# Singapore Space Challenge 2019

## Optimum Debris De-orbiting System (ODDS)

This mission report encompasses the description and details of Team ODDS' solution to the problem of clearing up space debris in Low-Earth Orbit by using a group of satellites working together. Cost effective, efficient, and at the same time a novel technique to resolve the problem was the endeavor. Our proposed solution consists of a satellite acting as a mothership, and two smaller satellites called deorbiters which will be responsible for targeting and removing space debris using the Ion Beam Shepherd Method.

Team ODDS

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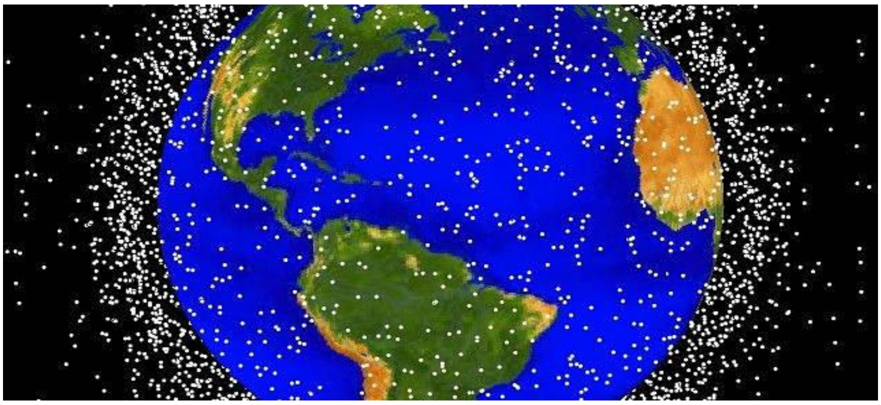
## A) Overview

### 1. Introduction to Problem

Space debris generally refers to all non-functioning objects, which includes non-operating satellites, fragments of spacecraft and other elements moving around in the earth's orbit. These objects can be found across almost all major, defined orbits around the earth[1]. Ever since the first space flights took place back in 1957, space debris has been growing in presence and volume.



Space debris can be broadly categorised into two forms, based on whether their existence can be tracked back to a specific event or not. The 'Identified' kind of space debris is as follows[2] :

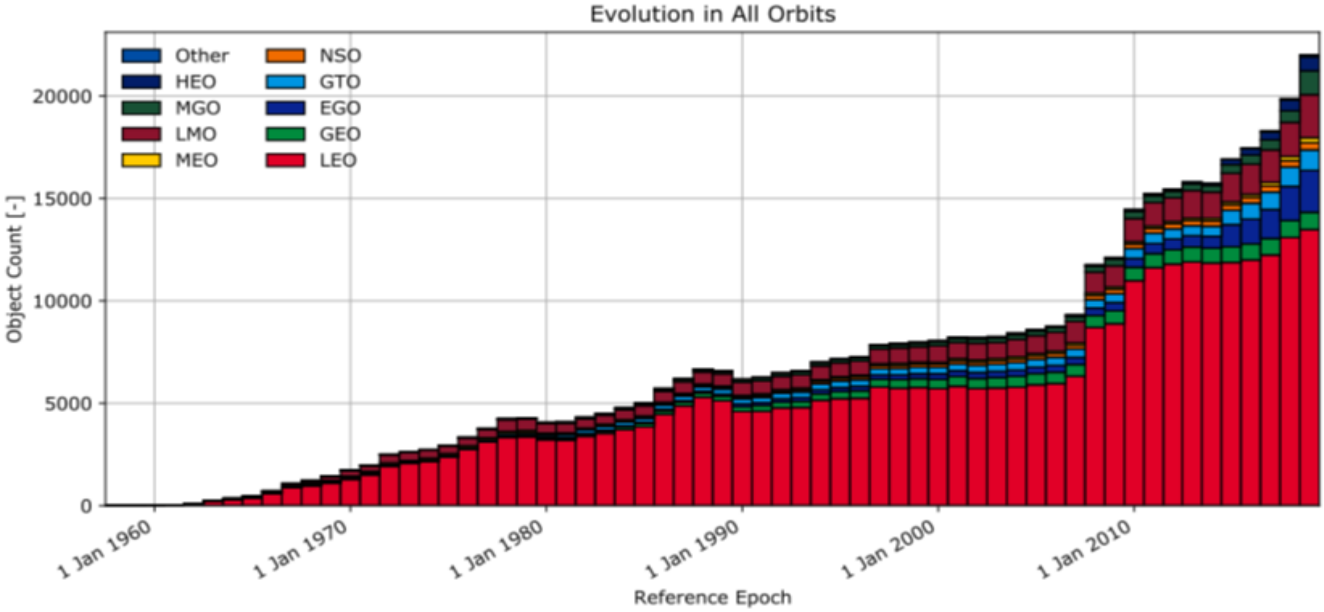
Table 1.1: Object Classifications.

| Type | Description |
|---|---|
| PL | Payload |
| PF | Payload Fragmentation Debris |
| PD | Payload Debris |
| PM | Payload Mission Related Object |
| RB | Rocket Body |
| RF | Rocket Fragmentation Debris |
| RD | Rocket Debris |
| RM | Rocket Mission Related Object |
| UI | Unidentified |

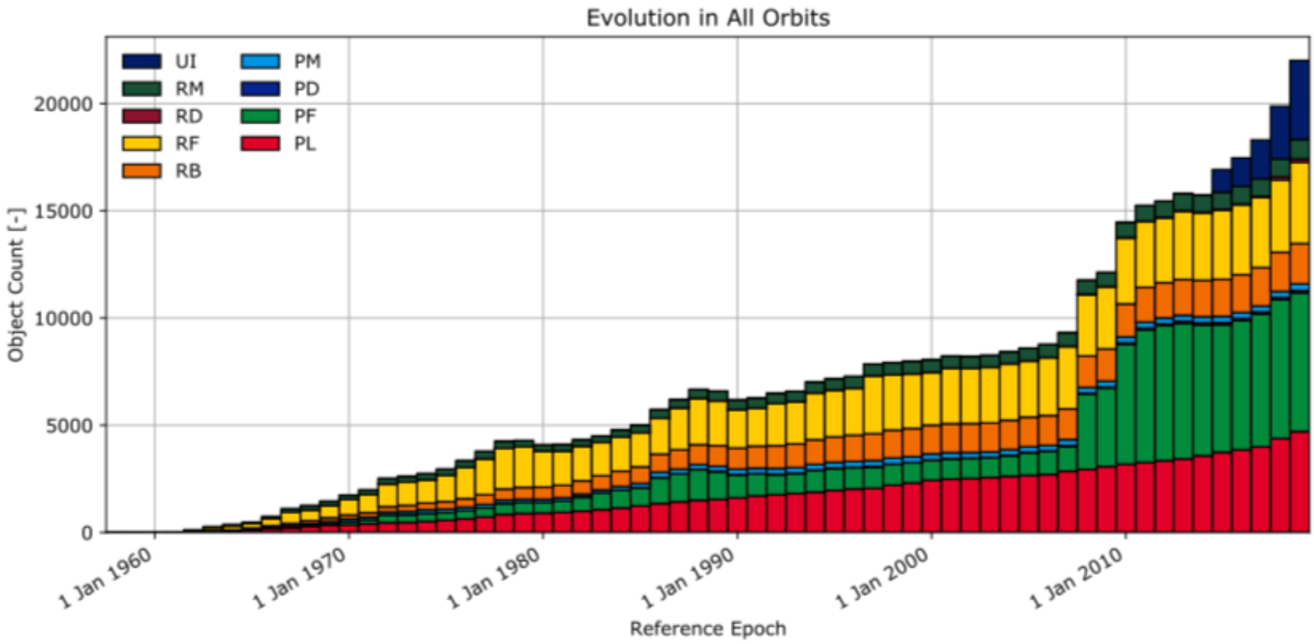
Orbiting objects can turn into debris due to different means, for example, satellites reaching the end of their servicing life and not having any option to deorbit themselves, docked payloads/parts of spacecraft detaching and falling into an orbit, fragments created by collisions of orbiting objects etc. A well-known in-orbit collision took place in 2009, when an inactive Russian satellite (Cosmos 2251) collided with an American communications satellite (Iridium 33), some 800 kilometres over Siberia, resulting in thousands of little fragments of space junk being created, many of which still hover over the planet[3]. These small fragments can be potentially very hazardous as they might collide with other satellites, or even the ISS and spacewalking astronauts aboard it, thus posing a significant danger.

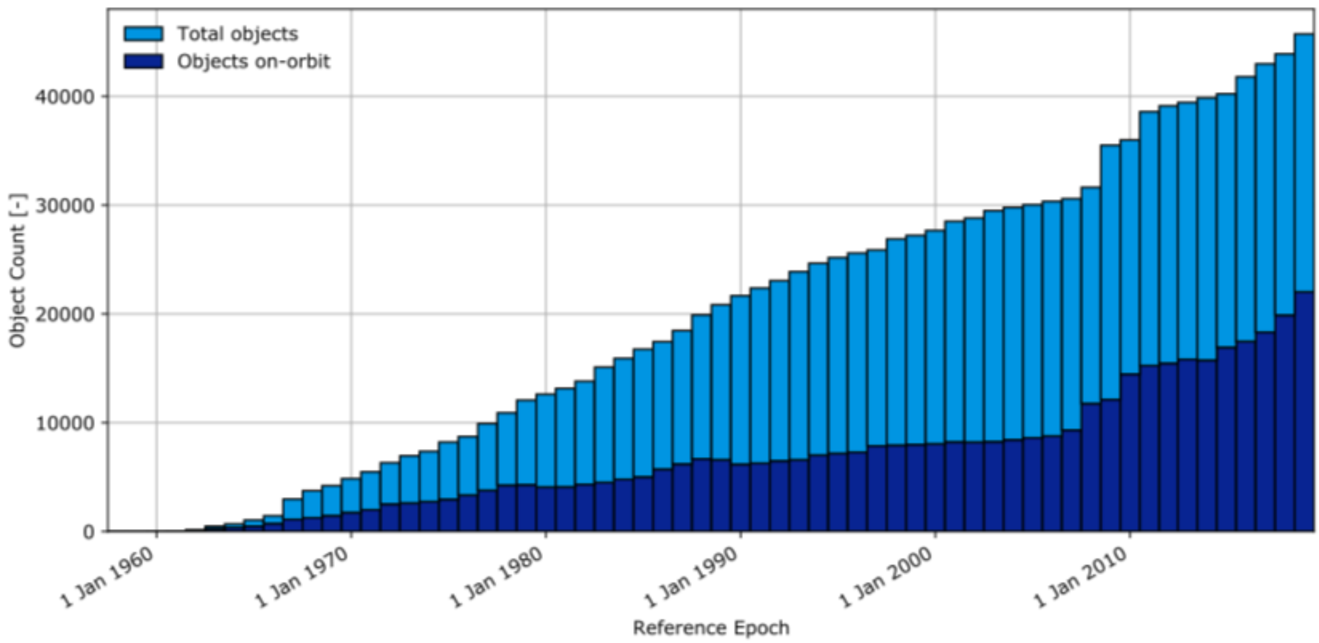
In 1978, NASA scientist Donald J. Kessler hypothesised an eponymous scenario[4] which painted a rather unsavoury picture of cascading collisions because of the Low-Earth Orbit (LEO) reaching a saturation point of having enough satellites and space debris, therefore setting in motion a chain reaction of collisions and creation of more debris.

The Low-Earth Orbit (<2000km, eccentricity<0.25) holds the highest amount of space objects and debris right now, indicated by the following graph[5]:




The following graphs indicate the type of objects orbiting and the total number of objects orbiting, including those not in-orbit [6]–






By all means, accumulation of space debris is a major issue requiring sustained attention and effective solutions, in order to avoid the Kessler Syndrome from actualising. Over recent years, a lot of agencies have stepped up efforts to deal with space debris, with some ideas already on their way to see fruition soon; for example, the European Space Agency has started working on projects involving robotic arms attached to servicer satellites which can latch onto targeted space debris and help deorbit it. To tackle this growing problem of space debris, we propose below a mission concept called the **Optimal Debris Deorbiting System (ODDS)** to deorbit multiple space debris using a novel method called the **Ion Beam Shepherd Method**.

## 2.Table of Innovations

Our design aims to use a combination of novel and flight-proven technologies in conjunction to develop a robust debris deorbiting satellite system. The table below presents a condensed list of the challenges faced in developing a satellite deorbiting system and our solutions to address those challenges.

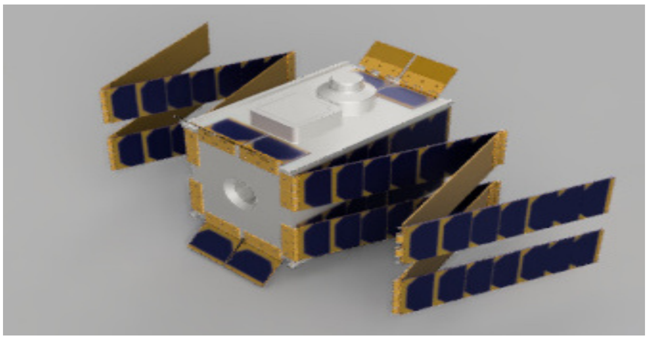
| Problem/ Challenge | Solution | Section |
|---|---|---|
| Selecting optimum inclination for deployment of the satellite system. | Using an algorithm to compute the inclination with high collision probability and highest density of debris. | C |
| Selecting 4/5 optimum target debris for each de-orbiter after selection of inclination. | Proposing an algorithm to compute sets of debris which meet the fuel constraints of the de-orbiter and have the highest combined threat factor. | C |
| Complications of docking/ rendezvous maneuvers with a non-cooperating object(debris) | Contactless active debris removal using Ion Beam Shepherd Method. | B |
| A fuel system to incorporate 2 ion thrusters located at opposite faces of the | Using a large novel centralized fuel tank with feed lines to accommodate the 2 ion thrusters. | E |

| | | |
|---|---|---|
| de-orbiter satellite. | | |
| Limited lifetime of de-orbiter satellites due to fuel constraints. | Using a mothership with multiple docking ports stationed at 900km for in-orbit refuelling of the 2 de-orbiter satellites. | D |
| | Using the end effector developed for maneuvers for the ASSIST system, maneuvers will be done by connecting the fluid connections from the end effector to the fuel lines on the docking mechanism of the mothership. | E |
| Continuous communication and contact with the LEO de-orbiter. | Using Addvalue's Inter-Satellite Data Relay System(IDRS) , which provide a constant on-demand communication link between the ground and de-orbiter by using a Geosynchronous satellite constellation system. | G |
| Tracking the space debris with high accuracy and rendezvous with the debris. | Using AGI's COMSPOC and Addvalue's IDRS to provide constant debris position updates to the de-orbiter satellite. | G |
| | Using ASC's DragonEye, a Flash LIDAR Space Camera for automated rendezvous and docking. | E |
| Robust Attitude Control System | Using a pyramid configuration of reaction wheels in combination with magnetorquers for high redundancy and precise control. | D |

## 3. Summary of the Optimal Debris Deorbiting System (ODDS)

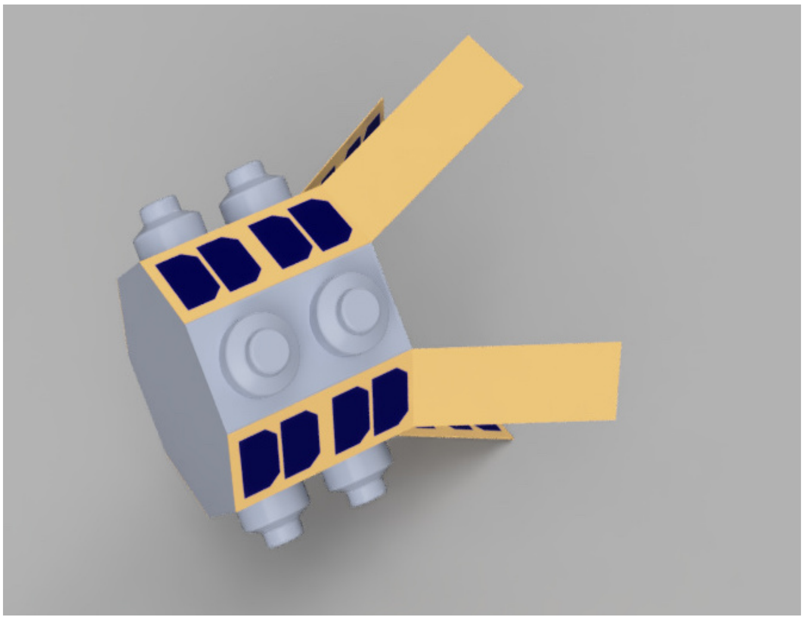
Our proposed satellite system consists of 3 satellites: 2 deorbiting satellites and 1 refuelling satellite.

- **Deorbiting satellite (Deorbiter)**: The purpose of the deorbiting satellites (**here on referred to as the deorbiter**) is to decelerate the debris to an orbit of 400km by imparting a constant force on it using bidirectional ion thrusters. This method is called the Ion Beam Shepherd Method and is elaborated in section B. The deorbiter will accompany the debris to an orbit of 400km after which it will perform a prograde maneuver to locate the next piece of debris. According to the fuel calculations in section, the deorbiter can typically deorbit 4-5 debris before it needs to be refuelled.



*Render of Deorbiter*

- **Refuelling satellite (Mothership):** The purpose of the refuelling satellite (**here on referred to as the mothership)** is to extend the lifestyle of the deorbiters by providing in orbit refuelling services through a mechanism called ASSIST. The docking mechanism in described in section E . The mothership will be stationed at a circular orbit of 900km **in the same inclination** as the deorbiters. The mothership will have enough fuel to refuel each deorbiter once and thus, effectively double the amount of debris that can be cleared. While it just serves 2 deorbiters in our mission concept, the mothership is envisioned to cater to a larger number of deorbiters once the concept of in-orbit refuelling is demonstrated.



