2] The debris orbits the Earth at very high velocities and thus a collision with a satellite has fatal consequences. Moreover, one hit satellite creates a cloud of dangerous debris that can lead to more collisions. NASA scientist, Donald J Kessler, proposed in 1978 that a chain reaction of debris exploding in space could result in impenetrable space debris concentration around the Earth that would not allow any spacecraft to safely fly to space. More detailed categorisation of space debris, as well as further assessment of its dangerous nature, can be found in the Appendix section "Problem definition".

The problem of space debris creates a need for a solution which the Czechmate start-up offers in a form of a space mission and a satellite. The mission plan is to launch the Czechmate satellite using a commercial launch vehicle, rendezvous with a dead satellite, safely deorbit the dead satellite and replace it with a new one and then deorbit itself.
The target customers are companies with the most active satellites since they have the highest chance of losing a satellite due to collision with space debris. The customers are hence predominantly commercial mega constellation providers such as SpaceX (Starlink), Amazon (Project Kuiper), Samsung or Boeing. As of 2021, Starlink operated the most satellites and therefore the mission is designed to service Starlink constellation satellites, however other players are likely to operate at similar altitudes and have comparable satellite specifications.
The conceptual design requirements are thus not only dictated by the nature of the problem, for example, a need for a payload that removes the dead satellite, but also by user requirements. The following user requirements were identified and are discussed in the Chief engineer's report.

- The total price of the system, including launch cost, must be competitive
- The system shall be reliable to successfully complete the mission
- The system shall replace the old satellite with a new satellite
- The Czechmate start-up shall offer a product that includes satellite as well as service that takes care of operation of the satellite

1.2 Final concept description

For the final design of the project, 8 stages are included, they are Launching, Separation, Launch and Early Operation, Rendezvous, Alignment, Capturing then finally Self-deorbit.
In this mission specifically, Starlink satellites in 550 km orbit at 53° inclination are chosen to be representation of the targets, removing debris at 1130 km could be the alternative mission, the 550 km orbit is the one being focused on. Since the "Czechmate" can be dropped at 600 km, 53° by Falcon 9 rocket, a change in inclination would not be necessary, but an inclination change is needed, the propulsion system will adhere to it. To transfer from 600 km orbit to 550 km orbit, a chemical fuel rocket motor, Waxwing, is chosen to be the propulsion system.
The main structure has been designed to withstand 19.8 g vertical and 9 g lateral load. The materials chosen are 6061-T6 aluminium and aluminium 5056 honeycomb. A passive thermal control system is used to keep the whole spacecraft between -20°C and 40°C. The deployment needs are met with a deployable composite boom, a hinge and a motorized mechanism. The thermal control subsystem can be adjusted to work at other altitudes by changing coatings.
For the satellite to process data onboard and communicate with the ground station, the total power required is 250W, the total power available from Solar arrays and the battery is 683 W. S-Band was the frequency chosen for communication purposes. The antenna was chosen based on this. The reaction wheel would be a relative simple and efficient method for attitude control. The Raspberry Pi Compute Module 4 computers are applied for the On-board Data Handling system using majority rule to increase reliability.
After reaching a position near the debris, the "Czechmate" satellite will turn on sensors (Radar and cameras) to target it. Once they are aligned within a certain range of direction, a net will be launched by a launching system to capture the debris. Using the momentum of the net, debris would be pushed to have a change in velocity, then both will deorbit and burn.
At the end of the mission, a new Starlink satellite carried by the "Czechmate" satellite will be released, then the later one would deploy a 10 m^2 solar sail to deorbit itself, the deorbit period would be 10 years approximately. Using propulsion system to deorbit would be a backup option in case the solar sail fails to function.

Mass (kg)	950(wet), 838(dry)
Volume (m^3)	5.85, 10(with propulsion and a new satellite)
Total Cost (£)	～40 million

Table 1: The "Czechmate" satellite mission

1.3 Design feasibility

Changing the weight of the system would mean a higher demand in terms of impulse and thrust however, the mission analysis engineer has assessed this issue by using a thruster that is easily capable of meeting the requirements for a more demanding mission and will therefore not affect the propulsion system or the chances of a successful mission.
The mechanical system has been integrated in accordance with the Falcon 9 Users Guide. The dimensions chosen for the adapter are standardized and the reliability of the interface is very high. The whole structure has been designed with a factor of safety equal to 1.8. This means that the structure should be able to withstand loads 1.8 times higher than the anticipated ones. Furthermore, the system structure should withstand loads associated with a small or even substantial change in the overall mass of the system. The change of mass of the system will only influence the speed with which the overall temperature changes, however, it should not influence the thermal equilibrium temperature. The system operates between 18°C and 20.5°C throughout the orbit. This means that the whole system's temperature could go up 19.5°C and all the devices would be in the operating range. Small changes in the heat input will not influence the thermal design that much and the whole system will remain operational. However large changes in altitude will have a considerable influence on the thermal equilibriums. Therefore, different coatings should be selected, if the altitude changes substantially. However, small changes in altitude will not change the overall spacecraft temperature greatly.
If the payload mass were to increase, the effect on the electrical system would be minimal. Save for changes in the power required to power the payload, the system should stay roughly the same. Should the target altitude increase, the size of both the solar arrays and the battery would decrease, as the time in daylight would increase, allowing more power to be generated, and the percentage of time in eclipse would decrease, meaning a smaller battery would be required. Should the target altitude decrease, the opposite would happen, and larger solar arrays and batteries would be needed.
For the debris targeting and capturing system, it has a relative large stability. If the mass of the net increases by less than 5 kg, the effect on the payload system is amlost negaligible Also, the net system is able to remove debris from 10 kg up to 300 kg, which is almost suitable for all medium large debris removal missions in LEO. The system also works at higher altitudes but the mass of debris that could be removed shall decrease.

1.4 User Needs

The user requirements listed in the introduction assumed that the customer is already selected, however, it is important to support the decision to target Starlink with evidence and compare it to other options.
Firstly, different orbits were compared in terms of the amount of space debris as shown in Figure 3. "The bar plot shows the sorted number of space objects into 1900 km width of altitude bands, and Low Earth Orbit (LEO) has the most space objects" [1] which yields the most potential customers and largest market.
Detailed market research including identification of similar missions and providers can be found in the section "*Market research*" in the Appendix. The market research suggests that from the various types of space debris in LEO, dead mega constellation satellites are the best category to focus on. This is because their size allows trajectory and position tracking using ground stations and due to the highest profitability. The aforementioned arguments shortlist potential customers to mega constellation operators from which SpaceX is targeted specifically as explained in the "Introduction". It is assumed that the SpaceX and Starlink company will remain on the market from 2025 to 2035 which is the period when the Czechmate satellite will be on the market.

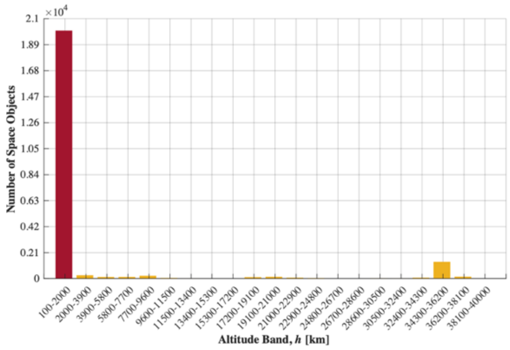


Figure 3: Distribution of space debris by altitude bands [1]

To be competitive on the market, the Czechmate start-up tailors its design to the following user requirements:

- The total price of the system, including launch cost, must be competitive.

Although the purpose of the Czechmate satellite serves a good cause and helps to solve a pressing global issue, it also needs to be financially feasible. To attract customers, the benefit of removing an old satellite must outweigh the price of the service and the satellite. Hence, the total price must be competitive and shall include the launch cost, the Czechmate satellite, the service that operates the satellite and margins.

- The system shall be reliable to successfully complete the mission

Mission design, quality assurance of components and communication with the customer must be of a high standard and maximize the probability of mission success. Companies with experience and successful missions are much more likely to be awarded contracts especially when it comes to governmental projects.

- The system shall replace the old satellite with a new satellite

As mentioned in the section "*Opportunities*" in the Appendix, there is no legal incentive for satellite operators to remove their dead satellites as of 2021. According to Blees et Al. in their research report called "Barriers to Entry Differences in barriers to entry for SMEs and large enterprises" (2003) [3], firms will enter the market only if the profit will give them a long-run competitive level.
The value of Czechmate could revolve around being a supplemental disposal method for an entire constellation. "It turns out that using an independent space tug for disposal maximizes the value of an entire satellite constellation." [4]
According to Business insider, 2,5 % of Starlink satellites might fail which means 10 500 satellites out of 42 000 satellite constellation. [5] It is estimated that the whole constellation will generate between 20 and 30 billion USD a year. [6] [7] It follows that each satellite generates around 500 000 USD a year. Removing a dead satellite and replacing it with a new one would therefore allow the company to generate 500 000 USD that would be lost otherwise and give a financial incentive to buy the Czechmate solution.

- The Czechmate start-up shall offer a product that includes satellite as well as service that takes care of operation of the satellite

It is important for the customer to buy a complete solution. Providing service that operates a spacecraft is common practice in the space industry.

2 System design reports

2.1 Chief engineer report

2.1.1 Introduction

The chief engineer is a multi-disciplinary technical leadership role overseeing the design of all spacecraft systems including mechanical, electrical and payload subsystems. [8] Furthermore, the chief engineer closely cooperates with the mission analyst to select the most feasible mission scenario

and with the project manager to distribute work packages to ensure all objectives are met effectively in a timely manner.
The individual work packages covered by the chief engineer in this project comprise of identifying and reviewing similar missions, identifying top-level system requirements, supporting the identification of suitable relevant technologies for subsystems, integrating subsystem models into complete system model, identifying and managing subsystem interface and coordinating overall system budgets for example mass, power, communication link, and budgets. Further chief engineer's responsibilities in this project included system-level feasibility analysis and system-level concept trade-offs.
The objective of the chief engineer's work is to evaluate the feasibility of different system designs based on trade-off analysis performed by subsystem engineers and integrate them into the overall system.

2.1.2 Requirements and System models

The work package requirements developed by the chief engineer are the user requirements and system requirements. The chief engineer also formulated the problem and a need for the mission which were outlined in the introduction and is further discussed in the "*Problem definition*" section in the Appendix.
System requirements are based on user requirements sheet and translate user needs into conceptual system level.

- The system shall be deployed by a reusable rocket

The launch service shall be cheap however at the same time, there shall be enough launches per year so that the Czechmate satellite could react quickly to remove a malfunctioning satellite. This is satisfied only by reusable rocket Falcon 9 v1.2.

- System components shall be preferably COTS parts

Many space missions use custom-designed components that perfectly fit the requirements, however, are very expensive. The Czechmate mission shall take advantage of COTS parts that are often standardised, have flight experience and are affordable. This also saves a lot of man-hours in the design process.

- The system shall be able to reach any satellite in a constellation after being released by a rocket

Since any satellite in the constellation may malfunction, the Czechmate service shall be able to reach any satellite and remove it. A disadvantage of this solution is that the spacecraft is too powerful for certain missions.

- The system shall be able to communicate with ground stations and receive and execute commands

The satellite is not designed to perform missions autonomously and therefore it shall be fitted with communication devices to send telemetry data and receive commands.

- The system shall use reliable space debris removal mechanism

Different capturing mechanisms are described in the section "*Ways of space debris removal*" in the Appendix. Reviewing capturing mechanisms of similar companies listed in the section "*Market analysis*" in the Appendix suggests that the most appropriate option is using a net. This is due to the relative simplicity of the mechanisms that do not require powerful electronics such for example laser or magnetic solution. Unlike claws, a net can be also used to capture rotating space debris. A net was tested as part of the RemoveDEBRIS mission with good results. [9]

- The system should have sufficient propellant and parameters to be able to carry a new satellite to the constellation

It was argued that the financial feasibility of the mission is conditioned on delivering a substitute satellite. The system requirements shall account for this and the Czechmate satellite shall be able to carry a new Starlink satellite and deploy it to substitute the dead satellite.
System requirements include requirement type, information about compliance and about who is responsible for each system requirement.

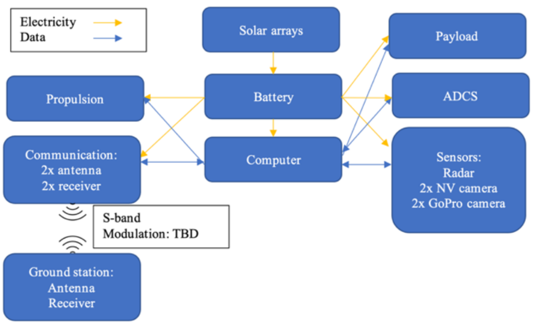


Figure 4: System diagram

Figure 4 depicts the system architecture of the Czechmate satellite and shows how are subsystems connected in terms of electricity and data stream. An important part of the satellite is a solar sail that is used for deorbiting. This system is critical, and therefore is not connected to any other subsystem. Requirements and system models include a number of assumptions and simplifications. Firstly, it is assumed that the Falcon 9 rocket will be available by the time of the Czechmate mission launch. Also, an important simplification in the system design is that it does not account for any losses due to resistance or battery deterioration. Furthermore, Czechmate assumes that the data produced by for example ADCS are readable by the computer. Last notable assumption is that the AI used for target recognition is fully functional.

2.1.3 Concept design options

On the conceptual level, three main scenarios were considered. The scenarios are numbered and described below with their advantages and drawbacks.

1. Removing a mega constellation satellite in LEO

This option was the initial plan which satisfied all user requirements except for replacing the old satellite with a new one. This would be a major drawback since, as it was argued in section 2.1.2, the added value to the mega constellation would be much lower. This is because the customer would have to pay for one launch for the removal mission and a second launch to replace the dead satellite. This design differs from the selected mission concept only in less powerful propulsion due to lower mass and in missing interface with the Starlink satellite.

- Advantages

This option has a much lower demand on propulsion system since the weight is reduced. Also, this type of mission could be used to remove any dead satellite and hence the start-up could focus on a broader spectrum of customers. The complexity of the system would be reduced due to the missing interface with the new satellite.

- Disadvantages

A large amount of evidence from multiple sources suggests that there is no reason for commercial customers to remove their malfunctioning satellites since space debris removal does not generate profit. Although ESA announced a policy that requires all new satellites to deorbit after their mission lifetime, there are no laws that would hold accountable owners of already existent space debris. Therefore, any mission concept that would be only removing space debris would be highly dependent on governmental and other financial incentives that do not exist as of 2021 and are unlikely to be introduced in the coming years.

2. Removing mega constellation satellite and replacing it with a new satellite

In this case, all user and system requirements are satisfied. It is also the most financially feasible option. For this mission concept, it is assumed that separation from the launch vehicle will occur at 600 km.

- Advantages

A mission concept that removes and substitutes a dead satellite is a novelty on the market which is likely to attract commercial customers as well as governments. Moreover, carrying a new satellite provides a complex solution to many customers who would otherwise not be interested in buying a space debris removal mission. As this mission targets space debris in LEO, it can take advantage of Earth's gravity and atmospheric drag when deorbiting the dead satellite and itself. Unlike for missions that operate in GEO orbit, the delay in communication in LEO is not significant which makes communication much simpler because relativistic effects do not need to be accounted for. Large propulsion means that the satellite does not have strict requirements on deployment altitude and thus can share launch vehicle with other missions. A smaller propulsion system would mean that the launch vehicle must deploy the satellite at the right altitude or even at the right time which implies that the price of the system would increase to cover the full price of the launch vehicle.

- Disadvantages

A major drawback of this concept is the need for a very powerful propulsion system since the satellite shall be able to remove any dead satellite in a mega constellation. Also, since a new satellite is carried, there is more at stake and an unsuccessful mission would lose two instead of only one satellite. This mission concept requires chemical propulsion which means that one mission can only remove one dead satellite. Having for example electric propulsion and solar panels could allow the satellite to stay in orbit longer and remove more space debris. Furthermore, "LEO systems operate in a much harsher environment compared to GEO including radiation and atmospheric drag that results in a much shorter lifetime (circa 5 years). " [10]

3. Removing GEO satellite

Geostationary orbit is highly sought-after and has a higher share of large satellites than LEO. Removing space debris from the GEO orbit implies different technical solutions for some subsystems. To give an example, the debris would not be pushed towards the Earth, rather it would be pushed towards the so-called graveyard orbit. Since this mission concept operates with a longer time scale than GEO orbit missions, different propulsion and removal systems could be introduced. In terms of propulsion, electric propulsion could be utilised and capturing system could use an energy-based solution such as magnets or lasers.

- Advantages

Since the GEO environment is not so harsh compared to LEO, satellites last longer, and one satellite could potentially remove multiple dead satellites. The satellite could travel to designated targets using electric propulsion which is relatively lightweight and is not dependent on the size of fuel tanks. Performing more tasks during one mission reduces the number of space launches per mission.

- Disadvantages

This mission concept requires a rocket specifically launched for this purpose which comes with a high cost. Despite its prominence, only a small amount of spacecraft is using this orbit as shown in Figure 3. Moreover, mostly governments operate satellites in GEO which brings considerable uncertainty to the business plan. Furthermore, due to the nature of space debris in GEO which is largely made of large satellites, the Czechmate system would need a powerful debris removal tool. This yields high energy requirements for an energy-based system such as lasers or magnets. Clamp technology is highly risky because it requires very precise targeting, cannot remove fast-spinning debris and can lead to cold-welding in space. Lastly, a net would need to be very heavy to deliver enough momentum to heavy satellites. Removing multiple dead satellites would imply having more nets and a reloading mechanism which further increases the complexity of the system. Lastly, deorbiting a satellite in GEO cannot take advantage of Earth's gravity.

Launch vehicle	VEGA	Soyuz	Falcon 9 v1.2
Payload to LEO	1500 kg to 700 km [11]	4400 kg from Baikonur	22800 kg [12]
Payload volume	Height: 7,88 m Diameter: 2,6 [11]	Height: 8,45 m Diameter: 3,7 m [13]	Height: 13,1 m [12] Diameter: 5,2 m
Reliability	90 % [11]	97 % [13]	98,5 %
Starts per year	3 – 4 [11]	10 - 15 [13]	11

Table 2: Launch vehicle trade-off

Table 2 shows the capabilities of different launch vehicles. The European VEGA rocket has a downside of too few starts per year which does not satisfy SR1. The Soyuz 2.1a is comparable in all parameters to Falcon 9 v1.2 however, it is an older design that might be retired in coming years and also has a much higher cost per launch that does not comply with SR1.

2.1.4 Final design option selection

The key design driver in the Czechmate conceptual design is feasibility. Although some mission concepts are feasible from the technical point of view, the financial balance is not positive which is the case for GEO orbit. As elaborated in the "*Market research*" in the Appendix, the most feasible design is such that removes dead mega constellation satellites. A mission that targets mega constellation space debris and substitutes it with a new satellite has the following key factors that were identified:

- Payload

This subsystem was constrained only by the maximum allowed size. Based on the payload design, which according to trade-off analysis in the Payload section of this report will use a net, all other subsystems were adjusted. For example, the battery is mainly powering the payload subsystem and therefore was designed based on payload requirements. A net is included in the conceptual design of the DeOrbit mission funded by ESA which proves the reliability of this system which thus satisfies SR 5. The payload has a targeting capability delivered by COTS cameras accounting for operation in sunlight as well as in an eclipse. Thus, both system requirements SR2 and SR8 are satisfied.

- Propulsion

Manoeuvring towards designated space debris in Earth's gravitational field that is present in LEO requires considerable ΔV and Isp capability. Once a propulsion system that meets the mission requirements is chosen, the wet mass of propulsion and mass of the whole system must be assessed to ensure that the overall system design is feasible, as depicted in Figure 5.

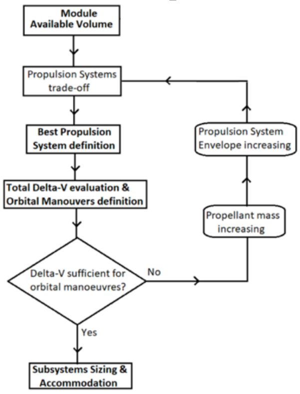


Figure 5: Propulsion system design

As per mission analysis calculations, it requires T of 569 N and Isp of 22000 s to reach the furthest satellite in the mega constellation. Parameters of the Czechmate propulsion system are T = 29400 N and Isp 2700 which demonstrates the feasibility of the design and satisfies SR3 and SR6.

- Energy budget

Solar panels are the main source of electrical energy in the system. The two main recipients of energy are the payload and the propulsion system. For this reason, solar panels were selected after both payload and propulsion were designed. As the satellite must operate through periods of solar illumination as well as during eclipse, a battery is designed to charge before the eclipse and provide energy in the eclipse. Since the total power required by components included in the conceptual design is 241 W, the battery can store 350 W and power supplied by the solar panels is 683 W the overall system design is clearly feasible and fully complies with SR7.

- Data budget

The main source of data is the payload subsystem, specifically the sensor part. All data is processed by an onboard COTS computer compliant with SR2. Since the targeting system uses AI to target the dead satellite the computer must provide sufficient computational power. After eliminating space-grade computers from the selection process due to their low performance it was decided to use multiple Arduino Raspberry-Pi processing units as explained in the Electrical engineer's report. The Arduinos process data from payload sensors and telemetry data and send it to Earth through the onboard antenna. The receiver in return receives commands from the ground station and the computer

interprets them to appropriate subsystems. All subsystems are integrated into the overall data stream architecture that satisfies SR4 and SR10.
An important part of data handling is the communication link. The antenna, as well as the receiver, must transfer a sufficient number of bits per second to prevent delays in communication. Moreover, modulation is used to minimize data loss and hence modulator, demodulator and clock must be present in the system. The exact modulation technique is to be decided in the next design stages as it is not in the scope of conceptual design. By comparing requirements on data to be transferred with properties of the Czechmate communication system design, for example, bandwidth, it can be shown that the system design is feasible. A possible vulnerability of the Czechmate design is its redundancy which increases overall complexity and requires error-proof programming that can quickly switch between input channels.
Finally, summing masses of all subsystems, including the Starlink satellite, gives 950 kg. Although this mass will increase due to for example wiring, it proves that the Czechmate satellite can be lifted by the selected launch vehicle that can carry up to 16800 kg to LEO.

2.1.5 Conclusions

A key finding in this report comes from the market research that shows that removing space debris is not a prosperous business. This is due to a lack of incentives to remove dead satellites and still a rather low number of private companies in the space industry. Assuming positive cash flow from governmental bodies proves to be highly uncertain and unlikely.
The selected mission concept design provides a cheap, affordable, and complex solution for the space debris problem. It was proven mathematically that all subsystems work together, and that the overall system architecture is feasible. Moreover, some parts are COTS which significantly reduces the cost of the system. To conclude, benchmarking with similar companies and critical review demonstrates the feasibility of the Czechmate system.

2.2 Payload Engineer report

2.2.1 Introduction

After the Czechmate satellite being sent to a position near the target debris (a few kilometres away), the “capturing system” starts to operate. According to the requirements, the job for payload engineering is to recognise and remove the target debris, which are Starlink satellites as well as other unfunctional satellites in the LEO.
The 3 main parts in this section are 1. Analysing the requirements from previous stages. 2. Comparing and selecting suitable components. 3. Combining components and determine the conceptual design.

2.2.2 Requirements and System models

Starlink satellites are chosen to be the targets in this mission. Each of the Starlink satellite has a mass of 260kg [14]. The initial plan is to use some sensors to locate the target debris, launch a net to capture it, then use the momentum of the net to decent it into a lower orbit. The target, together with the net, slow down and burn up in the atmosphere at the end.
There would be no propulsion system on the net. In this way, the net has to carry enough momentum. Momentum = m*v, the velocity could be a parameter to be increased to achieve the goal. However, if the velocity is too high, it might cut through the target and create smaller debris (since the material of the net has to be strong to withstand the impact). So, the velocity could be brought to a certain limit but not further. The other way is to increase the mass of the net, it seems to be a simple solution but it has a direct effect on the mass of the launching system and the propellant to be carried. After all, the point is to find the balance between the mass as well as the launch velocity of the net to meet the requirement.
The target debris, Starlink satellites, are typically deployed in the 550 km orbit. It is assumed that the front area of the target is 2 m^2, the mass is 260kg, m/A=130. The time for the target debris to decay is required to be 1 year or less (system requirement), the initial decay altitude could be calculated

according to Equation 1, 2 and 3 from Satellite Orbital Decay Calculations by the Australian Space Weather Agency [15]. To simply the estimation process, a Satellite Orbit Decay calculator is used (based on the principle of the Orbital Decay Calculations) [16], or it could be estimated from Figure 1 as well, they all approximately give the same value [17] [18].

$$D = (1/2)\rho v^2 A\ Cd \quad (1)$$

$$P^2 G\ Me = 4\pi^2 a^3 \quad (2)$$

$$dP/dt = -3\pi a\rho\ (Ae/m) \quad (3)$$

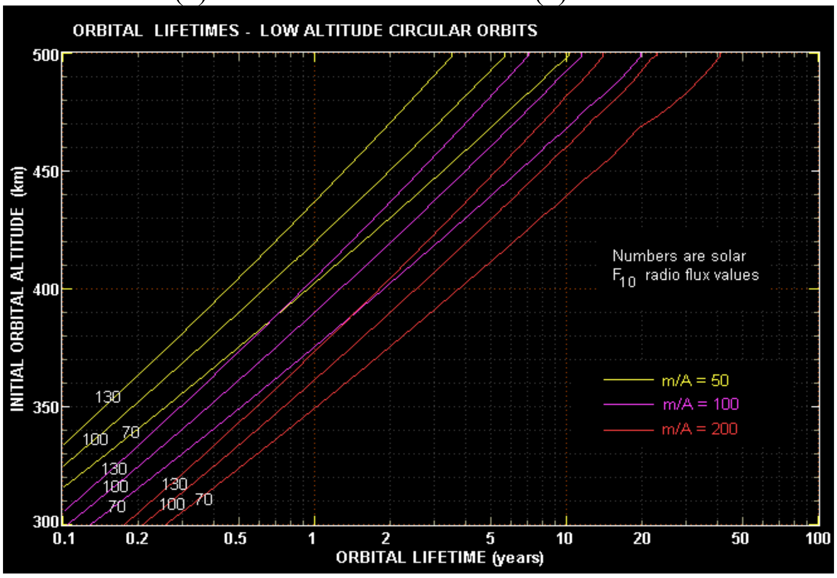


Figure 6: The graph of the orbital lifetime related to the initial orbit altitude of different m/A and solar radio flux.

