CZECH TECHNICAL
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F3

Faculty of Electrical Engineering
Department of Computer Science

Master's Thesis

Active technology for space debris removal

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Study program: Open Informatics
Branch: Software Engineering

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ZADÁNÍ DIPLOMOVÉ PRÁCE

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Studijní program: **Otevřená informatika**

Specializace: **Softwarové inženýrství**

II. ÚDAJE K DIPLOMOVÉ PRÁCI

Název diplomové práce:

Aktivní technologie řešení kosmického odpadu

Název diplomové práce anglicky:

Active technology for space debris removal

Pokyny pro vypracování:

Kosmický odpad vzniká z člověkem řízených misí na oběžné dráze. Tyto mise jsou buď po svém ukončení anebo během svého provozu selhaly. Kosmický odpad lze klasifikovat podle velikosti a původu. Na zemské orbitě se pohybují tisíce mrtvých satelitů, desetitisíce velkých úlomků a desítky milionů malých úlomků. Od roku 1999 musela mezinárodní vesmírná stanice ISS provést 25 úhybných manévrů kvůli kosmickému odpadu. I pokud pomineme zvyšující se riziko na oběžné dráze, začínáme se přibližovat ke Kesslerovu jevu, který popisuje stav orbity, která je fyzicky neprostupná z důvodu navzájem se odrážejících a kolidujících úlomků. Všechny kosmické agentury mají v tuto chvíli nějaký program na vesmírnou ekologii a existují také výzvy OSN, které dávají jasné pokyny pro likvidaci současných misí. Pořád tato situace ale neřeší selhání misí a odpad, který se ve vesmíru již vyskytuje. Existuje celá řada navržených řešení například na bázi sítě nebo repulzivních magnetů.
Cílem této práce je provést rešerši možných aktivních řešení problému. Tedy takových, kdy dojde k aktivní destrukci a čištění odpadu. V rešerši použijte veřejně dostupné zdroje a elektronické knihovny odborných článků.
Dále je cílem na základě této rešerše navrhnout možné řešení problému a prakticky předvést jeho fungování pomocí prototypu, modelu, animace nebo simulace.

Seznam doporučené literatury:

Weeden, B., 2011. Overview of the legal and policy challenges of orbital debris removal. Space Policy, 27(1), pp.38-43.

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III. PŘEVZETÍ ZADÁNÍ

Diplomant bere na vědomí, že je povinen vypracovat diplomovou práci samostatně, bez cizí pomoci, s výjimkou poskytnutých konzultací.
Seznam použité literatury, jiných pramenů a jmen konzultantů je třeba uvést v diplomové práci.

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Datum převzetí zadání Podpis studenta

Acknowledgement / Declaration

I thank doc. Ing. Miroslav Bureš, Ph.D., the supervisor of the master's thesis, for his valuable advice and suggestions, which were very helpful. Furthermore, I thank my family for their support during my studies.

I declare, that I have done the assigned master's thesis alone led by supervisor. I used only the literature that is listed in this work. Furthermore, I declare that I have no objections to lending or making public of my diploma thesis or it's part with agreement of department.

In Prague 12.8.2021

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Abstrakt / Abstract

Diplomová práca sa zaoberá problematikou odstraňovania vesmírneho odpadu. Prácu možno rozdeliť do troch častí. Prvá časť je teoretická a popisuje problematiku odstraňovania odpadu, ako napr. generovanie odpadu, Kesslerov syndróm, prevencia vzniku odpadu, detegovanie a sledovanie odpadu. Ďalej sa prvá časť venuje popisu ADR metód a ich porovnaniu. Metódy sú rozdelené do štyroch skupín podľa ich spoločných vlastnosti: drag augmentation systems, contactless methods, tether-based methods a contact removal methods. Následne sú metódy porovnané na základe vhodnosti orbitu, nákladov, technologickej úrovni metód a ich výhod a nevýhod v rámci skupiny. Druhá časť pozostáva z popisu matematických a fyzikálnych vlastností, ktoré je potrebné brať do úvahy pri simuláciách. Pre simuláciu bola zvolená laserová metóda, ktorá je podrobnejšie popísaná. Zároveň sú popísané orbitálne elementy, orbitálna zmena a priblíženie dvoch satelitov. Posledná časť pozostáva z popisu navrhnutej aplikácie pre simuláciu vesmírneho odpadu. Jedná sa o 3D simuláciu pohybu objektov. Užívateľ v rámci aplikácie môže simulovať orbitálnu zmenu zakladajúcu sa na Hohmann transfer, priblíženie prenasledujúceho satelitu a cieľového objektu a deorbitáciu pomocou laserovej metódy. Aplikácia je naprogramovaná v Jave a využíva JavaFX framework pre 3D grafiku a jednotlivé desktopové prvky.