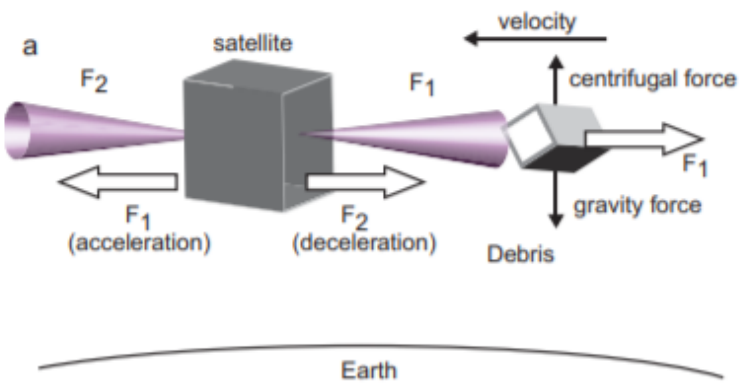
*Render of Mothership*

## B) Ion Beam Shepherd Method for Contactless Active Space Debris Removal

### 1. Introduction

The Ion Beam Shepherd Method uses a high-velocity ion beam produced in the ion thruster of a shepherd spacecraft (***in our case, the deorbiter***) flying in proximity of the debris. The ion beam is directed against the debris and is used to modify its orbit with no need for docking as shown in the figure below. In addition, a secondary propulsion system (***in our case, an identical ion thruster located on the opposite face***) is used to produce an equilibrium force which is needed to avoid the deorbiter from drifting away from the debris. The force transmitted comes from the transfer of momentum from the Xenon ions impacting against the surface of

Figure - Visualization of Ion Beam Shepherd



the debris. Since non co-operative objects are extremely difficult to dock with, the Ion Beam Shepherd Method is a novel way for contactless space debris removal[7]. the debris. Since non co-operative objects are extremely difficult to dock with, the Ion Beam Shepherd Method is a novel way for contactless space debris removal[7].

## 2. Rationale for choosing Ion Beam Shepherd Method

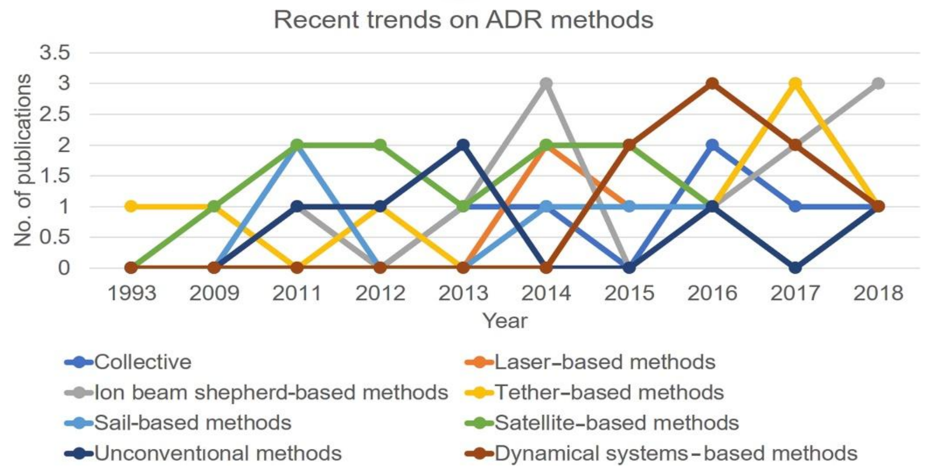
Almost all methods of debris removal require transmission of momentum from the removal system to the space debris. The most obvious way to do this would be to dock with the debris before deorbiting commences. However, space debris are non-cooperative objects generally characterized by a problematic attitude motion and are not easy to dock with. Docking, in general, increases the complexity of the method and reduces the reusability due to the increased risk factor.

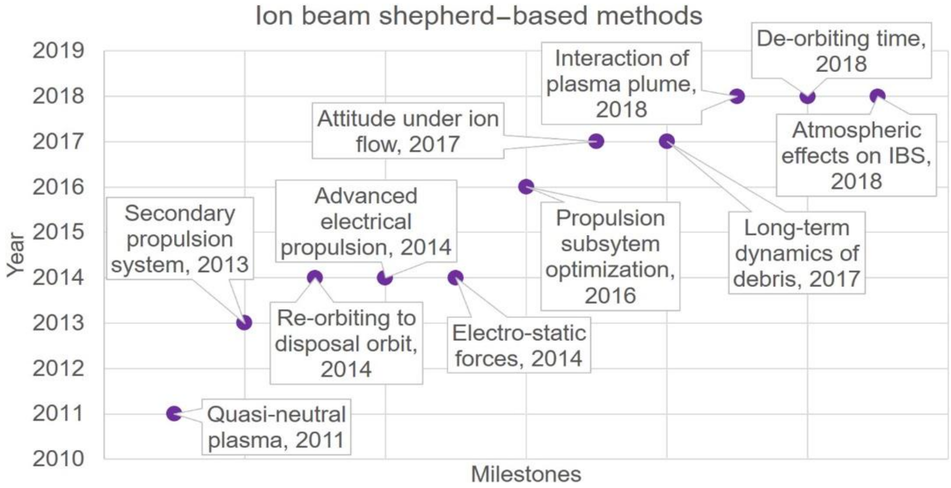
Another option is to perform a capture operation with an appendage released from the deorbiter. In this case, the major difficulty is the deployment of the capturing device. In addition to this, another downside of using a capture operation is that it would be difficult to reuse for multiple targets.

Debris removal concepts based on pulsed-laser ablation systems are another viable option as they can be operated far from the orbiting target, possibly even from the ground. Unfortunately, the small impulse obtained from material ablation cannot be effective against targets of larger sizes. Another limiting factor for ablation systems is the limited range and angle of operation.

Tether-based methods are another promising avenue in space-debris removal and a lot of work is being done on electrodynamic tethers (EDT), tethered throw net-systems and harpoons. Although promising in many aspects, they are much slower in action when compared to methods such as Ion Beam and laser ablation. With debris growing at a near exponential rate, a fast and highly reusable debris removal system is the best way to combat debris in the Low Earth Orbit.[8]

Due to high reusability, quick deorbiting time and reduced complexity in comparison to other methods, interest in the Ion Beam Shepherd method is growing rapidly and there has been an increased number of publications over the last decade. The Ion Beam Shepherd does have drawbacks such as ion beam divergence, sputtering and backflow. However, there have been rapid advancements in electrical propulsion over the last few years and it is expected that ion thrusters become even more robust and reliable over the coming years. Even in Singapore, companies such as Aliena [9] are leading the way forward in propulsive technology. Another drawback of the Ion Beam Method is the complexity of control in tracking and maintaining a constant distance & orientation from the debris over large durations. Again, there have been rapid advancements in control particularly in the fusion of AI and control to obtain highly reliable algorithms. In conclusion, our decision to choose the Ion Beam Shepherd Method was influenced by many factors such as the current trends, feasibility, deorbiting time and the up-and-coming technologies.




## 3. Calculations

In this section, we present some relevant calculations and figures to support the Ion Beam method. We try to fit the assumptions and formulae obtained from the "**ESA: Ion Beam Shepherd Report**" into the model of our satellite and present the relevant figures below.[10]

The assumptions used for the calculations are as stated below:

- The mission begins with the IBS co-orbiting with the debris on an initial generic orbit and ends when the two satellites have reached a common target orbit.
- The thrusters are operational for **16 hours each day** at maximum thrust and the remaining 8 hours at 25% of the maximum thrust.
- The primary and secondary propulsion systems employ ion thrusters with the same efficiency ($\eta 1 = \eta 2$) and exhaust velocity ($c1 = c2 = c$). (**In our case, BIT-3 Thrusters from Busek are used**).
- The target debris is in a circular orbit.
- The applied deorbit force is constant (**In our case 1.2mN**), fixed by the mission designer, and always directed along the tangent to the orbit.
- During the spiral-transfer the orbit evolves in a quasi-circular manner.
- The total efficiency of the momentum transfer is 75%.( Efficiency is decreased due to beam divergence)

Using the above ideal assumptions, the formula for the time taken to deorbit the debris from an initial generic orbit to a target orbit is given by –

$$\Delta t = m_d \frac{\sqrt{\mu}}{F_p} \times \frac{\sqrt{R} - \sqrt{r}}{\sqrt{rR}}.$$

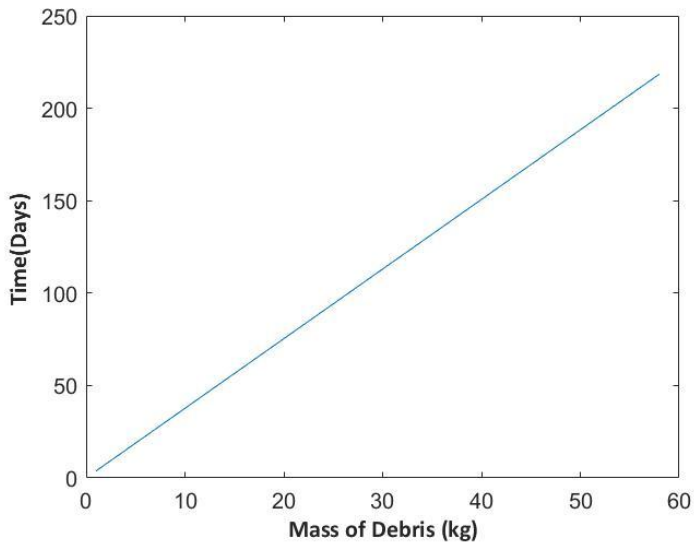
**$m_d$ = Mass of Debris   μ = Gravitational Constant (GM)   R= Initial Circular Orbit   r= Final Target orbit**

**Fp = Propulsion Force**

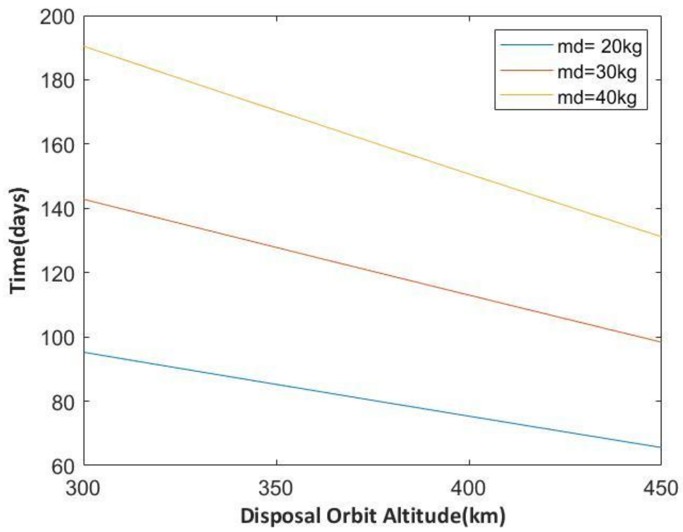
The BIT-3 Thruster used will provide a **constant thrust of 1.2mN** according to the datasheet released by Busek and is assumed to have a momentum transfer efficiency of 75%. Also, powering the thrusters at 25% of the maximum value for 8 hours is roughly equivalent to firing the thrusters at maximum value for 2 hours.

Using the formula for time taken adjusted with the above stated assumptions, the following figures are obtained using MATLAB code (Appendix A) -

**(1) Time to Deorbit vs Mass of Debris**
**Debris Initial Orbit: 800km**
**Debris Final Orbit: 400km**



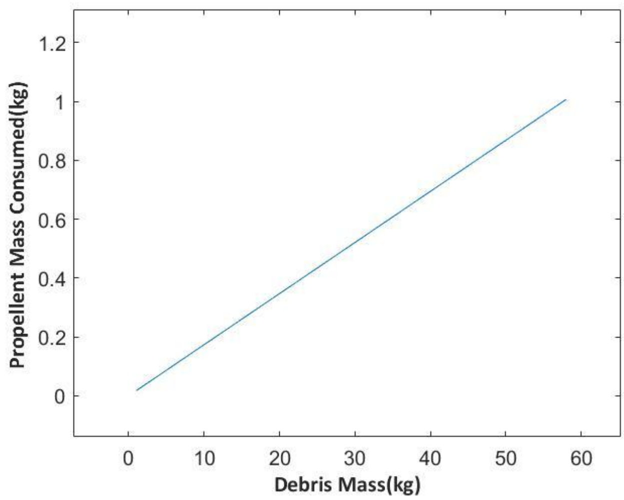

**(2) Time to Deorbit vs Final disposal Orbit**
**Debris Initial Orbit: 800km**
**Debris Final Orbit: Varying**




**(3) Propellant Mass Consumed vs Time to Deorbit Debris**

**Debris Initial Orbit: 800km**

**Debris Final Orbit: 400km**
**Mass Flow Rate: 40 µg/s (Approximated from BIT-3 datasheet)**
**Equation Used: Mass consumed = Mass Flow rate * Time**




**(3) Propellant Mass Consumed in Prograde Maneuver vs Distance to next target (ΔR)**
After the deorbiter has successfully deorbited a debris to the target orbit, it will perform a prograde maneuver to reach its next target. The graph presented below shows the amount of fuel consumed to perform a prograde maneuver from an altitude of 400km to the next debris target located at some constant circular orbit. Using the **Tsiolkovsy Rocket Equation**:

$$\ln(\Delta V/c) = m_0/m_f$$

*c= Exhaust Velocity   $m_0$= Initial Satellite Mass   $m_f$ = Final Satellite Mass*

- ΔV in the Low Earth Orbit (LEO) for an ideal Hohmann Transfer can be approximated as:

$$\Delta V = \Delta R/3.5$$

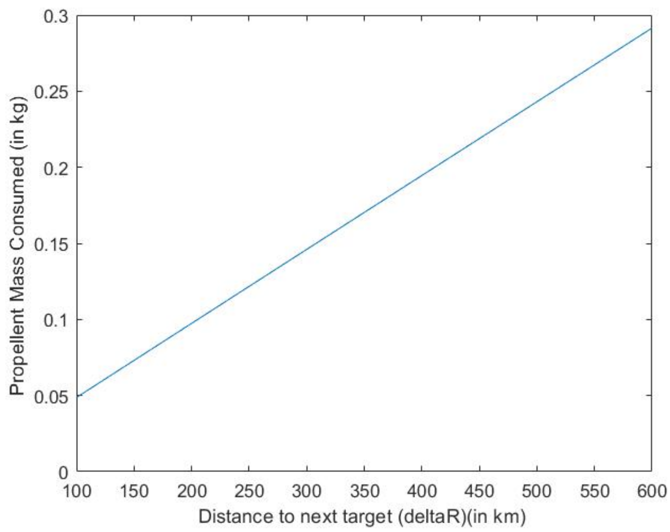
*ΔR = Difference in altitude (In km)*

- The exhaust velocity (c) is calculated using the datasheet of the BIT-3 Thruster by Busek:

$$c = I_{sp} * g_0$$

*c= Exhaust Velocity ; $I_{sp}$ = Specific Impulse ; $g_0$ = standard gravity*

Finally, the Tsiolkovsy rocket equation starts to lose accuracy when the burn duration is long, which is the case for electric propulsion. The loss of accuracy is primarily because the equation does not account for the action of gravity on the spacecraft during the duration of the maneuver. Thus, the actual fuel needed is higher than that calculated by the equation. To account for this loss in accuracy, we multiply the fuel by a conservative factor of 1.75 to obtain a more valid approximation.




**Though the propellent consumed is an exponential function of ΔR, the relation plotted appears to be a linear function. This might be due to the small variation in ΔR since only Low Earth Orbit is considered**. When larger values of ΔR are tested, the exponential curve starts to become apparent.

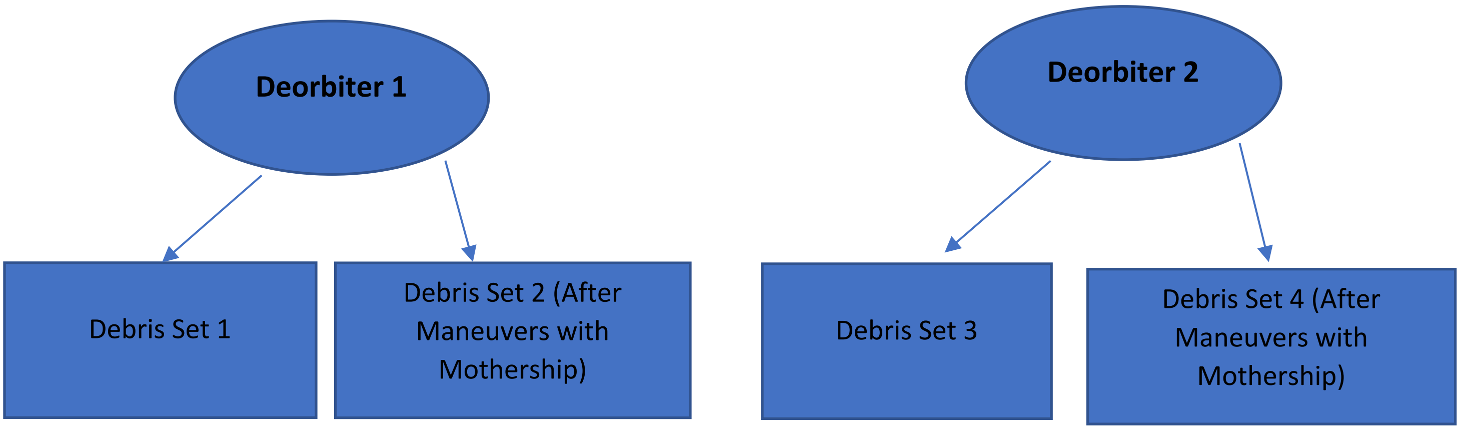
## 4. Conclusion

Concluding, the Ion Beam Shepherd Method is a novel way to remove multiple debris of large sizes from the Low Earth Orbit in a short span of time. The calculations presented above show that the fuel required to deorbit a 40kg debris in a constant circular orbit in LEO is as low 600 grams. This makes it possible to deorbit up to 6 large debris with a single deorbiter carrying just 5kg of fuel. The quick deorbit time implies that a substantial number of debris can be deorbited each year if multiple deorbiters are used. When combined with refuelling and integrated with technologies such as Dragoneye and IDRS, the reliability, robustness and lifetime all increase significantly. The above presented calculations and rationales make a strong case for the Ion Beam Shepherd Method as a plausible solution for the problem of space debris.