The initial decay orbit is calculated to be at 370 km, which means the target debris along with the net would have to travel from the 550 km orbit to the 370 km orbit. The change of velocity δv for this would then be calculated using Equation 4, where G is the gravitational constant, M is the mass of Earth, r is the distance between the satellite and the centre of the Earth, a and b are the long semi axis and the short semi axis of the elliptical orbit. For apogee, r is equal to a.

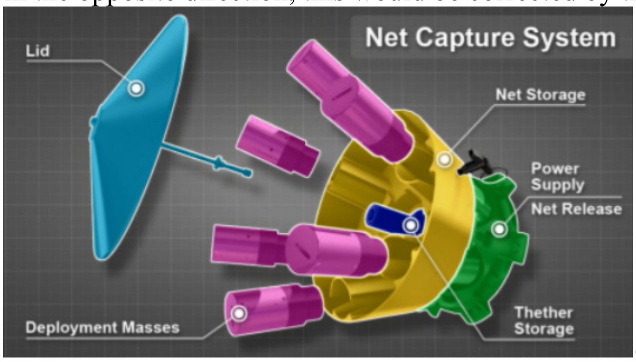
$$v^2 = 2GM\left(\frac{1}{r} - \frac{1}{a+b}\right) \quad (4)$$

The initial velocity in the 550 km orbit is 7589 m/s, the velocity at the apogee of the 370-550 km elliptical orbit is 7539 m/s, the δv would be 50m/s. However, considering that the net is not able to generate the second deceleration at the perigee to enter the 330 km circular orbit, the decay time in the 370-550 km elliptical orbit would be significantly larger than a year. So, it is decided that the δv would be set to 60 m/s to compensate (turns into the 320-550 km elliptical orbit). There could be the case that the target debris travels from 550 km to 600 km or even up to 1100 km. Once it goes over 700 km, the satellite would be almost impossible to decay. The system can remove the debris at 1100 km, but the net would need to be launch at a specific range of angle (perpendicular to the velocity). Since there are few demands in the 1100 km orbit, it is decided it might not worth to come up with a plan B though it is feasible.

2.2.3 Concept design options

At the current stage, propulsion systems on the net or the “robot arm” system would not be considered since they significantly increase the complexity of the control system, and the increase in component usually leads to the increase in error rate, shorter MTBF (Mean Time Between Failure).
The target debris is expected to have a δv of 60m/s, which means the required momentum is 19200 kgm/s. Assuming the mass of the net is 80kg, the launch velocity shall reach 240 m/s. To achieve this, a set of piston system would be applied. It is suggested that it includes 6 pistons, 6 weights would be attached to the net in a hexagon shape, each of the piston corresponds to accelerate one weight,

similar to the ones published by ESA Space Debris Office (Figure 2) but use chemical fuel instead of springs [19], in order to provide a much higher launch velocity.
Theoretically, the chemical fuel and the oxidizer will be stored in the piston, waiting to be excited. Energy required is 2304 kJ, for methane or ethane it takes 0.044 kg, for hydrazine (N_2H_4) it takes 0.119 kg [20] [21]. Hydrazine might be a good option since it is liquid at room temperature, easier to be stored. 0.6 kg of Hydrazine would be needed assuming 20% of the chemical energy is converted into kinetic energy. This satisfies the payload requirement 3.
According to the calculation, the net should be shot at the target backwards (opposite to the velocity direction) in order to deorbit (Appendix Figure 4), in the following discussion, it is suggested that shooting it vertically towards the Earth would decrease the deorbit time, but the vertical shooting deorbit time could not be accurately calculated.
Due to the reaction momentum of launching, the "Czechmate" satellite tend to have a change in velocity in the opposite direction, this would be corrected by the chemical propulsion system.



(a)

(b)

Figure 7. (a) The ESA Net Capture System Concept, springs are involved.

(b) The Model of the Net Capture System, ejection springs unloaded.

About the mass distribution of the net, Figure 3 shows the brief design. Two materials are used, steel wire with density of 8050 kg/m^3 and some polymer with density of 1200 kg/m^3. The diameter of the net is 9 m, the total mass is 80kg. 6 weights are uniformly placed at 6 outer corners, each has a mass of 9.3 kg. The grey lines represent sections of steel wires, these steel wires have a diameter of 4 mm and formed by multiple thinner wires, shown in Figure 4 [22]. The central hexagon has a radius of 2 m, in the yellow region, based on the steel wires, it is covered by 2 mm-thick polymers.
Cold welding is something that should be paid attention to as it does not occur on Earth. When it comes to the space, the surface of metal could not react with oxygen to form a thin coating of oxidation, the metal atoms tend to share electrons and stick when they touch each other. This could be dangerous. However, cold welding in space, does not happen quite often. "Typically, all metals launched into space have a thin coating of oxidation on them due to contact with the Earth's atmosphere." The coating still exists during the mission in space usually [23]. Here, the application of polymer is to ensure the cold-welding does not occur and also act as a bumper when the net hits the target debris.

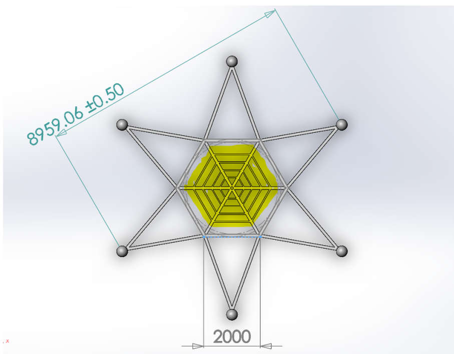


Figure 8: The CAD model of the net design. Figure 9: The steel wires.

The mass of the steel wires is 5.1 kg, the mass of the polymer is 19.2 kg. The mass of the polymer is estimated by calculating it as a thin sheet, the mass would be slightly less in reality.
The loading of the net during the impact mainly acts on the steel wires, by applying Equation 5, it could be calculated that the maximum axial force it could withstand is 3140 N assuming that the yield stress is 2.5e8 Pa for steel. This satisfies the payload requirement 1 and 4.

$$\sigma = \frac{F}{A} \quad (5)$$

To recognise the target debris, the detecting method as well as the equipment are essential. In Table 1, various parameters (such as Cost, mass, sample density, power required) are compared for Lidar, Radar and Camera.

	Advantages	**Disadvantages**
Lidar	(1)Short detecting time (data is collected quickly). (2)High sample density, high resolution.	(1)High cost. (2)Affected by high sun angles and reflections. (3)Power consuming. (4)Short detecting distance (<200m)
Radar	(1)Large detecting distance (>1000m) (2) Short detecting time. (3)Relatively low cost.	(1)Low resolution. (2)Unable to recognise colours. (3)Relatively large in size and mass. (4) There could be radio noise.
Camera	(1)Large detecting distance (>500m) (2)Low power requirement. (3)Relatively low cost.	(1)Long detecting time (data processing). (2)AI learning and a powerful processor are needed to recognise the target. (3)Light source (sunlight) is required.

Table 3: The advantages and disadvantages of different ways of measuring [11][12][13]

By dividing them into "Pros and Cons", the trade-off could be carried out according to the requirements. After the discussion, the combination of Cameras and a Radar would be the best option. Since the satellite is in eclipse during 40% of the mission period, a normal camera is less useful, a night vision camera would be added to compensate.
The major work (such as detecting the distance, locating) would be done by the radar, cameras are in charge of visualising and targeting the debris.
Initially, professional filming cameras such as RED HELIUM and ARRI ALEXA were on the list. Considering the high power required, high cost (over $30,000) and it would be used for a short period, 4 commercial cameras are considered (Table 2), the GoPro HERO10 Black, the Nikon D810, the Nikon D5 and the Rexing B1 (night vision camera).
The Mean Time Between Failure (MTBF) is considered, however, there is no any related information published by manufacturers, it might not be a good parameter to be compared at the moment.

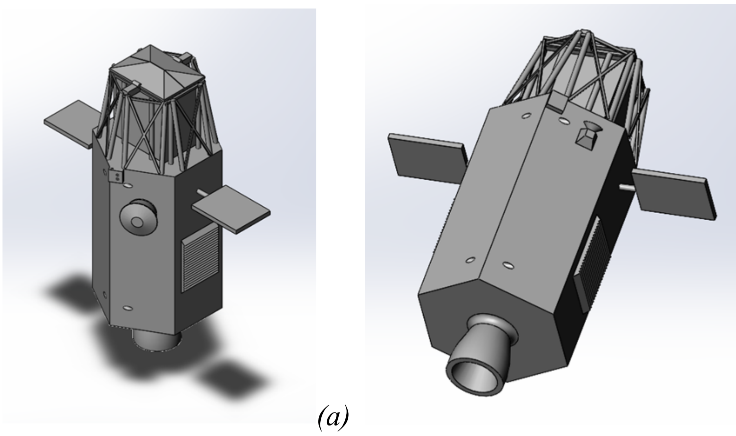
	HERO10 Black	**Nikon D810**	**Nikon D5**	**Rexing B1**
Mass (kg)	0.16	0.88	1.4	0.58
Power (W)	～2	～15	～15	～2
Resolution	3840 × 2160	7360 × 4912	5568 × 3712	1280 × 960
Dimensions (mm)	71×55×33.6	146×123×81.5	160×160×94	196×146×59
Price ($)	549	2900	6500	199

Table 4: The comparison of 3 cameras. [14][15][16][17]

By applying the Pugh method (decision-matrix method), the Nikon D5 is dominated by Nikon D810 so it could be removed. Although the Nikon D810 can provide high resolution, 8K video is not necessary considering the limited onboard processing ability. Afterall, the GoPro HERO10 Black and the Rexing B1 are chosen to be the main cameras to target the debris. This satisfies the payload requirement 2 and 8.

2.2.4 Final design option selection

The Net capturing system would be the final design option for this mission. The system includes an 80 kg hexagonal **net**, the **launching system** for the net, a **Synthetic Aperture Radar** (SAR) [24], 2 **GoPro** cameras, 2 **Rexing B1** night vision cameras (2 are applied as a backup, in case one fails), 2 Bipolar Stepper **Motors** and **gear boxes** [25]. After removing the target (unfunctional Starlink satellite), a **new Starlink satellite** (260 kg) would be deployed. The new satellite and the releasing mechanism (from Mechanical Engineer) are placed on the top of the "Czechmate", the propulsion system is placed at the bottom, the net launching system is placed on the side. A rough CAD model of the "Czechmate" satellite is developed (Figure 5).
By putting parameters of a GoPro camera into a ground sampling distance (**GSD**) calculator [26], the GSD of the satellite is 5800 m/px at 550 km altitude. However, neither mapping nor taking photos of the Earth surface is part of the mission, so this parameter would not be essential.
To ensure there is enough data processing power, 3 Raspberry Pi 4 Model B are chosen (with Electrical Engineering) to be the Command and Data Handling System (**CDH**) instead of Nano Mind A3200 [27] [28].



(a) *(b)*

Figure 10: (a) (b). The CAD model of the "Czechmate" satellite.

Power estimation

The power required would be approximately 120 W. The power distribution: the **On Board Data Handing system** (OBDH): 10 W [27], the **Radar**: 15 W [24], the **4 cameras**: 8 W (2 GoPro, 2 Rexing), the cargo door and the **releasing mechanism** (motors and gear boxes): 80 W [25] (the gear box is used to decrease rotation speed and increase the torque), the **light-up system** for the chemical fuel to launch the net: <10 W.

Mass estimation

The mass would be approximately 100 kg. The mass distribution: the **net and the launching system**: 90 kg, the **OBDH**: <1 kg [27], the **4 cameras**: 0.7 kg [29], the **Radar**: 1.4 kg [24], **2 motors** (and gear boxes): 2.8 kg (+4 kg) [25].
To obtain the **feasibility**, simulations of net systems capturing space debris done by P. M. Trivailo and H. Kojima would be a good reference [30]. According to the simulations, the net system is capable for capturing targets in various situations such as a rotating target, two close targets or a target in a shape of a bar.
In terms of the cost, the hardware of the targeting system (OBDH, Radar, cameras) would be about £ 300000 [24]. Although it seems a bit expensive, by applying a powerful radar, the requirement for cameras (AI, image processing) is lower, lighter processors are preferred, the deadweight decreases, a lot more budget could be saved on the propellant part for the whole mission (Mission Analysis).

2.2.5 Conclusions

After taking the position of Payload Engineering in the Conceptual Aerospace System Design, a deeper understanding is developed, about how the whole system works and the part that Payload Engineering engages. Although a large amount of research was done, it is more important to first figure out the requirements, have a big picture of the mission, then combine the elements from previous research and realise how the result satisfies the requirements overall.

2.3 Mechanical Engineer report

Before the lunch of the mission the whole system has to be integrated with the launcher. The interface between the system allows for the separation of the spacecraft at the chosen altitude. It also provides an umbilical connection that carries the necessary power or information. It is this work package's task to choose the most feasible separation system and integrate the spacecraft with the launcher. The launcher payload envelope is directly associated with the structure of the system as it provides necessary information for its design, such as dimensions or loads experienced during the launch. The structure of the system has to withstand all the loads during the launch and be able to fit in the fairing of the launch vehicle. The structure allows for the mounting of the necessary onboard equipment and heat transfer between components, depending on the chosen material. The thermal control of the system has to be precisely designed to allow for the equipment to function in the range of the operating temperatures. The Payload and electrical subsystems will require the necessary thermal control. Additionally, the mechanical work package has to design the deployment mechanisms based on other subsystems' requirements.

2.3.1 Requirements and Systems models

2.3.1.1 Launcher Integration

The separation system has to allow for spacecraft separation with preferably low shocks. It also has to be integrable with the selected launch vehicle and allow for the attached system to weigh around 950 kg, since this is the estimated mass of the system [MR1]. Mission analysis has deemed Falcon 9 1.2 as the most appropriate rocket for this mission. The Falcon 9 payload user's Guide [12] provides information about the necessary interface for the system and the Fairing dimensions.

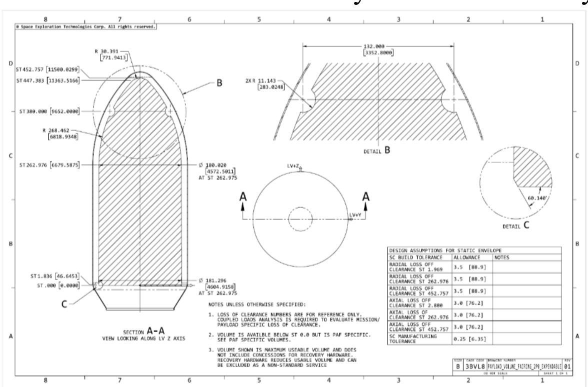


Figure 11: Payload static envelope [12]

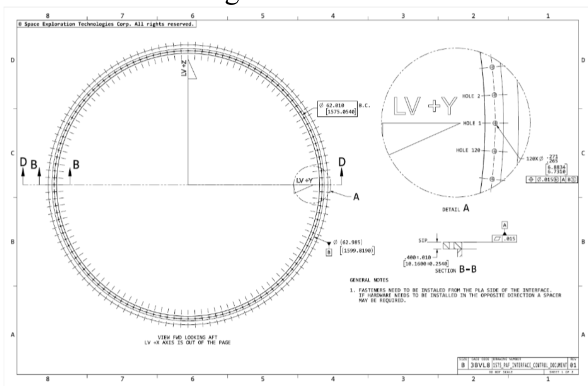


Figure 12: 1575 mm interface drawing [12]

For the adapter to be bolted to the falcon 9 interface it has to be 1575 mm in diameter. The overall width of the satellite cannot be larger than 5.2 m and the height has to be smaller than 13.2 m.

2.3.1.2 Structural requirements and system model

The main structure of the system has to withstand all the loads throughout the mission [MR2]. It also cannot be larger than the fairing of the Falcon 9 vehicle but it has to accommodate all the components that need to be fitted inside the system [MR6]. Therefore, the volume of the whole structure will have to encompass around 5.85 m3. This estimation is based on the volume estimation of other subsystems' component volumes. The material of the main structure should also be made from a material that allows good thermal conductivity. Falcon 9 user's guide [12] provides the typical loads that payloads under 1814 kg experience during the launch. Taking into account the mass estimation of the Spacecraft, these structural requirements are:

- The minimum Spacecraft resonant frequency – It is the minimum resonant frequency of the system. Anything below could resonate with the launch vehicle and result in a catastrophic event. The minimum axial and lateral frequencies for the Flacon 9 launch vehicle are 25 Hz and 10Hz respectively.
- Static loads – these are the loads that are a result of the thrust of the rocket and can act in the lateral and axial directions.
- Sine vibrations – These loads predict the maximum possible loads at the top of the payload attach fitting. The maximum axial and lateral loads during the launch are 0.9g and 0.6g respectively.
- Random loads – They represent the higher frequency loads and have less effect with the increasing size of the payload. As our satellite is fairly large we do not have to consider these loads during this stage of design.

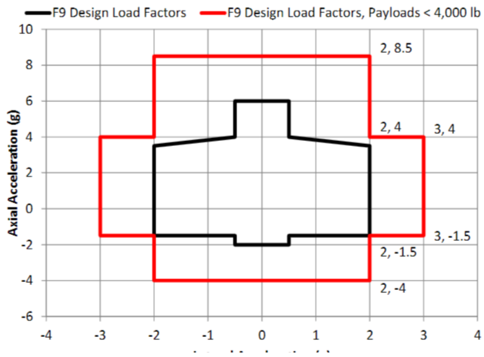


Figure 13: Experienced loads envelope during launch

The maximum loads experienced in the vertical and lateral direction have been estimated by the addition of the above requirements and their values are 11g and 5g respectively. The safety factor, that is chosen based on the fact that the analysis done in this document is only analytical and not experimental, has been estimated at 1.8. This means that the main structure of the spacecraft will have to withstand 19.8g in vertical and 9g in the lateral direction. The vertical stress has been calculated according to the buckling theory. The following equation that calculates the maximum possible force that a column will be able to withstand has been used:

$$F = \frac{\pi^2 EI}{KH^2} \qquad (1)$$

Where E is the Young's modulus of the chosen material, I is the moment of inertia of the cross-section of the column, K is the variable that is equal to 2 since the column is fixed at both points and H is the height of the column.

2.3.1.3 Thermal control requirements and system model

This subsystem has to provide necessary cooling or heating for the onboard components to allow for their operation [MR3]. All the subsystems have provided the necessary operating range for their components. Based on this information, the operating range of the whole satellite has been estimated to lie between -20°C and 40°C [MR5]. The thermal control system has to provide the spacecraft with a thermal equilibrium between these temperatures. From the information provided by the mission analysis, for nearly half of the time in orbit, the system will experience an eclipse. Therefore, the hot and cold temperatures have to be calculated. The hot temperature is assumed to be influenced only by, direct solar radiation, albedo radiation, Earth's shine and power dissipation from onboard devices [31]. However, the cold temperature is assumed to only include the Earth's shine and power dissipation. For both cases, the system is assumed to be a sphere. The equation for calculating the temperature is:

$$T_{eq} = \frac{Q_{in}}{\varepsilon\sigma A} \qquad (2)$$

where Q_{in} is the heat input from all the sources, ε is the overall emittance of the system, σ is the Boltzmann constant and A is the external surface area of the spacecraft [31].

2.3.1.4 Mechanism system model and requirements

This subsystem allows for the deployment of the necessary devices and other components [MR10]. The need for the deployment is specified by other subsystems' heads. It also allows for the system to occupy the lowest volume inside the fairing. The electrical engineer has requested for the solar arrays to be deployed, while the solar sail deployment is needed by the mission analysis engineer. As part of the mission, the system will carry another spacecraft based on the customer. For the purpose of an example, we have chosen the Starlink satellite. This satellite is to be deployed as a replacement for the removed satellite. Therefore, there has to be a system that will be able to hold it down or attach it to the satellite and then, deploy it at the required time [MR8]. There will also be two solar panels on opposite sides of the hexagonal structure to ensure the stability of the system and for the centre of gravity to align with the centre of the structure cross-section.

2.3.2 Concept design options

2.3.2.1 Launcher Integration

There are many different separation systems that are available on the market, however, only one can be selected based on the payload's size, mass, and loading thresholds. One of the possible separation systems that allow such mass is a clamp-band system. This system is commonly used in many interfaces, is amongst the most reliable on the market, and imposes low shocks on the structure. Another possible separation system that could be used is bolts. Its dimensions can also be adjusted to fit the falcon 9 interface and it allows such high mass vehicles to be mounted to it. However, its shock output on separation can be higher than that of a clamp band system. It is also specified in the Flacon 9 payload guide [12] that the mounting of a clamp band separation system is provided as a standard service and mounting of any other separation system would incur higher costs.

2.3.2.2 Structural concept trade-offs

The main structure has to accommodate a specified estimated volume. For this, we have to specify the cross-section of the structure. There are 3 different concepts that have been considered. The first one is a rectangular shape. It is the simplest cross-section to be manufactured but it also presents the highest stress at the joints between the wall panels out of the 3 concepts. Furthermore, it has the lowest efficiency in terms of the used volume as the space near the corners might not be properly utilized. The other section to be considered is a hexagon. This section has lower stress at the joints and distributes them better than a rectangular. The overall volume is utilized more efficiently and the manufacturing process is simple. The last cross-section to be considered is a circular one. This shape has the best load distribution and volume utilization. However, it is harder to manufacture it and the mounting of the components to the side walls might be more complicated.
Another aspect of the structure that needs to be considered is the material that it would be manufactured from. The material has to be strong enough to withstand all the loads while being as lightweight as possible. The following materials were considered for the main structure:

Alloy	Density(kg/m^3)	E (GPa)	γ
6061-T6 aluminium	2770	68.3	0.33
Beryllium	1850	241	0.08
Invar	8069	141	0.23
6AL-4V Titanium	4432	110.3	0.31

Table 5: Material Properties [32]

To form a whole structure that encompasses all the components that need to be inside, wall panels will have to be attached to the main structure. This structure should be as light-weight as possible, while also maintaining fairly high strength.

2.3.2.3 Thermal Control concept design

The most essential choice for the thermal control system is between the active and passive systems. A spacecraft would preferably be based on a passive control system since it would not require any power from the electrical subsystem. However, the problem for the Czechmate is that its time in eclipse is quite long and thus it can achieve extremely low temperatures if no heating is provided. Every heating device will require power therefore the thermal control system would be counted as an active subsystem. To avoid this, a low emittance coating has to be applied to the spacecraft. It is apparent that emittance value directly influences the eq. 2 and the higher it is the lower the thermal equilibrium temperature is. Therefore, to achieve an equilibrium temperature that will allow for every device to operate, a coating with very low emittance has to be used. Another property of the coating is the absorptance that will influence how much heat is received. Materials with higher absorptances will cause the spacecraft to achieve high temperatures in the sunlight, but will also heat up the spacecraft in the eclipse, which would allow for the use of higher emittances. A lower absorptance will act otherwise. It will provide smaller temperatures in the sunlight but also very low temperatures in the eclipse.

2.3.2.4 Mechanisms concept design

There are different deployment devices available. A hinge uses a coiled spring on a shaft, which provides a positive torque in a fully deployed state [32]. This torque must be larger than any in orbit loads to keep the solar panels deployed. The advantage of this mechanism is that it does not need electrical energy to be deploy. Deployable booms are a mechanism that uses electrical energy to slowly deployed a rolled material to which components can be attached. Another deployment device could use a motor to open some structure or a door.

2.3.3 Final Design selection

2.3.3.1 Launcher Integration

The separation system selected for this mission is a clamp band system due to the high reliability and low shocks associated with the separation. This system is feasible for such high-weight satellites and is used commonly in many launches. Additionally, SpaceX will provide the mounting of this subsystem as a standard service, which would not be the case if some other adapter was selected.

2.3.3.2 Structure

After careful consideration, the cross-section of a hexagonal shape has been selected. This structure provides a good load distribution and is fairly easy to produce. The attachment of components to the wall panels is also simple as the surface is straight and the volume will be used efficiently since a circular fuel tank will be fitted inside. The material selection has been based on equation [1]. The main structure columns have been modelled in Matlab and tested with various materials. Beryllium was the best performing material with a really high Young's modulus and low density. However, Beryllium has a low shear strain and therefore could fail due to the high lateral loads experienced during the launch. Therefore 6061-T6 aluminium has been selected as it provides good vertical strength and is also the best performing in shear stresses. Its mass to strength ratio is the best out of all materials when considering the vertical and shear stresses.
The most feasible choice of material for the side panels is a honeycomb structure. The strength efficiency compared to the weight that this structure carries is high making it the most feasible choice for the walls. The material used for them is a aluminium 5056 honeycomb.

2.3.3.3 Thermal Control

Different coatings with different absorptances and emittances have been tested out it a Matlab program that has been developed for the purposes of this conceptual design. The emittance has always been kept low while the absorptances were allowed to be in the range of low to high. It has been determined that low absorptances will be the most appropriate for this system. There have also been two coatings selected and the one with even lower absorptance will be placed where the spacecraft will be exposed to the solar radiation the most. This will decrease the temperatures while the spacecraft is in the sunlight. However, these temperatures are still too high for the onboard devices. Therefore the system will need thermal control that will radiate around 660 W of heat from the spacecraft. Radiators of the appropriate size have been selected for this. The structure materials are also highly heat-conductive which will allow for better heat distribution. The overall temperature of the system during two full orbits is represented on the graph on the right.