Kľúčové slová: vesmírny odpad, ADR metódy, laserová vesmírna metóda, JavaFX

The master's thesis deals with the issue of space debris disposal. The thesis can be divided into three parts. The first part is theoretical and describes the issue of debris disposal, such as debris generation, Kessler syndrome, debris mitigation, and how to detect and track debris. Furthermore, the first part deals with a description of ADR methods and their comparison. The methods are divided into four groups according to their common properties: drag augmentation systems, contactless methods, tether-based methods, and contact removal methods. The comparison of methods consists of comparing the suitability of orbit, cost, technological readiness level of methods, and their advantages and disadvantages within the group. The second part consists of a description of the mathematical and physical properties that need to be considered. The laser method was chosen for the simulation, which is described in more detail. At the same time, orbital elements, orbital change, and rendezvous are described. The last part consists of a description of the designed application for space debris simulation. The application is a 3D simulation of the movement of objects. Within the application, the user can simulate an orbital change based on Hohmann transfer, rendezvous of a chaser satellite and a target object, and simulate deorbitation using a laser method. The application is developed in Java and uses the JavaFX framework for 3D graphics and individual desktop elements.

Keywords: space debris, ADR methods, Laser space-based method, JavaFX

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Chapter 1
Introduction

1.1 Motivation

For more than a half century, we have been discovering the universe using artificial space satellites. Thanks to satellites, we are expanding our knowledge of the planets in our solar system and our galaxy. In addition to discovering and using satellites for scientific purposes, we also use satellites for common use. Thanks to the launch of more and more satellites into the Earth's orbit, which form a network or constellation, we can use services such as Internet access, finding our location (GPS), or make phone calls. More satellites help improve service availability.

Satellite lifespan is around fifteen years. After that time, it is necessary to replace the satellite with a newer satellite to maintain the service. This is not because the 15-year-old satellite will no longer be able to orbit the Earth or carry more fuel for more prolonged operations. The problem is that new technologies are coming to the market quickly, and a satellite orbiting the Earth for two or three decades would no longer be able to cope with the demands of the time.

Replacing an old satellite with a new one does not mean removing the old satellite from Earth's orbit and launching a new satellite. It is only launch of a new satellite, which is placed in a suitable orbit to not collide with each other and preserve its purpose. In this example, it is possible to see a problem that has not been taken into account from the beginning that a number of nonfunctional satellites arise in orbit over time.

During the operation of satellites, errors can occur, which can also cause the satellite to malfunction or become uncontrollable. This carries a high risk of collision with other satellites or small debris (e.g. created by collisions) and polluting the Earth's orbit. In order not to reach the point where the Earth's orbit will be so polluted that it will not be possible to launch a new satellite into space, it is necessary to prevent the creation of debris but also to remove existing debris.

1.2 The aim of the work

This work aims to describe the issue of space debris, why debris is generated, how we can mitigate its creation, and how we can detect and track it. Describe existing active debris removal methods for the removal of existing debris. Summarize their advantages and disadvantages and technical parameters. Furthermore, compare these methods from different points of view, such as Technology Readiness Level, price, and characteristics.

From the described methods, choose one method, which will be described in more depth. This means explaining the mathematical model of the physical quantity of the method. Describe and explain the parameters and visualization of orbits and satellite motion.

Create a desktop application for space debris simulation based on the selected ADR method. The application should consist of a 3D simulation of the movements of satellites or objects in Earth's orbit. Furthermore, it should allow the addition of a satellite or fragment to the Earth's orbit by adjusting the altitude, direction, or inclination. Implement a simulation of space debris removal with its visualization. Describe and explain the individual components of the application.