## C) Selection of Target Debris

### 1. Introduction

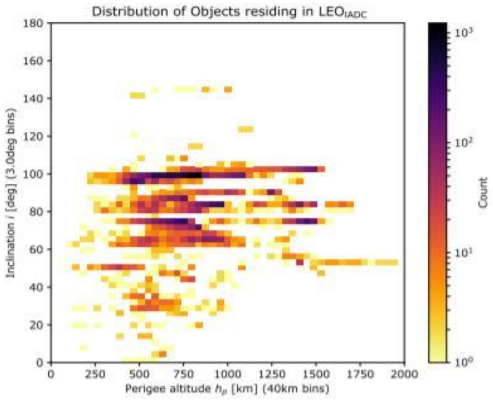
Each deorbiter in our proposed solution deorbits up to 6 medium sized debris (20-60kg) before it needs to be refuelled. The debris' selected in this mass range will ideally be defunct satellites so it is easy to obtain parameters such as the mass and area of the satellite. It is also required to select a set of debris targets instead of just a single debris. Each set of debris selected for a deorbiter can contain between 4-6 targets. All the targets must be at the same inclination (maximum variation in inclination ~ 1-2 degrees) since plane change maneuvers cost a lot of fuel [11]. Thus, it is required to select sets of debris between the altitudes of 400-1000km in the mass range of 20-60kg all located on the same inclination. Four such sets of debris are required as explained below.

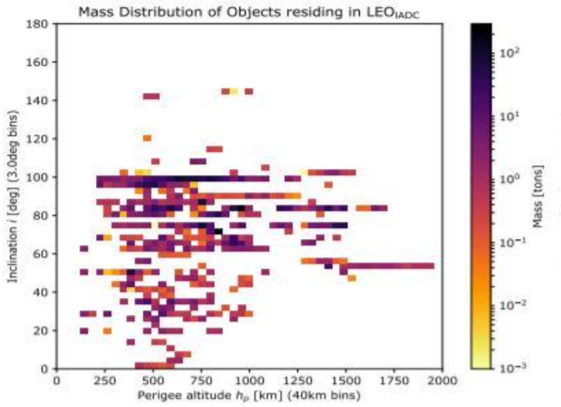



Summarizing, 4 sets of 5-6 debris each are required. Thus, our solution clears as many as 20 debris for an inclination range of 2 degrees (Example: 60 -62 degrees). The selection of the sets of debris and the optimum inclination are described in the following sections.

### 2. Selection of Optimum Inclination

According to ESA's Space Environment Report 2019 [12], the distribution of objects residing in LEO vs inclination is given by the following figures.



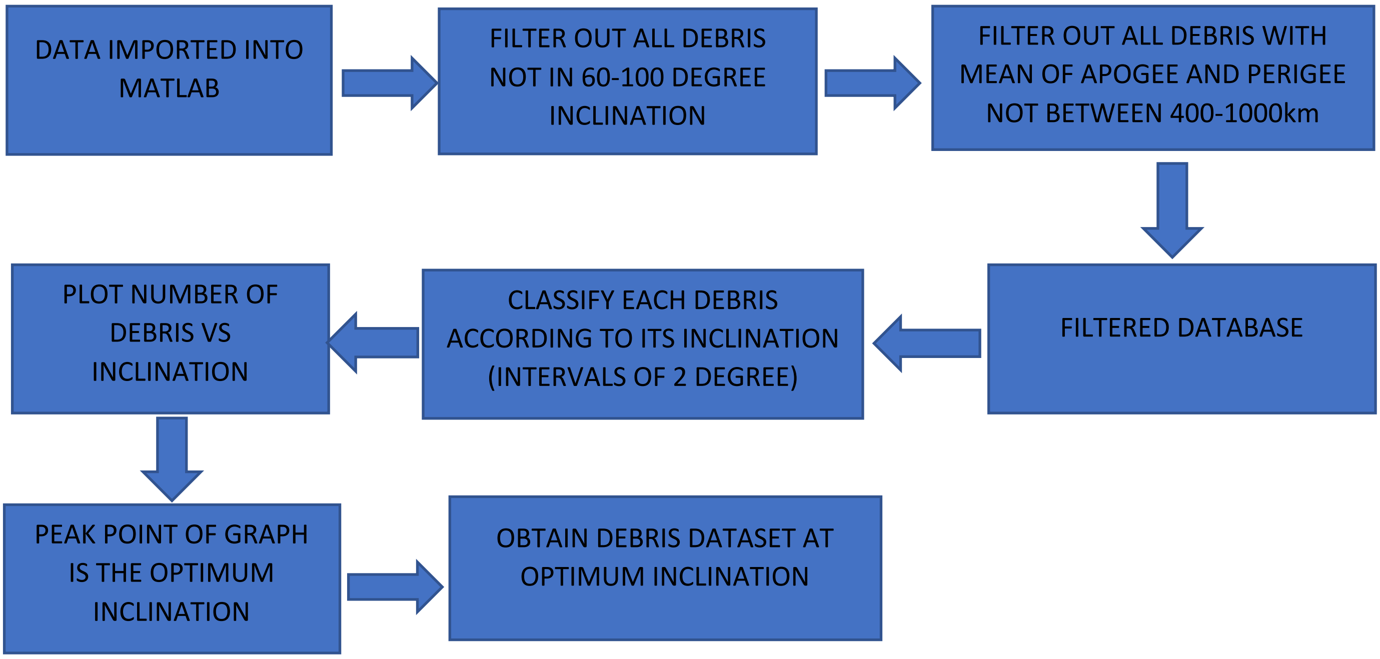




From the above figures, it can be concluded that the highest concentration of objects is between the inclinations of 60 – 100 degrees and that the optimum inclination lies between 60 and 100 degrees. Although the above figures show the combined distribution of both active and debris objects, we assume that the distribution over inclination of only debris objects is very similar to the above shown distribution. Looking roughly at the graph, the inclination with highest concentration of debris (darkest colour) appears to be 97-100 degrees.

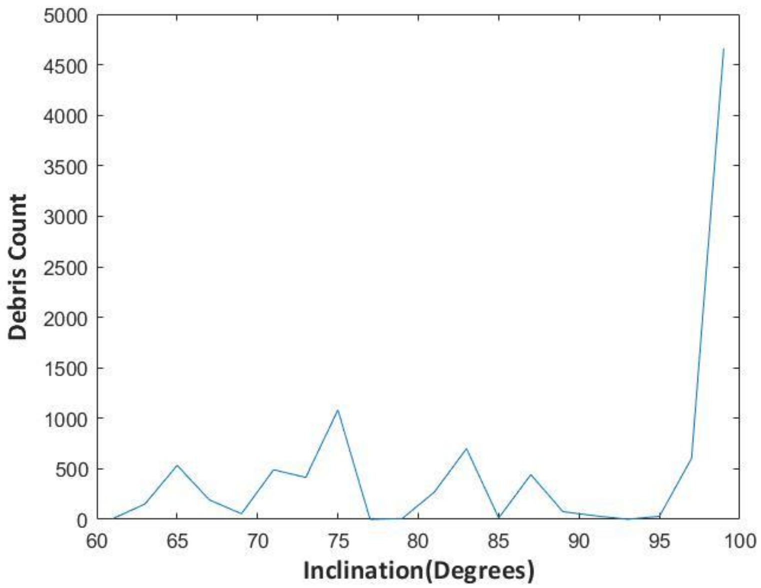
The next step is to numerically select an inclination between 60-100 degrees. For the mission to have the highest impact, the inclination with the maximum density of debris and thus, the highest risk of collisions should be chosen as the optimum inclination. To do this, we download a database of the existing debris from space-track.org[13]. A sample of the data downloaded is shown in the figure below.

| A | B | C | D | E | F | G | H | I | J | K | L | M | N | O | P | Q |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| INTLDES | NORAD_CAT_ID | OBJECT_TYPE | SATNAME | COUNTRY | DECAY | PERIOD | INCLINATION | APOGEE | PERIGEE | RCSVALUE | RCS_SIZE | FILE | LAUNCH_YEAR | LAUNCH_NUM | LAUNCH_PIECE | CURRENT |
| 1983-101H | 15110 | DEBRIS | COSMOS 1501 | CIS | 4/2/1985 | 88.69 | 82.83 | 218 | 202 | 0 | MEDIUM | 1 | 1983 | 101 | H | Y |
| 1984-058G | 15068 | DEBRIS | COSMOS 1571 | CIS | 1/7/1984 | 89.06 | 70.02 | 241 | 215 | 0 | LARGE | 1 | 1984 | 58 | G | Y |
| 1984-054N | 15062 | DEBRIS | COSMOS 1568 | CIS | ######## | 87.68 | 72.81 | 167 | 152 | 0 | LARGE | 1 | 1984 | 54 | N | Y |
| 1984-063C | 15059 | DEBRIS | SL-12 PLAT | CIS | ######## | 87.79 | 51.61 | 168 | 163 | 0 | | 1 | 1984 | 63 | C | Y |
| 1984-054L | 15049 | DEBRIS | COSMOS 1568 | CIS | ######## | 88.96 | 72.77 | 237 | 210 | 0 | SMALL | 1 | 1984 | 54 | L | Y |
| 1984-054K | 15048 | DEBRIS | COSMOS 1568 | CIS | ######## | 88.41 | 72.79 | 201 | 190 | 0 | LARGE | 1 | 1984 | 54 | K | Y |
| 1984-054J | 15045 | DEBRIS | COSMOS 1568 | CIS | ######## | 91.34 | 72.86 | 361 | 320 | 0 | | 1 | 1984 | 54 | J | Y |
| 1984-054H | 15044 | DEBRIS | COSMOS 1568 | CIS | ######## | 92.34 | 72.62 | 437 | 341 | 0 | | 1 | 1984 | 54 | H | Y |
| 1984-054G | 15043 | DEBRIS | COSMOS 1568 | CIS | ######## | 87.31 | 72.78 | 147 | 135 | 0 | LARGE | 1 | 1984 | 54 | G | Y |
| 1984-058H | 15069 | DEBRIS | COSMOS 1571 | CIS | ######## | 87.34 | 69.97 | 147 | 138 | 0 | LARGE | 1 | 1984 | 58 | H | Y |
| 1984-058J | 15072 | DEBRIS | COSMOS 1571 | CIS | 8/9/1984 | 91.28 | 70 | 365 | 310 | 0 | MEDIUM | 1 | 1984 | 58 | J | Y |
| 1982-033E | 15105 | DEBRIS | SALYUT 7 DEB | CIS | 7/8/1984 | 90.83 | 51.6 | 323 | 307 | 0 | MEDIUM | 1 | 1982 | 33 | BL | Y |
| 1984-064F | 15104 | DEBRIS | COSMOS 1575 | CIS | ######## | 88.06 | 82.3 | 203 | 154 | 0 | | 1 | 1984 | 64 | F | Y |
| 1984-064E | 15103 | DEBRIS | COSMOS 1575 | CIS | ######## | 87.97 | 82.33 | 213 | 135 | 0 | | 1 | 1984 | 64 | E | Y |
| 1984-064D | 15102 | DEBRIS | COSMOS 1575 | CIS | ######## | 87.69 | 82.32 | 172 | 149 | 0 | | 1 | 1984 | 64 | D | Y |
| 1984-064C | 15101 | DEBRIS | COSMOS 1575 | CIS | ######## | 87.42 | 82.33 | 155 | 138 | 0 | LARGE | 1 | 1984 | 64 | C | Y |
| 1984-071C | 15097 | DEBRIS | SL-6 PLAT | CIS | ######## | 86.96 | 62.83 | 131 | 116 | 0 | LARGE | 1 | 1984 | 71 | C | Y |
| 1984-060D | 15088 | DEBRIS | COSMOS 1572 | CIS | 9/7/1984 | 88.67 | 82.34 | 220 | 197 | 0 | MEDIUM | 1 | 1984 | 60 | D | Y |
| 1984-061E | 15083 | DEBRIS | COSMOS 1573 | CIS | ######## | 87.99 | 72.87 | 186 | 164 | 0 | | 1 | 1984 | 61 | E | Y |
| 1984-054F | 15042 | DEBRIS | COSMOS 1568 | CIS | ######## | 88.65 | 72.8 | 215 | 200 | 0 | MEDIUM | 1 | 1984 | 54 | F | Y |

The debris catalog contains information on 29860 debris objects (Note: Some of the debris in the catalog has already burned out due to atmospheric drag). To filter the data, we import it into MATLAB. The following flowchart briefly illustrates the algorithm used to process the data to get maximum inclination. The entire MATLAB code can be found in Appendix A.




The graph plotted after the following algorithm is executed in MATLAB is plotted below. As can be seen from the graph, the maximum density inclination is indeed from 98-100 degrees. Thus, we have numerically derived that the inclination for deployment of the deorbiting system should be 99 degrees.

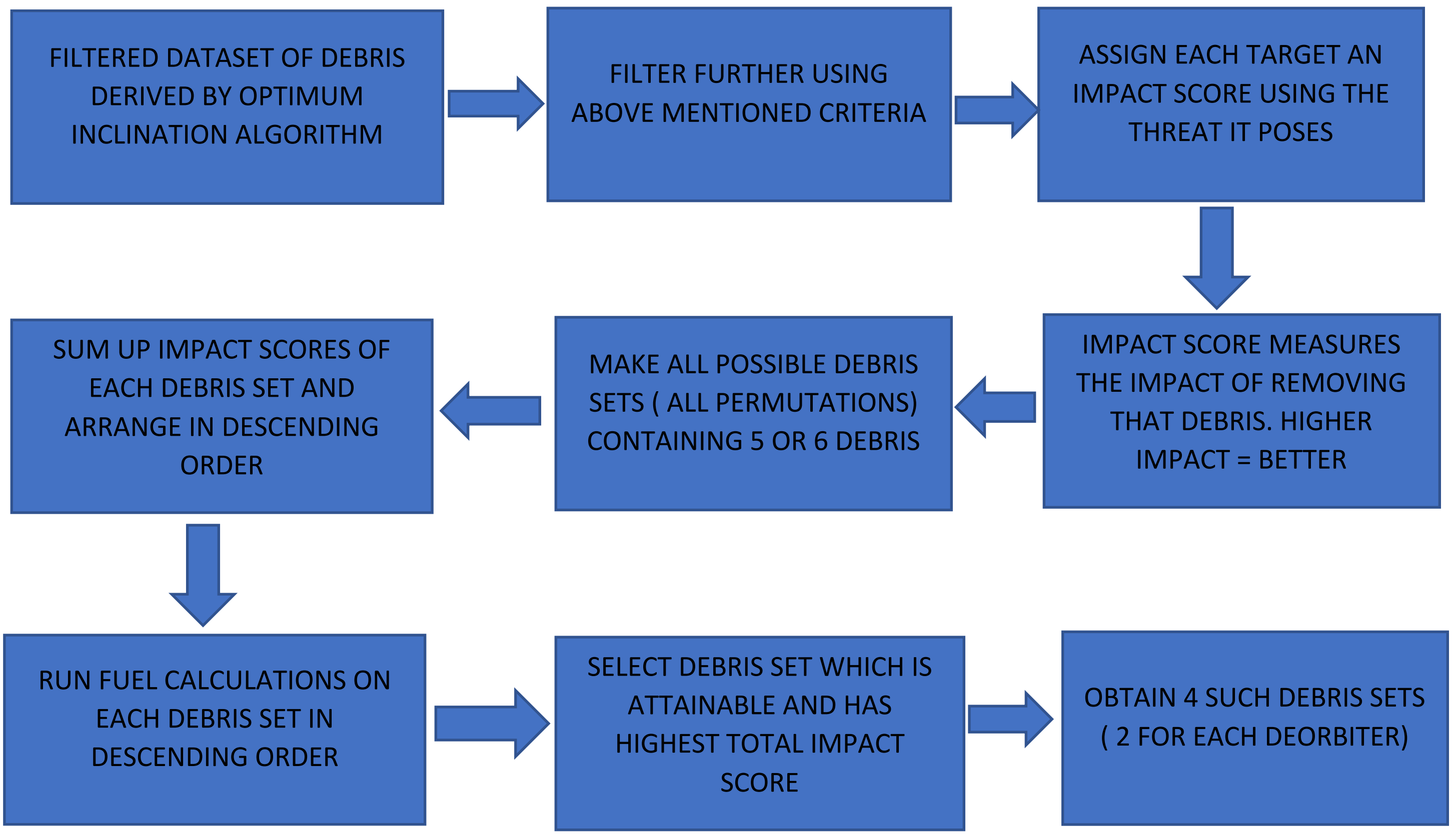



### 3. Selection of Target Debris Set at Optimum Inclination (99 degrees) [14]

As stated in the introduction, 4 sets comprising of 4-5 debris each are to be selected for inclinations between 98-100 degrees. Although we have obtained the list of potential debris and their basic orbital characteristics using data processing detailed above, debris characteristics such as mass and cross- sectional area aren't available in the public domain. Hence, here we propose the flowchart of an algorithm which can compute optimal debris sets once additional data about the debris is obtained. The filtered dataset obtained in deriving optimum inclination will be filtered further using the following criteria –

| Sr NO. | Factor | Reason |
|---|---|---|
| 1 | Mass | As described in Section B, the mass of the debris would be critical in deciding how much fuel the deorbit maneuver takes. It is desired to have the debris mass between 20-40 kg. |
| 2 | Area (RCS) | The higher the Radar Cross Sectional Area (RCS) of the debris (defunct satellite), the better is the efficiency of the Ion Beam Shepherd Method since higher area implies better momentum transfer from the deorbiter to the debris. At the same time, a higher area is also an indicator of higher collision risk. Thus, a debris object with high area to mass ratio (A/M) will be preferred. |
| 3 | Altitude | Debris at a low altitude which is expected to decay soon on its own due to atmospheric drag will be given less importance. Satellites located at higher densely populated altitudes (500-800km) with high risks of collision will be given priority. |
| 4 | Eccentricity | Highly eccentric orbits will make the deorbiting maneuver more complex and inefficient. Thus, it is preferred to select debris whose orbits are close to circular (low eccentricity). |
| 5 | Legal Barriers | If legal barriers exist for a particular debris, then it will be removed from the list of potential debris targets. |

A simplified algorithm for computing the 4 optimal debris sets is described below. While it is possible to add many more intricate details to this algorithm, the general crux of it remains the same as the one presented below. The basic idea behind the algorithm is to obtain a set of debris which is feasible to deorbit given the fuel constraints and at the same time has the highest positive impact on the space environment**.**



This section focused on obtaining the optimum inclination for our satellites and selecting the target debris sets at that inclination. Ideally, the target debris are defunct satellites in the mass range of 20-60kg. A numerical analysis was conducted to obtain the optimum inclination of 99 degrees which was roughly verified using a graph released by ESA. An algorithm was also described on paper to compute the optimal target debris sets at that inclination. However, the algorithm couldn't be run numerically due to its complexity and due to lack of data on the debris**.**

## D) Satellite Design

### 1. Concept of De-orbiter[15]

According to mission report of NASA, CubeSats are generally used by commercial companies, educational institutions and non-profit organizations to conduct scientific experiments and technology demonstrations because they are cost effective, timely and relatively easy to accomplish the mission.