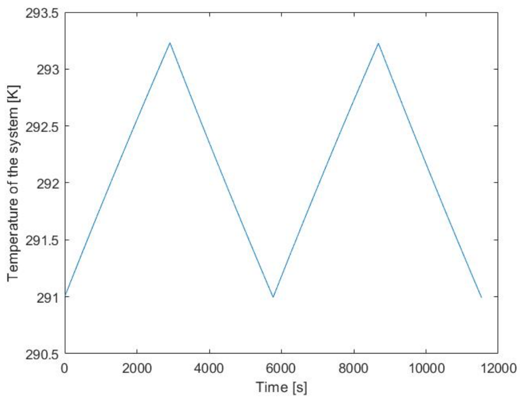


Figure 14: System Temperature during 2 orbits

2.3.3.4 Mechanisms

The most commonly used mechanism for a solar sail is a hinge. It has the advantage of not using electrical power to deploy the solar array and is really lightweight compared to the other options discussed. This is the mechanism that will be used to deploy the solar panels. As for the solar sail, the most appropriate mechanism is the deployable boom. They are lightweight and small considering the amount of material stored inside of them. There is a deployable composite boom currently in development that is even more lightweight than the original one [4]. These devices can also unfold in all 4 directions at once which is feasible for the solar sail. The hold down and release mechanism for the mega constellation satellite will have to be a motorized mechanism that will allow for the release of the satellite. The whole mechanism will have to be of a large size, compared to other mechanisms, as the satellite itself will carry a high mass. This mechanism will also have to be adjustable to the needs of the customer and a motorized device will be the best option in terms of customizability.

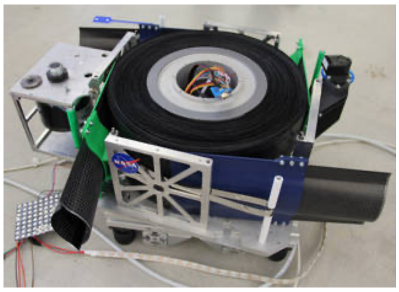


Figure 15: Deployable composite boom *[42]*

2.3.4 Conclusions

The mechanical design subsystem is capable of withstanding loads throughout the mission and keeping all the components within their required operating Temperatures for this mission. All the deployable devices are accounted for and they are able to stay stowed. The whole system has been integrated with the Falcon 9 vehicle and is feasible to be launched inside of it. Since the mission might change, depending on the customer, the altitude at which the system would operate could become larger. In this case, the whole mechanical design is feasible apart from the thermal control system. The temperature would become too low for the components to function. However, this could be adjusted by the usage of different coatings.

2.4 Project manager report

2.4.1 Introduction

For the course unit MACE31521 Conceptual Aerospace design, a business plan had to be created to tackle the space debris issue growing throughout the last decade; in the appendix, research proves this. Alongside the chief engineer, it concluded that as one of the only mega-constellations that were actively launching hundreds of satellites were Stalink by Space X; as seen in the appendix, thus our project is based on the removal of Space X's Starlink and replacing them with a new one, dead satellites also are part of the space debris thus providing the service to remove these satellites for Space X is a feasible idea as we only depend on Space X, it also concluded that no government was going to pay for this project as they do not have any issue with the space debris at the moment. Although for Space X, dead satellites mean space that is not generating any revenue, and that is where this project comes into place and provides Space X to remove and replace these satellites.
As a Project Manager, the preliminary work was to do market research alongside the Chief Engineer and allocate our potential customers, and from there, a business idea was developed. Thus to have a good project, this project was managed in a very structured way maintaining high communication levels and dividing the work into reasonable time frames, alongside the feasibility and resources were managed to ensure that the project was concluded into a realistic conceptual plan, this will be demonstrated in the following paragraphs.

2.4.2 Task planning, monitoring and action tracking

As mentioned earlier, this project is meant to be for Starlinks mega-constellations, and thus all this project is based on all the values for a Starlink satellite. However, it was made so that if there are other mega-constellations dead satellites, they could be tackled as well. Intense research was done on how many satellites fail and how many this project can tackle per year to make sure that the project is being feasible enough; after all of this plan was made, the project preparations began where the workload was identified alongside the requirements to begin the project.
To begin the project, Gantt charts were chosen to be used for better task planning, monitoring, and action tracking of the project; thus, a preliminary Gantt chart was created as seen in image 1, which included most of the work packages for each subsystem and a rough deadline for each work package. Hence, the project worked in a smooth manner, this was ensured in meeting after the lectures to ensure everyone was working on the right thing and they had sufficient time to advance with their work packages.

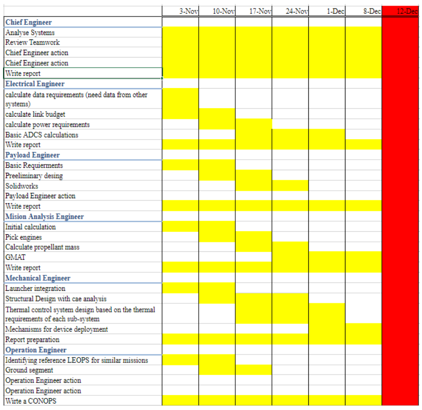


Figure 16: (Initial Gantt Chart)

Although not all subsystems could work as they depended on other subsystems to advance with their work, thus a design matrix was created first to see which subsystem depends on which work packages, as seen in image 16. This was done by creating each engineer on the x-axis and y-axis to then followingly add their dependencies and interactions on the work packages in the x and y-direction; this helped the team to know from whom to get their information and from the Gannt chart, when.

DESIGN MATRIX						
X	Payload Engineer	Mission Analysis	Mechanical Engineer	Electrical Engineer	Operations Engineer	Chief Engineer
Payload Engineer	X	A1	A2		A3	O
Mission Analysis		X				O
Mechanical Engineer	A5	A6	X	A4		O
Electrical Engineer				X		O
Operations Engineer	A7	A8		A9	X	O
Chief Engineer	A10	A11	A12	A13		X

Figure 17: (Design Matrix)

A1	The mass and size of the propulsion system
A2	The mass and shape of the structural frame
A3	Time peroid and budget of the debris removal mission
A4	Volume, mass, thermal requriments
A5	Volume, mass, thermal requriments
A6	Volume, mass, thermal requriments
A7	The capturing mechanism
A8	The starting altitude, rendezvous process and the required individual mission time.
A9	The communication method, ADCS
A10	Power requirements, type of current used (AC/DC), data budget, mass, limitations (temperature, vibrations, loads) of paylaod, redundancy of payload equipment
A11	Power requirements, type of current used (AC/DC), data budget, mass, limitations (temperature, vibrations, loads) of propulsion system, requirements on the launch vehicle
A12	Mass, dimensions, SpaceX attachment mechanism, attachments mechanism with Falcon 9 rocket, grade of protection (thermal, radiation, vibration resistance)
A13	Power requirements, type of current used (AC/DC), data budget, mass, limitations (temperature, vibrations, loads) of computer and ADCS, redundancy of payload equipment
A14	Implicatiopns of all possible modes of operation on the system, peak energy and datastream requirements, constrains and limitations of all modes of operation

Table 6: (Values of the Design Matrix)

Followingly the Gantt chart was improved with more days, as seen in image 3, with better time management and with better consideration with busy times of all engineers as any quizzed or coursework that our course was experimenting or on a more personal level like, study days, free days or even days dedicated to their societies work, as well all work packages were added which helped the team to understand better which work package was done on which day and when to claim their information from other engineers. Throughout the Gantt chart and the project, one of the project manager's responsibilities was to check that all the requirements were being met and not overpassed; this was done by checking each member's numbers and working on comparing them to the requirement to see if these are met. The shades of orange tell the amount of work that is going to be done; the darker it is, the more work will be done, light green means that they have other societies work or social time, darker green means that there are other deadlines from other courses, pink means the deadline the of the work package, red means a whole day group meeting where the group works together to finalize the project and work on the poster, and finally yellow means work when available time, which only applies to the report writing as it is a more individual work package.

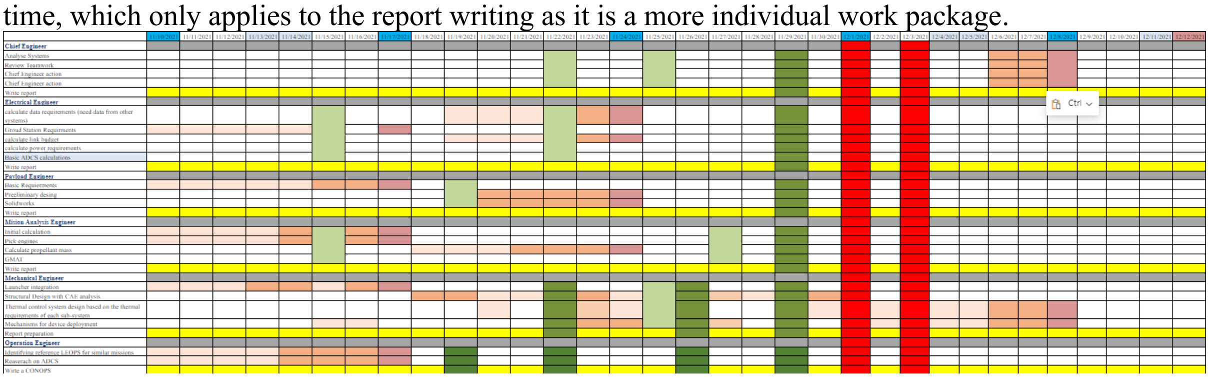


Figure 18: (Finalized Gantt Chart)

2.4.3 Resource management and communication

Communication was handled with the design matrix mentioned earlier. The first issue was that some subsystems could not work until another subsystem gave them some work advancement. This caused delays in our initial phase of the project. This was solved by creating the design matrix, which made the project much more straightforward and alongside the design matrix and Gantt charts, it was pretty simple for the engineers to contact each other depending on when the work was finished looking at the Gantt chart and what they required from that engineer as seen in the design matrix.

Communication was made effective in all stages by the project manager. This began with organizing an hour meeting for preliminary notes and business analysis of the project after each lecture that was provided. After-hours group meetings were organized in-person and on Zoom later during the term to analyze and create the requirements required for this project; these were then finalized by the project manager and the chief engineer. After creating the design matrix, one-on-one meetings were arranged from a prior week so that information could be exchanged and explained in a proper manner by the engineer providing them. These were done for most of the components of the design matrix. Finally, group meetings were organized to achieve the best of the team for the project; these were also organized to update each other where they had advanced with their sections and if any engineer needed help to make critical decisions. These group meetings also helped the team understand in depth what all engineers were working on and where to have a better idea of the project; this made other engineers improve other engineers with their part of the project. This only made the project better more efficient.
The main methods of communication used in this project were, in-person meetings which were organized as mentioned earlier, Zoom meetings were also organized, which became more regular towards the end of the project to make sure everything was going as planned, and finally, a Facebook messenger group was created to keep all team members informed and aware of the meetings that the project manager organized.
The resources of this project were managed in a more theoretical way, as the cost is something that is calculated in the next phase of the project, although for this project to be feasible, the cost was made as realistic as possible, for example, the cost of the whole project was tried to keep it lower than the cost of similar projects and in the same way than the price of the boosters and cameras and similar items were kept to a competitive price, the whole point of this was to try and make this project as tentative for our buyers as possible. Other resources were also managed in the same way; a trade-off template was provided to all engineers to make more accessible the decision making for their subsystems alongside making all the values as realistic as possible.

2.4.4 Requirements development, tracking, and compliance

As mentioned earlier, the requirements were made alongside the chief engineer, the user requirements were made for the user to be, in this case, Space X as this project is made with the intention to remove the defunctioning Starlink satellite, this is because as a business point of view Space X can pay this start-up for this service and this will be beneficial for them because as mentioned earlier Space X loses revenue on satellites that are not working in orbit, as well as Space X, get a new satellite in place, and this can benefit even more as they cost as this project uses a falcon nine which is a Space X rocket. Before getting the user requirements, the fundamental problem and needs were thought thoroughly to get this project right from the first step. From there, the system requirements were thought of and finalised alongside the engineer's compliance and primary responsibility. However, some of the responsibilities of the requirements were divided between two engineers. The one with higher responsibility was chosen. The project manager created followingly the requirements tree to ensure that the requirements could be visualised more straightforwardly, which helped the project look simpler.
From the system requirements, each subsystem analysed and created its subsystem requirements which were then developed even more during the project. The duty of the project manager is to make sure that these requirements are met. This was done by analysing each subsystem work step by step to ensure that each subsystem was doing the right job and no engineer was steering in another direction; this was done in the group meetings where each action was supervised to ensure that everything was in order. During the project, some requirements were changed to a more realistic requirement, but this only made the project more feasible. However, most of the requirements were followed to the letter to ensure the project was taken as seriously and professionally as possible.

2.4.5 Conclusions

It can be concluded that this project was managed in a very structured way obtaining high communications levels between the engineers with formal group meetings and other methods mentioned above, alongside all the work packages were completed, and a very successful presentation was obtained. As well, all the requirements were met of this project for the user, system and each subsystem, and finally, this project is feasible, and it can be taken to the next stage if sufficient funding is provided.

2.5 Mission analyst report

2.5.1 Introduction

This section will cover the mission analysis department for the mission to collect space debris on behalf of project Czechmate; this section will consist of a step-by-step rundown of each section behind the mission followed by a trade-off for that part as well as a feasibility analysis to fully explain why that option was elected. The user requirements for this mission, which were developed from inputs from the chief engineer, are for the company to provide a cheap, reliable, and effective way to replace commercial satellites in a Low-Earth-Orbit; therefore, a single mission needs to be cost-effective and use components that have proven to work often, whilst offering a lot of information to aid with the design. From payload, the method intended to deorbit a satellite is to position itself above the target satellite and fire a net towards the target, capturing it and pushing it towards Earth to be burned up in the upper atmosphere. Subsequently, also releasing another, functioning, satellite into the same orbit as the non-functioning satellite was placed in. The mission analysis will therefore need to consist of a well-planned and well-executed operation in which the payload can successfully be taken to an orbit, perform the necessary manoeuvres to get itself aligned with the target and then perform a safe de-orbit and re-entry.
Deciding which altitude to have the mission to focus around will be the altitude with the highest demand for space debris removal as this would allow the business to be aimed at a lot more potential customers, and after conducting some market research, as denoted from the previous sections, it was uncovered that most of the space debris and dangerous collisions are occurring at an LEO orbit (>2000km); therefore, that would be the orbit the mission will be targeting.

2.5.2 Launch Vehicle

Firstly, a suitable launch vehicle needs to be elected to transfer the payload to the correct orbit to be deposited, the table below denotes some of the rockets that were considered for this operation as well as their capabilities and prices.

Rocket:	Falcon 9	VEGA	Soyuz
Max payload to LEO:	22,800Kg	1500Kg	4400Kg
Cost:	$62 million ($28 million operational)	$37 million	$52 million
Ease of Access to information	Excellent	Excellent	Good

Table 7: Launch vehicle comparison

From this table, it was decided that the satellite will be carried by the Falcon 9 since, while it is a lot more expensive as a single launch than the other rockets, it can carry a much higher payload and therefore the payload can launch as an addition to a pre-existing SpaceX mission to deposit its own satellites, subsequently, on its own power, move to its correct inclination and it is a much cheaper option compared to the other rockets, as well as still being capable enough to carry our payload along with their own mission so the business would only need to pay a fraction of the price to launch since it would be co-using the rocket. Also, since we will be releasing a new functional satellite, SpaceX will pay for the mission to remove and replace their satellite. Furthermore, there is a lot more information available from SpaceX on their Falcon 9 which will strongly aid in developing accurate calculations to (a huge launcher spare capacity) it gives the possibility of launching several missions simultaneously if the business proves to be successful; however, for this analysis, only one mission will be considered.

2.5.3 Launch Site

For the launch site, the mission requirements were to be able to reach most LOW-EARTH-ORBIT satellites within a reasonable range so that this mission can be targeted to multiple clients. Upon assessment, one of the main customers in mind was Starlink satellites that find themselves incapable of deorbiting themselves within a timeframe of fewer than 5 years. SpaceX has currently placed their Starlink satellites in three categories called "shells" the first being at 380Km, the second at 550Km and the last which will be gradually put up at 1130Km. To demonstrate how this mission will operate, this report will focus on the 550Km bracket and how the payload will approach it. For this mission, the satellite is placed at an inclination of 53° and we will need to either be deposited at this inclination or perform an inclination change. Kennedy space centre is one of the sites where SpaceX currently launches from which is beneficial to the mission spec since from KSC, the Falcon 9 can launch us at multiple inclinations over a wide range (including the 530 needed for this mission which means in this particular case we will not need any propellant for an inclination change) the other sites include Cape Canaveral and Vandenburg Space Force base (both of which are capable of reaching multiple inclinations as well) but the current CEO of SpaceX, Elon Musk, has planned to move most of the future launches to KSC and has a 20-year exclusive lease on the launch pad until 2034; so it would be more feasible to plan the mission from there in anticipation for the mission in 5 years. The launchpad that will be used for this mission will be the Kennedy Space Centre Launch Complex LC-39A and a launch from this site has the capability of delivering the satellite to an inclination between 28°-62°. An important factor to our launch is the launch azimuth (the angle between the north direction and the projection of the initial orbital plane onto the launch location) as it is essentially the compass heading for what direction the satellite has when launched. To calculate this, equation (1) was used; the platform has a latitude of 28.5°, so to get to an inclination of 53o the launch azimuth was calculated to be 70.42^0 Although most of this will be handled by our carrier, SpaceX.

$$\beta = arcsin\left(\frac{\cos(i)}{\cos(\emptyset)}\right) \qquad (1)$$

2.5.4 Mission

For which specific orbit that should be used from LOE to conduct this mission, a trade-off between all the possible deposit orbits the Falcon could release the payload was conducted from an inclination of 53 degrees. Obviously in this case it can be assumed that SpaceX can deposit us at the right inclination and altitude however we cannot assume this if the mission target was a satellite independent of the Starlink system and the falcon can only deliver us near to the final target. The requirements of the orbit must be to be within a Low-Earth-Orbit range and an inclination that would not require a lot of propellant (fuel) to be on track for the target. Furthermore, the payload should be able to manoeuvre itself towards the target satellite to capture the inactive one as well as deploy the new, functioning satellite.

Orbit height:	450km	600km	1200km
Shell focus	1st	2nd	3rd
Distance from target	100km	50km	650km
Delta-V requirements	high	average	high

Table 8: Mission analysis data

After comparing potential altitudes to be deposited, a 600km starting point was elected to analyse this mission, whilst the other orbits can test a much higher propulsion requirement than 600km, it is a good average to use to demonstrate how all the components in the mission will operate since analysing the distance from this point to any Starlink satellite will enable the model to be adaptable enough to work with any LEO satellite that needs to be removed.

2.5.5 Delta-V budget

Since they have a much higher delta-V requirement to reach the furthest satellite from them, it is unreasonable to assume such a manoeuvre would be necessary especially since SpaceX makes regular trips to multiple altitudes in LEO, therefore the altitude that would lead to the more accurate calculations to analyse the feasibility of our mission would be at 600km from the surface of the Earth. In addition, since the lower and higher altitudes are much further away from each other in comparison to 600km and need a lot more delta-V to travel to when scaled it makes more sense to make sure the fuel used up in the manoeuvre is as minimal as possible to ensure there is enough propellant as a precaution in the likely outcome that there needs to be an inclination/ RAAN change after separating from the main rocket. Additionally, orbit manoeuvres take a lot more energy going from a smaller orbit to a larger one than going from a larger orbit down to a smaller one and since most LEO satellites are around or below 600km, this is a good altitude to model from to ensure the mission can be as adaptable and as feasible as possible.

The requirement from the orbital manoeuvre is to aid our payload is transporting itself from the position the Falcon 9 deposits us in orbit, to its target satellite. Due to the requirements from how the payload will deploy the net towards the inactive satellite, capturing it and sending it on a trajectory towards Earth, it was assessed that the orbit manoeuvre should align the payload directly above the target at a range of roughly 100m facing the Earth. For this mission, orbital inclination changes would not be considered but for other potential missions, there is an anticipation that such a manoeuvre will be necessary.

Manoeuvre:	Hohman Transfer	Single burn transfer: circle to elliptical	Single burn transfer: elliptical to elliptical	Bi-elliptical transfer
Delta-V requirements	low	high	moderate	Very low
time	long	short	moderate	Very long

Table 9: Transfer comparison

A Hohmann transfer involves initiating a burn sequence at one orbit, travelling towards the target orbit where the second burn begins, and the satellite begins to slow down as it settles into its new orbit. A circle to elliptical involves commencing a single burn to go from a circular orbit to an elliptical one that should have a different orbital time and allow the chaser to catch up with the target and is one of the simplest transfers to operate. An elliptical-to-elliptical transfer involves a single burn from the thrusters to get the payload into an orbit with a different velocity to capture the target. Finally, a bi-elliptical transfer involves moving from a circular to an elliptical, then back to a circular and involves three burns; two to change altitude and eccentricity and a final burn at the end to maintain the orbit. It is typically the least delta-V demanding manoeuvre out of all of them.

After conducting extensive research, which is illustrated by the table above, it was decided that a Hohmann transfer would be the most appropriate option for this specific mission since it would align itself on the same plane as the target satellite and based on the calculations using equations (2) with the reference altitude difference from 600km to 550.1km it would use the less delta-V to perform this transfer than a single burn transfer yet it will still be faster than a triple burn transfer based on equation (3), which measures how long the transfer will take. In another case, if the payload was to be deposited at the same altitude and inclination then an eccentricity or inclination change to alter the speed of the payload to catch the target it would be more appropriate to use a single burn transfer to move from a circular to an elliptical orbit since it's the most efficient way to have the two objects travelling at separate speeds to compensates for the phase angle difference.

(2) $\Delta V = |\Delta V_1| + |\Delta V_2|$ (3) $T = 2\pi\sqrt{\frac{a^3}{\mu}}$ {'a' is semi-major axis 'μ' is the gravitational constant

For this mission, two delta-V values are needed for the payload to align itself correctly with the target satellite. The first burn will be to begin the Hohmann transfer, and the second will be to keep the payload into the new orbit of 550.1km above sea level, where the satellite will hopefully be placed 100m directly away from the dead satellite whilst facing the Earth. Based on typical values and information from previous, similar missions, it can be estimated that such a mission has a total Delta-V of around 4.24km/s. the transfer should resemble figure (1) and the actual model of the transfer produced on an orbital simulation software is demonstrated in figure (2)

Using equations (4) and (5) for this mission, the delta-Vs required for this trip were calculated however, in the tool that was used to analyse data through the usage of these equations, equation (6) was also considered in preparation for any potential inclination changes as well as altitude changes to have the chaser satellite rendezvous with the target satellite. These calculations depicted that for this mission the delta-V budget would be 3.9826 x 10^4 for the two burns necessary to perform the Hohmann transfer to the desired orbit; based on assumptions about the furthest the payload would have to travel in terms of altitude and inclination (7), the most, total, delta-V that would be required should be roughly 4.9763 x 10^4. This is important to consider since the propulsion method needs to be able to handle a mission with much higher requirements. The delta-V requirements to get the satellite into the correct orbit would be down to SpaceX so this business would not need to conduct any research in that department.

(4) $\Delta V_1 = V_{transA} - V_0$ (5) $V_0 = V_{circular} = \sqrt{\frac{\mu}{a_0}}$ (6) $V_{transA} = \sqrt{\frac{2\mu}{r_x} - \frac{\mu}{a_{trans}}}$ (7) $\Delta V_i = 2VSin(\frac{\Delta i}{2})$

2.5.6 Mission Timeline

Something that came into consideration while designing this mission was how could this system handle complications and increase the chances of a successful mission? Excluding the time taken to get the satellite to the correct altitude, the mission would only take a few hours, and for more demanding missions it should not take more than a day; however, this is assuming that the launch happens exactly on time without any complications caused by uncontrollable factors such as the weather. Therefore, to counteract the effect of this the mission timeline was set to a month to give the payload sufficient time, after it does eventually launch, to stay in its deposited altitude until the time for the manoeuvres to occur come around; in addition to further time for it to stay in its new altitude until it realigns with its target once again. Because the orbital periods at this height are relatively short it is reasonable to assume that the satellites can realign themselves in a reasonable amount of time and increase the chance for a successful mission whilst not affecting the amount of propellant that would be needed for the mission and thus reducing the weight of the whole payload. The payload will also spend roughly 40% of its orbit in eclipse. This information is very important for the mechanical and electrical engineers as it will impact their design heavily.

2.5.7 Propulsion

Now that the delta-v budget has been established for this mission, as well as for the most demanding case, a suitable propulsion system now needs to be chosen that will be capable of meeting the performance requirements for this mission. To help select an appropriate propulsion system, using the delta-V values that were calculated along with equations (8) to find the specific impulse, several different propulsion systems were compared. The two main power types that met these requirements were chemical and electrically powered thrusters. However, since electric thrusters (such as an ion thruster) would require a lot of power, far more than what the payload would be using, and the technology is very new so there isn't much information to help aid the system model design, it was decided that the propulsion system would be a chemical one.

$$(8)\ \Delta V = I_{sp} g_0 log \frac{m_0}{m_1}$$

$\{'I_{sp}'\ is\ specific\ impulse\ 'g_0'\ is\ acceleration\ due\ to\ gravity\ 'm0'\ is\ wet\ and\ 'm1'\ is\ dry\ mass\}$

Name:	Weight:	Length:	Diameter	Max Thrust	Cost:
RD-170	9750kg	4.0m	3.8m	7900000N	N/A
RS-25	3177kg	4.3m	2.4m	2279000N	$40 million
Waxwing	87kg	1.3m	0.71	29400N	$7million
Merlin 1c	630kg	2.92	2m	480000N	$9million

Table 10: Rocket engine comparison

The main requirements other than performance that went into choosing propulsion was the weight and cost since this mission would be a one-time launch and not reusable; therefore, the mission would need to be as cheap as possible. The option that was elected for this mission was the Waxwing (rocket motor) since its power performance met the minimum that was necessary for the mission and it was the smallest, lightest, and most importantly, the cheapest. The waxwing does not have any additional power requirements, instead, it just uses a dual tank consisting of an oxidiser and refined rocket propellant, which mostly consists of kerosene.