At the end of the work, describe how the application was tested.

Chapter 2
Background

Dozens of satellites are launched every year. They are employed in both the military and civilian sectors. Their tasks include providing communication, determining the device's location, accessing the Internet in places where it is still unavailable by another method, and allowing satellite telephony. Among other things, they help with weather forecasting. The continuous increase of satellites in space and not addressing the satellite's problem after completing the operation or after malfunction are causing **space debris**, **space junk**, or **orbital debris**.

2.1 A brief history of space debris

More than 63 years have passed since the first entry into space. Almost from the first mission, space junk began to be generated. The Cold War and the race to achieve control of space between the United States and the Soviet Union helped create debris. In the past, they have not addressed any measures against debris prevention. On the contrary, thanks to anti-satellite weapons, they contributed to the creation of more fragments. Below we can see important milestones in the history of space debris.

October 4, 1957 The first satellite to be launched into space. It is a Soviet Union artificial satellite called Sputnik 1. Sputnik 1 contained only one device, a transmitter that kept sending a beeping signal for less than a month. After three months, the satellite entered the atmosphere, where it burned up[1].

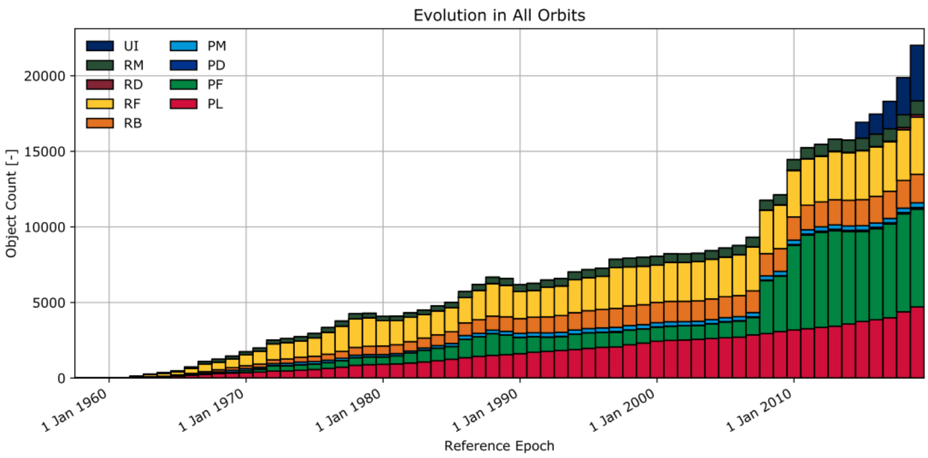
March 17, 1958 The fourth satellite to be launched into space is the American satellite Vanguard 1. To this day, it is still located there and is the oldest satellite and space debris. The previous three satellites did not move too far from the Earth's surface because they used smaller rockets to launch into space. Therefore, after a few months or years, they returned to the atmosphere. Vanguard 1 did not return to the atmosphere similar to the previous three satellites because the designers used a three-stage launch vehicle. It was a groundbreaking satellite because it contained solar cells that gained energy. These two factors caused Vanguard 1 managed to move further away from the Earth's surface. Scientists estimate that Vanguard 1 will be in space for 240 years[1].

1960s - 1990s Anti-satellite weapons (ASAT) caused a significant increase in the number of fragments. ASAT had produced 7 percent of the total number of space debris at the time[1].

1970s With the increasing number of objects in orbit, which the Cold War also helped, databases were created that kept information about currently occurring objects in space. The Space Object Catalog by the US Air Force was one of them. If a new satellite fragment

	was detected, then a new identifier was created and assigned to the fragmented satellite piece[2].
1979	NASA established the Orbital Debris program to solve the problems caused by space debris. They tried to mitigate the causes of the creation of the remains. Therefore, they eliminated residual propellants in the tanks to prevent explosion and the generation of new fragments[3].
1984 - 1990	Long Duration Exposure Facility (LDEF) mission. The LDEF gathered information on micrometeorites and very small space fragments. NASA tested the suitability of using different materials for missions[4].
1995	NASA was the first space agency to issue guidelines for reducing the generation of space debris[5].
January 11, 2007	The destruction of the Chinese weather satellite FengYun-1C created many fragments in near-Earth orbit. At the time, the number of space debris increased by 30 percent. 1967 pieces were newly catalogued[6].
February 10, 2009	The first accidental hypervelocity collision of two satellites. The first was the American communication satellite Iridium 33. The second was the Russian communication satellite Cosmos 2251. Iridium 33, unlike the Russian satellite, was still active. After the collision, 1366 pieces of debris were detected[7].