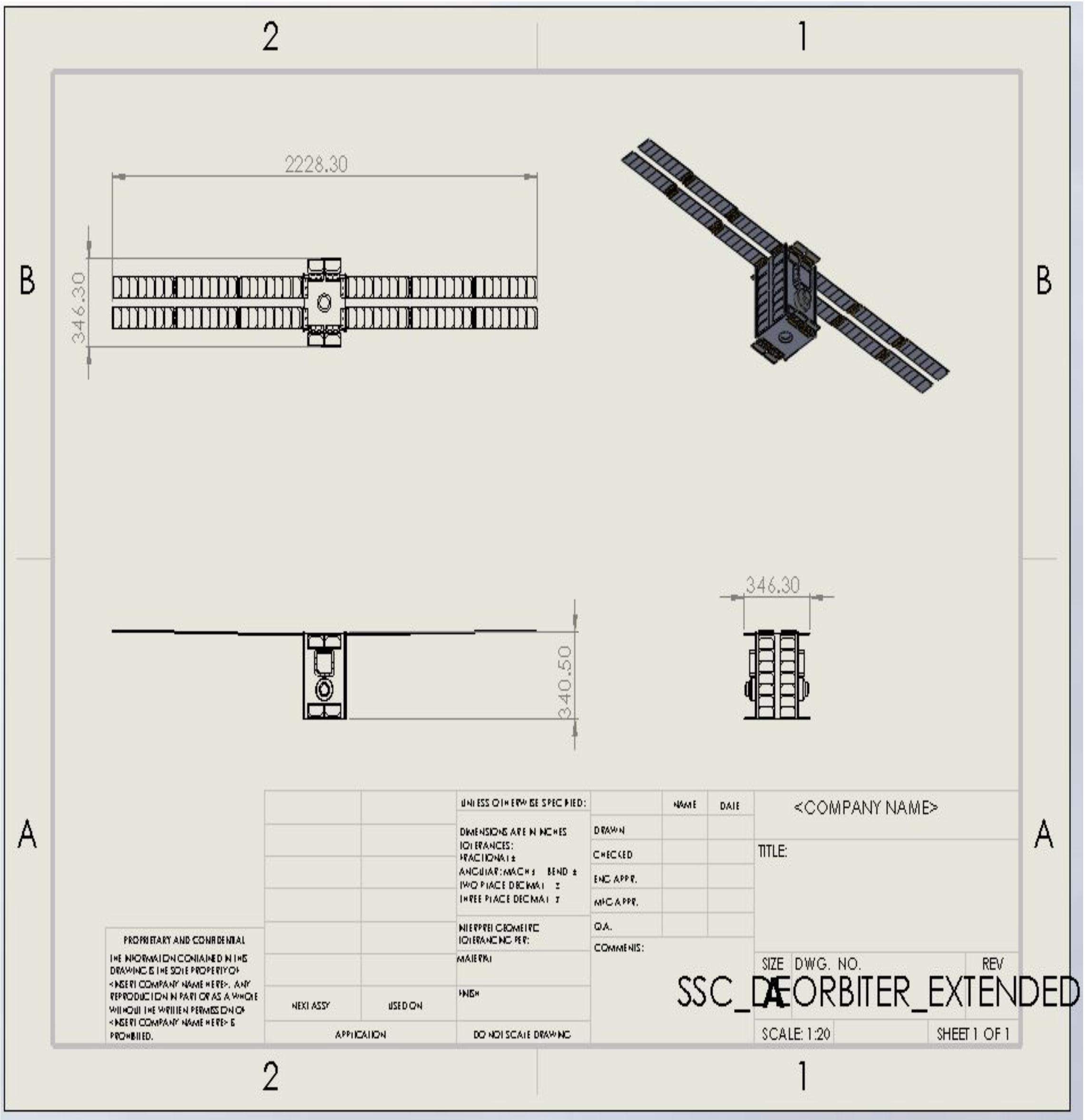
The CubeSats are available in various dimensions like 1U,2U,3U,6U,12U...etc. We are using 2 standard 12U CubeSats to carry our mission of de-orbiting space debris. The reasons for using a 12U CubeSat is to accommodate the following:

1)Two ion thrusters positioned opposite to each other for propulsion as well as to deorbit the debris. Since we are using electrical propulsion system, the power requirement for the de-orbiter is as high as 56-80 Watts per thruster.
2)In order to meet the power requirements, we are using 25 solar arrays 127g each. Since the total weight of our system is about 22 Kg, and 12U allows a maximum payload of 20Kg, we chose 12U[16].

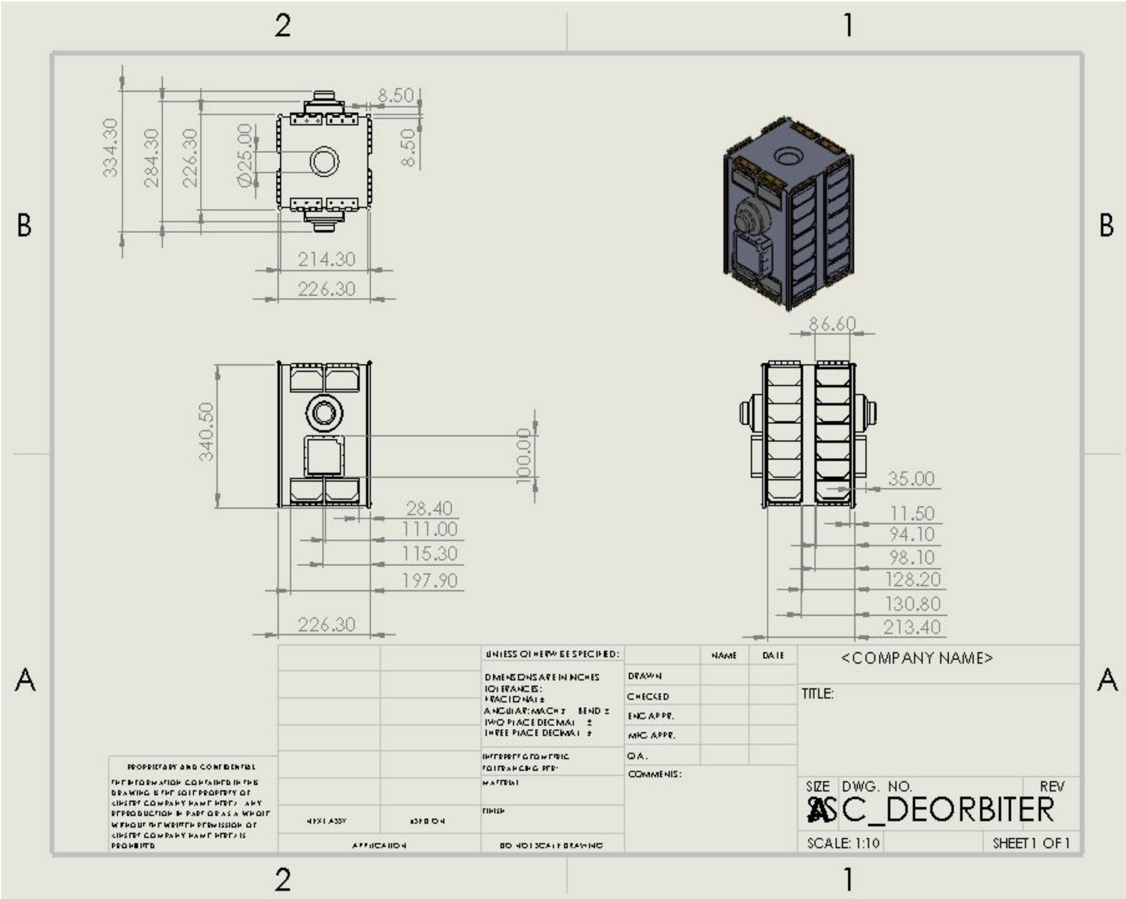
The deorbiter is fitted with two BIT-3 Xenon ion thrusters which are commercially available. The purpose of using the 3U Endurosat X/Y solar panels is because of its ability to produce 8.3 Watts of power nominally using 7 high efficiency(30%) triple - junction GaAs cells over a single array with an added advantage of being lightweight. This solar panel is fully compliant with the CubeSat standard.

| Sr NO. | Parameter | Dimension(mm) |
|---|---|---|
| 1 | Deorbiter[17] | 226.3 x 226.3 x 340.5 |
| 2 | Deorbiter after deploying Solar Panels | 2228.3 x 346.3 x 340.5 |
| 3 | Solar Panels[18] | 82.6 x 325 x 1.6 |
| 4 | ADCS[19] | 43.5 x 43.5 x 24 |
| 5 | CDHS[20] | 96 x 90 x 12.4 |
| 6 | EPS(with battery) [21] | 89.3 x 92.9 x 25.6 |
| 7 | Rail[22] | 8.5 x 8.5 x 340.5 |

A simple design of the latch can be seen in the drawings. The latch serves as a target to dock with the mothership. The deorbiter has the benefit to dock to the Mothership using any of the 2 latches available which facilitates the docking procedure.




*Drawing of Deorbiter with Solar Panels extended*




*Drawing of Deorbiter with Solar Panels Closed*

## 2. Concept of Mothership

On-orbit servicing is part of a future major disruption in the space landscape. Along with the normal 12U deorbiter, we will be using a spacecraft called "Mothership".

We conceptualised a preliminary design of the Mothership whose primary goal is to act as a refuelling station for the deorbiter to re-fuel. However, the Mothership can be made a lot more advanced for a complete satellite servicing(which will be discussed later in the Future plans and Scalability) using advanced robotics, electric propulsion, vision based navigation and rendezvous and proximity techniques.

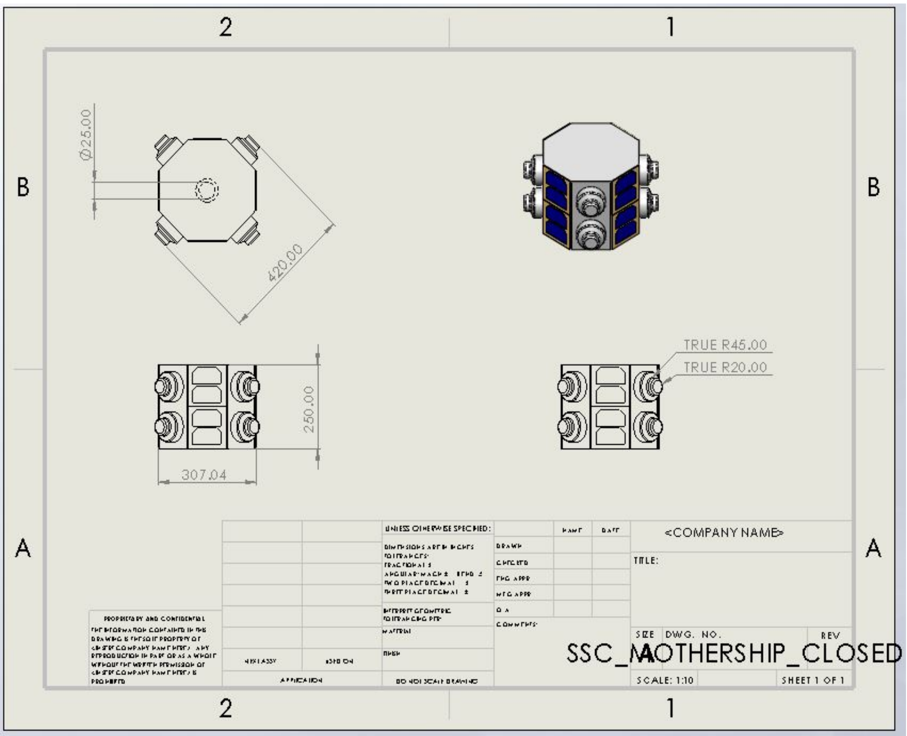
The idea of providing on-orbit servicing for satellite in both GEO and LEO is one of the upcoming missions of airbus. However, this would be made in general to accommodate servicing for most of the satellites in space. In our idea, Mothership is designed to tailor the needs of the 12U deorbiters which we are deploying for space debris removal[23].

**Design of Mothership**: We have made a fundamental design of Mothership with a current aim of acting as a refuelling centre to increase the life of the de-orbiter. The mothership will be stationed at a constant circular orbit of 900km and will station-keep using an ion thruster.

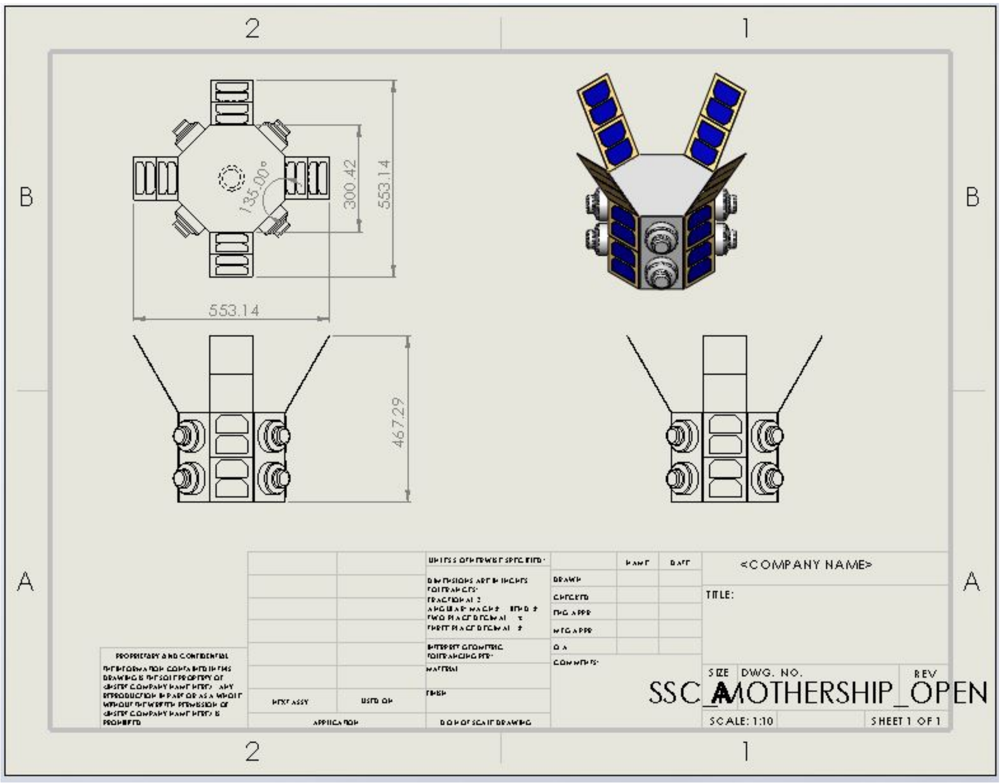
The main aim of the mothership is to refuel, it consists of all the general components every satellite has along with an additional storage tank for the fuel storage and docking ports. The following table has the breakdown of the major parts of the Mothership.

Many docking ports are incorporated on the mothership which makes it easier for the deorbiter to dock to any of the docking ports as the mothership without significant maneuvering required from the mothership.

| Sr NO. | Parameter | Dimension(mm) |
|---|---|---|
| 1 | BIT-3 Xenon Ion thruster[24] | 180 x 88 x 102 |
| 2 | Solar Panels | 82.6 x 325 x 1.6 |
| 3 | Attitude Determination and Control System ADCS | 43.5 x 43.5 x 24 |
| 4 | Command and Data Handling System(CDHS) | 96 x 90 x 12.4 |
| 5 | Electrical Power System(EPS) | 89.3 x 92.9 x 25.6 |
| 6 | Mothership dimensions(closed solar panels) | 307.4 x 307.4 x 250 |
| 7 | Mothership dimensions(opened solar panels) | 553.14 x 553.14 x 467.29 |




*Drawing of Mothership with Solar Panels Closed*




*Drawing of Mothership with Solar Panels Open*

## 3. Command and Data Handling

The command & data handling system includes their own computer that interfaces with various components present on the deorbiter and performs all the calculations required., data analysis, data comprehension of locating the space debris. The ISIS on-board computer(iOBC)[25] is a flight proven, high performance unit based on ARM9 processor with a clock speed of 400MHz and offers a multitude of standardized interfaces. The iOBC has a compact robust design which fits in well for the deorbiter requirements. Its power efficient and is compatible with multiple OS options.

**De-orbiter**: De-orbiter needs a high processing computer to track the debris location, managing constant communication, inputs for firing thrusters and precision in maneuvers and docking. The daughter board complements the main board which provides us with huge flexibility of customizing the board. We would be tailoring the daughter board to locate the space debris using the data received by the AGI Comspoc.

**Mothership**: Mothership uses the same CDHS module which is used in the de-orbiter. The mothership needs has similar requirements to that of the requirements like managing communications, inputs to stabilize in the station keeping orbit and docking when necessary.

Electronic components used in spacecraft must be built to survive the harsh space environment and function reliably in it. They are exposed to high energy radiations, which includes gamma radiations. Hence, it is very important to use the components which are certified for operating in space. However, taking into consideration our mission requirements, using flight proven components which are readily available in the market increases the success rate of the mission.

*On Board Computer*

## 4. Electrical Power System

*Electrical Power System*

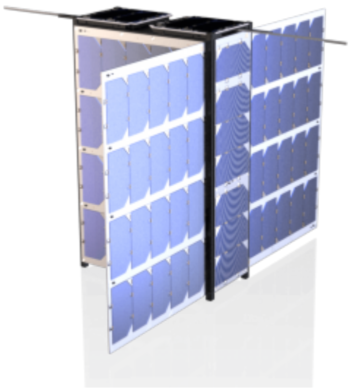
The electrical power system(EPS) encompasses electrical power generation, storage and distribution. EPS is one of the fundamental subsystem which plays a crucial role in the success of the mission. Our mission uses  Gomspace NanoPower P31u Electrical Power System  which facilitates the power distribution on the deorbiter. The Endurosat 3U X/Y solar panels[26] used for the deorbiter are supported by this EPS system.
Since we are using electric propulsion, our mission needs a lot of power to achieve the desired functionality. Multiple solar panel arrays will be used to generate the required power. One array will have 7 solar cells, producing nearly 8.3W of power, hence we will be using about **25 solar arrays on each deorbiter**.

The Gomspace NanoPower P31u is tailored for small-satellite/cubesat missions, with one unit providing upto [27]30W of power. **Due to its compact size, light-weight and the given power requirements,, we will be using 10 units of these on each deorbiter and 5 on the mothership to generate enough power.** The P31u fits to PC104 embedded systems standards, using triple junction photovoltaic cells and a highly efficient boost converter. Note that the system used is with an on-board battery.