Material:	Yield stress:	Density:
Steel (alloy)	250MPa	7800kg/m^3
Aluminium (alloy)	276MPa	2710kg/m^3
Carbon fibre	2500MPa	2000kg/m^3

Table 11: Material comparison

When conducting on potential materials to make the fuel tank of the propellant from, an interesting discovery was made. Both the ESA and NASA have been working on carbon fibre tanks and both have proved to work in testing which is very practical for this mission. As the table demonstrates, carbon fibre is a lot stronger and lighter than most other materials that were typically used in this industry and therefore it would be a very straightforward trade-off to favour carbon fibre over any other material.
In order to work out the dimensions of the tank first, equation (9) was employed to calculate the mass of the propellant needed to conduct the necessary manoeuvres, this came down to 112kg. subsequently, equation (10) was used to determine the volume the tank would need to be to store that much fuel. Finally using equations (11), (12) and (13) to find the radius, wall thickness and mass of the tank, respectively. These values are very important for the mechanical engineer as they will heavily impact the weight and dimensions of the payload. From these calculations' it was discovered that the mass of the tank should be 8kg and the volume is 5.3m^3 (5.51mm thickness and 1.438m radius). To further reinforce the trade-off for materials that could be used for the fuel tank; it was discovered that the other material would mean a larger and thicker tank was necessary, thus resulting in a much heavier tank. The final mass for the propulsion system including oxidiser, fuel tank, rough piping based on some assumptions and nozzle should come down to 397kg.

(9) $m_p = m_1 \left(e^{\frac{\Delta V}{I_{sp} g_0}} - 1 \right) \{'mp' is\ mass\ of\ the\ propellant\}$ (10) $V_{tank} = \frac{m_p RT}{P_{max} M_{Xe}}$ {'Mxe' is molar mass constant, 'Pmax' is maximum power}

(11) $r_{tank} = \sqrt[3]{\frac{V_{tank}}{\frac{4}{3}\pi}}$ (12) $t_{wall} = Safety\ Factor \cdot \frac{P_{\max}\ r_{tank}}{2\sigma_{yield}}$

(13) $m_{tank} = \frac{4}{3}\pi[(r_{tank} + t_{wall})^3 - r_{tank}{}^3]\rho_{tank}$

2.5.8 End-of-Life

At the end of the mission, whether it was successful or not, the satellite needs to have a safe way to begin de-orbiting and safely re-entering the Earth. The minimum required delta-V required to start a satellite on its decay orbit is 100ms^{-1}. A potential way that can be utilised to deorbit the satellite is to use some propellant to angle the payload towards the Earth and give it an impulse helping it to come down into Earth's atmosphere. Although, in some cases for this mission, it may be that there won't be enough propellant, depending on other factors in the mission such as issues with alignment, so there may be insufficient fuel left after a mission to begin deorbiting and caring extra propellant just for de-orbit increases the delta-V budget and thus means the payload is heavier and will need a bigger tank with is the opposite of what the business is trying to achieve. Furthermore, there is a higher chance of not being able to perform re-entry if the payload is only relying on propellent to get itself back to Earth otherwise, it will remain in its orbit as space debris and the entire mission will be null and void. Therefore, after conducting some research, an interesting idea was proposed to tackle this issue. Instead of using the thruster to perform the decaying orbit, the payload will employ a solar sail. Based on information from some articles and using equation (14), a solar sail of 10m^2 can deorbit the satellite in a little over 10 years. This amount of time, whilst being an extensive period, is within the

rules and regulations set by NASA & ESA for maximum time to de-orbit, consequently, that is not an issue. Moreover, the solar sail would weigh roughly the same as the maximum mass increase needed for a deorbit at the highest point, yet because the sail is only dependent on a deploying mechanism and has nothing to do with the actual mission, it is a far more reliable method for de-orbiting and one that is easier to model mathematically through equation (15) since so long as the mechanism worked it would be able to successfully de-orbit.
As for a safe re-entry, several papers proved that Starlink and other similarly sized LEO satellites will burn in the Earth's upper atmosphere and fully disintegrate before reaching the surface; this means that, through already tested methods, the returning dead satellite and the net from the payload will burn up completely and hopefully not cause any damage to life irrespective of what trajectory it has. As for the payload that will be sent up, it is heavier and larger than the Starlink so, in order to make sure it would also have a safe retry, research was conducted regarding debris re-entry with similar dimensions. Some papers revealed that it should be perfectly fine but to further aid the feasibility of this option, equation (16) and (17) was used to model the speed of the payload on re-entry and the effect the upper atmosphere it would have, and the results showed that it will still burn up in the atmosphere, just slightly longer than the Starlink. The alternative option would be to wait until the satellites are aligned in such a way that after firing the net, the trajectory will lead the satellite towards an unpopulated area. A model of where the satellite would be at any time was made using appropriate software as shown in figure (2). The issue with this method was that the satellite was not big enough or travelling slowly enough for it to reach the surface, but it would be beneficial to have the satellite perform its de-orbit in time to land in an unpopulated area just for safe measures.
(14) $F = q_s \frac{1+k}{c}$ {'qs' is illumination and 'c' is speed of light} (15) $\Delta V = \frac{2q_s tA}{Wc}$
$\{'A' is\ area\ of\ solar\ sail\ 't'\ is\ time\ and\ 'W' is\ weight\}$
(16) $E = \frac{1}{2}mv^2$ (17) $W = \frac{1}{2}c_d A\rho v^2 s$ $\{'cd' is\ drag\ coefficient,\ 's' is\ didstance\}$

2.5.9 Conclusion

Using all the equations mentioned before, in conjunction with some further equations obtained from multiple trustworthy sources, a tool was developed to aid the decision-making process and test the feasibility of all the final decisions. All decisions were made based on their suitability to the mission and the requirements set out initially, that the system should have. As this mission should follow, the payload with the new Starlink Satellite will be loaded onto the Falcon 9 along with potentially other satellites for SpaceX's mission, making the mission cheap since it will be employing a reusable rocket with a high payload. The rocket will fly to an orbit of 600km and release the payload into an LEO orbit at a 53° inclination. The payload will then, using the power of its propulsion system, begin a Hohmann transfer to 550.1km to position itself directly above the inactive satellite where the net can align itself to the target and capture the satellite whilst simultaneously launching the new working satellite. After the mission has been completed, the solar sail mechanism will be deployed and using the power of illumination from the sun, the payload will begin its decay orbit towards Earth where it will disintegrate completely and safely after 10 years; in which case several other missions can launch between that time since each mission will be mutually exclusive.

2.6 Operations engineer report

2.6.1 Introduction

This Concept of Operation describes how the system work and the mission timeline. A typical operation scenario is described. The potential risk of the system and its impact to the environment is discussed.

2.6.2 Goals and Objectives

The main goal of Czechmate is to capture and remove an out of service satellite of a constellation out of orbit and replace it with a serviceable one. Since non-serviceable satellites cannot generate any revenue, Czechmate can maximize the profit of the constellation at a much smaller cost with only needing to launch both satellite and the capturing device at once. The satellite is captured using a net and brought toward the earth. The satellite is then burnt when entering the atmosphere. A new satellite is also placed after the satellite is captured.

2.6.3 Overview of system and key elements

The system is a carrier that can carry the capturing device and the replacement satellite. It catches up with the satellite at a higher orbit and conducts Hohmann transfer to a lower altitude that the satellite is on. The system can deorbit itself.

Pre-launch calculation

The coordinate of the defunct satellite is obtained from an open-source database. The mission duration is decided. The power system is chosen based on the mission duration

Launching

The system is launched by a reusable rocket to lower operation costs.

Communication

The system sends its telemetries and the videos from the go pro back to the earth ground station. The ground also updates the coordinates of the defunct satellite constantly.

Rendezvous

Since the system needs to catch up with the satellite, the system is designed to be released at a higher altitude than the target satellite because the system can travel faster at that altitude. The system conducts the Hohman transfer and brings itself just 100 meters above the satellite at the right time so that they can alight after transfer.

Capturing

The targeting system first targets the satellite. Once, it is targeted, the ground control has the final decision on starting the capturing. After the decision has gone through. The capturing device launches a net to apply momentum that can deorbit the target satellite. A booster is also activated to cancel out by reaction force when launching the net.

Releasing replacement satellite

A new replacement satellite is released after capture.

Deorbit

The system deorbits itself afterwards. The system deorbits itself using a solar sail and booster if there are enough fuel remaining and re-enter the atmosphere.

2.6.4 Modes of operation

- Launch mode
  When the system is in the launching vehicle, it is launch mode. The system is connected to the launch vehicle to receive power.
- LEOP mode [33]
  Once the system is separated from the launcher, the system enters separation mode. The solar panels are extended and oriented to the sun. A system check and instrument calibrations are conducted after a stable power supply is received. The ADCS is turned on to make sure the system is stabilized. Then, the system establishes a connection with the ground station. Telemetries of the system are sent to obtain the location. The system then orbits once for the solar panel to obtain enough power for the mission.
- Standby Mode
  If the system is required to stay at the first orbit and wait for the right timing to conduct a Hohmann transfer. The system enters standby mode, where most of the non-essential components like cameras are turned off to preserve power.
- Rendezvous Mode
  The system conducts a Hohmann transfer to an orbit that is 100m above the defuncted satellite. Once they are rendezvoused and aligned, the system enters capture mode
- Capture Mode

The detection system activates and targets the defunct satellite. Once, the targeting system is locked onto the satellite. The ground control decides when to start the capturing. If the capturing command is received. The capturing device launches the net from above and toward the target satellite. The momentum of the net brings the target satellite towards the earth. The satellite is deorbited and will enter the atmosphere in at most 1 year. The satellite is burnt and destroyed.

- Release mode
  After the launch of the net, the system enters Release mode. The replacement satellite is released from the carrier.
- Self-deorbit mode
  The solar sail on the system deploys and deorbit itself. The booster also activates if there is remaining fuel. It will re-enter the atmosphere in approximately 10 years.

2.6.5 Operations Scenarios

In this scenario, the Czechmate is set to remove a defunct Starlink satellite from SpaceX. The defunct satellite in the first orbital shell is at an altitude of 550km and an inclination of 53 degrees. That is where most of the Starlink satellites are located. The launch vehicle is Flacon 9 from Space X. This is the only reusable rocket that is operational commercially and can travel to LEO. The launch site is Kennedy Space Center which is going to be the launch site that SpaceX mostly use for launching their Starlink satellite [2]. The timing of the launch is calculated based on having the least time required for this mission. The mission timeline is based on the launch of Starlink 23 on the 7th April 2021 [34].
T= (Hours:Minutes)

1. Liftoff T=00:00
2. Separation T=01:03
   The system separates with the launch vehicle at 600km.
3. LEOP (First stage) T=01:05
   The system extends its solar panel, establish ground communication, perform system check and instrument calibration.
4. LEOP (First Orbit-600km) T=02:05
   The system orbits once on this altitude for getting enough power with the solar panel.
5. Hohmann transfer T=03:42
   The system conducts the Hohmann transfer to an orbit of 500.1 km using boosters. The time required is 3.2 hours.
6. Second Orbit-550km T=06:52
   At 550.1km, the satellite is aligned with the Starlink satellite.
7. Capture Satellite T=06:55
   The Starlink satellite is targeted and is captured by the net.
8. Release new satellite T=06:57
   The replacement Starlink satellite is released at the same altitude. The mini booster on the new Starlink satellite brings itself back to the orbit at 550km.
9. Deorbit T=07:00
   The system deploys the solar sail to deorbit itself (30mins)
10. Re-entry T=07:30
    The system is planned to re-enter the atmosphere in 10 years. The system will be fully burnt and disintegrated in the atmosphere.

The total mission time expect re-entry is expected to finish in 7 hours. If there are any delays with steps 1-3. The system will miss the time window to conduct the Hohman transfer. The system can wait on this orbit to catch up with the satellite. In this case, the mission can take up to 1 to 2 months.

2.6.6 Potential Impacts

The system and Starlink satellite are decided to disintegrate and burnt up completely when re-entering the atmosphere [35]. There should be no debris hitting the ground.
The system and the captured can collided with other space satellites or other space debris.

2.6.7 Risk and potential issues

- The solar panels fail to extend and have no power for the system.
- The ADCS is not on and the satellite is not stabilized
- The system fails to align or target the satellite.

If those events occurred. The mission should be considered a failure and the self-deorbit phase would start. Since the replacement satellite will also go to waste if we deorbit it, it would be worth considering adding redundancy to our system.

- The system fails to establish a connection with the ground
- The solar sail fails to extend

If those events occurred. Our system cannot deorbit and is at risk of colliding with other satellites. The booster can still work and act as a backup plan if there is still fuel remaining. Moreover, if the system fails to establish a connection with the ground. Consider adding a timer for starting the self-deorbit phase if there is no connection to the ground after a certain amount of time.

2.6.8 Feasibility Analysis

The LEOP can take longer than designed to. The system may fail to conduct the required system check, fail to extend the solar panel etc. Longer time is required for the system to identify and troubleshoot the problem. That would cause a delay. The system cannot conduct the Hohman transfer at the designated time. Then, it needs to wait at the higher orbit and catch up with the defunct satellite. The system can enter standby mode to preserve power. The solar panel are used so that the power supply can last up to one or two months if a longer waiting time is required.

The Hohman transfer is the most fuel-efficient way for a plane change. There are also extra fuel with the satellite at a higher altitude and inclination if needed.

Launching the net from above the satellite is the most effective way in removing the satellite compared to launching the net sideways and slowing the satellite down. It also ensures any debris created from the capturing process are moving toward the earth.

The system can be deorbited using a booster and solar sail. Since solar sail does not require any fuel, the system can still be deorbited when the fuel is all used up.

2.7 Operations engineer report

2.7.1 Introduction

The electrical subsystem consists of the following subsystems: power, guidance, navigation, and control (GNC), communications, and in-board data handling (OBDH). The power subsystem is responsible the generation, distribution, and maintenance of power for the duration of the mission. It is a vital subsystem as a power-system failure results on the loss of a space mission (Fortescue et al., 2011) The guidance, navigation and control subsystem is in charge of calculating the position, velocity, and orientation of the system, and adjusting these parameters as required to complete the mission [2]. The communications system allows the system to work by transmitting tracking, telemetry, and command data between its elements, which consist, in this mission, of ground stations and the spacecraft [2]. Finally, the on-board data handling system oversees the gathering of data from the different sensors and cameras and processing it, so it is transmission ready. It also handles the processing and routing of the command data provided by the ground station (Fortescue et al., 2011).

2.7.2 Requirements and System models

2.7.2.1 Power

For the design of the power subsystem, it was determined that the main requirement if the subsystem was to be able to provide power to all components for the duration of the mission, as can be seen in requirement ER1. Since the system was to be designed to be able to perform its mission at a range of orbits, instead of a specific one, no strict power values were set, due to possible changes in payload power requirements, which is accounted for in ER2. For the purposes of the system model development, the variables pertaining to the most common orbit the mission would be targeting were used (altitude: 550.1 km and inclination: 53º). The first step in designing the power subsystem was to decide the type of power source to be used (Fortescue et al., 2011). Solar arrays were determined to be the best option for this mission. The decision process will be discussed in further detail in Section 4. The next step was sizing the batteries needed (Fortescue et al., 2011). To do so equation 2.1 was used to determine battery capacity:

$$C_r = \frac{P_e T_e}{(DOD)n(1-battery\ degradation)} \quad (2.1)$$

Where C_r is battery capacity, P_e is power required during eclipse, T_e is eclipse period, DOD is depth of discharge and n is the battery efficiency. After having determine the capacity, the batteries can be sized using the energy density value. Afterwards the power that the solar array needs to generate during is determined. To calculate this the power needed to charge the battery was determined and, in conjunction with the power that the system required to operate during daylight was used to calculate the energy the power source needed to generate over the daylight period. This was then divided by the daylight period, obtaining the power needed to be able to generate to power the satellite. Then, using equation 2.2, the power the system would be able to generate per square meter of solar panel was calculated.

$$P_0 = I_r n \quad (2.2)$$

Where P_0 is the power per square meter, I_r is the solar irradiance and n is the energy conversion efficiency of the solar panels. By dividing the power that the solar array needed to generate to power the satellite by the power per square meter, the area of the solar array was determined, and by dividing it specific power the mass of the solar array was determined [2]. After all this, a power distribution system was selected based on power requirements and the number of interfaces.

2.7.2.2 Guidance, Navigation and Control

In order to design the guidance, navigation and control system model the first step was to determine the type of stabilization. It was determined that for the payload to work, the spacecraft needed 3-axis spin stabilisation (requirement ER10). After this, the various torques affecting the system were calculated. To calculate manoeuvre torque, the diagonal components of the mass moment of inertia were multiplied by the angular velocity of the manoeuvre and divided by the desired time to stabilise to obtain each directional component. The gravitational torque was calculated by using the following equations:

$$T_{G,x} = \frac{3\mu}{2\times(RE+h)^3}\left|I_{yy} - I_{zz}\right| \sin(\phi)\cos^2(\theta) \quad (2.3)$$

$$T_{G,y} = \frac{3\mu}{2\times(RE+h)^3}\left|I_{zz} - I_{xx}\right| \sin(\theta)\cos(\phi) \quad (2.4)$$

$$T_{G,z} = \frac{3\mu}{2\times(RE+h)^3}\left|I_{xx} - I_{zz}\right| \sin(\theta)\sin(\phi) \quad (2.5)$$

Where μ is the gravitational parameter of the Earth, I_{xx}, I_{yy} and I_{zz} are components of the mass moment of inertia, RE is the radius of the Earth, ϕ is spacecraft roll and θ is spacecraft pitch [3]. The magnetic field torque was calculated using an estimated value of the residual magnetic dipole of the spacecraft and the magnetic moment of the Earth. To calculate the solar radiation torque, the following equation was used:

$$T_{sr} = \frac{I_r}{c} A(1+R)\left(C_{ps} - C_g\right)$$

(2.6)

Where I_r is the solar irradiance, c is the speed of light in vacuum, A is the cross-sectional area of the spacecraft, R is the solar reflectance factor and $\left(C_{ps} - C_g\right)$ is the distance between the centre of pressure and the centre gravity [3]. This last value was approximated. The drag torque was taken as zero for the purposes of this calculation, since at the orbital altitudes the system would be at, its value can be considered negligible (Fortescue et al., 2011).

After having the calculated all the torques the reaction wheels (section 4 will explain why reaction wheels were chosen in further detail) were sized to be able to counter the effects of the largest one.

2.7.2.3 On-board data handling

The first step in the development of the on-board data handling system handling (OBDH) was to identify the functions it would have to perform. After having done this, it was possible to estimate the number of channels the OBDH system needed to manage all subsystems and maintain on-board data control, thus fulfilling requirement ER8. The next step was to perform calculations to determine the estimate amount of processing power the system would need to process telemetry data, command data, housekeeping data and the video feeds, radar data and image recognition software that the payload system requires.
After determining the required processing power estimate, a computer was chosen based on its ability to comply with these requirements

2.7.2.4 Communications

The main assumption taken during the communications system model design was that the variables pertaining to the orbit the mission would be targeting were to be used (altitude: 550.1 km and inclination: 53º)
The development of the communications system model started by determining the link requirements, such as the access time, quantity of data per unit time (from OBDH and payload systems), sampling rates and bits per sample. Afterwards, using a modulation scheme plot (figure 2.1), a desired link budget was determined. Then, using the frequencies available to the band to be used in the mission, S-band, the variables pertaining to the antenna of our choice, a CubeSat patch antenna, and the transmission rates necessary to transmit and receive the data needed for the mission to run smoothly (making the system complaint with requirements ER4, ER5 and ER7) the power required to transmit the required data was determined for both uplink and downlink (more information on the choice of communication band and antenna in section 4). First, the gains, such as antenna gains, and the losses, such as free space loss, were calculated. To calculate free space loss, for example, equation 2.7 was used.

$$L_{fs} = 20\log_{10}\left(\frac{4\pi S f}{c}\right) \tag{2.7}$$

Where S is the slant range, f is either uplink or downlink frequency, depending on which one is being calculated and c is the speed of light in vacuum. After having calculated all the losses and gains, the power required was calculated using the following chain of equations [3]:

$$C = \frac{C}{N} + N_0 + 10\log_{10}(R) \tag{2.8}$$

$$EIRP = C - G_r + L_{fs} + L_i + L_{add} \tag{2.9}$$

$$P = EIRP - G_t \tag{2.10}$$

Where C is the signal strength, $\frac{C}{N}$ is the carrier-to-noise-ratio, R is the transmission data rate, $EIRP$ is the effective isotropic radiated power, G_r is the receiver gain, L_i is the line losses, L_{add} is the additional losses and G_t is the transmitter gain. This value was then converted from decibels to watts to check if the options were valid.

2.7.3 Concept design options

2.7.3.1 Power

When designing the power subsystem, it was necessary to determine the type of power source best suited for the mission, which would require it to fulfil requirements ER1 and ER2. Three different types were considered: RTGs, solar arrays and fuel cells. Radioisotope thermal generators (RTGs) convert the thermal energy caused by the decay of radioactive isotopes into energy using the thermoelectric effect [3]. Solar arrays convert solar electromagnetic radiation into electrical energy

using semi-conductors [3]. Finally, fuel cells work by generating electrical energy from the reaction between a fuel and an oxidiser through an electrolyte [3].
Following this, the type of battery had to selected. The battery type to be used would have to allow the fulfilment of requirements ER1 and ER2. Batteries can be divided into two types: primary and secondary. Since this mission would used a separate power source, only secondary (rechargeable) batteries were relevant. The types of secondary batteries considered were nickel-metal hydride, nickel-hydrogen, lithium-ion, lithium polymer. Nickel-metal hydride and nickel-hydrogen rely on the electrochemical processes between nickel and hydrogen to store power [4]. Where they differ from one another is that nickel-hydrogen batteries use compressed gaseous hydrogen. Lithium-ion and lithium polymer batteries rely on the movement of lithium-ions between two electrodes (energy.gov, 2017). They differ from one another by the former uses a liquid electrolyte and the latter uses either a solid or semi-solid electrolyte

2.7.3.2 Guidance, Navigation and Control

For the design of the GNC subsystem, it was determined that the system would be 3-axis spin stabilised, as per requirement ER10. As such, the use of reaction wheels and momentum wheels was considered for the system. Reaction wheels react to changes in vehicle-pointing error, by the speeding up the wheel, which initially had a speed of zero. This torque corrects the orientation of the spacecraft, leaving the wheel spinning at a low speed, until another error either further increases or decreases the speed at which the wheel is spinning. Secular disturbances can cause reaction wheels to be saturated, requiring an external torque, through a process called momentum dumping, to bring the wheel back to rest [2]. A momentum wheel, on the other hand, is spinning at a constant, high speed. This provides the spacecraft with gyroscopic stiffness. The attitude of the spacecraft can be adjusted, in a system using momentum wheels, by torquing the wheels, either increasing or decreasing the speed at which they spin. Periodically, the wheel needs to be desaturated, by bring its speed back to the nominal value [2].
Afterwards, the sensors to be used by the system to determine attitude, thus fulfilling requirement ER11, had to be selected. The following types of hardware were considered: Sun sensors, star sensors, horizon sensors, magnetometers, gyroscopes. Sun sensors work by measuring one or two angles between the incident sunlight and their mounting base. Despite being very accurate and reliable, sun sensors require clear fields of view [2]. Since scanners type star scanners are used on spinning spacecraft, only trackers were analysed. Tracker type star scanners work by tracking one or more stars, with some modern versions tracking star patterns, to obtain attitude information. [2]. Horizon sensors are infrared sensors that are used to detect the contrast between the temperature of Earth and of deep Space. These sensors provide Earth-relative information for Earth-pointing spacecraft [2]. Magnetometers are used to measure the direction and size of the Earth's magnetic field. By comparing it to Earth's known field, it allows the system to determine attitude. [2]. Finally, gyroscopes are used to measure speed or angle of rotation from an initial reference. They are used to attain precision attitude control when combined with external references [2].

2.7.3.3 On-board data handling

The main design decision that had to be done for the on-board data handling system was the type of computer that was going to be used by the system for it to comply with requirement ER8. The options considered were commercial of-the-shelf components, unlike the other systems, where general options were analysed. This was done to reduce analysis time, by not having to investigate every individual component that constitutes the on-board computer. In the end, two options were considered. The Gomspace NanoMind A3200 space grade computer and the Raspberry Pi Compute Module 4

2.7.3.4 Communications

For the communication system, two main decisions had to be made. The first one was to decide on the kind of operating frequency the satellite would be using. S-Band, K-Band and L-Band were the

operating frequencies analysed for this mission. The S-Band ranges from 2-4 GHz, K-Band ranges from 12-40 GHz and L-Band ranges from 1-2 GHz
The other decision was the type of antenna to be used. Patch antennas and parabolic reflector were considered options for the system. Patch antennas are small, lightweight antennas that can be attached to the structure of a spacecraft. They can have a large beamwidth for their size [6]. Parabolic reflectors are usually best suited for missions with high peak gains and low beamwidths [2].

2.7.4 Final design option selection

2.7.4.1 Power

When deciding on power source to be use in this system, the main considerations were the peak power required, the mission lifetime, as stated in requirements ER1 and ER2 (figure 4.1). It became immediately apparent, by analysing the operational envelops of the power sources considered, that fuel cells were not a viable choice of power source, since the mission may take up to month, whereas fuel cells usually operate to a maximum of a couple of weeks. Choosing between solar arrays and RTGs, the main deciding point was their difference in cost and COTS parts availability, as per requirement SR2, since both sources were able to generate enough power to sustain the system. Since the method of satellite disposal chosen is to have it disintegrate on re-entry, using and expensive power source such as RTGs, is not viable, thus solar arrays were the chosen power source. As for batteries, Lithium-ion batteries were chosen over nickel-metal hydride and nickel metal due to their higher specific energy, higher energy density and wider temperature range. This allowed the development of a lighter power subsystem without compromising electrical performance. They were also chosen over Lithium-polymer batteries, despite the latter being more efficient since their improvements are not significant enough to justify the increase in cost [7]. In addition, Lithium-ion batteries are more commonly used, and COTS components are widely available, thus making it compliant with requirement SR2.

2.7.4.2 Guidance, Navigation and Control

The main design driver in the guidance, navigation and control was the debris removal system, in this case, a net fired from the satellite. As such, the system world required the ability to do precise targeting, in addition to being able to counteract the momentum from the net. As such, reaction wheels were chosen over momentum wheels since they would be better suited to react to the immediate change in momentum, instead of relying on gyroscopic stiffness. Reaction wheels are also more appropriate to small changes in orientation, such as those needed for targeting, than momentum wheels, since the latter is designed to resist attitude changes [2]. In terms of sensors, it was decided that the system would be equipped with a star sensor, a magnetometer and a IMU gyroscope. The star sensor would provide general attitude data, with the magnetometer provide complementary data. The gyroscope would monitor acceleration data Sun sensors were discarded because of the target orbits, which are in LEO, since the system spends a large percentage of time in eclipse. Horizon sensors were not very appropriate, since they’re better used for nadir pointing satellites[2].