In the graph 2.1, we can see the evolution of space debris in all orbits since the launch of the first satellite. The number of payloads is constantly growing. Over the last two decades, there has been a significant increase in fragments, which was helped by the Chinese ASAT test and the accidental collision of the Iridium 33 and Cosmos 2251 satellites. We also see that the number of unidentified objects is increasing significantly.



(a) Evolution of number of objects.

Figure 2.1. Space objects evolution in all orbits[8]

There are several sources of debris[8]:

- **UI - Unidentified** - Objects to which we cannot be assigned when they were launched into space.
- **RB - Rocket Body** - refers to launch vehicles used to launch satellites into orbit.

- **RM - Rocket Mission Related Object** - objects associated with the launch of satellites into space. These parts were intentionally released. Examples are shrouds and engines.
- **RF - Rocket Fragmentation Debris** - are parts of the rocket that spontaneously released from the rocket, and we can identify them back to where the fragment came from.
- **RD - Rocket Debris** - similar to RF, but we cannot identify exactly where it comes from.
- **PL - Payload** - refers to objects that perform a specific task. Examples are active satellites or calibration objects
- **PM - Payload Mission Related Object** - are objects that were used in space missions but have been released, such as astronaut tools.
- **PF - Payload Fragmentation Debris** - similar to RF, however, this is the case of a satellite in which a part has been released or due to a collision with another object. In retrospect, we can identify when the release occurred.
- **PD - Payload Debris** - similar to PF, but we cannot retrospectively identify when the fragment was released.

In the picture 2.2, we can see the density of contamination Earth orbit, satellites, or space debris. Each orange dot represents one satellite, which can be either active or inactive. In the middle of the picture, we see the Earth covered with a dense network of objects. This network is located only a few hundred kilometers from the Earth's surface.

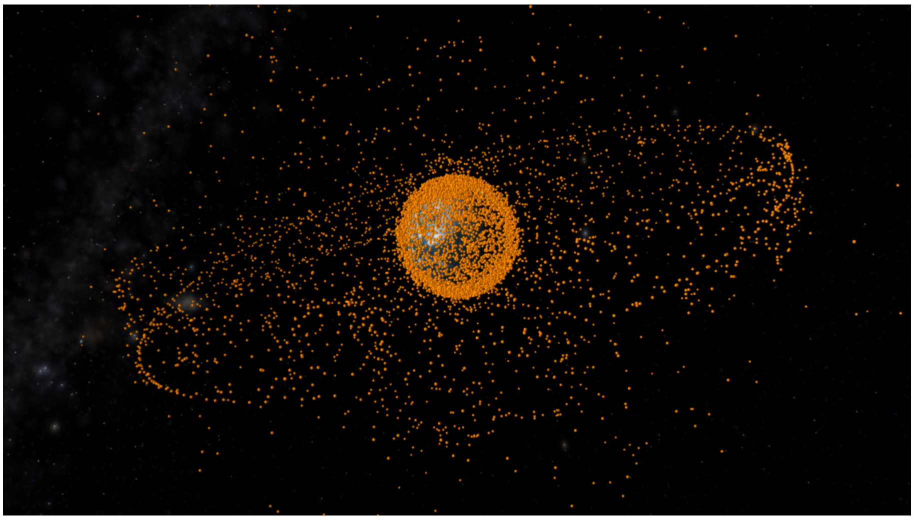


Figure 2.2. All active or inactive satellites currently in space[9]

2.2 Basic terms

Target satellite or object - is any artificial satellite or part of a satellite that is to be removed from space. It is space debris.
Chaser satellite - is a satellite that either tries to approach a target object or, in the case of a contact method, to capture or attach a device so that space debris can be removed.
Deorbiting or reentering - is a way to remove malfunctioning satellites or rockets from a nearby orbit. The main idea is to use the atmosphere, when the object that enters

the atmosphere burns in it.
Reorbiting - is a change in the altitude of a satellite.