## 5.Power Generation

As of 2010, approximately 85% of all nanosatellite form factor spacecraft were equipped with solar panels with rechargeable batteries[28]. Hence, we are using the typical method of using solar panels for generation of the power needed for the functioning of the deorbiter.

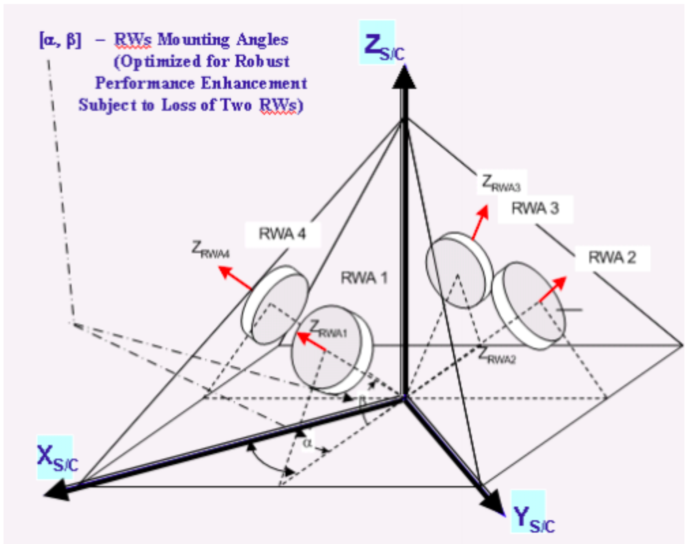
The mission uses the new 3U Endurosat X/Y solar panels which are complementary with the Gomspace NanoPower P31u Electrical Power System card and  the accompanying battery Unit. The space qualified  panels produces 8.3 Watts of power each using 7 high efficiency (29.5%)[29] . Along with a very high efficiency, it is integrated with temperature sensor and a heater to maintain a threshold temperature in order to charge the lithium ion batteries.



***Endurosat Solar Panels***

## 6. Attitude Determination and Control System (ADCS)

It is imperative for the deorbiters to have precise attitude control so that the thrusters are oriented in a way which maximises momentum transfer to the debris. Additionally, it is also essential for the deorbiter to have the desired attitude during the refuelling maneuver. Thus, the ADCS of the deorbiters should be precise as well as robust.




*Reaction Wheel Configuration*

The attitude of the satellite will mainly be controlled by a set of 4 reaction wheels[30] mounted in a pyramid[31] configuration as shown in figure -. 4 reaction wheels are chosen primarily to impart redundancy to the system. It is also a well-known fact that reaction wheels saturate over time and thus, using 4 reaction wheels will decrease the decay rate to saturation. The primary reason for choosing a skewed pyramid configuration is the fact that even if 2 reaction wheels fail due to ageing, a pyramid configuration still makes it possible to provide weak attitude control along all 3 axes, and thus is expected to have a longer mission life than the standard 3 reaction wheels orthogonal configuration. The impact of reaction wheel failure for pyramid configuration is illustrated in the table below.

| Reaction Wheels Failed | Impact |
|---|---|
| 1 | No impact, Attitude Control still has full functionality |
| 2 | Weak control over all 3 axes still possible. |

In addition to the reaction wheels, the ADCS will consist of 4 magnetorquers (1 for redundancy). The major function of the magnetorquers is to desaturate the reaction wheels by generating an equal countertorque. The magnetorquers can also be used for minute attitude corrections in combination with reaction wheels. Magnetorquers are most efficient at low altitudes due to the relatively high magnetic field there and hence are ideal for LEO. Three magnetorquers will be placed in an orthogonal configuration with the fourth magnetorquer placed in a way such that it is symmetrical with respect to the three magnetorquers' axes.

**The same attitude determination and control system will be used on both deorbiters and the mothership.**

### 7. Launch Plan

The deorbiter and mothership have been designed like traditional cubesats, hence they will be compatible with almost all launching platforms like Tesla, NASA, JAXA or ISRO rockets. For this mission, we have chosen Indian Space Research Organisation's (ISRO) Polar Satellite Launch Vehicle (PSLV) rocket. This choice was made on the basis of cost-effective launch plans offered by ISRO, which are remarkably cheap, and their proven track record of successful deployment of cubesats, with one of their many successful launches being the latest one of Cartosat-3 from Sriharikota[32].

The launch inclination will be 99 degrees and will take place from Satish Dhawan Space Centre in Sriharikota, with a launch azimuth of 140 degrees. The cubesats will be launched on a ride- share basis (the cubesats will be a part of a bigger cohort of satellites being launched together, in this case courtesy the PSLV), thus further reducing the launch costs, which are expected to be around S$844,000 for one cubesat.[33]

The cubesats will be housed in QuadPack CubeSat Deployers developed by ISIS[34]. While being launched, the cubesats will be enclosed in the deployer, and will be released into orbit only when signalled by the launch vehicle. The cost of the deployers is included in the miscellaneous costs.

A limitation of this method of launching is that the rockets being relied on are single-use rockets, thus driving up the costs. However, as the ODDS concept is improved upon and more deorbiters are launched with every attempt, the launch costs will be offset by the increased lifespan of the deorbiters being deployed and their own increased servicing period as well.

### 8. De-orbit Plan

The mission has been planned in such a way the deorbiters and the mothership will deorbit themselves to a graveyard orbit (350km) at the end of their lifetime. Both the deorbiters and the mothership have propulsive capabilities which makes it easy for them to execute a retrograde maneuver to deorbit themselves to a graveyard orbit. At the graveyard orbit of 350km , atmospheric drag will take over and cause the satellite to burn out in the atmosphere.

## E) Propulsion

### 1. Electric Propulsion

Electric propulsion has been in use for quite some time now but its relevance is growing day-by-day as the human race tries to find new ways to propel its spacecrafts through the vast universe. The propulsion technique employs the use of electric power to accelerate the propellant by electrical or magnetic means thus generating a thrust according to Newton's Third Law of Motion.

The fundamental idea of electric propulsion is that the propellant of the propulsion system is converted to a plasma using a process such as heating or electron bombardment. The plasma is the working fluid of the electric engine. A plasma[35] is a homogenous mixture of electrons, ions and neutral particles. If neutrals are absent in the plasma, the plasma is said to be ideal. Electromagnetic fields can act on these charged particles and expel them out much in the same way that exhaust gases are expelled from a chemical rocket engine. This in turn produces a thrust given by the following equation:

$$F = \sum \dot{N}_i M_i C_i$$

The subscript $i$ in the equation refers to the species of charged particle which is being accelerated and C is the exhaust velocity of the charged species.

There are many different kinds of electric propulsion methods which have been proposed and used over the years. The one that is of particular interest to us is ion propulsion. Ion propulsion is a type of electric propulsion which accelerates ions in order to produce a thrust. The ions are produced by ionizing a gas like xenon using energetic electrons. The positive ions formed by this process are accelerated using an electric field. One might argue that this process will cause the spacecraft to develop a net charge. This is easily rectified by using another cathode placed near the emitted beam of positive ions which emits electrons into the accelerated beam of ions to render it electrically neutral.

## 2. Performance Parameters

While selecting the optimum engine to use for the deorbiter and mothership, we considered certain performance criteria to help ease the process of selection. The propellant flow rate, thrust produced and ΔV[36] were of particular interest to our deorbiting system as the main deorbiting mechanism depends on the amount of thrust that is provided to the debris, we need sufficient amount of ΔV to complete maneuvers between two different orbits after deorbiting and for maneuvers and the propellant flow rate must be low enough that the lifetime of the deorbiter is long enough to deorbit multiple debris objects and be able to complete a refuelling maneuver.

Our mechanism of deorbiting, which uses our thrusters, requires that our thruster produces a few millinewtons of thrust in order to send the debris object into a decaying orbit around the Earth. The ΔV consideration is important because of in-space orbital maneuvers that our spacecraft will have to perform after the deorbiting is complete and for maneuvers from the mothership. A propellant flow rate which guarantees a lifetime of 1.5-2 years of the deorbiter on one full load of fuel will be sufficient and one refuel maneuver will be sufficient to extend the life of the deorbiter to almost 4 years which is our aimed mission length.

Luckily for us, we found the perfect thruster in the form of the Busek BIT-3 Thruster.

## 3. Busek BIT-3 Thruster

Based on these performance parameters, the ion thruster that we have selected for our deorbiters and mothership is the BIT-3 ion thruster which was developed by Busek. Busek[37] has been known to develop ion thrusters of supreme quality and their thrusters have been used on other missions such as the NASA/ESA LISA Pathfinder mission.

The specifications of the BIT 3 ion thruster are as given below:

| | |
|---|---|
| **System Power** | 56-80 W |

| | |
|---|---|
| **Ion Beam Current** | 9-17 mA |
| **Propellant mass flow** | 40 µg/sec |
| **Thrust** | Up to 1.25mN |
| **Specific Impulse** | Up to 2300 sec |
| **Delta V** | Up to 2.5 km/s<br>(14kg CubeSat) |
| **Operating Temperature Range** | -10 to $45^0$ C |
| **Dry Mass** | 1.28 kg<br>(without gimbal) |

**Table**: Specifications of BIT 3 Ion thruster for Iodine as the propellant [2]

In order to proceed further in our calculations, we required the specifications of the BIT-3 thruster when Xenon is used as the propellant but we could only locate the datasheet for the BIT-3 thruster with iodine as the propellant. Therefore, we have assumed similar specifications for Xenon and proceeded to calculate the required values.

The BIT 3 thruster meets all of the specifications required for the deorbiting mission. The system uses an input power of approximately 56-80 W. The maximum thrust produced by the thruster is approximately 1.25 mN which is more than sufficient for our deorbiting system to be effective. It has a high maximum specific impulse of approximately 2300 sec which is ideal for our mission.

## 4.Propellant: Xenon or Iodine?

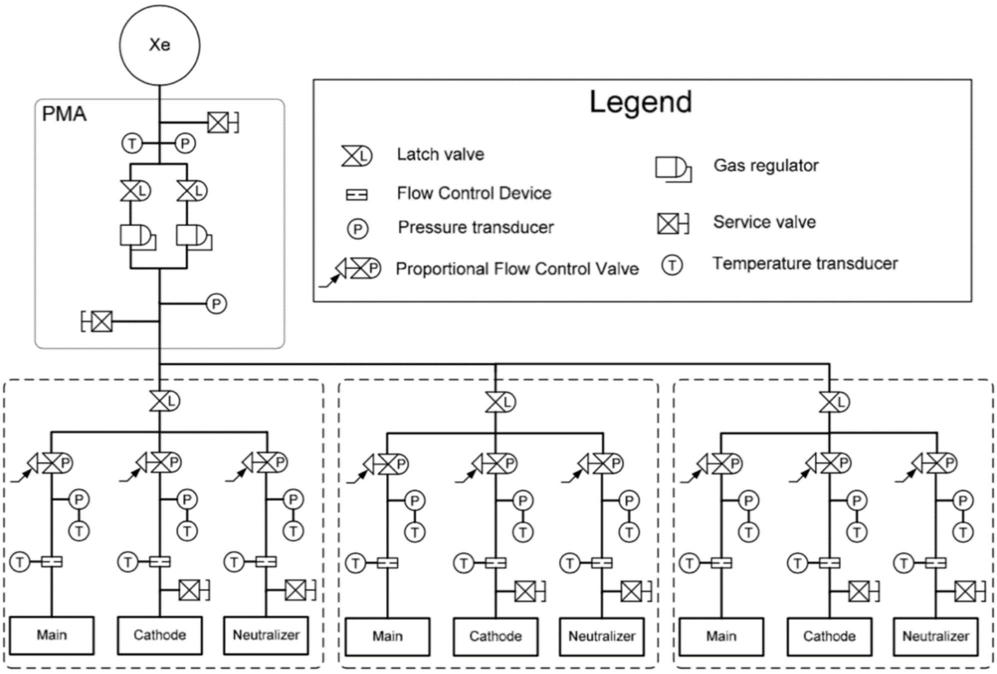
The propellant that we will be using for the thruster will be Xenon[38] gas. The BIT 3 thruster was originally designed by Busek to be a state-of-the-art iodine thruster and revolutionize propulsion technology by actually using a solid propellant in an ion thruster. Even though the iodine propelled BIT 3 engine has a higher specific impulse of 2300 sec and produces a higher thrust of 1.25 mN, there are several potential risks associated with using iodine as a propellant. These are the following:

1. Iodine must be sublimed[39] in order to use it as the propellant of the ion thruster.
2. There is a risk of iodine deposits being formed in the propellant system which can cause problems to the working of the system. Iodine can react with the materials of construction of the propellant system and affect the structural integrity of the structures.
3. Iodine, being a solid propellant, will be harder to refuel than a gaseous propellant like Xenon which can easily be refuelled through fluid connections.

## 5. Propellant Feed System

As mentioned previously, our deorbiter is designed to have two thrusters. The front and back face of the deorbiter each have one thruster attached to it. Accordingly, a propellant feed system had to be designed in order to transfer the propellant from the fuel tank to the thruster and neutralizer[40] of the BIT-3 thrusters. The feed system and the propellant system of the mothership will be the same as the one designed for the satellite with a few changes to the layout of the feed system and the dimensions of the tank. In particular, the mothership requires only 1 ion thruster for stationkeeping and thus will be simpler than the one described below.

The parameters that influenced the design of the feed system were the mass of the fuel to be carried on the deorbiter, the pressure at which to maintain the Xenon gas, the flow rate of the propellant and the thrust generated by the thrusters. The design was also influenced by a feed system that was designed by NASA for an ion engine operated on Xenon for their missions. The figure below shows a simplified schematic diagram of the propellant feed system:

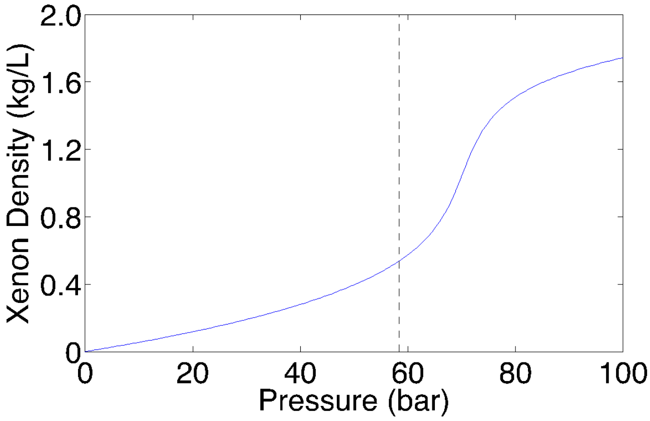



*Simplified schematic diagram of the Xenon Feed System developed by NASA which is used in our satellite*

The schematic shows one Xenon tank which provides fuel to three thruster systems, however we will only be using two thrusters and thus require only two main fuel lines which will transport the propellant from the tank to the thrusters. In addition to the two main fuel lines, we will require another two fuel lines which will connect the docking port of the deorbiter to the fuel tank. This will allow us to refuel the fuel tank when the deorbiter docks with the mothership. Another difference between the schematic shown and the design of our fuel system will be the use of the third fuel line branching off from the main fuel line. This fuel line will be used to power the backup neutralizer present on the propulsion plate of the fuel system rather than the cathode of the system as the BIT-3 thruster, being an RF ion thruster, does not require a cathode to operate. The backup neutralizer will be used in the event of a failure of the original neutralizer present on the propulsion plate.

The design of the fuel tank is dependent on the volume of the fuel that we want to carry. As we know the density of the fuel as well as the mass we want to carry, we can easily estimate the volume to be 3 litres. Now while designing the fuel tank itself, we were faced with the choice of using a cylindrical fuel tank or a rectangular fuel tank. Of course, the deciding factor for this particular dilemma would be the dimensions of the tank which are further constrained by the dimensions of the satellite mentioned in Section D. If we assumed the tank to be cylindrical, we found that to encompass 3 litres of fuel we would need a tank with a diameter that is too large to fit into our satellite and hence a rectangular tank is the more viable option. The dimensions of the rectangular tank were decided to be 120x100x200 mm. However, due to the rounded edge design of the fuel tank these dimensions will be reduced but considering the tank as rectangular is an appropriate assumption for our design calculations. The thickness of the tank was calculated according to the basic stress analysis of a thin walled cylindrical[41] pressure vessel. Our assumption of the tank behaving as a cylindrical pressure vessel is valid as the tank has rounded edges and the approximate shape of the tank can be regarded as a cylinder. According to our calculations, the required thickness for a stainless-steel tank will be 3 mm. Therefore, the mass of the tank without fuel will be 1.5 kg.

The mass of the fuel which is to be carried on the spacecraft is 5 kg. The pressure of the tank is fixed to be at 100 bar. This particular pressure has been chosen because Xenon exists in a supercritical state[42] at this pressure which means that it has a very high density of 1.75 kg/L and we will only require a small volume of 3 L to carry 5 kg of fuel which is ideal for a CubeSat. These calculations are easily inferred from the graph shown in the figure which is a plot of the relationship between density of Xenon and the pressure of Xenon.




*Graph showing the relation between Pressure and density for Xenon gas. The dotted vertical line indicates supercritical pressure*

The temperature of operation of the propellant system must be above the critical temperature of Xenon. Therefore, keeping in mind the operation temperature of the other systems present, the temperature of operation of the propellant system during full operation of the thruster is 300 K. This is well within the temperature of operation of the BIT-3 ion thruster as mentioned in the BIT-3 datasheet above .

The mass flow rate of Xenon is 40 µg/sec and the thrust produced by the engine is 1.2 mN. Therefore, the exhaust velocity of the engine can be calculated to be 23 km/s. Using these values, we have obtained a linear relation between the mass of the debris we want to deorbit and the mass of the propellant that will be consumed while deorbiting the debris which has been previously mentioned in Section B. Using these specifications, we can calculate the exact diameter of the fuel lines that will transport the fuel from the fuel tank to the thruster and neutralizer.