2.7.4.3 On-board data handling

The main constraint of the on-board data handling system was the processing power needed to run the image recognition software, in addition to handling telemetry and command data, plus any house keeping data necessary. The main advantage of the Gomspace NanoMind A3200 is its reliability in the space environment, but it lacks in processing power. As such the Raspberry Pi Compute Module 4 was chosen for the system and, to get around any possible component failures, the majority rule would be implemented in the system, by using 3 Compute Modules 4 receiving the same input data. If one the computers outputs the wrong data, the system will be able to detect which one it is and reset it, thus circumventing their lower reliability.

2.7.4.4 Communications

For communications, S-Band was chosen to be the communication frequency. S-band frequency are widely used for LEO applications and, as such, have a wide variety of COTS components, allowing the fulfilment of requirement SR2. A patch antenna was chosen over a parabolic reflector due to its small size and large beamwidth, as this allows a reduction of the system weight and allows a large coverage time [6].

2.7.5 Conclusion

Using the equations mentioned in this report in addition to few others from various sources, a system model was developed to aid in the decision process. All decisions were made based on the calculations performed on the system model, in order to ensure their suitability for the mission and to comply with the requirements set out initially.

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Appendix

Table of contents of Appendix

List of figures

List of tables

Nomenclature

EOL	End Of Life
ESA	European space agency
GEO	Geosynchronous earth Orbit
ISS	International Space Station
LEO	Low Earth orbit
StDev	Standard deviation

Problem definition

Types of space debris

Space debris is a term referring to objects in Earth's vicinity that do not serve any purpose. Space debris can be divided into two categories - natural and artificial.
Natural space debris consists of small pieces of cometary and asteroidal material called meteoroids. Natural debris does not pose a significant threat to human space activities due to its size and quantity.
Artificial space debris is any non-functional man-made object in space and its size can range from pieces of a few millimetres to satellites as big as a school bus. As of 2021, the European space agency (ESA) registers around 9500 tonnes of all objects in the Earth's orbit. An overview of all objects and their sizes can be found in the table below.

Origin	Category	Count
Number of rocket launches since the start of the space age in 1957	Total number excluding failures	About 6090
Number of satellites these rocket launches have placed into Earth orbit	Still in space	About 7510
	Still functioning	About 4500
	Total	About 12000
Number of debris objects regularly tracked by Space Surveillance Networks and maintained in their catalogue	Total	About 29110
Estimated number of break-ups, explosions, collisions, or anomalous events resulting in fragmentation	Total	More than 570
Total mass of all space objects in Earth orbit	Total	More than 9500 tonnes
Number of debris objects estimated by statistical models to be in orbit	Size greater than 10 cm	About 34 000
	Size between 1 cm and 10 cm	About 900 000
	Size between 1 mm and 1 cm	About 128 000 000

Table of space debris in orbit as of 9.7.2021 [1]

There are many sources of man-made space debris, however some of the most common ones include:

- Satellites that have reached the end of their life
- Satellites and spacecraft that have failed
- Rocket stages that have launched satellites into space
- Nose cones, payload covers, shrouds, bolts and other launch hardware
- Solid propellant slag

- Space activity cast-aways (accidental or deliberate), eg wrenches, human waste
- Deterioration fragments, e.g., peeling paint
- Fragments from exploding batteries, fuel tanks
- Fragments from collisions, both accidental and deliberate

In the future, the situation is likely to worsen as the amount of space debris is predicted to grow in all orbits. If there were no new satellites launched into orbit in the future, the current pollution has already crossed a threshold and would slowly grow even without human intervention. It is in the scope of this report to provide a prediction of space debris in the future. NASA's LEGEND model is a full-scale, three-dimensional, debris evolutionary model that is the NASA Orbital Debris Program Office's primary model for study of the long-term debris environment projection. It covers the near-Earth space between 200 and 50 000 km altitude, including low Earth orbit (LEO), medium Earth orbit (MEO), and geosynchronous orbit (GEO) regions.
LEGEND-simulated historical LEO environment and results from three different future projection scenarios. Each projection curve is the average of 100 Monte Carlo runs. The effective number is defined as the fractional time, per orbital period, an object spends between 200 km and 2000 km altitudes. Credit: NASA ODPO. [2]

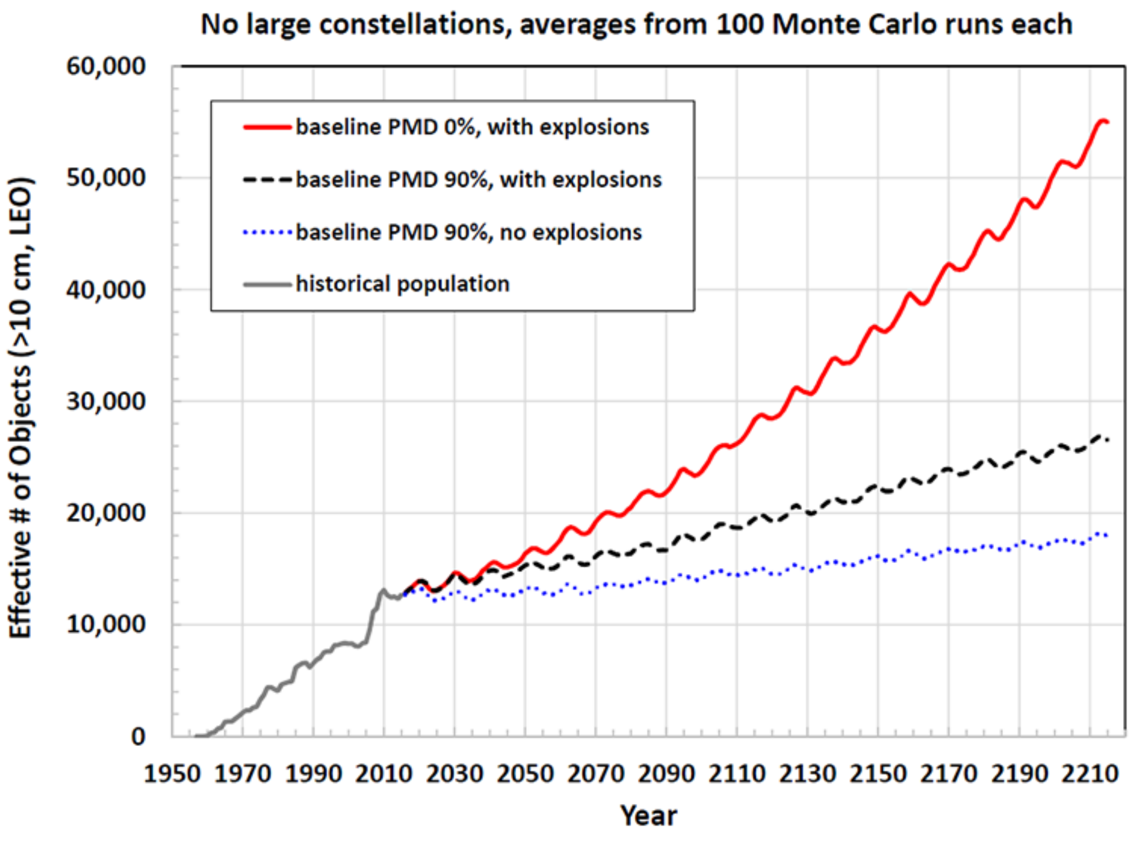


Chart showing number of objects >10 cm in LEO [2]

Dangerous nature of space debris

Space is a hostile place and objects that can be found in orbit have extreme velocities. A typical impact occurs at a closing velocity of 10 km/s. The laws of physics dictate that an object must have a particular

velocity in order to stay in Earth's orbit and this velocity decreases with altitude. Therefore, in LEO velocities are extremely high.
Such impacts are called hypervelocity impacts to indicate their extreme nature. At these velocities a piece of debris has more kinetic energy than an equal mass of high explosive. To give an example, an impact with a one-kilogram object will cause more damage than the explosion of one kilogram of TNT as can be seen in the figure below.

Block of aluminium hit by a 14-gram piece of plastic travelling at 28000 km/s in space [3]

We identify what size of space debris can be found in Earth orbit, at what altitudes and inclination are the highest concentrations later in this report, however we need to assess the probability of collision in orbit and identify the most dangerous inclinations and altitudes here.
According to a study P.H. Krisko, who also utilised the NASA's LEGEND model, it is possible to assess future risk for spacecraft posed by space debris in LEO orbit. The study indicates a LEO environment that is already highly collisionally active among orbital debris larger than 1 cm in size. Most of the modelled collision events are non-catastrophic (i.e., They lead to a cratering of the target,

but no large-scale fragmentation.). But they are potentially mission-ending, and take place between impactors smaller than 10 cm and targets larger than 10 cm. Given the small size of the impactor these events would likely be undetectable by present-day measurement means. Impact rates of about four per year are predicted by the current study within the next 30 years, with the majority of targets being abandoned intacts (spent upper stages and spacecraft). Still, operational spacecraft do show a small collisional activity, one that increases over time as the small fragment population increases.
Those collision rates are averaged over 200 Monte Carlo iterations of the LEGEND process and tabulated in the table below. [4]

Time period	**Total period 1957 to 2035 (79 years)**
Ave # collisions by impactor/target size Target > 10cm, Impactor > 10cm Target > 10cm, Impactor < 10cm Target < 10cm, Impactor < 10cm	 5,03 98,32 1,18
Ave # collisions (All) Catastrophic	104,53 (StDev 26,76) 5,75 (StDev 6,75)
Ave # collisions (both objects >10 cm) Catastrophic	5,03 (StDev 2,58) 2,4 (StDev 1,96)
Ave # collisions (Target > 10cm, Impactor < 10cm) Catastrophic	98,32 (StDev 26,57) 1,68 (StDev 1,38)
Ave # collisions (operational spacecraft) Catastrophic	5,02 (StDev 2,49) 0,30 (StDev 0,57)

Table of average collision events [4]

To summarize a few main points, the averaged LEGEND result with the stated study parameters shows that during the study period:

- The number of collisions among 1-cm and larger objects passing through LEO is nontrivial at present in the future.
- Collisions between small impactors (< 10 cm) and large targets (> 10 cm) make up nearly 95% of all events.
    - Of these events, about 98% are non-catastrophic.
    - Of these events, between 5% and 9% involve targets that are operational spacecraft. This relative rate appears to be dropping over time.
- Nearly 5% of all collision events are catastrophic.
    - Of those events, 30% involve small impactor on large target pairings. [4]

Moreover, the danger of hitting space debris does not encompass only the debris that is already in orbit. When two larger objects in space collide, they create an enormous number of small parts that subsequently collide with each other and create a chain reaction which greatly amplifies the damage caused by one event of space debris collision. This phenomenon was described by Donald J Kessler.

The Kessler Syndrome

NASA scientist, Donald J Kessler, proposed in 1978 that a chain reaction of debris exploding in space could end up making it impossible to use satellites, and ending many other space activities. Should this happen, the region above our atmosphere could become unusable for generations. The idea is that the extensive number of things we launch into low earth orbit (LEO), could create such a dense environment, that an inevitable collision could create a cascading effect, with each collision adding to the growing crescendo. Kessler proposed that with enough collisions, debris could overwhelm the LEO region entirely. [3]

Critical Mass

Some experts at NASA already suggest that we are already at critical mass in the low-Earth orbital zone 900 – 1000 kilometres (560 – 620 miles) over our heads. One frightening prospect raised by Kessler, is that a cascade might not be obvious, until it was well under way, a process that could take years. Though unlikely, it's conceivable that this has already begun.
In 2011 a report by the U.S. National Research Council warned that the amount of orbiting space debris had reached a critical level. Computer models were showing that the amount of space debris "has reached a tipping point, with enough currently in orbit to continually collide and create even more debris, raising the risk of spacecraft failures". The NRC report stressed the need for international regulations limiting debris and increased research of disposal methods [3]

Distribution of space debris in Earth's orbit

The key to effective space debris mitigation is assessment of its distribution density in Earth's vicinity. Considering the mission envelope of the CzechMate, objects in LEO are in focus i.e., up to altitude of 2000 km. This would make sense in general, since LEO is the main space debris field followed by the ring of objects in GEO, according to ESA's model visualized below.

ESA – distribution of space debris around the Earth [5]

Generally, space debris can be divided into 3 categories. Although we could see in the table "Table of space debris in orbit as of 9.7.2021" that the number of particles increases dramatically as their dimensions decrease, with mass it is quite the opposite. The table demonstrates that the majority of mass of all space debris is made of objects that fall in the category "large". [6]

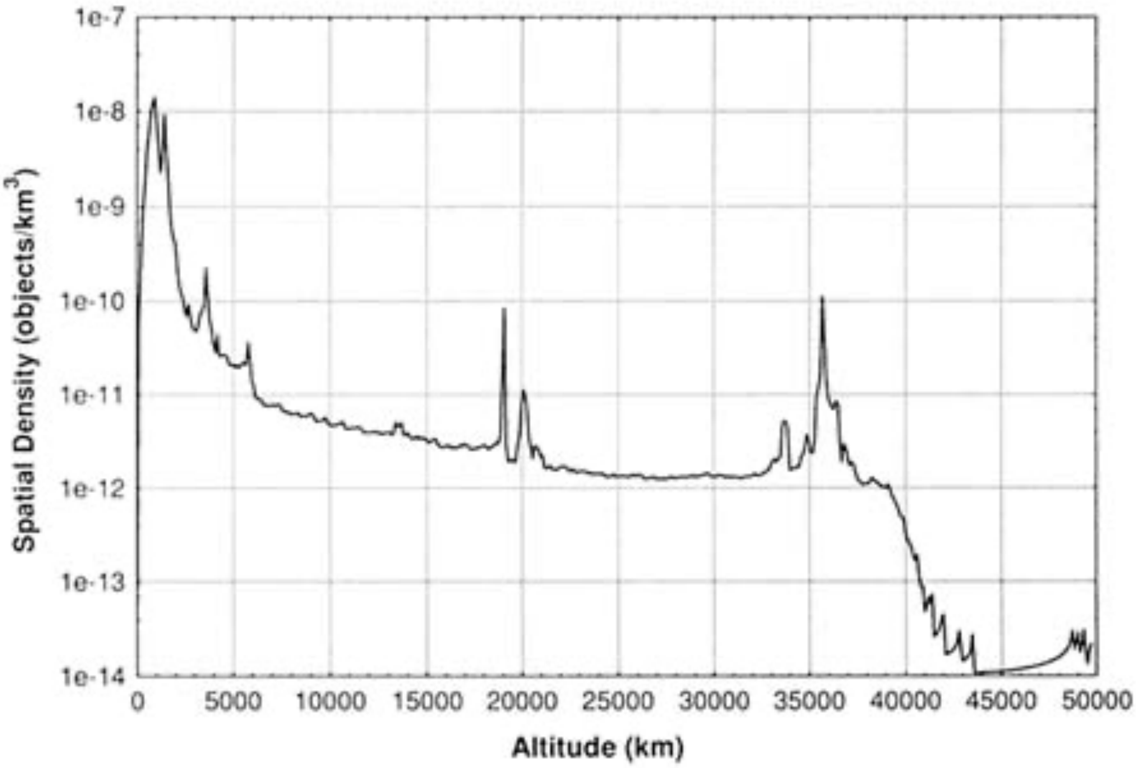
Category	Size	Percentage of total mass
Large	Larger than 10 cm	99,95
Medium	Between 1 mm and 10 cm	Less than 0,05
Small	Smaller than 1 mm	Less than 0,01

Space debris categorisation [6]

While extensive data have been acquired on the cataloged population, cataloged objects represent only a small fraction of the debris in orbit; estimates of the populations of uncataloged debris are based on a limited number of sampling measurements tied together with models. Any estimates of the overall debris population are thus uncertain; they are likely to change as new data are acquired. [6]
We adapted a systematic approach and considered each category of space debris in LEO separately.

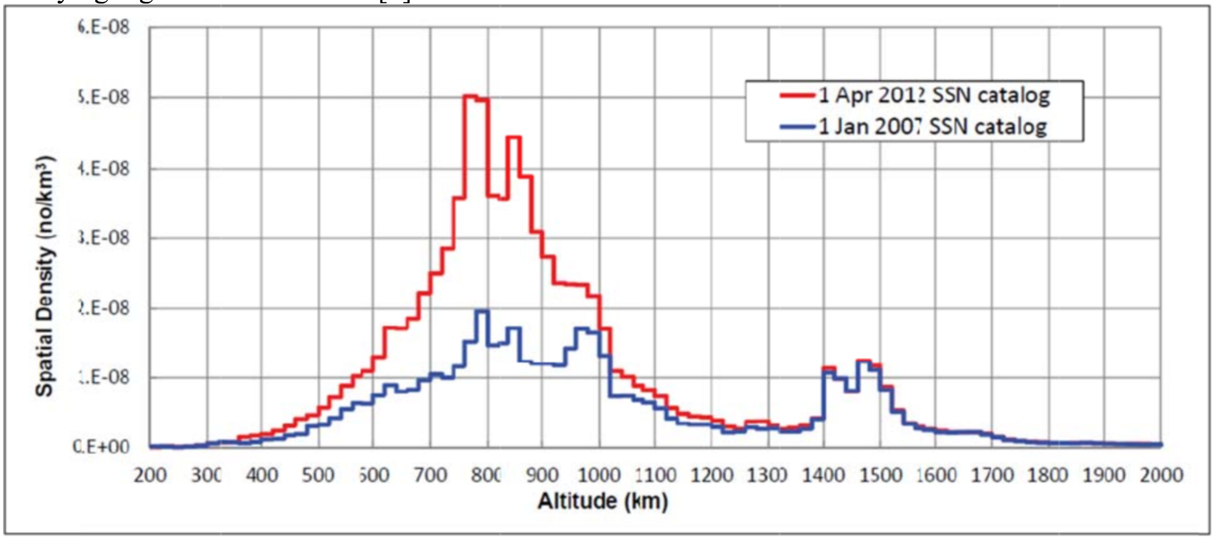
Large debris

The best-known segment of the debris population is the population of catalogued large debris. Some features of the distribution of the cataloged debris population can already be seen in the figure "ESA – distribution of space debris around the Earth", including the concentrations in the GEO ring and in LEO. The figure below quantifies the data by portraying the approximate spatial density of cataloged objects at various altitudes. Clear concentrations can be seen at less than 2 000-km altitude (LEO), around 20 000 km (semi synchronous orbit), and at 36 000 km (GEO). This justifies our approach revolving around the LEO. [6]



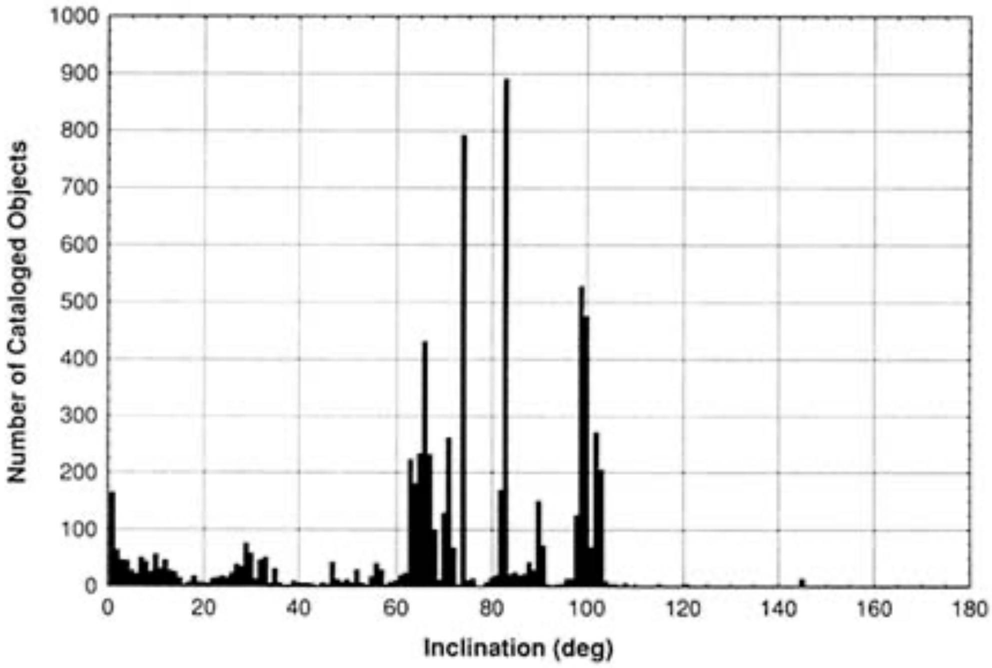
Spatial density of the 1994 U.S. Space Command Satellite Catalog [6]

These concentrations of higher spatial density are due to large numbers of objects in near-circular orbits at or near these altitudes. The lower background level of spatial density visible in the figure above at altitudes up to 40 000 km is due to objects in highly elliptical orbits with perigees in LEO and apogees up to 40 000 km. This background spatial density also exists in LEO, where most highly elliptical orbits have their perigee. Most objects in highly elliptical orbits are either rocket bodies that placed spacecraft in semi synchronous orbit or GEO or objects in Molniya-type orbits. Few objects are catalogued in orbits higher than 40 000 km.
Within the region below 2 000 km, the distribution of catalogued objects by altitude is highly nonuniform, with peaks around 760 to 800 km and 830 to 860 km. Although objects in the lowest altitude orbits eventually re-enter the atmosphere, this population is augmented by objects from decaying higher-altitude orbits. [6]



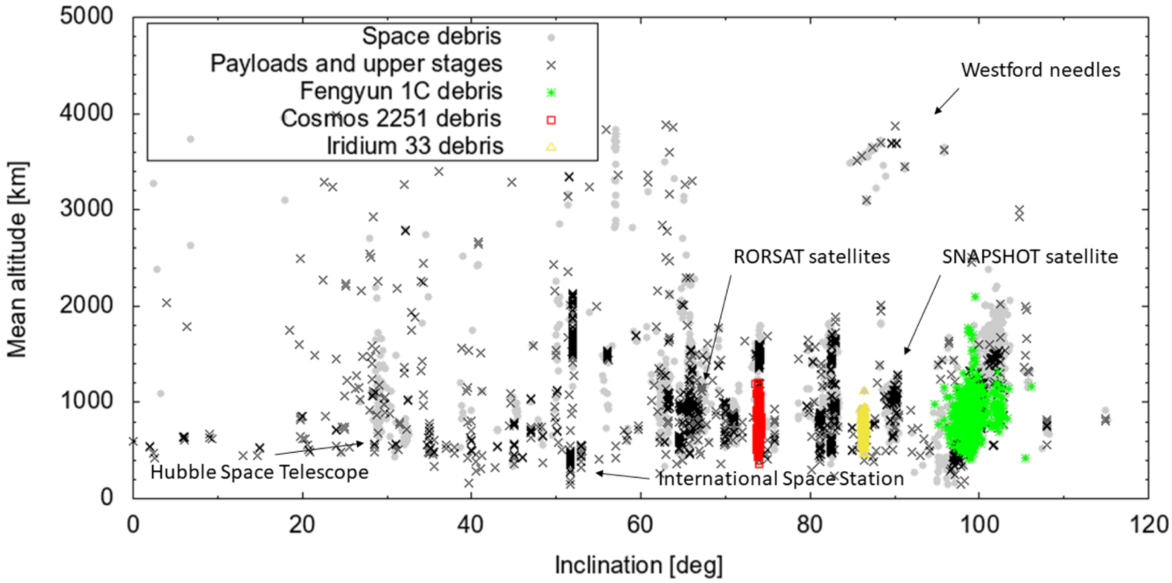
Distribution of space debris according to altitudes [7]

The aforementioned figure presents the spatial density distribution of the catalogued objects on 1st April 2012 (red histogram), and also shows that the space debris at 1000 km altitude have more than doubled since the beginning of 2007 (blue histogram). Fragments generated from the anti-satellite weapon test conducted by China in 2007 and the collision between Iridium 33 and Cosmos 2251 in 2009 were major factors in the jump in the numbers of space debris. [7]
Although altitude is the most important factor influencing feasibility of space debris removal, inclination is very important too and cannot be neglected as it dictates the delta v requirement and thus propellant capacity that needs to be accommodated on board of the CzechMate satellite. Most catalogued objects are in orbits with fairly high inclinations. This means that relative collision velocities for these objects will be generally higher than orbital velocity. Differing orbital inclinations also cause asymmetric distributions in the LEO satellite population by latitude. For example, objects in low-inclination orbits do not contribute to the apparent congestion or bunching of objects in the higher temperate zones, and since few objects are in truly polar orbits (with inclinations of 90 degrees), "holes" in the space object swarm appear over the Earth's poles. (This does not, however, mean that high-inclination orbits will have a lower collision probability; any two circular orbits at the same altitude will intersect at two points, irrespective of their respective inclinations.) Following figure shows the inclination distribution of catalogued large space objects. [6]



Inclination distribution of catalogued population according to Kaman Sciences Corporation, 1994 [6]

It was argued that the amount of space debris is rising every year and so although the figure above does give a good overview, we include more recent figure that depicts inclination vs mean altitude of catalogued object orbits with altitudes less than 4000 km. It is possible to distinguish intact objects, such as payloads and rocket bodies, fragments from satellites Fengyun-1C (mean inclination of 99,0 deg), Cosmos 2251 (mean inclination of 74,0 deg), Iridium 33 (mean inclination of 86,4 deg) and other catalogued debris. Marked are also orbits of the International Space Station (ISS), Hubble Space Telescope (HST), Russian RORSAT satellites, US SNAPSHOT satellites and Westford Needles clusters. [8]



Inclination vs mean altitude under 4000 km (2019) [8]

Medium-sized debris

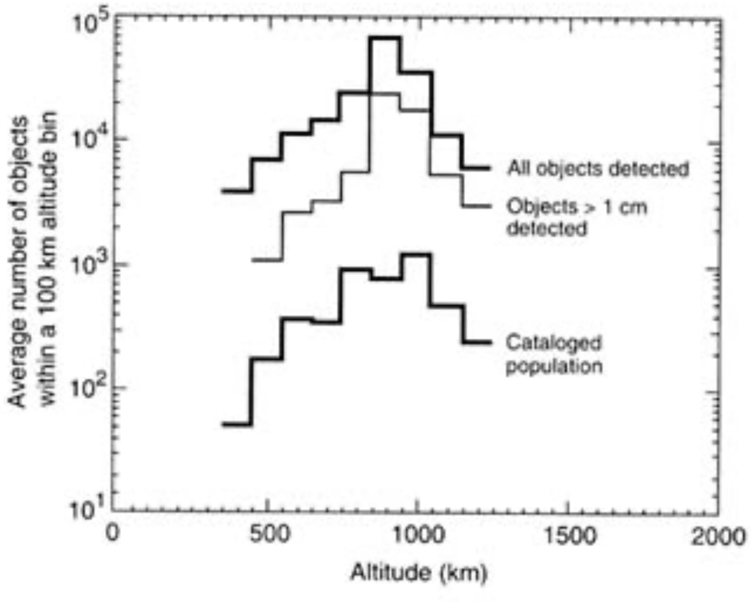
The population of medium-sized (approximately 1 mm to 10 cm in diameter) debris is not nearly as well known as the population of large debris. As described above the only measurements of the medium-sized debris population come from sampling of lower-altitude, higher-inclination LEO orbital regions with ground-based sensors. All other estimates of the size and characteristics of the medium-sized debris population are based entirely on extrapolations.

To a first approximation, it might be expected that medium-sized debris would be found in about the same orbits as large debris, since most medium-sized debris originates from large objects. However, all large objects may not contribute equally to the medium-sized debris population; some types of large object (such as rocket bodies that have been a source of explosive fragmentation) may produce much more debris than others.

Medium-sized debris, which often has a higher ratio of cross-sectional-area to mass than large debris, will often be more strongly affected by atmospheric drag and thus will experience more rapid orbital decay.

Although there is no measurement data proving the origins of medium-sized debris, most likely the population is composed of fragmentation debris and mission-related objects (since non-functional spacecraft and rocket bodies are obviously large debris). The number of medium-sized debris objects detected is large compared to the number of large objects.

Ground-based sensors, particularly the Haystack radar, have provided the most detailed information to date on the population of medium-sized debris objects. The following figure shows the estimated population distribution of objects detected by the Haystack radar when parked vertically, as compared with the population distribution of objects in the U.S. catalogue. [6]

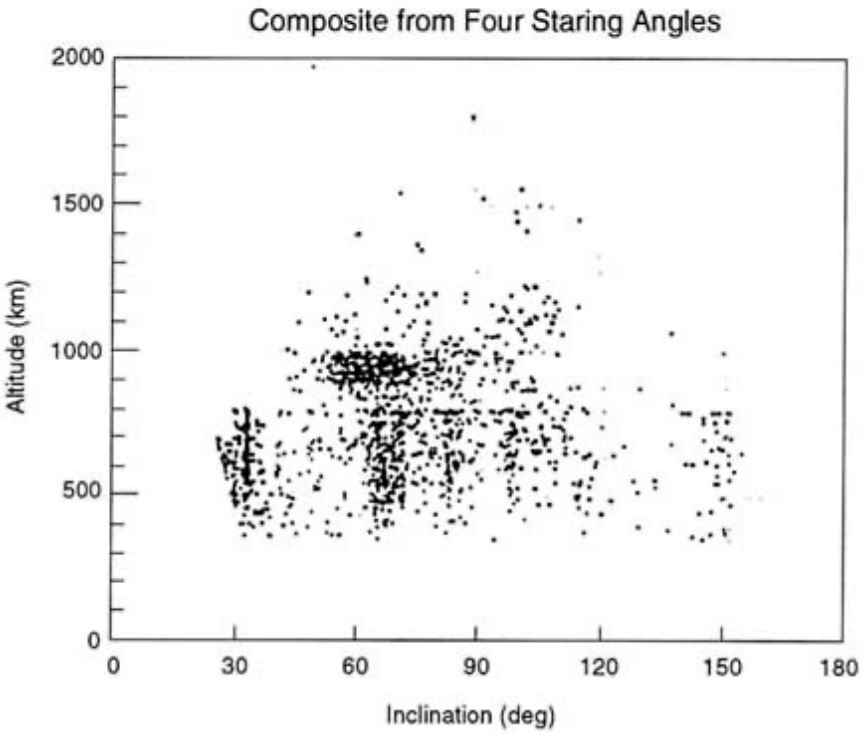


Estimate of LEO mid-sized orbital debris population from Haystack radar (2003) [9]

Interestingly, the data show that for the region measured, the altitude distribution of medium-sized objects is similar to that of the larger objects included in the U.S. catalogue. There are, however, two significant differences: (1) below about 1000 km the population of medium-sized objects detected by Haystack declines with decreasing altitude faster than the population of large, catalogued objects; and (2) around 900 to 1000 km there is a large peak in the population of medium-sized objects detected by

Haystack with no corresponding peak in the population of large, catalogued objects. The first difference is consistent with the expectation that medium-sized pieces of debris are more strongly affected by atmospheric drag than larger debris. The peak in the medium-sized population around 900 to 1000 km, however, points to a source of debris other than previously recorded breakups. [6]

The eccentricity and inclination of many of the medium-sized objects detected by Haystack can also be determined. These measurements show that medium-sized
Space debris is more frequently found in low inclination and eccentric orbits than catalogued large debris and that the large number of objects detected between 900 and 1000 km are in near-circular orbits with inclinations around 65 degrees (Stansbery et al., 1994). The reported detection of objects with inclinations greater than 110 degrees may be a result of the high uncertainty in determining inclination for objects that are barely detectable.
As mentioned previously, the Haystack data suggest that there may be major sources of centimetre-sized orbital debris other than previously recorded breakups. The large number of objects in orbits between 900 and 1000 km with orbital inclinations between 60 and 70 degrees suggests that there is a significant source of debris in this area. If this source were breakups, however, the debris would have been spread over a much wider area than is evidenced by the data. It thus seems possible that some of this debris may be the result of a previously unmodeled source. This possibility is supported by the polarization data from Haystack, which suggests that the objects have relatively smooth and spherical shapes, rather than the irregular shapes that would typically be created in a breakup. A combination of orbital and physical characteristics can be interpreted to suggest that these objects may be tens of thousands of 0,6-2,0 cm diameter liquid droplets of a sodium/potassium coolant leaking from the non-functional cores of dead satellites such as the Russian Radar Ocean Reconnaissance. Furthermore, the data from Haystack do not include the debris that was generated by the two events involving Iridium and Fengyun satellites which undoubtedly contributed to medium-sized debris population especially at inclinations around 84 and 100 degrees. [6]



Altitude vs. inclination of medium-sized space debris (2003) [9]

Small debris

Although the mission of CzechMate will likely not encompass removal of small space debris, we dedicate this section to it for the sake of completeness of this report.
There is an extremely numerous populations of small (<1 mm in diameter) debris particles in Earth orbit. Like medium-sized debris, small debris is all either mission-related objects (e.g. aluminium oxide particles expelled from solid rocket motors) or fragmentation debris (the product of either breakups or surface deterioration). Aluminium oxide particles from solid rocket motor exhaust are generally believed to be approximately spherical in shape with a maximum diameter of about 10 microns.
These particles are initially ejected from rocket bodies at velocities from about 1,5 to 3,5 km/s, depending on the particle size (smaller particles are generally ejected faster). Most of these particles rapidly re-enter the Earth's atmosphere, while others (typically larger particles) are typically sent into a variety of elliptical orbits, depending on where the rocket was fired. Paint chips and similar products of deterioration are usually much larger than the aluminium oxide particles, averaging hundreds of microns in diameter. Such debris particles are released from spacecraft or rocket bodies with virtually no initial ejection velocities and thus initially share nearly identical orbits with their parent object. Finally, the products of breakup span the entire range of small (as well as medium and large) debris sizes and exhibit a variety of shapes. Small breakup fragments likely experience a larger range of ejection velocities than medium or large fragments, placing them in a wider range of initial orbits.
Below 2000 km Earth-altitude, pieces of debris are denser than meteoroids; most are dust from solid rocket motors, surface erosion debris like paint flakes, and frozen coolant from RORSAT (nuclear-powered satellites).
Small space debris mostly appears in “clouds” and is far less dangerous than the other two debris categories. It can, however; interfere with measurements by absorbing or reflecting radiation. [6]

Intact dead satellites in LEO

We recognised in this report that the category of interest for the CzechMate mission is large debris in general. However, the category “large space debris” encompasses rocket stages, shrapnel, and dead satellites. Dead satellites are the type of space debris the CzechMate shall focus on from the following reasons:

- Availability of data e.g., Altitude, inclination, mass
- Easily tracked due to their size
- Relatively simple to catch with variety of techniques (unlike clouds of space debris)
- Owner can be identified and notified about the mission

Therefore, we exploited an open-source database by the Union of Concerned Scientists. The file originally listed over 4000 satellites, both active and inactive. Our report worked with the last version of the file at that time that was last updated on May 1st, 2021.
Firstly, we filtered out satellites that were missing some of their key parameters. Then, we applied a filter to the parameter “class of orbit” to include only satellites in LEO. Subsequently, a function was written that combined the data about date of launch and life expectancy to decide whether a satellite is still active. This allowed us to create a list of satellites with the exact parameters we were looking for. That is, they are inactive, in LEO and include all technical parameters. That concluded the pre-processing stage of our data analysis at the end of which we have a list of satellites with detailed data which could be used in the future for planning of an actual de-orbiting mission for CzechMate.
Secondly, we calculated altitude for each satellite by averaging the perigee and apogee and calculated the following quantitative data as a general overview of our data set:

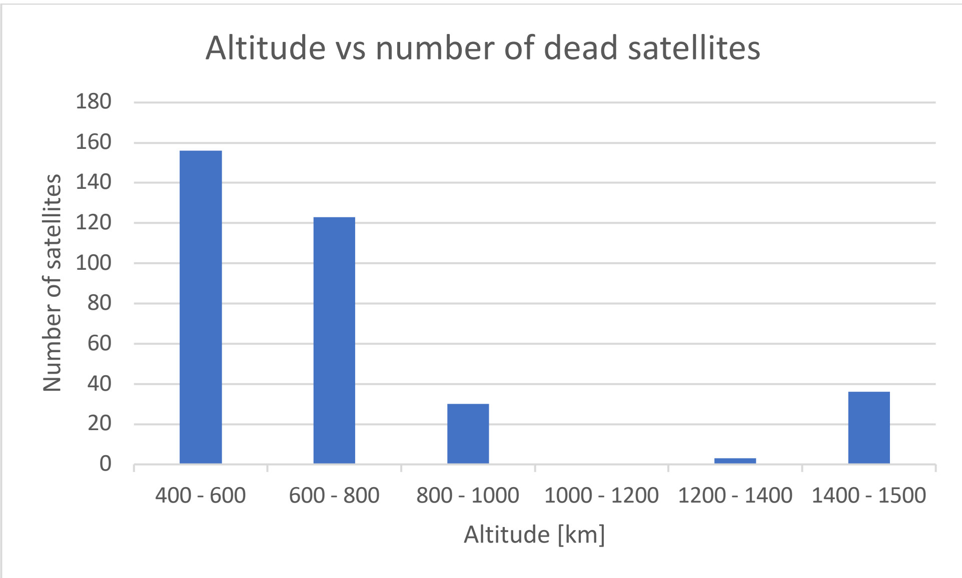
Parameter	Value	Unit
Modus of Average altitude	499	km
Average of Average altitude	724,1	km
Modus of Inclination	97,5	degrees
Average of Inclination	84,48	degrees
Modus of Launch mass	4	kg
Average of Launch mass	876	kg
Number of satellites after applying filters	354	[ ]

Dead satellites quantitative summary

For the space debris removal mission of CzechMate it is crucial to assess which altitude in LEO is most populated with dead satellites, what is the most used inclination and satellite mass. To account for different parameters for each satellite, we divided the aforementioned areas into intervals and counted number of satellites that fall into each category. The resultant data are tabulated below, and each table is visualised as a graph.

Altitude interval [km]	Count
400 - 600	156
600 - 800	123
800 - 1000	30
1000 - 1200	0
1200 - 1400	3
1400 - 1500	36

Altitude vs number of dead satellites table

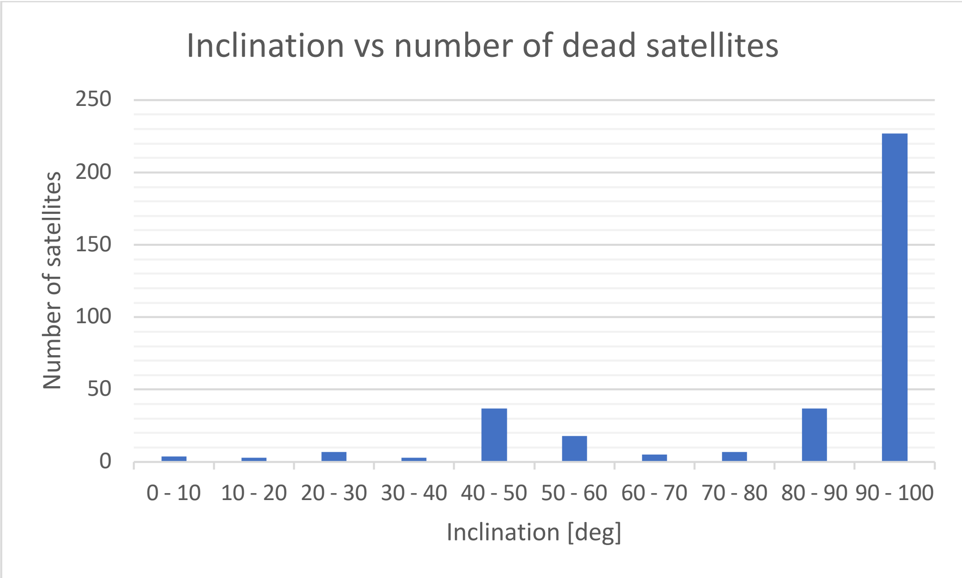


Altitude vs number of dead satellites graph

Inclination interval [deg]	Count
0 - 10	4
10 - 20	3
20 - 30	7

30 - 40	3
40 - 50	37
50 - 60	18
60 - 70	5
70 - 80	7
80 - 90	37
90 - 100	227

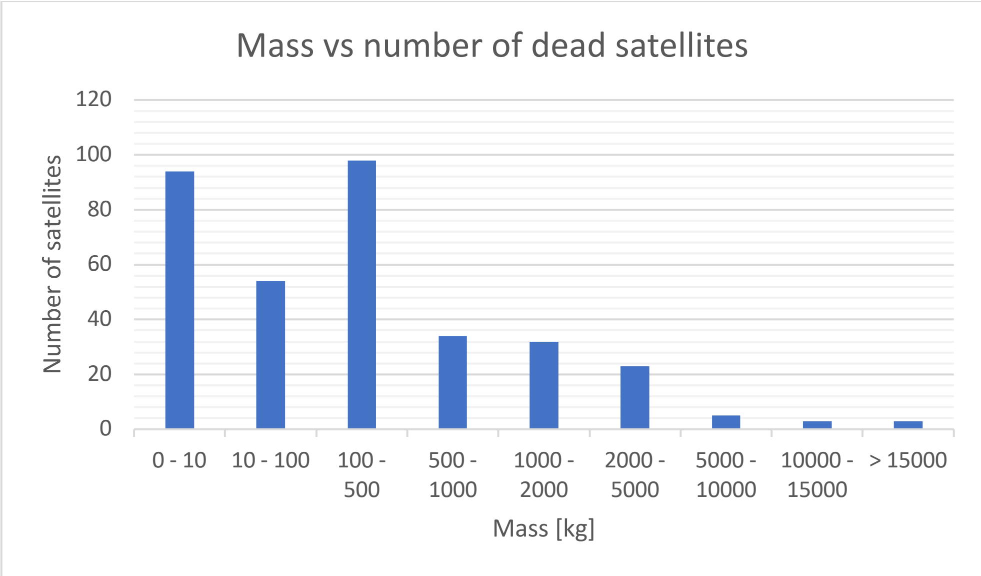
Inclination vs number of dead satellites table



Inclination vs number of dead satellites graph

Mass interval [kg]	Count
0 - 10	94
10 - 100	54
100 - 500	98
500 - 1000	34
1000 - 2000	32
2000 - 5000	23
5000 - 10000	5
10000 - 15000	3
> 15000	3

Mass vs number of dead satellites table



Mass vs number of dead satellites graph

The presented data suggest that the most used altitude is between 400 and 600 km and the majority of dead satellites is found between 400 and 800 km. Furthermore, the most used inclination is clearly between 90 and 100 degrees. There are almost 100 satellites lighter than 10 kg and in the category 100 – 500 kg.

Rocket stages in LEO

Rocket stages are the next category of large space debris which could be of potential interest for CzechMate mission for the same reasons that we listed for dead satellites. Rocket stages are parts of launch vehicles that are necessary only for certain part of ascend to an orbit and are ejected as soon as they are not needed to dispose unneeded ballast on the launch vehicle. There is, however; one downside which is less available data. The open source CelesTrak [10] database that was used for this report does not include information on mass of the rocket stages.
This report worked with the last version of the file at that time that was last updated on August 18th, 2021.
Using the same methodology as for dead satellites, we firstly filtered out satellites that were missing some of their key parameters. Then, we applied a filter to the parameter "object type" to include only rocket stages. Subsequently, a function was written that averaged the perigee and apogee to give average altitude which was used to filter out any object that was not in LEO. This allowed us to create a list of satellites with the exact parameters we were looking for. That step concluded the pre-processing stage of our data analysis at the end of which we have a list of rocket stages with detailed data which could be used in the future for planning of an actual space debris removal mission for CzechMate.
Secondly, following quantitative data was produced as a general overview of our data set:

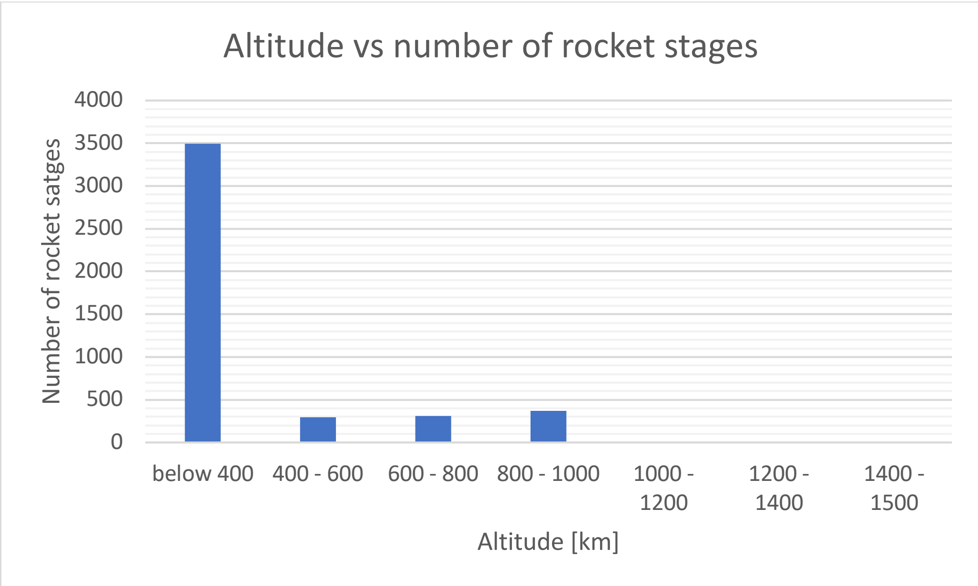
Parameter	Value	Unit
Modus of Average altitude	150	km
Average of Average altitude	297	km
Modus of Inclination	52	degrees
Average of Inclination	63	degrees
Number of rocket stages after applying filters	4470	[ ]

Rocket stages quantitative summary

For the space debris removal mission of CzechMate it is crucial to assess which altitude in LEO is most populated with rocket stages and what is the most used inclination. Exploiting our previous work on dead satellites, we divided the aforementioned areas into intervals and counted the number of satellites that fall into each category. The resultant data are tabulated below, and each table is visualised as a graph.

Altitude interval [km]	Count
below 400	3495
400 - 600	297
600 - 800	308
800 - 1000	370
1000 - 1200	0
1200 - 1400	0
1400 - 1500	0

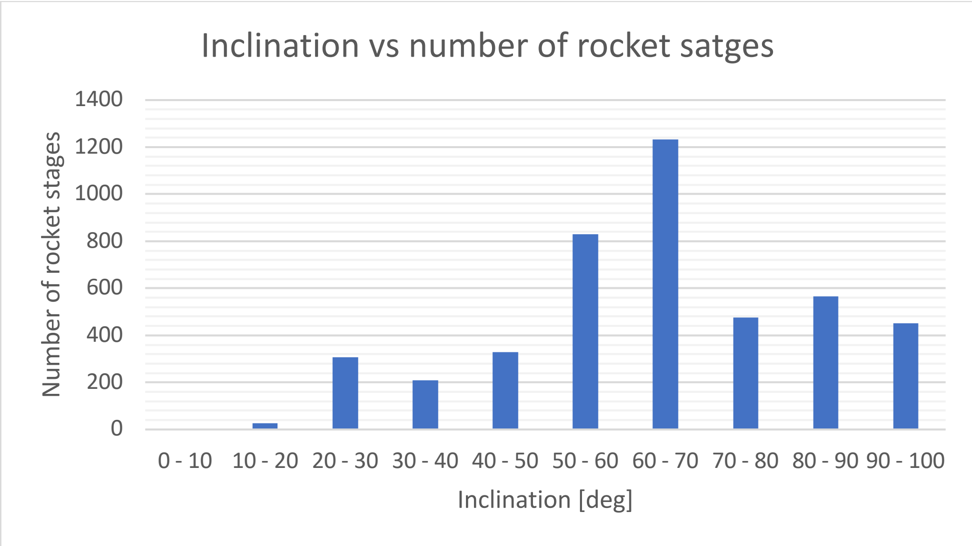
Altitude vs number of rocket stages table



Altitude vs number of rocket stages graph

Inclination interval [deg]	Count
0 - 10	3
10 - 20	26
20 - 30	308
30 - 40	210
40 - 50	330
50 - 60	831
60 - 70	1233
70 - 80	475
80 - 90	566
90 - 100	451

Inclination vs number of rocket stages table



Inclination vs number of rocket stages graph

The data differs from the dead satellites category. The graphs suggest that the altitude with the most rocket stages is below 400 km. This is likely because a lot of the objects were ejected at the beginning of the space race at relatively low altitude and have descended meanwhile. Furthermore, there are more than twelve hundred rocket stages circulating the Earth with inclination between 60 and 70 degrees.

Market research

Opportunities

As there is no incentive for businesses to fund a space debris removal missions as of now, funding of these missions will typically come from governmental bodies or space agencies. According to 2012 ISU Symposium in Strasbourg France on space debris removal: "*Today not only is there no mechanism in existence to pay for removal of space debris but there is no mechanism that could decide what is a fair payment for such activities and whether there are different payments for those removal of space debris from different orbits, etc. Innovative thought and national initiative by one or more countries or a region could help show the way forward.*"

For example, ESA has signed an €86 million contract with an industrial team led by Swiss start-up ClearSpace to purchase a unique service: the first removal of an item of space debris from orbit.

As a result, in 2025, ClearSpace will launch the first active debris removal mission, ClearSpace-1, which will rendezvous, capture and take down for reentry the upper part of a Vespa (Vega Secondary Payload Adapter) used with Europe's Vega launcher. This object was left in a "gradual disposal" orbit (approximately altitude 801 km by 664 km), complying with space debris mitigation regulations, following the second flight of Vega in 2013.

Paying for such a service contract rather than directly procuring and running the entire mission represents a new way for ESA to do business – intended as the first step in establishing a new commercial sector in space. [11]

Recently the UK Space Agency has released a new funding up to 800 000 GBP to support new space cleaning start-ups. [12]

This suggests that the number of funding opportunities as well as the budgets will rise in the future. In general, there are three ways how incentives for space debris removal might emerge in the future:

Regulation

National or international policy can require end-of-life disposal. The same policies could add stricter consequences for non-compliance. This, however, requires a delicate balance for two reasons. First, it is unclear which governing body has the authority and capability to create a universal policy. Second, overly harsh requirements and consequences could do as much harm to orbital innovation as space debris. Still, ideas like Aerospace Corporation's de-orbit credit trading scheme may allow the market to determine an appropriate balance on regulation. [13]

Liability insurance

Already today satellite operators are theoretically responsible for damage caused by satellites they launch. The United Nations' "Convention on International Liability for Damage Caused by Space Objects" established that nations are liable for damages caused by satellites launched from their soil.
In the U.S., the first $500 million of that liability is passed directly to satellite operators. Insurance is sold to cover the potential damage a satellite could cause as debris. The cost of that insurance is only continuing to go up, and eventually it will be more affordable to de-orbit than pay for insurance indefinitely. [13]

Cleaning own orbit

Space debris is often considered a tragedy of the commons, where the pain is spread across too many parties for anyone to pay for debris removal. For many mega-constellations though, their debris primarily threatens their own satellites. As large satellite constellations are deployed, they become incentivized to protect their own infrastructure and ensure the disposal of their own satellites.
One solution is using a space tug to assist satellites. This is a task for CzechMate.
Other companies such as Astroscale and ClearSpace are pursuing space tugs for debris removal too.
The value of CzechMate could revolve around being a supplemental disposal method for an entire constellation. "It turns out that using an independent space tug for disposal maximizes the value of an entire satellite constellation. Having a guaranteed back-up disposal method allows constellations to operate through more risk than they could otherwise tolerate.
To demonstrate that on an example, let us take a constellation with 100 satellites. Suppose the satellites have a 100% chance of surviving to five years, but degradation means only 90% will last seven years. Suppose also that each satellite is generating $1 million in value per year.
As a baseline scenario, all of the satellites would de-orbit after five years. We will call this scenario A.
Now, let us add a space tug to the mix. Suppose a space tug can de-orbit these satellites for $5 million per satellite. In this scenario, the constellation satellites could operate for seven years and use a space tug to dispose of the 10% that do not last to dispose on their own. We will call this scenario B.
Now compare the two scenarios. In scenario A, no added value is possible. All satellites must be decommissioned and removed before they run out of fuel.
In scenario B, although it costs $50 million to use the space tugs, it also adds $180 million in value from extended satellite operations. Overall, using a supplementary disposal service is a $130 million windfall for the constellation." [13]
Scenarios such as this show that space debris removal is more than a nice environmental mission. It is good business that can provide direct value to commercial satellite customers.
It is also possible an incentive will emerge that we have not recognized yet. [13]

Risks

According to Blees et Al. in their research report called "Barriers to Entry Differences in barriers to entry for SMEs and large enterprises" (2003), firms will enter the market only if the profit will give them a long-run competitive level. But several mechanisms can prevent firms from entering the market. There can be barriers to their entry that do not go along with the industry benefits. One of the main barriers preventing the CzechMate entering the market is the legal and political framework which limits the spacecraft's capabilities to almost exclusively European satellites.