The final element of the design of the fuel system is the propulsion plate which has the proportional flow control valves mounted on it and it will also have the thrusters and neutralizers mounted on it. The pipeline after the flow control valves will have a flow control device, pressure transducers, temperature transducers and service valves present on it.[43]

The figure shown on the next page is a simplified 3D model of our propellant feed system along with the mechanical drawings of the feed system and propulsion plate:

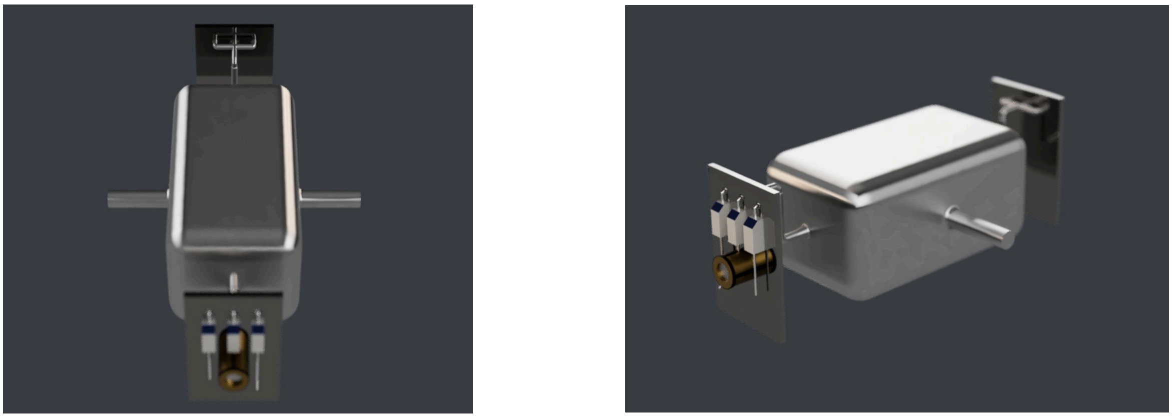


**Figure:** Simplified 3D model of the propellant feed system which will be used on our deorbiter

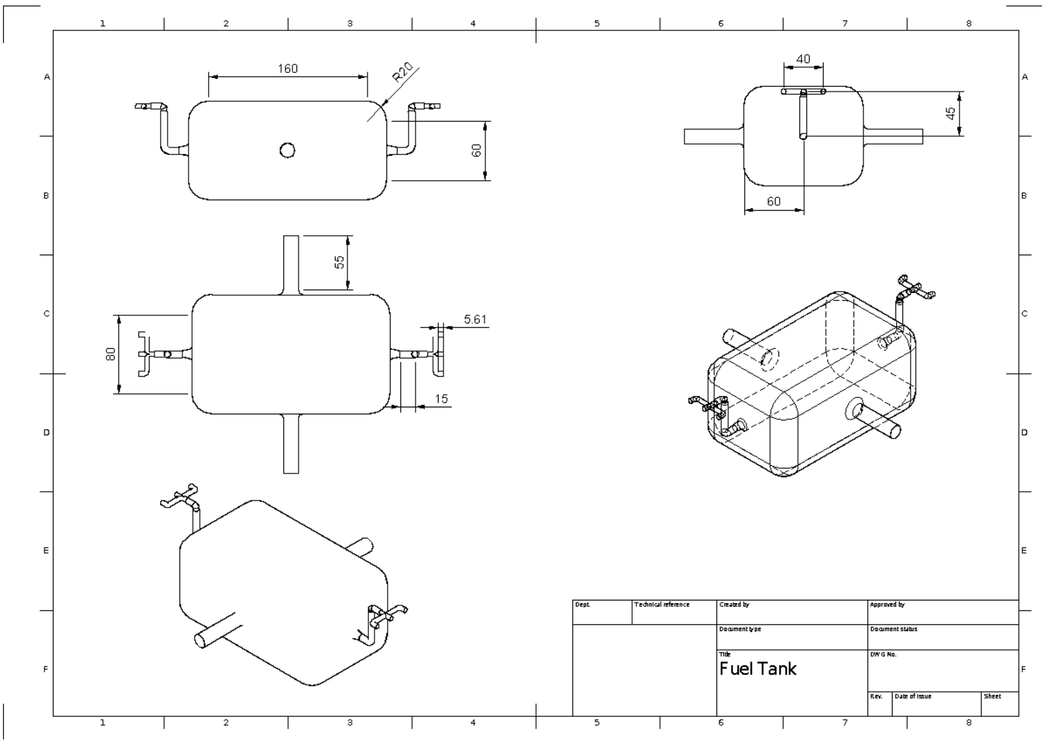
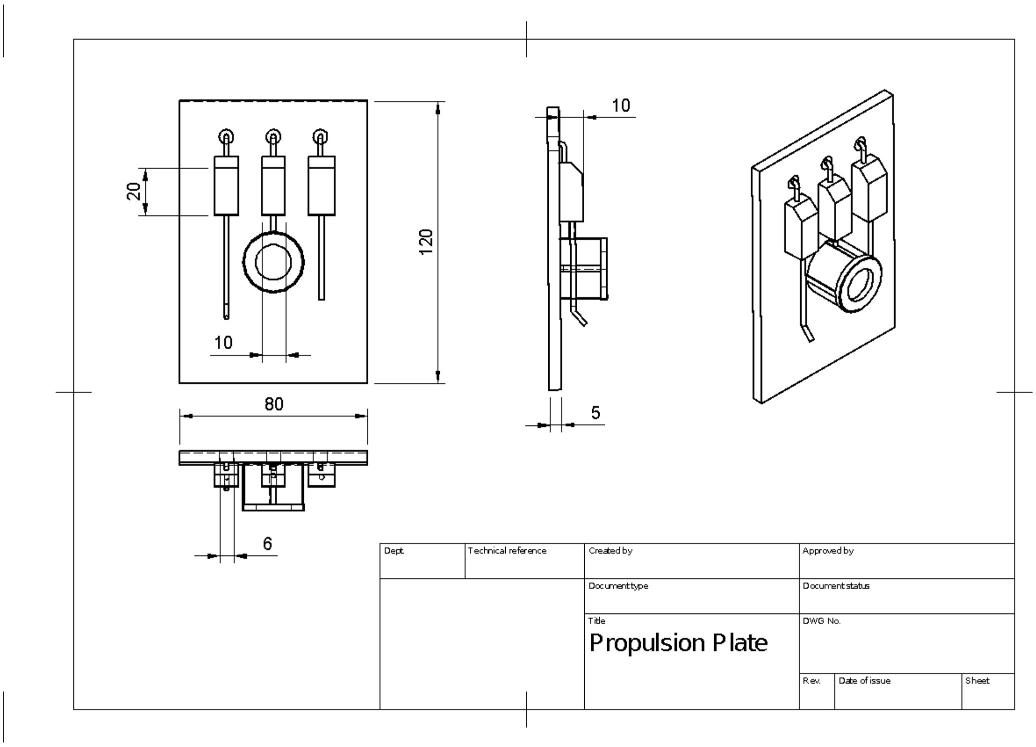



**Figure:** Mechanical Drawing of Fuel Tank




*Mechanical Drawing of Propulsion Plate*

## 6. Docking System: TriDAR or DragonEye?

Rendezvous and docking have been an integral part of many space missions ever since the day that Neil Armstrong and Dave Scott performed a manual rendezvous and docking maneuver of the Gemini spacecraft with the unmanned Agena target vehicle. However, these days autonomous rendezvous and docking systems have been developed which eliminate the need for any human intervention in the process of rendezvous and docking. These methods have been developed on the basis of imaging and ranging systems which can create 3D visualizations of the target body for docking.

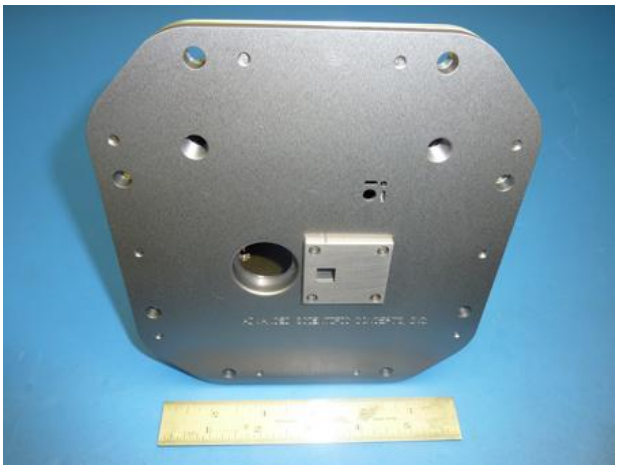
For a long time, the available systems were built for rendezvous and docking with cooperative targets. For example, the placement of retro-reflectors on the target bodies to enable rendezvous and docking with the target body. However, it is not always practical to place a mechanism on the target body which can convert it to a cooperative body enabling us to perform rendezvous and docking maneuvers. A prime example of this is space debris. Space debris has no mechanism which will help us to rendezvous and dock with the space debris. We found two systems which can help us to dock with such targets and we were confused as to which one to pick for our CubeSat. The TriDAR system developed by Neptec and funded by NASA and the Canadian Space Agency, and the DragonEye developed by Advanced Scientific Concepts Inc for docking with the International Space Station and on-orbit satellite servicing. The two systems are discussed below and the merits of each system are taken into account while selecting the optimum autonomous rendezvous and docking system.

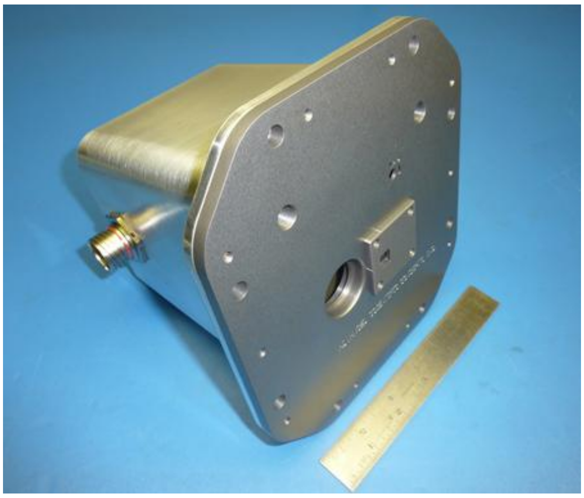
TriDAR is a combination of two systems: triangulation[44] and LIDAR. The LIDAR system is a surveying method which uses laser light to measure the distance to a target. During the operation, the target body is illuminated with laser light and the reflected light is measured with a sensor. The differences in return times of the laser light and wavelengths can be used to create 3D representations of the target body. Essentially, it is a radar which uses laser light instead of radio waves. TriDAR combines this new form of radar technology with a thermal imager to create a system which uses laser triangulation technology and LIDAR to provide enhanced docking and rendezvous capabilities. The system uses the information contained in successive 3D images and matches it with the known specifications of the target object to obtain the position and orientation of the target body. TriDAR provides the capabilities of two 3D sensors in one and therefore, it is the economical choice to use TriDAR on our deorbiter. It has a very wide range of operability. Thus, it is useful in the short-range docking phase of the satellite as well as in the long-range approach phase.

Even though TriDAR is an apt rendezvous and docking system to use onboard our CubeSat, the system is too large and bulky to fit in the satellite that we have designed. Thus, we must make use of a different system which can easily fit on our satellite and provide us with a reliable service. Thankfully, we found the perfect replacement in the form of DragonEye.

The DragonEye[45] system was developed by Advanced Scientific Concepts Inc for NASA's Commercial Orbital Transportation Services. The DragonEye is lightweight and has a small size, making it the perfect candidate for our rendezvous and docking system as we are using a CubeSat. The DragonEye system uses a Flash LIDAR camera to perform proximity operations in space. Flash LIDAR is essentially the same as a normal LIDAR system. However, a flash LIDAR only uses a single pulse of laser light in order to map a target region. It has the capability to capture 128x128 range 3D pixels per frame up to 30 frames per second, allowing 3D data to be generated in real time. Other than its smaller size, using a Flash LIDAR system[46] has several advantages such as improved spatial resolution and it can provide platform relative position and attitude angles. It has a 45x45 degree field of view and a range of more than 1.5km. It is easy to interface with the command and data handling interface and has also been flight tested several times. It was flight tested on the STS-127 and STS-133. Recently SpaceX also used the DragonEye system on their Dragon Vehicle for performing proximity operations. It also uses cutting edge technologies such as Class I eye-safe lasers for illumination, real time images without motion distortion, a non-mechanical camera which works via an ethernet connection.

The DragonEye system is lightweight and has a small form factor which means it can fit well with our CubeSat System. In view of these advantages, the DragonEye will be deployed on our deorbiter to facilitate rendezvous maneuvers of the deorbiter with the target debris and docking maneuvers of the deorbiter with mothership for refuelling.

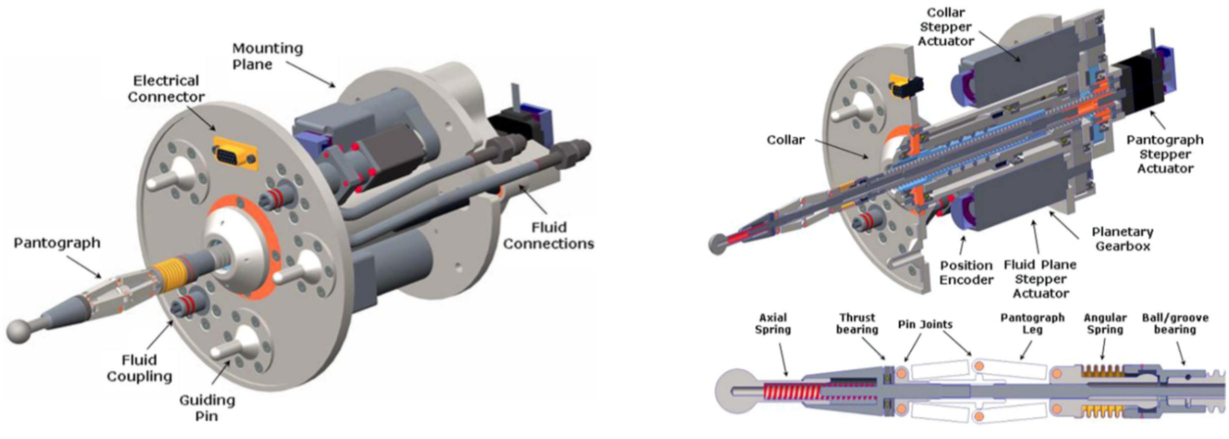




***ASC DragonEye***

## 7. Refuelling system: ASSIST

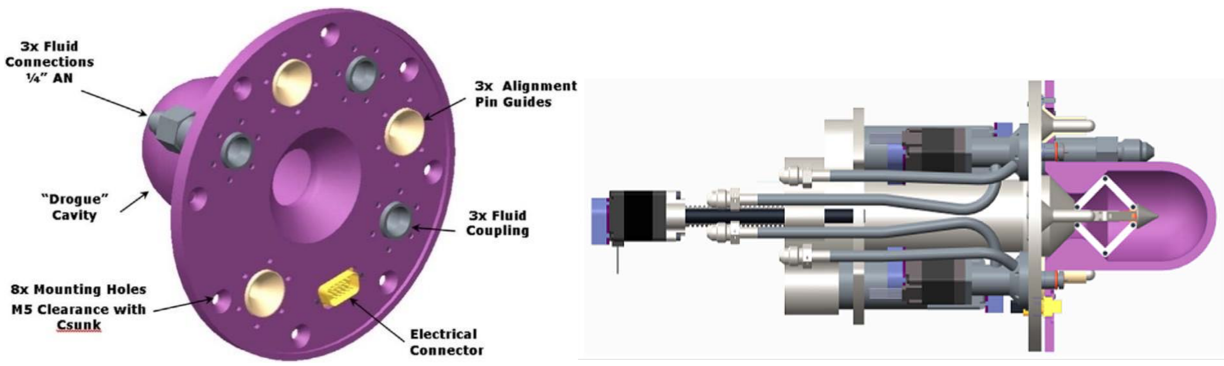
The refuelling system which will be used by us is similar to the one which was proposed by the European Space Agency a few years ago. The system is called ASSIST. The aim of ESA was to create a standard refuelling provision which could be installed on satellites and other spacecraft to enable in-orbit refuelling of these spacecrafts. Our method used for docking will be different from that proposed by ESA as has been mentioned in the previous subsection. However, our main interest is the end effector as that is what will be used by us to transfer the fuel from the mothership to the deorbiter.




*3D Model of the end effector that will be used on our docking system in order to carry out maneuvers, operations on the deorbiters to help increase their lifetime.*

In the original concept, the authors mention that the end effector will be attached to a robotic arm which will facilitate the docking of the servicing and serviced spacecraft. However, we will be attaching the end effector at the end of the fuel line joining the centralized fuel tank to the docking port of the deorbiter. The most important aspect of the end effector is what is referred to as the fluid plane[47]. This is the plane of the end effector on which the fluid coupling, alignment pin and electrical connector are present. A collar present on the fluidic plane allows to ensure the final and hard docking process. Once the fluid plane has been transferred and the preload has been applied, the system will be secured and fluid pressure or external torque will not be able to separate the fluid planes.

The end effector of this system has fluid connections attached to it which will connect to fluid couplings on the berthing fixture present on the mothership. After successful docking takes place, the fluid coupling on the end effector on the satellite will be securely fastened to the fluid coupling on the mothership. Each of the couplings will be pressurized with a gas like Nitrogen or Helium and the pressure decay will be monitored to detect any leakage. Now, Xenon gas will be transferred from the fuel tank of the mothership to the fuel tank of the satellite through the fluid connections which connect the pipeline of the mothership and satellite.