The legal and political barriers

Space Law includes five major treaties; two of them are addressing the Active Debris Removal issues:

- The Treaty on Principles Governing the Activities of States in the Exploration and Use of Outer Space - the Outer Space Treaty (OST) of 1967
- The Convention on International Liability for Damage Caused by Space Objects of 1973 (the Liability Convention), defining the liable party as the "Launching State".

Paul Kallender-Umezu, in his article called "*A market for cleaning up space junk?*" analysed the main weaknesses of these two treaties. According to him, one of the main issues is that there is no clear definition of space debris; nor any binding legal definition of them. But the most important problem is that there are no regulations mentioning who has to remove debris. Indeed, there is no legal document or treaties that clearly explain who must remove their satellites, or their fragments, and the space law did not mention any obligation to do so. The debris and fragments are considered as individual objects. Then, in the case of an active debris removal process, these two treaties (the Liability Convention & the OST) do not tell who is responsible if a third party like CzechMate removes a piece of debris by mistake, or cause trouble to a piece of debris which can damage another satellite. [14]

Technical and economic barriers

In addition to the political and legal framework that needs to be consolidated, there are also some technical and economic barriers that could prevent CzechMate being used for space debris removal purpose. Although the exact concept of space debris removal mechanism for Czechmate is still unknown and in the early design phase, most of the potential debris removal concepts considered rely on one or more systems that are unproven technology. In space activities, Technology Readiness Level (TRL) is used to assess the maturity of evolving technologies (devices, materials, components etc.). A new technology is not suitable for immediate application. Indeed, the system/technology is subjected to experimentation, maturation and realistic testing of the concept before getting the qualification and authorization to go in space (Emanuelli et Al. 2013). This TRL procedure can be long (scale going from 1 to 9) [14], and development costs are very high in order to reach the TRL 9.
Although some space debris removal systems are already in development and for example nets were successfully tested, high non-recurring costs need to be taken into consideration. [15]
According to the definition of Lisa Guerra from NASA: "Non-recurring costs include all costs associated with the design, development and qualification of a single system. Non-recurring costs include the breadboard article, engineering model, qualification unit and multi- subsystem wraps". Then, once the first CzechMate debris removal system would have been in space to prove the concept feasibility, the follow-on mission will only support the recurring costs, which represent the costs associated with the production in serial of the approved system (Lisa Guerra, 2008).
The implementation of a space debris removal on CzechMate system would include these non-recurring costs related to the development of the technology maturation, but also to the mission cost.

As an example, SSTL (a satellites' manufacturer) has estimated the cost model to remove Envisat with the chaser concept. The prices reach around €65M per debris removed with a chaser, and €30M per debris removed via a Shuttle. To these removal costs, €135M need to be added to finance the initial spacecraft manufacturing to launch the system, and the price of the launch itself (SSTL, 2013).
Therefore, development costs are too high to be supported by equipment or satellites' manufacturers themselves. If technical barriers can be easily overcome, economic barriers will require the financial support of institutions to make industries involve in this market. [14]

Market analysis

Several companies have been increasingly offering active spacecraft-based deorbit systems. Space startups such as AstroScale, ClearSpace, and D-orbit have long-term plans and have already started initial technology demonstrator missions. These systems consist of separate, dedicated spacecraft that attach to decommissioned satellites to place them into decaying or graveyard orbits. In December 2019, Iridium stated that they would like to pay for an active deorbit system to remove 30 of their defunct satellites. [16] In addition, NASA STD-8719.14A stipulates that all spacecraft using controlled reentry processes have to be within 370 km of the target when landing. [17] Therefore, future concepts such as sample return missions are going to need active re-entry devices to satisfy these requirements.
This section covers some of the main stakeholders in the industry that are working towards the implementation of active space debris removal, as well as some other promising technologies that can potentially be used for actively deorbiting spacecraft in the future. [18]

TechEdSat Series Exo-brake

The exo-brake has passive systems and also active control capability. The TechEdSat-6 mission was the first one implementing this technology, on a 3,5U CubeSat with a mass of 3,51 kg that deployed its exo-brake from the rear of the satellite. It targeted a reentry over Wallops Flight Facility by modulating the drag device to adjust the ballistic coefficient as orbital determination about the satellite state became available over time. The Iridium gateway enabled the command of the brake, which proved to significantly affect the reentry time and consequently, the location of the Wallops target area. The spacecraft overshot the intended target range slightly, since it could not achieve a lower 4 – 5 kg m^{-2} ballistic coefficient configuration, which would have yielded suitable results if placed at 300 km. However, the mission demonstrated successfully the reentry experiment and the command/control capability by overflying Wallops right before reentering. This technology was going to be demonstrated again in the TechEdSat-8 mission. Although the exo-brake was successfully deployed, a power system failure occurred before the targeting process started. The TES 8 exo-brake was an improved version of the previous TES 5 and TES 6 devices. The ballistic coefficient range was larger (6 – 18 kg/m^2) which allows better control authority for targeting. TES 10 and upcoming TES 11 are also incorporating this design. [19] [18]

RemoveDebris Consortium Partners

The RemoveDebris mission carried two 2U CubeSats that were ejected from the mothership to simulate space debris and demonstrate active deorbit capabilities. The first CubeSat, known as DebrisSat-1, deployed at a very low velocity from the main spacecraft and subsequently inflated a balloon that provided a larger target area. A 5 m diameter net was ejected from the main spacecraft just 144 seconds after deployment, capturing the CubeSat at a distance of ~11 m from the mothercraft. The object, once enveloped in the net, re-entered the atmosphere in March 2019. The RemoveDebris mission also carried another active debris technology consisting of a harpoon. In this scenario, a target platform attached to a boom was deployed from the main spacecraft. The mothership then released the harpoon at 19 m/s to hit the platform in the center. Once that occurred, the 1.5 m boom that connected the two objects snapped on one end. However, a tether secured the target in place, avoiding the creation of new debris. This resulted in the first demonstration of a harpoon technology in space. The harpoon target assembly had a dry mass of 4,3 kg. [17] [18]

Astroscale

Astroscale is a company founded in Japan and with offices in the UK, the US, and Singapore. Their two main objectives are to provide services to address the end-of-life scenario of newly launched satellites, and to proactively remove existing space debris. They collaborate with a variety of governmental and international organizations around the world (such as the US government, ESA, the European Union or the United Nations) in order to position themselves as leaders of a more sustainable low-Earth orbit environment. [18]
As part of the EOL campaign, the ELSA-d mission, which is scheduled to launch in 2020, will consist of two spacecraft, with one acting as a „servicer“ and the other as a „client“. They will have launch masses of 180 kg and 20 kg respectively. The concept of operations is to perform rendezvous maneuvers by releasing the client from the servicer repeatedly in order to demonstrate the capability of finding and docking existing debris. The technology demonstrations will include search and inspection of the targets, as well as rendezvous of both tumbling and non-tumbling cases. [20]
Regarding their active debris removal campaign, Astroscale is also working with national space agencies to incorporate solutions to remove critical debris such as rocket upper stages or defunct satellites. This campaign started with a partnership with the Japanese Space Agency (JAXA) in February 2020. This collaboration will result in the implementation of the Commercial Removal of Debris Demonstration project (CRD2) which consists of the removal of a large space debris object performed in two mission phases. Astroscale will be involved in the first part, with a satellite that identifies and acquires data from an upper stage rocket object from Japan. The company is responsible for manufacturing and operating the satellite to complete these tasks, with a planned demonstration in 2022. [20]
In June 2020, Astroscale acquired the intellectual property of the Israeli company Effective Space Solutions. This company developed the Space Drone servicing vehicle, which is capable of providing active debris removal. The Space Drone will mature into an Astroscale program. [21] [18]

ClearSpace

ClearSpace is a Swiss company founded as a spin-off from the Ecole Polytechnique Federale de Lausanne (EPFL) research institute. Their plans also include service contracts for active debris removal. One of their proposed missions, ClearSpace One, which has been backed by ESA, will find, target, and capture a non-cooperative, tumbling 100 kg VESPA (Vega Secondary Payload Adapter) upper stage. The chaser spacecraft will be launched into a 500 km orbit for commissioning and initial testing before raising its altitude to 660 km where the VESPA is located, where it will attempt rendezvous and capture. ClearSpace One will use a group of robotic arms to grab the upper stage and then both spacecraft together will be deorbited to a lower orbit for a final disintegration in the atmosphere. The mission is planned to launch in 2025 to help establish a market for in-orbit servicing and debris removal. [22] [18]

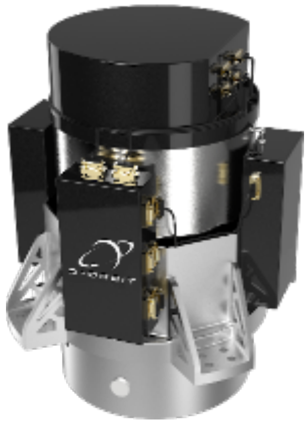
Illustration of ClearSpace 1 spacecraft [23]

Momentus

Momentus is a company founded in 2017 and based in California that operates space transportation systems that can propel or deorbit other spacecraft. Their Vigoride platform is capable of carrying satellites with masses up to 250 kg. With a wet mass of 215 kg, it is capable of providing up to 1,6 km s^{-1} for 50 kg payload, through a water plasma propulsion system. Although the main objective of this system is to provide enhanced propulsive capability to their customers, the platform is suitable for active deorbiting. Momentus has booked several Vigoride missions on Falcon 9 launches through 2020 and 2021. [18]

D-orbit

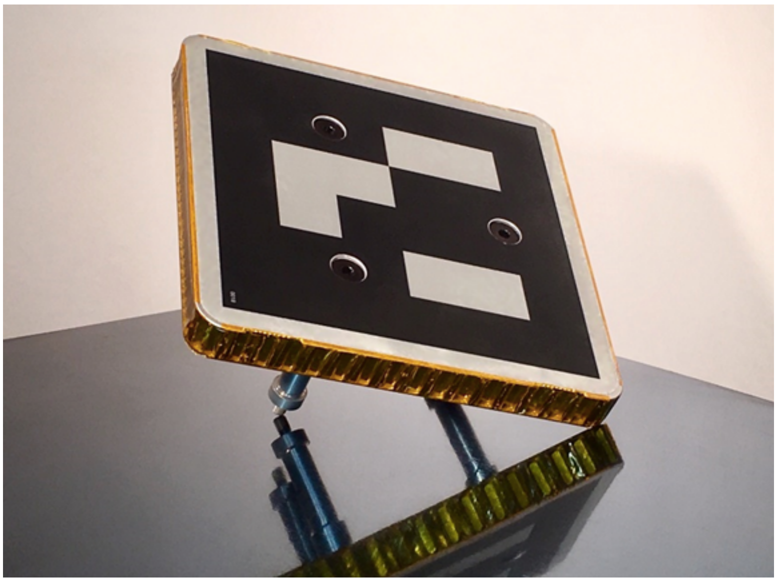
D-orbit is a space transportation company founded in 2011 in Italy, with subsidiaries in Portugal, the United Kingdom, and the United States. It provides transportation services onboard their ION CubeSat carrier platform that can provide precision deployment and is able to host satellites from 1 to 12U. Two initial flights are already scheduled for 2020 onboard the ArianeSpace Vega SSMS POC flight and the SpaceX Falcon 9. The first mission will carry 12 Doves from the Earth-observation company Planet, and future versions of this technology will consider other applications such as retrieving orbiting spacecraft to deorbit them. In addition, D-orbit provides an external solid motor booster specifically for deorbiting purposes. This independent module, known as D-Orbit Decommissioning Device (D3) shown in the figure below, is a proprietary solution that is optimized for end-of-life manoeuvres. [24] [18]



D-Orbit D3 module [18]

Altius Space Machines

In 2019, the satellite constellation company OneWeb signed a partnership with Altius Space Machines from Boulder, Colorado, to include a grappling fixture on all their future launched satellites in an effort to make space more sustainable. The Altius DogTag consists of a universal interface for small satellites that is inexpensive and lightweight. The fixture design enables various grappling techniques to enable servicing or decommissioning. It uses magnetic capabilities as its primary capture mechanism but is also compatible with other techniques in an effort to accommodate other potential customers and act as a standard interface. [25] More specifically, it is compatible with magnetic attraction, adhesives, mechanical, and harpooning captures. Figure below includes an image of the prototype. [18]



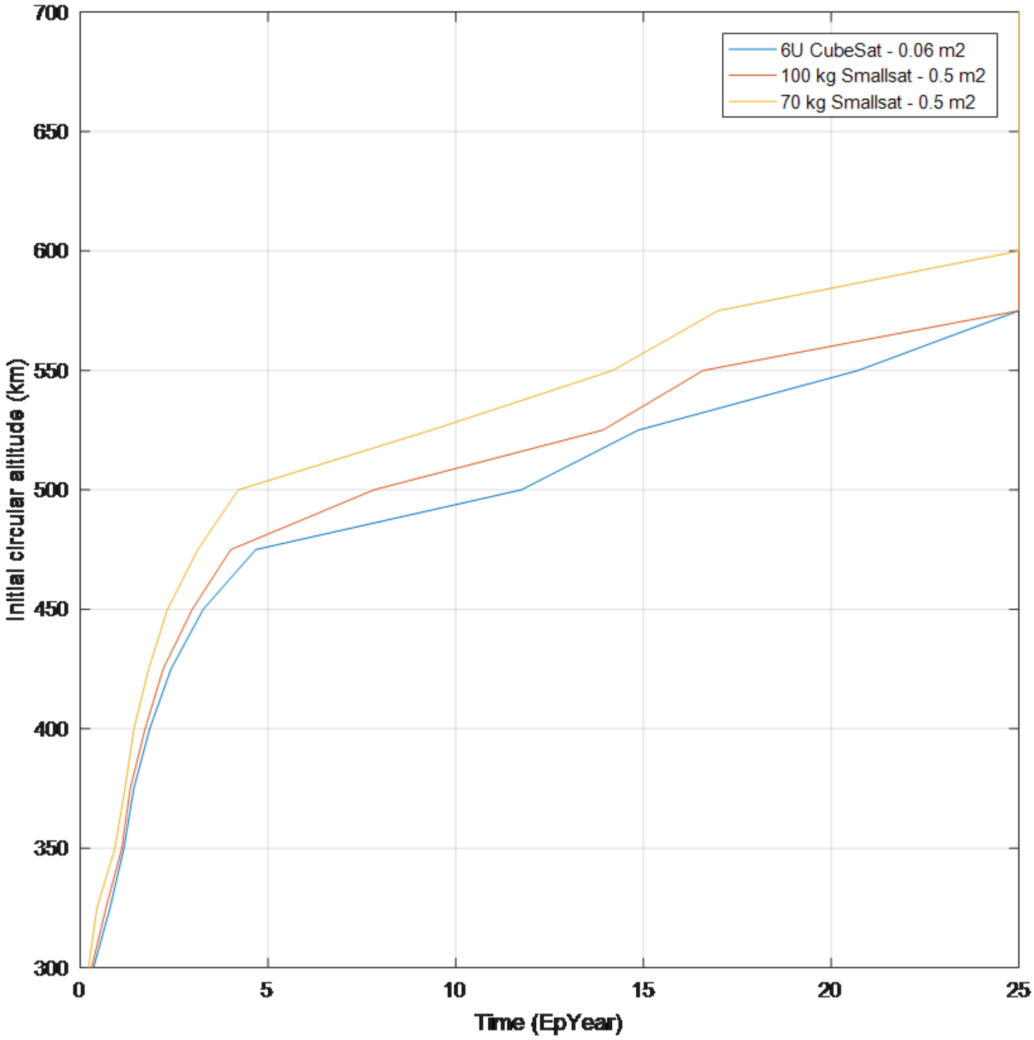
DogTag prototype [26]

Market analysis - summary

Active systems that include commanded and modulated systems, as well as independent servicing spacecraft, are also maturing and will play a fundamental role in the upcoming years. A version of the Exo-brake with pointing capabilities has been demonstrated in the TechEdSat-6 mission, while the RemoveDebris mission has successfully tested two different active methods, a net and a harpoon, for future implementation in active debris removal operations. Companies such as Astroscale, Momentus, D-Orbit, or ClearSpace are already developing and planning to launch servicing spacecraft that can attach to decommissioned satellites to bring them down to a graveyard orbit or disintegrate in the atmosphere. In conclusion, this technology has increased significantly in maturity since the last iteration of this report and is expected to grow as the demand for deorbiting services increases with additional launches. [18]

Ways of space debris removal

Small spacecraft launched at or around the International Space Station (ISS) altitude (400 km) naturally decay in well under 25 years. However, at orbital altitudes beyond 800 km, there is no guarantee that a small spacecraft will naturally decay in 25 years due to uncertainties in atmospheric density and the differences in ballistic coefficient, as seen in figure below. [18]



Graph of initial circular altitude vs time to deorbit 2020 [18]

In this image, a representative 6U CubeSat with 0.06 m2 drag area and 14 kg of dry mass decays at different rates depending on several initial circular orbits. The results differ from those achieved with another representative spacecraft of 100 kg and 0.5 m2 of drag area, showing the important effect the ballistic coefficient plays in the orbit propagation. The majority of launched small spacecraft do not carry on-board propulsion, making them unable to achieve graveyard orbits for decommissioning. Therefore, they need to rely on deorbit techniques such as increasing the drag area by rotating the spacecraft with their Attitude Determination and Control System (ADCS) module if they are in low altitudes. For some spacecraft, their exposed drag area is not enough to meet the 25-year requirement. They can use deorbit devices such as drag sails (passive systems) or even hire external deorbit services (active systems), in order to deorbit. This is a “blue ocean” for the CzechMate system. [18]

NASA’s Space Debris Elimination (SpaDE) program is investigating the possibility of firing pulses of atmospheric gasses in front of targeted debris, slowing their speed, and increasing the rate at which the debris deorbits and falls back to Earth, generally burning up in the atmosphere. [3]

Active debris removal involves the rendezvous and docking with, and ultimately removal of, an item of space debris. There are several types of technology that could be used for docking.

Mechanical

Mechanical solutions to grapple space objects have long been used, for example the Canadarm on the Space Shuttle and ISS. Simpler models such as 'tentacles' or claws have also been proposed. Mechanical manipulators are the most technically advanced solution to capturing space objects, involving advanced robotics and many moving parts. Mechanical solutions can be specialised to a single task, such as docking with a liquid apogee engine seen in the MEV-1 mission, or more general. However, increasing complexity increases the risk of failure and the cost of such a device. For the more delicate manipulation required in more advanced in orbit servicing, they are a requirement, but for simply capturing space debris, other solutions may be preferable, depending on the target. [27]

Magnets

Astroscale is using magnets in its demonstration mission ELSA-d. This requires fewer moving parts (and hence failure modes) and is likely to be cheaper than a mechanical mechanism. However, it requires a magnetic surface on the target satellite. Furthermore, the magnetic surface must be positioned in such a way that the approaching satellite can manipulate the two satellites once connected. It is possible with specially made magnetic docking points but these would have to be integrated into the design of the spacecraft before launch. [27]

Nets

Nets have been proposed as a way of capturing, in particular, non-cooperative debris. The capturing spacecraft would not dock with the debris, and could capture from a distance. This technology was tested as part of the RemoveDEBRIS mission, where a target piece was ejected from the satellite, before being captured by the net. However, this has some downsides. The unpredictable capture of the satellite means it could be positioned in a way that makes manoeuvring once captured difficult. Furthermore, there is only one chance to capture the debris, and any failure could exacerbate the debris problem. [27] [28] [29]

Harpoons

Harpoons have been proposed as a mechanism for capturing space debris. Indeed, the RemoveDEBRIS mission tested a harpoon, firing it into
a target 1,5 m away. Use of a harpoon, as with a net, means there is reduced risk from collision. However, harpoons have significant risk. Only one attempt at capture can be made, there is a significant risk of creating more debris, and once captured the target may be difficult to manipulate. [27]

Lasers

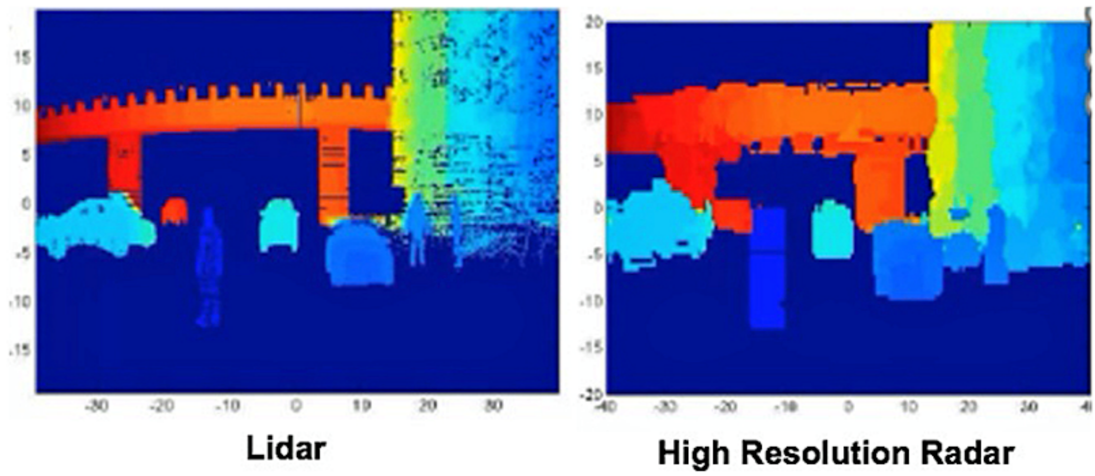
NASA's Space Debris Elimination (SpaDE) program is investigating the possibility of firing pulses of atmospheric gasses in front of targeted debris, slowing their speed, and increasing the rate at which the debris deorbits and falls back to Earth, generally burning up in the atmosphere. [9]
All of these schemes will be very expensive and use technology that is still to be developed. The laser scheme looks as if it might be the most promising. However, in this case, political problems might outweigh technical problems.

De-orbiting of large-scale objects

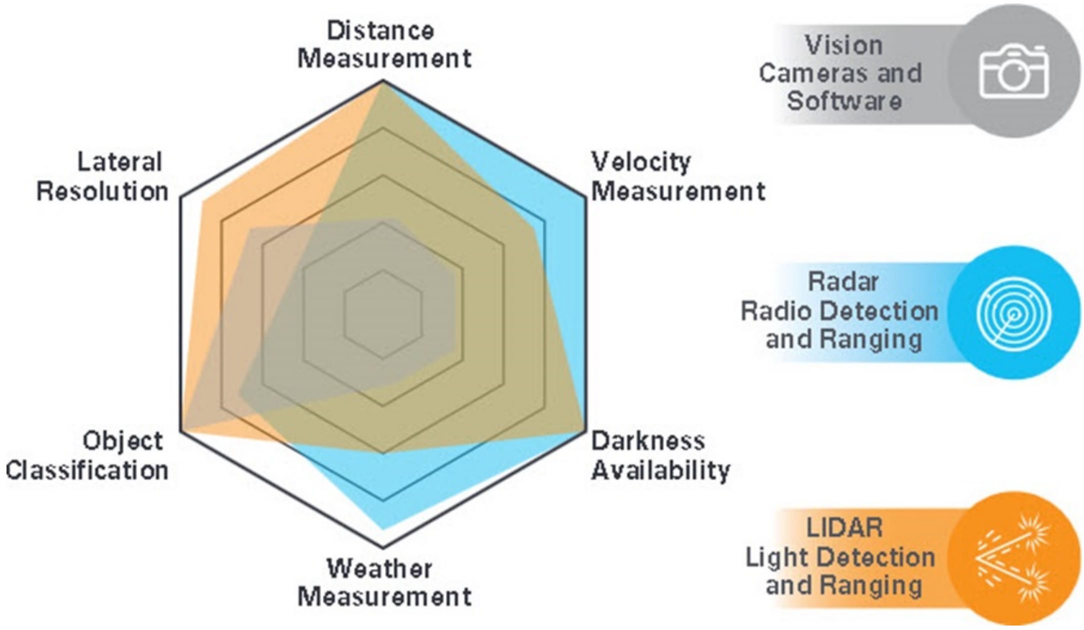
Large scale objects such as dead satellites in orbit are especially dangerous because they are a potential source of space debris. On the other hand, these objects are fairly well mapped and can be tracked easily due to their size. Therefore, removal of such objects could be one of the main tasks of a space debris removal mission.
Large satellites as for example the Envisat do not require a sophisticated technology to deorbit them and even a small momentum change in the correct direction is sufficient for the atmospheric drag to increase insofar that the satellite slows down, loses altitude, and eventually falls and burns in the atmosphere. Time that it takes for an object to descend from a certain altitude is a function of many variables such as the mass of the object, its cross-sectional area, drag coefficient, atmospheric rotation.