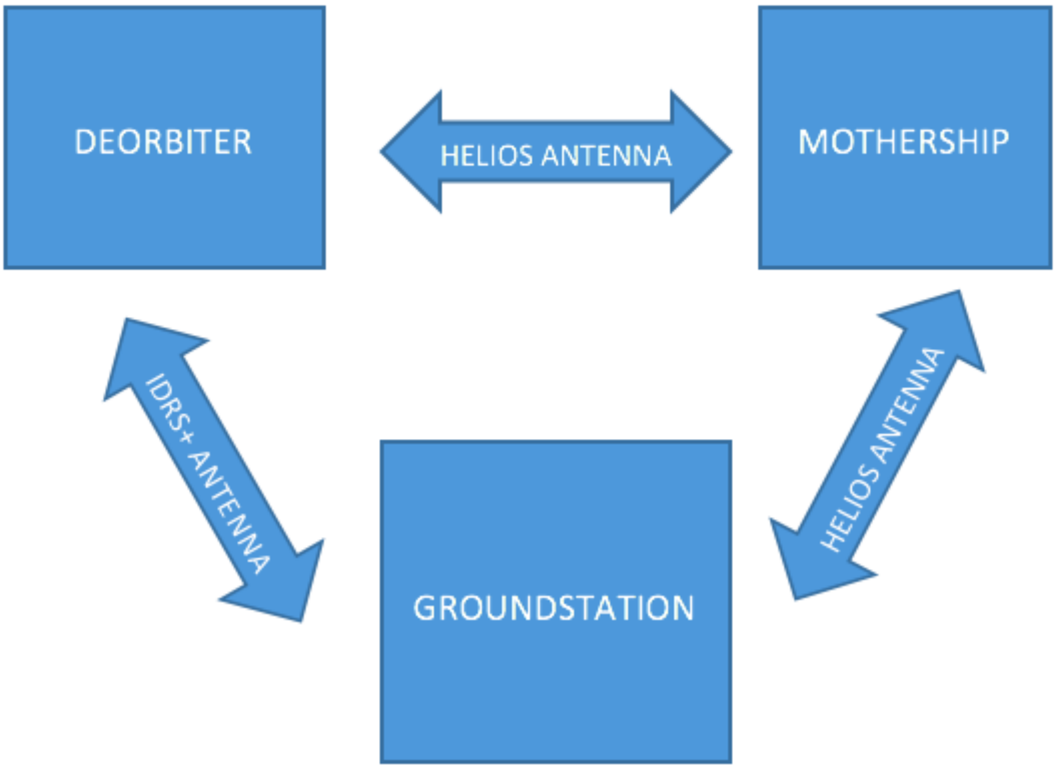



*Berthing mechanism which shall be used on the mothership and the view of the system after full docking has taken place.*

This system is being considered by Airbus for their on-orbit servicing spacecraft SpaceTug[48] and it is well on its way to becoming an international standard for in-orbit refuelling operations. That is why we have decided to use it for our satellite-mothership system as it is a simple mechanism which can be easily placed at the end of the docking structure of the satellite and mothership to give a simple and elegant means of propellant transfer.

## F) Communications

In a mission requiring identification of the location of debris (with further tracking of it) and careful control of the orbits of the mothership and the deorbiters, proper and continuous communication is imperative in order to minimize errors, which can have potentially debilitating consequences. There will be three channels of communication in the ODDS mission- between the de-orbiter(s) and mothership, between the de-orbiter(s) and ground stations and between the mothership and ground stations.




*Communication Setup*

### 1. Processes and Types of Communication Setups Required

There are 4 major activities requiring an efficient and reliable communication setup during the mission, which can be classified on the basis of their need of the type of communication – constant or non-constant:

**i.**
**ii.**

**Non-Constant Communication –**

- **Debris De-orbiting** – This is the process where the deorbiter will be using ion thrusters to push the target debris off its existing orbit into a spiral towards the earth, which is the key act of the mission. Once the

  deorbiter completes rendezvous with the debris and stabilises itself at a constant distance of 3-4m from it, the deorbiting maneuver begins. The deorbiting maneuver is a fairly long process (50-100 days) and doesn't require constant communication feedback. Thus, the deorbiting process (**once the deorbiting maneuver begins and is confirmed to be stable**) will not require a constant relay of information with the ground station, and the communications made when the deorbiter is in line-of-site with the ground station are sufficient.

- **Maneuvering** – This is the process wherein the satellite will be changing orbits through prograde or retrograde maneuvers (**prograde maneuver when deorbiter goes from the graveyard to the next debris target, retrograde maneuver when deorbiter refuels at mothership and goes to the next debris target**) ; Since this process is again a standard idealised Hohmann Transfer, the trajectory of the maneuver can be easily and accurately calculated on paper. Established protocol can be executed without need for continuous data relay. The altitude in which the debris has to be targeted it pre-determined, and the deorbiter will be using its own tracking features to lower or raise itself into those orbits.

**Constant Communication –**

- **Debris Rendezvous and Deorbiter-Mothership Rendezvous** - Rendezvous with the debris will involve the deorbiter locating and then adjusting itself into a suitable distance of 3-4m from the debris so that it can use its ion thrusters to initiate the debris de-orbital, while the deorbiter-mothership rendezvous involves the deorbiter docking with the mothership to refuel before getting dispatched for another round of de-orbiting. Both the processes are encompassing some critical maneuvers, and at times it might become necessary to abort the maneuver or make required changes to the control algorithm by uplinking immediately from the ground, and hence a passage of constant communication is imperative to avoid collisions/incorrect docking sequence(s). Constant communication is achieved using Addvalue's IDRS[49], which is described below.

## 2. Rationale behind Selecting Different Methods of Data Relay

a) The sensitivity of the processes in terms of the room for safe maneuvering and scope of unwelcome physical contact between deorbiter-mothership or deorbiter-debris is an important consideration. The consequences of any of these objects coming into contact will be quite disastrous, and at the same time the two are intrinsically important to the key objective of the mission, hence require continuous data relay and monitoring to ensure a smooth execution.

b) Constant communications provisions for all four processes is possible in theory, but given the size of the mission and the instruments involved, it would be unviable to have such a system as it would eat into the available power unnecessarily, hence reducing the power available for other systems which need it more, and will also in turn affect the battery's durability due to uninhibited use and the constant heat generation it will create.

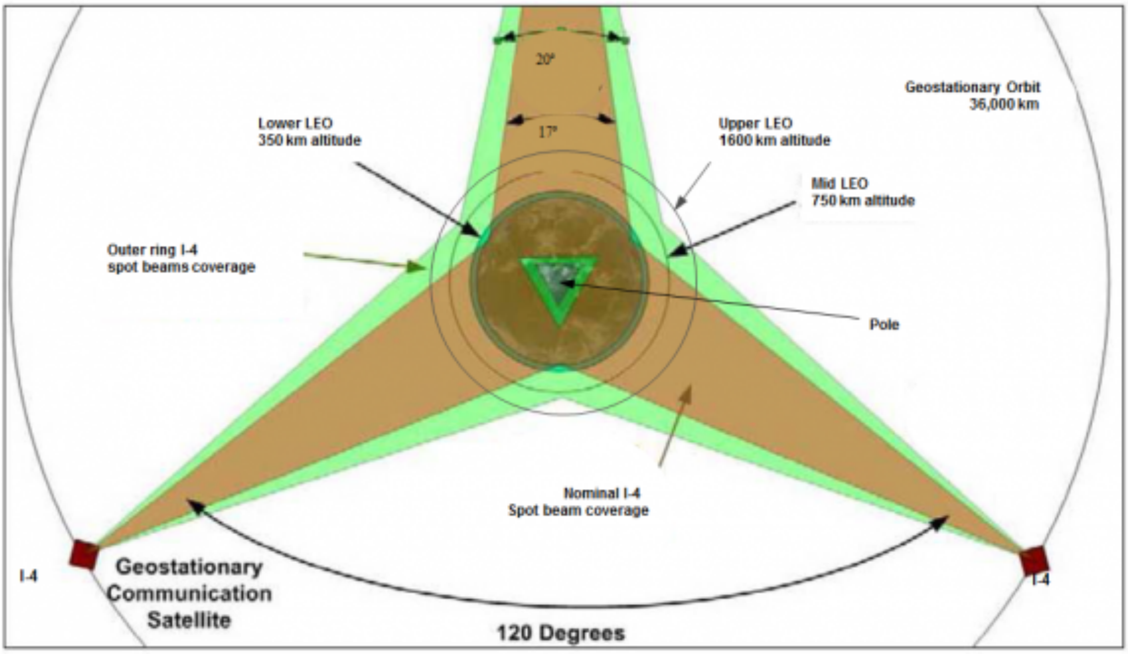
c) Financial and ergonomic prudence would suggest against having a constant communication setup for the entire payload when it is not needed, therefore budgetary constraints also play a role in deciding which processes get which type of data relay passage.

## 3.TDRS & IDRS

The Tracking and Data Relay System(TDRS)[50] consists of a constellation of 10 satellites in geostationary orbit around earth, of three different generations possessing multiple receiving and transmitting antennae, allowing for constant communication of LEO satellites with the ground. This system was developed by NASA back in 1983, and has had, since then, 3 iterations. Addvalue's Inter-Satellite Data Relay System (IDRS) borrows its fundamentals from the TDRS playbook, with some differences. TDRS is not available for commercial use currently,

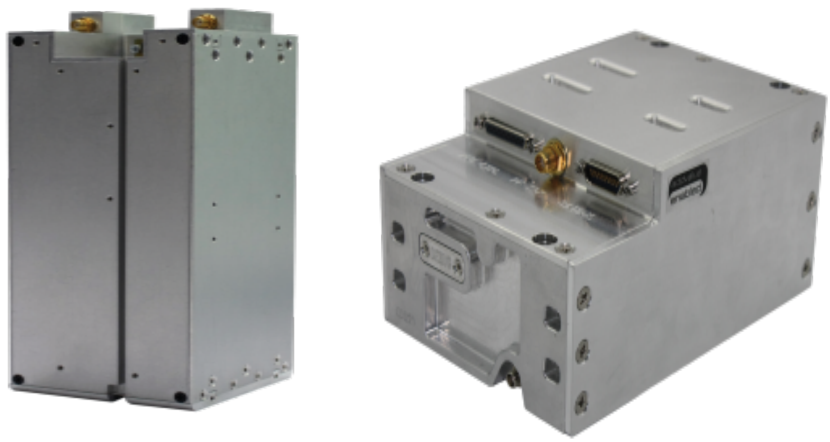
while IDRS has been built for commercial use. IDRS offers on-demand real time communication with LEO satellites and will be imperative for our mission.

The IDRS serves as an on-demand communication provision system with the help of a transceiver installed on the deorbiter. This transceiver links up with INMARSAT-4 geostationary satellites, which are in a geosynchronous orbit and provide Broadband Global Area Network coverage. The communication is two-way, with minimum latency. In the LEO, data transfer rates can be in excess of 200Kbps with the help of this setup[51]. The INMARSAT's BGAN is a tried and tested network[52], thus providing great reliability. The efficient real-time connectivity will be crucial for the debris rendezvous and the deorbiter-mothership rendezvous which have been described above in detail.



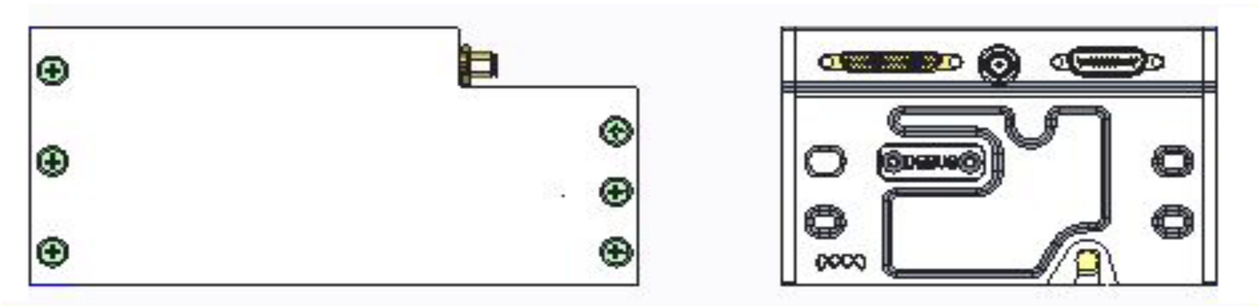

*IDRS System Concept*

The system will consist of an IDRS terminal functioning as a full duplex modem/router attached to the deorbiter, with the size of the terminal being around 2U form factor. This terminal consists of two BGAN transceivers for redundancy, which connect to the INMARSAT-4 satellites in GEO whenever commanded to, operating in the L-band. The INMARSAT-4 satellites can in turn contact the ground station, and vice versa. The weight of the transceiver[53] will be 2.3 kilograms in total, with a peak power of 22W. The key tasks like telemetry or tracking and command will consume only up to 10W of power. Latency time will be 0.5-1.5 seconds, with an expectation of short interruptions during satellite-satellite handovers and registration/reacquisition to a BGAN network. Network availability is present for 99.5% of the orbit, with forward link speeds for data transfer ranging from 200-300Kbps and return link speeds from 96-250 Kbps[54]. On-demand real-time communications are integral for some stages of our mission and thus Addvalue's IDRS in an invaluable component of the deorbiters.



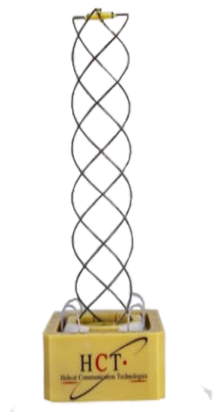
*IDRS Terminal and Transceiver*

The IDRS will have an ethernet interface supporting all major protocols and provide streaming, assured and background IP, with no interruption during spotbeam handovers. A typical spotbeam is expected to last for 2-3 minutes[55]. The use of BGAN network will also allow multiple deorbiters in orbit to communicate simultaneously.



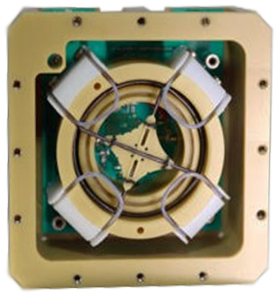
*Mechanical Drawing of the transceiver*

## 4. Communication Band and Antenna Used




The band to be used for communications linking will be the L band[56] (operating frequency of from 1-2 GHz). The primary reason for using L band is that it is much more efficient for tropical environments (akin to Singapore) as it can penetrate rain, fog and cloud covers. L-band is a tried and tested frequency for CubeSat communications, and it also is compatible with the IDRS being deployed. This frequency will be used for all channels of communication: ground-mothership, ground-deorbiter, mothership-deorbiter and INMARSAT-(deorbiter+mothership). Two antennae for communication will be deployed for redundancy as described below.

The deorbiter and the mothership will be equipped with Helios Deployable Antennae[57] for communications between them. This is a Quadrifilar Helical Antenna (RF Configuration) with both left- and right-hand polarization of antennae). The size is less than that of a 1U CubeSat, therefore being an ideal choice due to special restrictions. For further antennae properties, refer to Appendix (A). This antenna will be deployed through an electrical mechanism.



*Helios Deployable Antenna*

### 4. AGI ComSpoc

The data regarding debris location will be relayed to the deorbiters through AGI's ComSpOC[58] service, using the IDRS communication channel. The Commercial Operations Space Centre, or ComSpOC, is a facility created by AGI which tracks the real time location of over 9000 satellites in orbit (which it stores in a database), including defunct ones which we will be targeting. This facility uses a large variety of radio-frequency, radar-and-space and optical sensors[59] , located in diverse spots geographically which constantly keep track of the 9000+ satellites in its database, while also updating it for new additions to the existing satellites. The SSA Software Suite[60] fuses this data and produces a catalog whose data will be communicated to the deorbiters using IDRS, so that they can avoid collisions as well as track the debris constantly. This catalog will be available on a subscription basis, whose costs have been covered under miscellaneous cost (Appendix A).

## G) Conclusion

### 1. Limitations

Some of the limitations of our mission concept and their suggested remedies/ countermeasures are presented below-

**1) Limited lifetime of ion thrusters:** Given that the entire mission time is in excess of 3 years, there are doubts on the operational lifetime of the Busek BIT-3 Thrusters. The primary reason for the limited factor is the grid erosion of the thrusters. However, we are confident that with the increased research being done in grid-less thrusters, many state-of-the-art sensors will soon be able to replace BIT-3. NASA's Evolutionary Xenon Thruster(NEXT)[61](Too large for CubeSats) demonstrated operational lifetimes in excess of 48000 hours. Even in Singapore, Aliena is leading the way forward in electric propulsion. Thus, at the current rate of technological advances in propulsion, the lifetime of thrusters will soon be more than that needed for our mission.

**2) Ion Beam Interaction with the Debris:** In the case of our proposed deorbiting method, the Ion Beam Shepherd method, the impact of the ion beam on the debris can cause the formation of a low density plasma region behind the debris [62]. The non-neutral effects in this region can alter the force acting on the debris object and this has not been accounted for in our calculations. With increasing interest in the Ion Beam Shepherd method according to current trends, research interests in plasma physics are also increasing. In addition to this, the mission itself will serve as an experiment to understand the impacts of the low density plasma region. By comparing the deviations of the theoretical momentum transfer from the actual momentum transfer, we can get a rough estimate of the impacts of the low density plasma region. This will help us to account for any loss of force in our calculations and adjust the thrust required accordingly to deorbit the debris.

**3) High Precision Control Algorithms (Proximity Formation Flying):** In the execution of the Ion Beam Method, each deorbiter will have to rendezvous with debris and maintain a constant distance of 3-4m from the debris throughout the deorbiting maneuver. In addition to this, the deorbiter will also have to rendezvous with the mothership for refuelling. Thus, very high precision and novel, strong algorithms need to be developed and tested. Since the algorithms and control structure will be new, some shortcomings are expected. To account for these, we have included high R&D costs (Appendix A) so that comprehensive ground testing can be done. In addition to this, the use of IDRS will allow us to abort maneuvers by uplinking from the ground itself in case we feel there is a high risk.

**4) Power requirements during absence of sunlight:** While the quantity of solar panels used is more than sufficient to power the deorbiter and its thrusters when the solar panels are facing the sun, the thrusters can't be fired continuously during absence of sunlight as their high power requirements will quickly discharge the batteries. Hence, it is planned to keep them powered at a minimal thrust during this duration. This has been taken into account in our calculations in Section B by assuming that the thrust is powered down to 25% of its normal value during the absence of sunlight.

## 2. Additional Applications

Some of the additional applications of our mission concept are presented below-

**1) In-Orbit Servicing:** Our concept proposes using a mothership to refuel the deorbiters. However, the capabilities of the mothership can be extended much further to include in-orbit servicing. In-orbit servicing is a vast domain and some of the functions that ODDS can incorporate are refuelling services on a commercial basis, upgrade services such as payload swap to increase the satellite mission objectives, relocation services to offer a better optimisation of fleet operations, inspection services and life extension services.[63]

**2) Emergency Response:** There are many satellites in LEO which are defunct/ have no maneuvering capabilities. In a scenario where two such satellites are on a collision course, it is imperative to intervene and deflect one of the satellites from its current trajectory. The deorbiters in ODDS can thus serve as an emergency response system to keep two satellites on a collision course from colliding.