Appended Figures

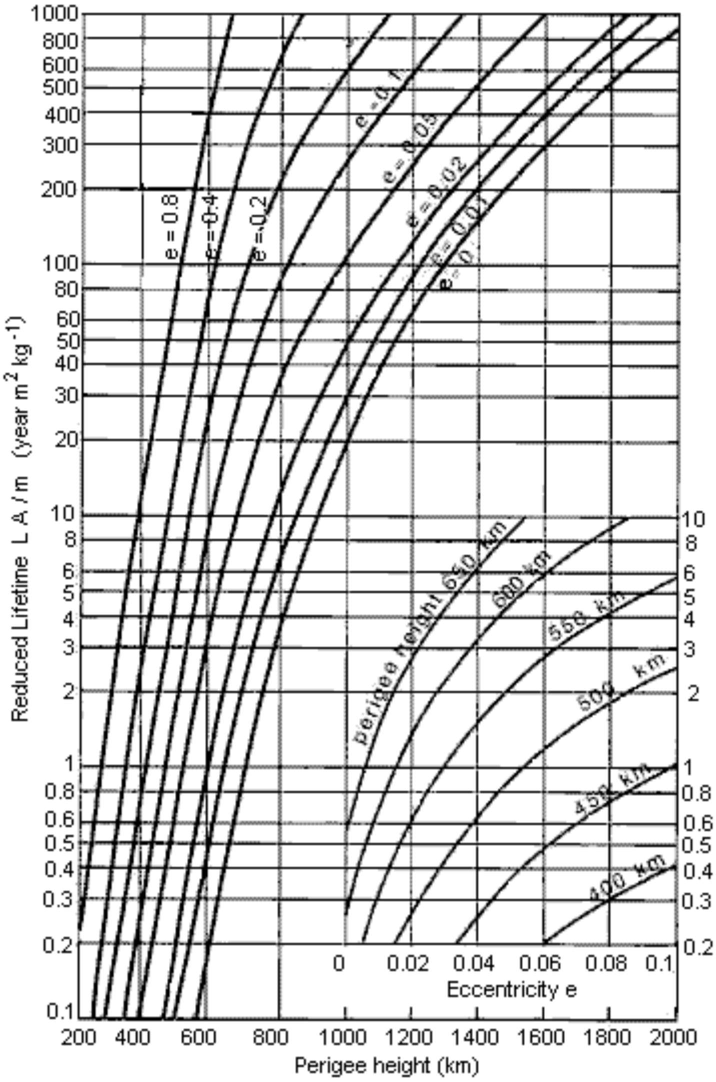
Payload report



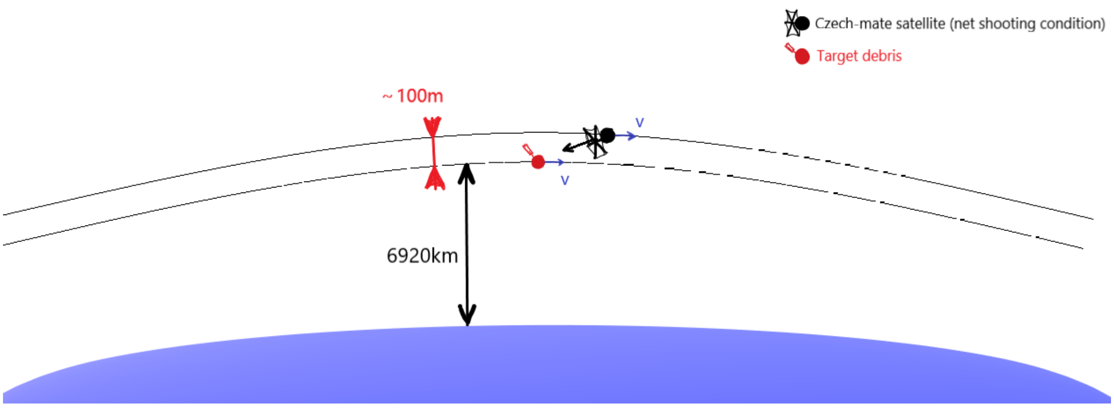
Appendix Figure 1. The comparison of resolutions between a Lidar and a High-Resolution Radar. [30]



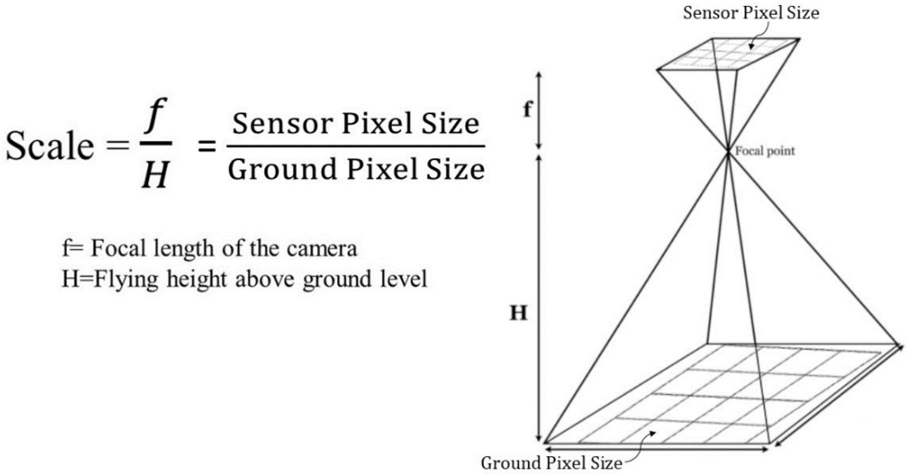
Appendix Figure 2. The hexagon diagram comparison of Camera, Radar and Lidar. [31]



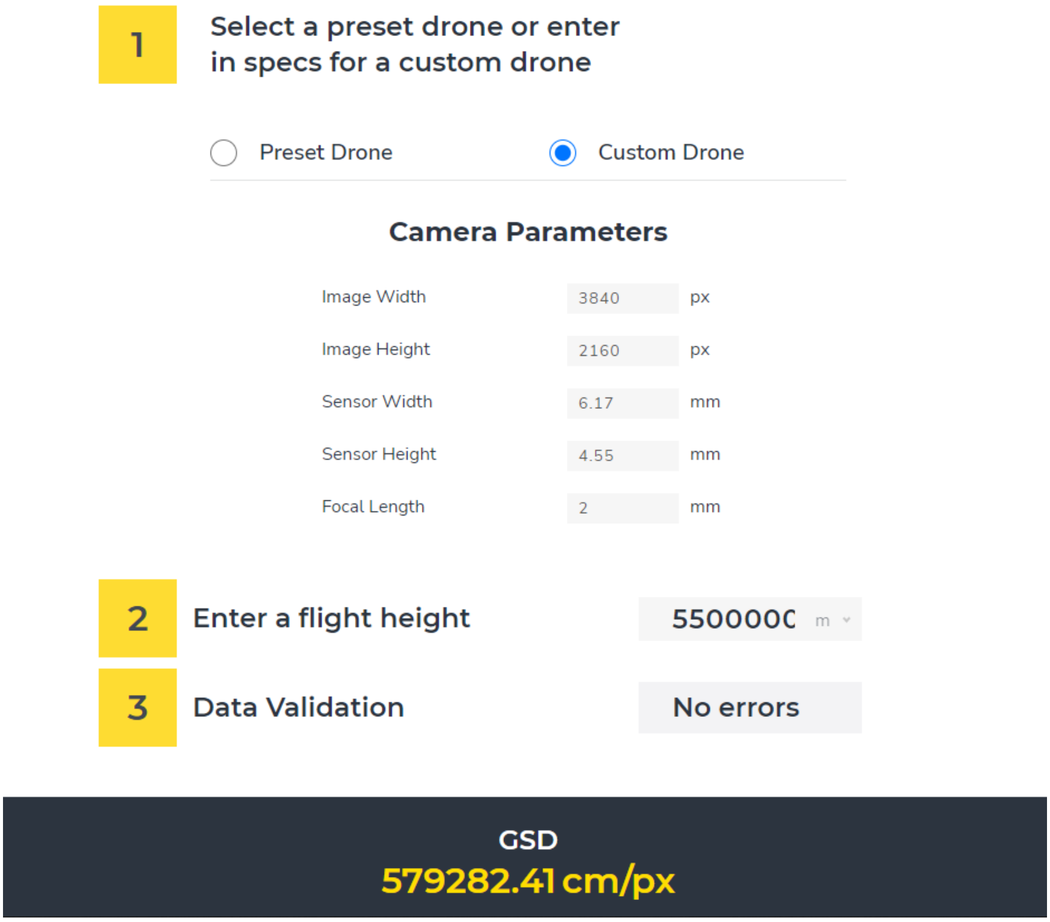
Appendix Figure 3. a normalised or reduced orbital lifetime (L) for a wide range of perigee heights and eccentricities.* [32]



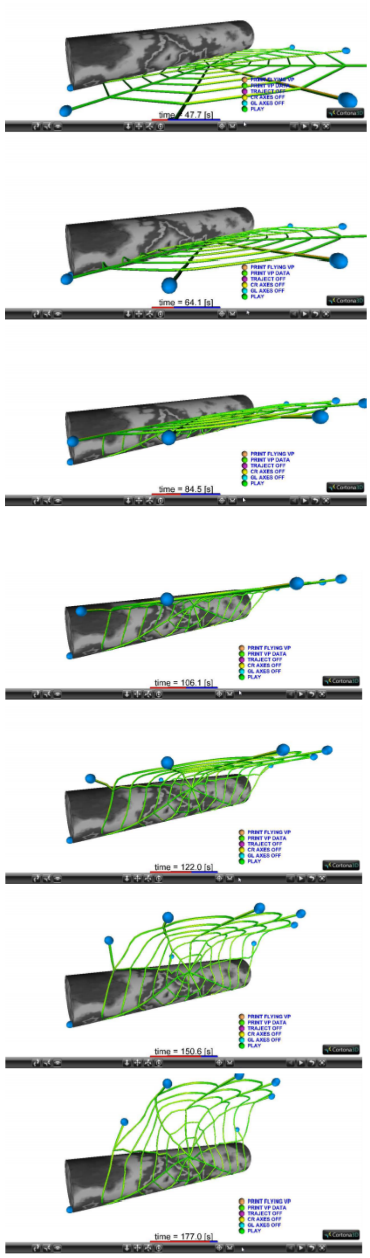
Appendix Figure 4. The net is shot backwards, opposite to the travelling direction.



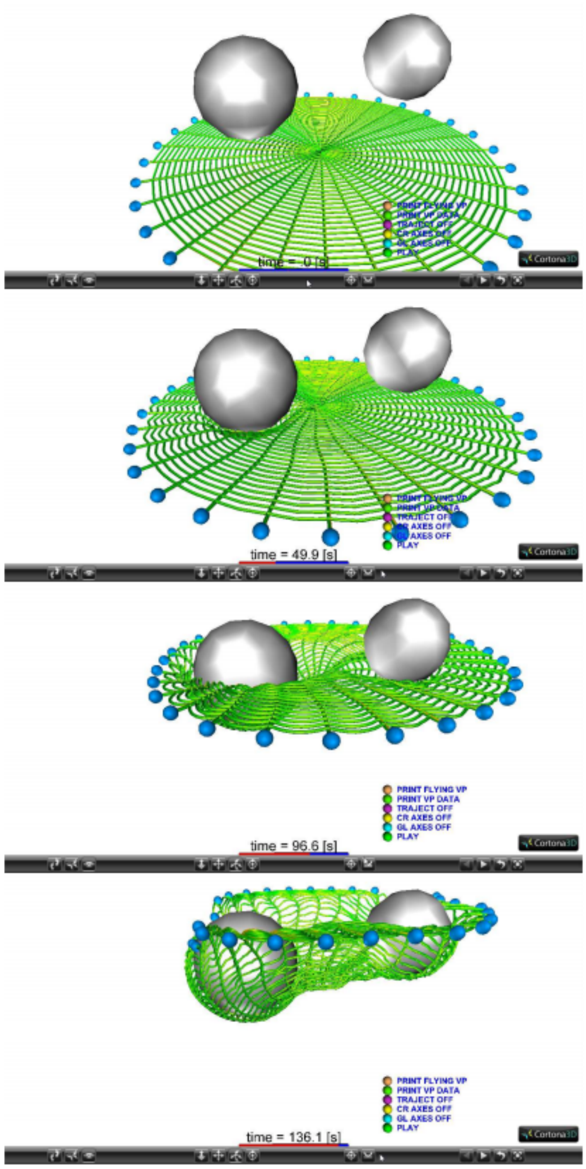
Appendix Figure 5. The theory of GSD.



Appendix Figure 6. The GSD calculator. [33]

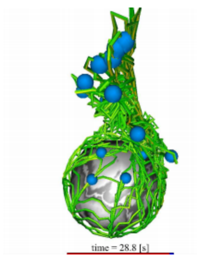


(a)



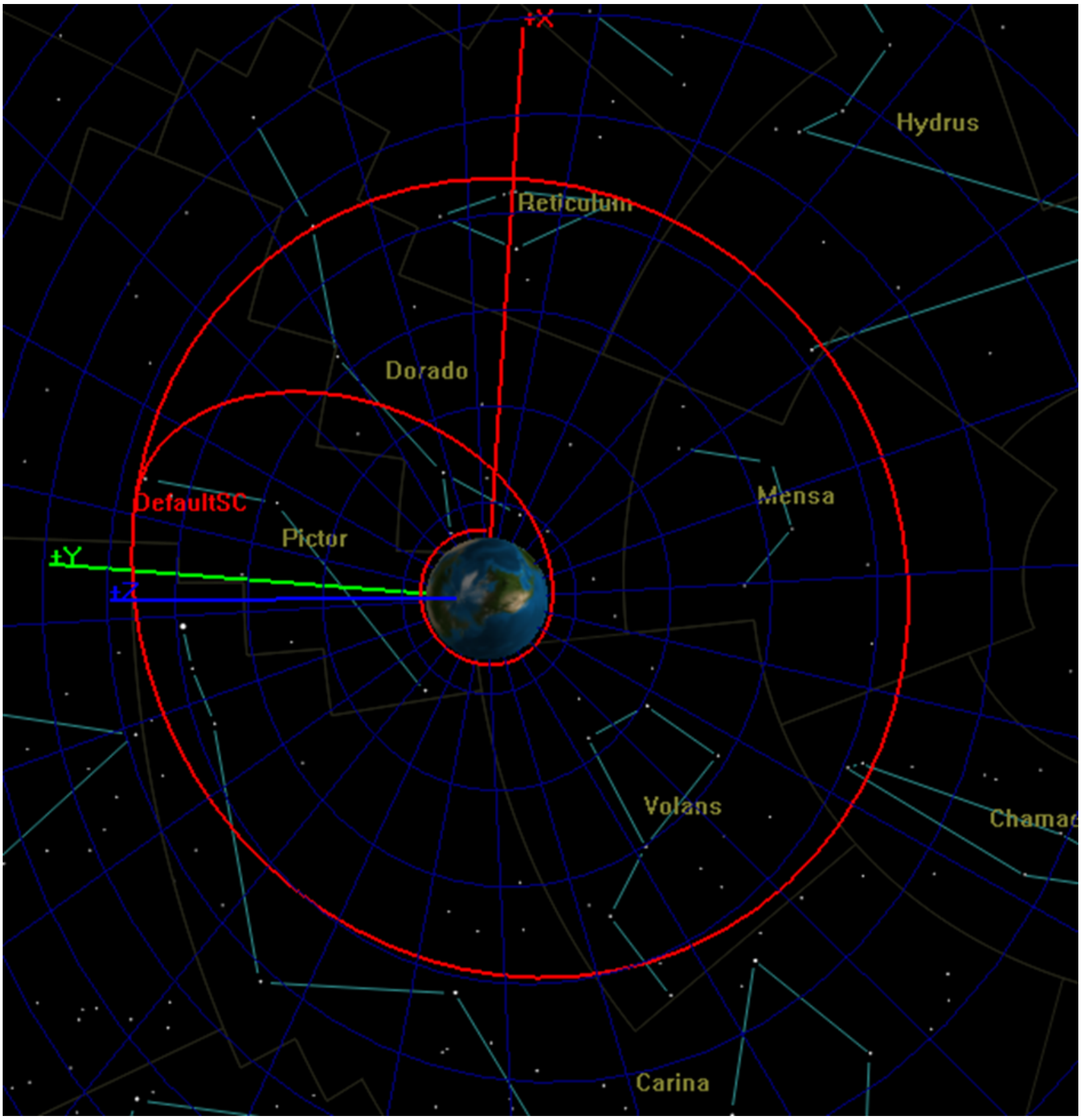
(b)

Appendix Figure 8. The research on the impact situation.
(a) Time history of the collision of the flying web with a long Cylinder.
(b) Collision of the flying web with two spherical debris. [34]

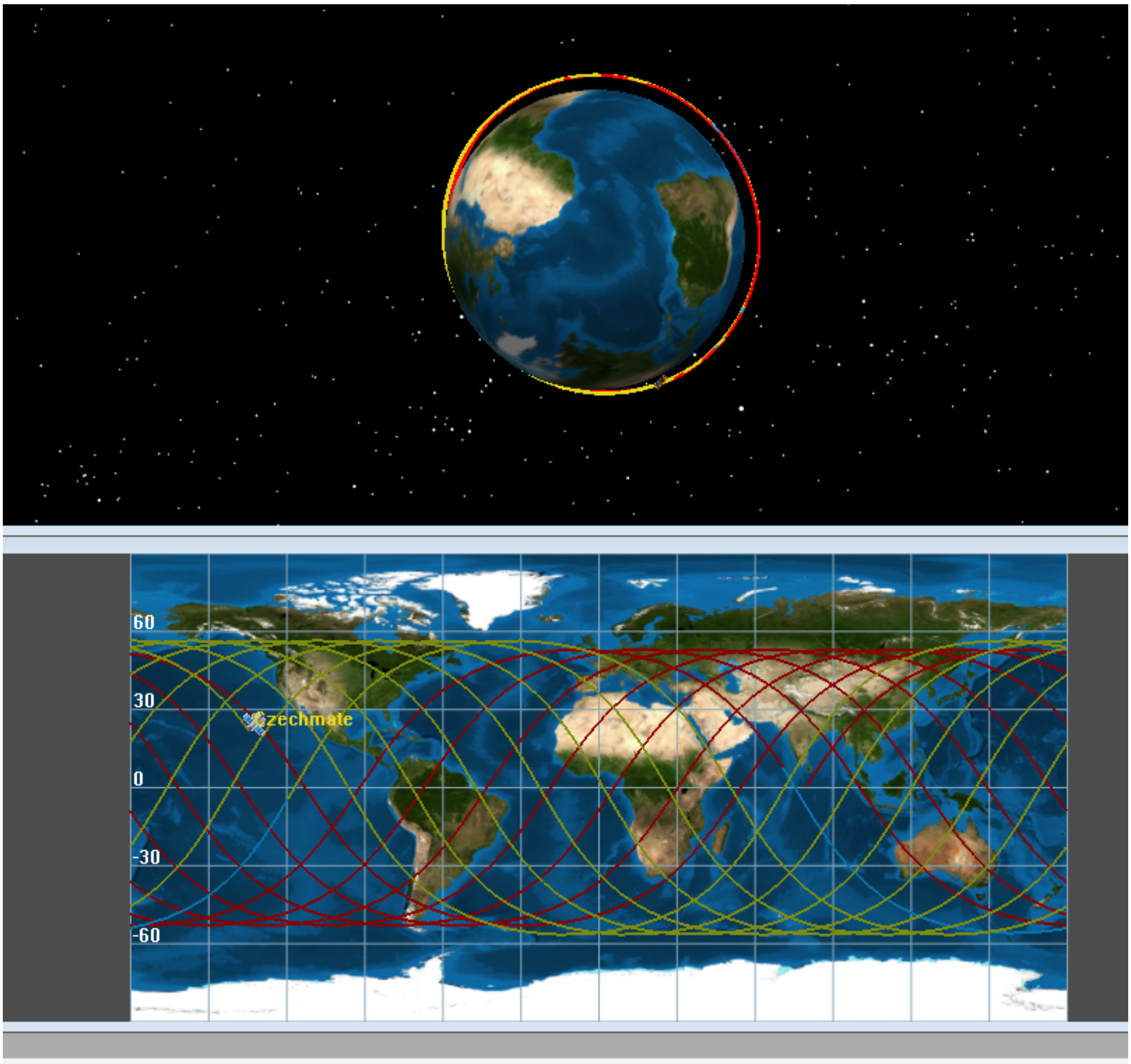


Appendix Figure 9. Non-rotating flying net wrapping an off-set spherical obstacle. [34]

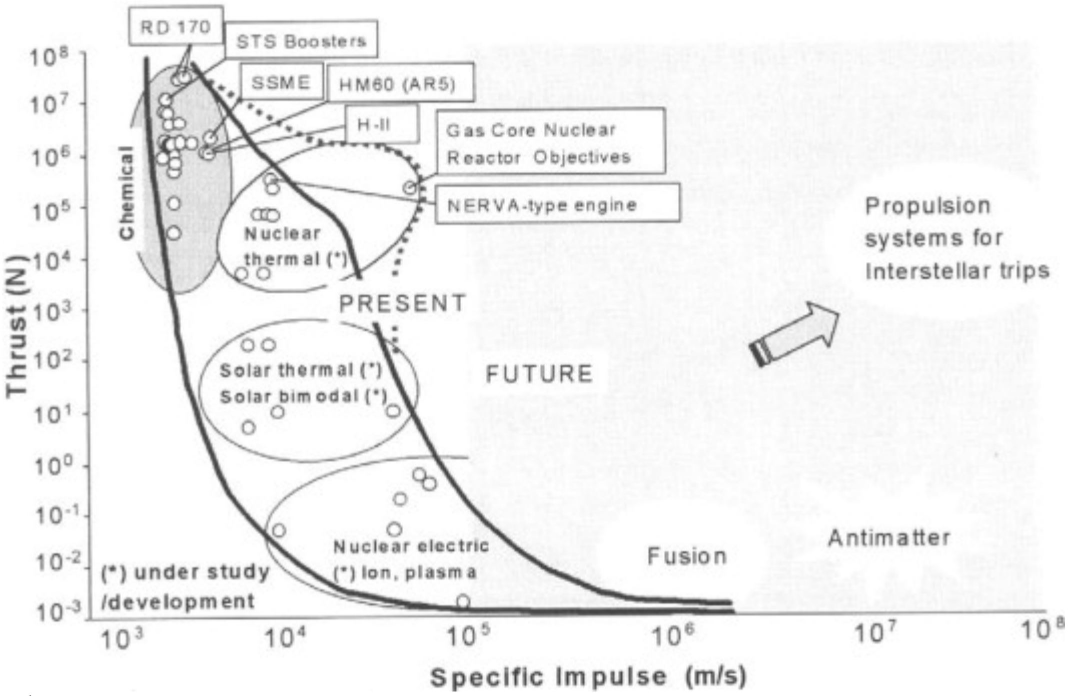
Mission analyst report



1st picture



2nd picture



3rd picture

Electrical engineer report

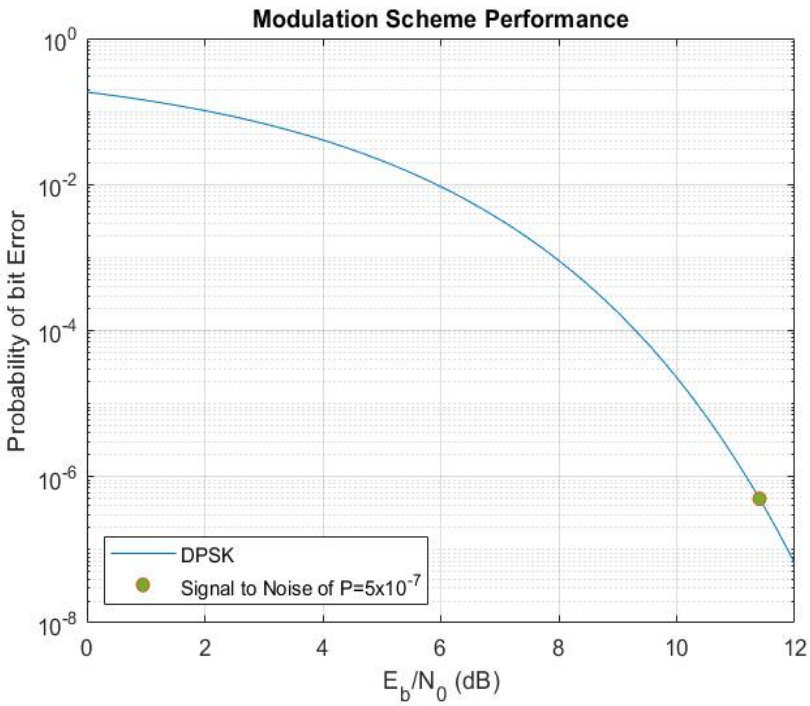


Fig. 2.1 – Modulation Scheme Performance

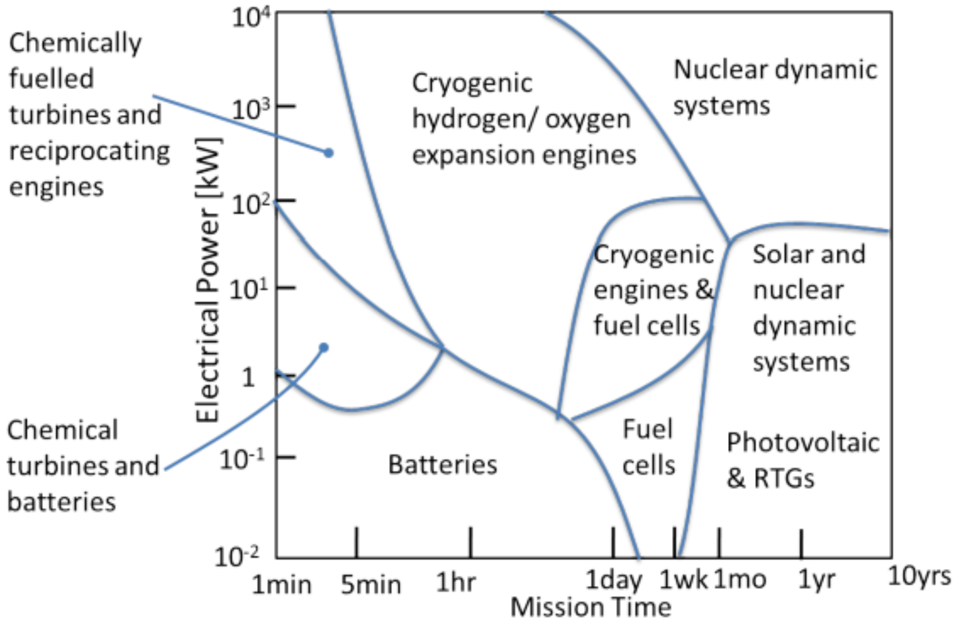


Fig 4.1 - Typical operational envelopes of primary power sources [3]

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