3) **Debris Attitude Motion Measurements and Modelling**: Till date, very little information is available on the attitude state of decommissioned intact objects in LEO. As the deorbiter maintains a constant distance from the debris (defunct satellite) for the duration of the entire deorbiting maneuver, it will be able to track and model the attitude motions of the debris with the help of DragonEye. Not only will this help strengthen our control algorithms for future missions, but it will also help other groups pursuing Active Debris Removal to better understand the complex motions of debris.[64]

4) **Large Number of Deorbiters**: Currently, a system of 2 deorbiters and 1 mothership has been proposed to demonstrate the feasibility of the concept. Once the concept is demonstrated, this system can be expanded to have a large number of deorbiters(10-20) and 1 very large mothership acting as a refuelling station. The large number of deorbiters could effectively clear a very large number of debris. With such a large number of deorbiters which results in increasing returns to factor, small plane change maneuvers (4-5 degrees) also become viable and hence this combined system could cover up to 10 degrees of inclination (4-5*2) with the mothership stationed at the central inclination.

## 3. Concluding Remarks

In conclusion, ODDS is a system to deorbit a large number of space debris in Low Earth Orbit. The mission report above has tried to demonstrate the feasibility of the concept and address the challenges faced. From time to time, it has tried to convey and defend the rationale behind the authors' decisions through factual as well as numerical analysis. The objective for ODDS was to integrate existing technologies with some novel technologies

to build a highly reliable, efficient and cost-effective deorbiting system and the authors feel that it has been achieved successfully.

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## Appendix (A): Breakdown of parts and costing analysis

| Subsystem | Name | Power | Dimensions | Weight | Cost | Quantity |
|---|---|---|---|---|---|---|
| Ion Thrusters | Busek BIT-3 | 80W per thruster | 180*88*102 mm | 1.28kg | Approximately S$ 75354 per thruster | 5 |
| Solar panels - solar cells with integrated magnetorquer , sun sensor and gyroscope. | Endurosat 3U Solar Panels X/Y | 8.43W per array | 325*82.60*2.5 mm | 127 g | S$ 6000 per set of 3U panels | 30*2(deorbiter) + 1*10(mothership) such sets |
| Communications Terminal | IDRS POC Terminal | 22W max. | 96*89*220 mm | 2.3kg | Unknown (Miscellaneous) | 2 pieces |
| Communications Transceiver | IDRS i100 d Transceiver | 8W | 125*96*70 mm | 1kg ( Included in terminal weight) | Unknown (Miscellaneous) | 2 (in the terminal) |
| L-band Antennae | Helios Deployable Antennae | 1W max. | 100*100*35 mm | 1kg | S$11000 per unit | 4 units |
| Propellant Feed System – consists of fuel tank, latch valve, temperature and pressure transducers, service valve and flow control device, proportional flow control valve | Custom Designed | | - | 1.5kg | Pressure Transducer - S$183*3<br>Temp. Transducer - S$152*6<br>Latch Valve - S$272*1<br>Flow Control Device - S$27*3<br>Service Valve- S$5*2<br>Proportional Flow Control Valve- S$408*3 | 2 assemblies |
| Electrical Power System | Gomspace NanoPower P31u | Unknown | 89.3*92.9*15.3 mm | 200g for one unit with battery, total 10*2(deorbiters)+ 5*1(mothership) needed | Unknown (Miscellaneous) | 15 units |
| Refuelling System | ESA ASSIST** | Unknown | Unknown | Unknown | Unknown (Miscellaneous) | 2 assemblies |
| Docking Rendezvous System | ASC DragonEye | 35W | 112*119*122 mm | 3kg | Unknown (Miscellaneous) | 2 (one for each de-orbiter) |
| Attitude Determination and Control System - includes reaction wheels and magnetorquers and control board | -NCTR-M002 Magnetorquer<br><br>-SatBus 4RW0 Reaction Wheels | Magnetorquer - 0.2W<br><br>Reaction Wheels- 3.2W | Magnetorquer-70 (length)*9(diameter) mm<br><br>Reaction Wheels - 43.5*43.5*24 mm | Magnetorquer- 30g<br><br>Reaction Wheels- 155g | Magnetorquer- S$1200<br><br>Reaction Wheels- S$6000 | 3 sets of 4 pieces of each |

| | | | | | | |
|---|---|---|---|---|---|---|
| | -Control board | Control Board-0.45W | Control Board - 86*88*2 mm | Control Board-47g | Control Board-S$20000 | |
| Command and Data Handling Systems (In-flight Computers) – ARM9 processor | ISIS iOBC | 0.4W | 96*90*12.4 mm | 76g | S$6630 per unit | 3 sets |
| Thermal Louvers | NASA CubeSat Form Factor Thermal Louvers | Unknown | 300*300*10 mm | 110g | $600 | 3 sets |
| 12U Bus Assembly | ISIS 12U CubeSat Structure | Unknown | 226.3*226*341 mm | 1.5kg | S$18000 | 3 sets |
| Research and Development | - | - | - | - | S$2,000,000 | For all the components, instrumentation. Includes cost for structural integrity testing and dry runs/on-ground simulations for the satellites. |
| Launch of Satellites | Using ISRO PSLV | - | - | - | S$844,000 for one satellite. | Three satellites in total. |
| Manpower/Labour Costs | | - | - | - | $7,00,000 | |
| Fuel Requirements | Xenon | - | - | - | S$163 for 100g | 25kg in total. |
| Miscellaneous Costs- includes company collaborations, sourcing of components and unknown costs - | | - | - | - | S$2,000,000 | - |
| Total Cost for one deorbiter, including all components, not including R&D, miscellaneous costs etc. - | | | | | S$503,215 | - |
| Total Cost - | | | | | S$9,019,858 approx (9 million SGD) | - |

## Appendix (B) – MATLAB Code

The MATLAB code used to run Ion Beam Shepherd calculations and find the optimum inclination is posted below.

```
%% Debris Mass vs Time Taken to Deorbit for Constant Thrust
G = 6.67 * 10^-11 ;           % Gravitational Constant
M = 5.972 * 10^24 ;           % Mass of Earth
R= 7171*10^3 ;                % Debris Initial Altitude (800km)
r = 6771*10^3;                % Debris Dispose Altitude (400km)
```

```
u = G*M;                    % Earth gravitational Constant
F = 1.2*10^-3;              % Thrust of BIT 3 Thruster
md=1;
efficiency= .75 ;
t = (efficiency*md * sqrt(u) * (sqrt(R) - sqrt(r)) )/(F*60*60*24*sqrt(R*r)) ;
% Time taken to complete Debris Deorbit (Obtained from Research Paper)
i = 1 ;
% Iterating over different debris masses
for md = 1:3:60
      t = (efficiency*md * sqrt(u) * (sqrt(R) - sqrt(r))
)/(F*60*60*24*sqrt(R*r)) ;
      t= t*(4/3) ; % Accounting for fact that thruster will function for 18
hours a day
      time(i) = t ;
      massd(i) = md ;
      i=i+1;
end

%% Time taken to deorbit vs Final Disposal Orbit
i=0 ; % New counter to ierate over debris disposal orbits
% A nested for loop is used to produce three datasets of different debris
% mass vs disposal altitude
for r = 300:10:450
 i = i+1 ;
 ro(i)= r ;
 r= (6371+r) *10^3;
 for md= 20:10:40
      t = (efficiency*md * sqrt(u) * (sqrt(R) - sqrt(r))
)/(F*60*60*24*sqrt(R*r)) ; ;
      t= t*(4/3) ; % At night thruster will be off, accounting for that
      if md==20
            td1(i) = t ;
      end
      if md==30
            td2(i) = t  ;
      end
      if md==40
            td3(i) = t ;
      end
 end
end
% Some corrections made to post process data cleanly



td1(17) = td1(16);
ro(17) = ro(16);
td2(17) = td2(16);
td3(17) = td3(16);


%% Exhaust Velocity of Satellite
Massflowrate = 40 * 10^-9 ; %% Mass Flow Rate of Propellent
Thrust = 1*10^-3 ;          %% Thrust
```

```
c= Thrust/ Massflowrate ;  %% Calculation and verification of exhaust
velocity

%% Mass of Propellent Consumed for debris deorbital vs Debris Mass
Massflowrate = 40* 10^-9 ;   % Mass consumed= Mass flow rate* Time
Massconsumed = 2*(2/3)* time * Massflowrate *3600*24 ; % Using variable
'time'

%% Mass of propellent consumed for posigrade maneuver
% Using Tsiolkovsy Rocket Equation
%In LEO, deltav=~ 3.5(deltar)
% delta r = 400 km , delta v = 114.28 m/s
% ln(deltav/exhaustv)= m0/mf
m0= 22; c = 22555;
i=1
deltartable=[] ; reqfuel= [ ]; % Computing empty arrays to store data
for deltar = 100:50:600 % Iterating over different delta r's
      deltav = deltar/3.5 ;
      mf = (m0/(exp(deltav/c))) ;
      reqfuel(i) = (m0-mf)*1.75 ; % Multiplying by factor 1.75 to account for
inaccuracy of Tsiolkovsy equation for ion thrusters
      deltartable(i) = deltar ;
      i = i+1  ;
end

%% Finding Optimum Algorithm
% Please note that the database of debris must be imported into MATLAB
% before running the simulation
inclinationarray = []; % Creating empty inclination array to store data
count=1 ;
for iteration = 61:2:99 % Labelling of Inclination array
 inclinationarray(count , 1) = iteration ;
 inclinationarray(count,2) = 0 ;
 count= count+1 ;
end

for counter =1:1:height(debriscatalog) % Iterating over the entire debris
catalog
 inclination = debriscatalog{counter,10} ; % Obtaining inclination
 apogee = debriscatalog{counter,11} ;
 perigee = debriscatalog{counter,12} ;
 meanaltitude = (apogee+perigee)/2 ; % Taking mean altitude as average of
apogee and perigee



 if ((60<= inclination) && (100>=inclination)) % Eliminating inclinations
below 60 and above 100

      if (400<= meanaltitude) && (meanaltitude<= 1000) % Adding required
debris to inclination array
            arraynumber = floor(((inclination -60)/2)) +1 ;
            if inclination==100
                  arraynumber= arraynumber -1;
```

```
            end
            inclinationarray(arraynumber,2) = inclinationarray(arraynumber,2)
+ 1 ;
      end
 end
end
plot(inclinationarray(:,1) , inclinationarray(:,2)) ; % Plot of inclination
array
xlabel("Inclination(Degrees)") ; ylabel("Debris Count") ;
```

## Appendix (C) – STK Simulation

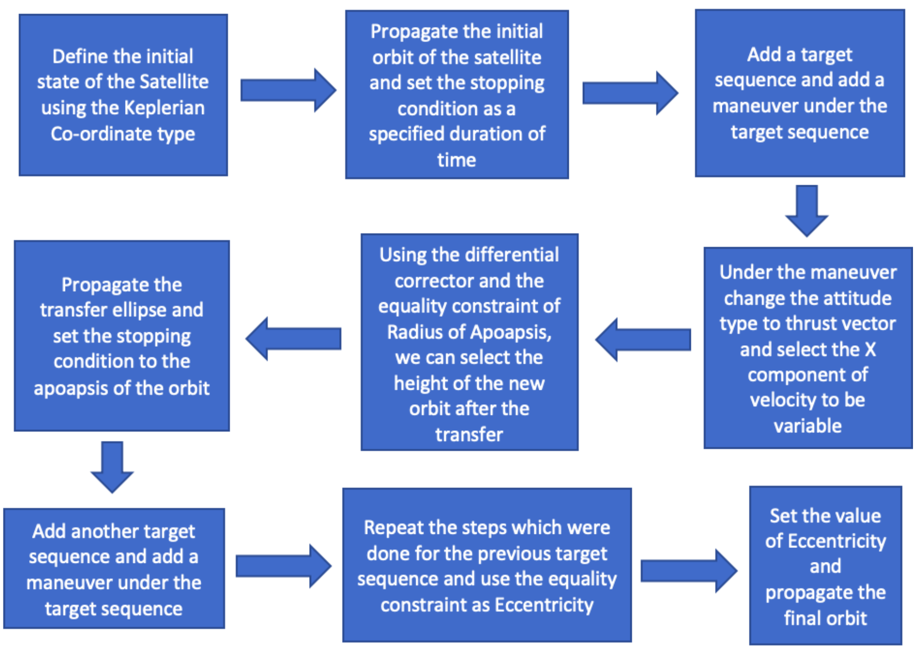
The simulation was created using the Systems Tool Kit software developed by Analytical Graphics Inc. We are grateful to them for having provided us with the software as well as the license to create the simulation for Singapore Space Challenge 2019/20.

Our simulations display several maneuvers that our satellites will perform during the mission. The orbits in the simulations have been extended a bit beyond the range of Low Earth Orbits to properly show the maneuver taking place. The docking maneuvers have been displayed by bringing the two objects very close together and keeping them 2 meters apart. The CAD models made by us for the report have been included in the simulation wherever it was necessary and possible to do so. The rendezvous maneuver was built using the pre-installed scenario called LEO Rendezvous.

Since the maneuvers required defining parameters and performing calculations, it was helpful to use Astrogator in the Basic-Orbit page of our satellites in order to model the orbits of our satellites. Now, we will carefully describe some of the maneuvers that we displayed in our simulation.

**1) Hohmann Transfer Orbit**

The Hohmann Transfer orbit is one of the simplest orbit transfer maneuvers performed by satellites. It constitutes of a burn at the beginning of the maneuver pushing the satellite in an elliptical orbit referred to as the transfer ellipse. Another burn is performed at the apoapsis of the ellipse to inject the satellite in another orbit around the same central body. We used Astrogator to make a Hohmann Transfer Orbit in the following analysis



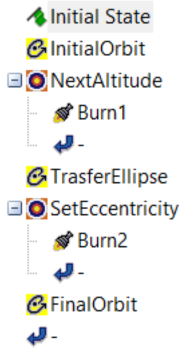
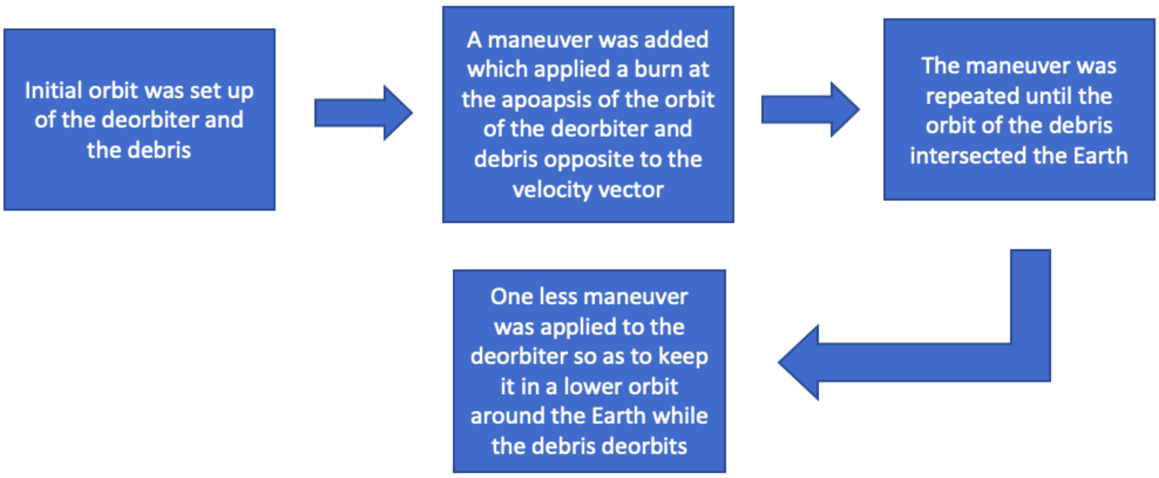
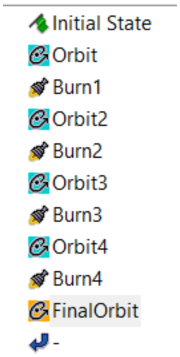



*Figure Mission Control Sequence of Hohmann Transfer Orbit*

## 2) Deorbiting Maneuver

Our deorbiting maneuver is based on the principle of applying a force on the debris object in a direction opposite to its motion so as to slow it down and send it into a lower orbit. Eventually, the debris object will enter the atmosphere and burn up in the atmosphere thus, completing the deorbit maneuver. This situation can be treated equivalently as performing a burn opposite to the velocity vector at a particular point in its orbit thus, sending it into an approximately spiral orbit and eventually deorbiting debris when its trajectory intersects the Earth. An initial orbit was set up at a specified altitude and the true anomaly of the debris object and deorbiter were set in a manner to keep them a small distance apart. The altitude used in the simulation is higher than the one we will use in the actual mission. This has been done in order to properly display the spiral orbit and subsequent transfer of the deorbiter to a higher orbit. The deorbiting maneuver was created as follows:

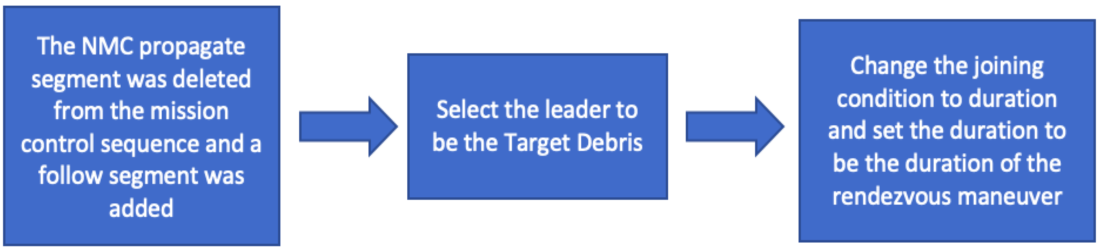


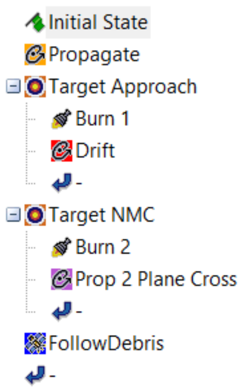



*Mission control sequence for deorbiting maneuver of Target Debris. The deorbiting maneuver of the Deorbiter will have one less Burn segment.*

### 3) Rendezvous and Docking

Before we can deorbit the target debris, we must rendezvous with the target debris before applying a force on it using our thrusters. It is also necessary for docking with the mothership and performing refueling maneuvers. Thus, it is important to create a LEO rendezvous maneuver. The docking maneuver shown is the placement of the mothership and deorbiter two meters apart from each other. Subsequently, they will orbit in that position for a specified duration while refueling takes place. The maneuver was created using the pre-installed scenario file called LEO_Rendezvous. The mission control sequence was edited to remove the final orbit around the target and replace it with a follow segment where the deorbiter will move closely behind the target debris and initiate the deorbiting sequence.




*Mission control sequence for LEO Rendezvous of Deorbiter with Target Debris*