2.3 Space debris generation

We know from the introduction that space debris is any inoperative device or part of a device that has completed its mission and remained in orbit. Exists several reasons why a device becomes debris. The first is that a satellite without a built-in mechanism for returning to the atmosphere cannot return after its mission. Nevertheless, gravity acts on the satellite, pulling it back to Earth. Therefore, the satellites must have a correctly set velocity that allows them to resist gravity's effects. As a result, satellites can be in space for many years but not forever. The time of orbit occupation depends on the distance from the Earth's surface. The satellites with an altitude of up to 2,000 km pass through a thin layer of atmosphere that creates drag. After some time, they need to use engines to maintain their trajectory. Satellites with an altitude of more than 2,000 km can be affected by natural satellites, planets, the sun, or other space bodies. Hence, they also use rocket engines to maintain their orbit. The second reason is failure. It can be mechanical or software. In this case, devices with a built-in satellite return mechanism do not have to go back. They travel through space. The last reason is a fragment. Collisions of two objects, testing of anti-satellite weapons, satellites' explosion, or fragments themselves are sources of other fragments formation. A collision can be accidental or controlled. In both cases, there is an enormous increase in small pieces in space. Testing of anti-satellite weapons has been typical in the past, especially during the Cold War. The explosion of rockets near the satellite or the rocket's direct hit to the satellite also creates much debris. When a malfunction occurs, the residual fuel in the fuel tanks can cause an explosion, which will cause many pieces to disperse into the surroundings. The fragment itself may cause additional fragments due to collisions between the fragment and the satellite.

2.4 Types of orbits

When removing space debris, we also consider the type of orbit. The Earth's orbit is divided into:

- Low Earth Orbit (LEO)
  - Polar orbit
  - Sun-synchronous orbit (SSO)
- Medium Earth Orbit (MEO)
- Geostationary transfer orbit (GTO)
- Geostationary orbit (GEO)
- Graveyard orbit

These types of orbits are sometimes called **spatial distribution**. Satellites launched into space are usually placed in LEO. LEO is located at an altitude of 160 km to 2,000 km. LEO is a commonly used orbit because satellites do not always orbit the equator, as in GEO. It is possible to tilt the plane. Launching a satellite closer to the Earth's surface is more accessible than farther from the Earth. The objects are protected from cosmic rays owing to electromagnetic radiation, which still acts in LEO.The International Space Station (ISS) is located in LEO. Its velocity is 7.8 km/s. Similarly, other objects in LEO have approximately the same velocity. This means that one circle around the Earth takes about 90 minutes. It is a drawback for single telecommunication satellites because they circle around the Earth very fast, making communication difficult. Therefore, multiple satellites are launched into orbit with different positions. They create a `net` or constellation.

Both polar orbit and SSO belong to LEO. The difference between the polar orbit and other types of orbits(such as SSO or equatorial orbit) in the LEO is that the satellites pass through the south and north poles and do not move around the equator in the polar orbit. SSO is a type of polar orbit. Its peculiarity is that it passes through any point of the Earth at the same time. This brings advantages in the weather forecast or capturing the same place at the same period[10].

MEO is located between LEO and GEO (from 2 000 km to 35 786 km). MEO is mainly used for navigation satellites such as Glonass or Galileo. One circle can take 2 to 24 hours.

GEO is located at an altitude of 35 786 km. One circle around the Earth takes the same amount of time as one rotation of the Earth. It is 23 hours, 56 minutes, and 4 seconds. The speed of satellites in GEO is about 3 km/s. GEO is used for satellites that need to copy the same position relative to the Earth. Examples are telecommunications satellites. Unlike LEO, it is enough to place only a few satellites to achieve complete Earth coverage[10].

GTO is a special orbit used to transfer satellites into the correct orbit. This is used, for example, when launching a satellite to GEO. The launch vehicle is enough to bring the satellite to the perigee. The **perigee** is a point in an orbit that is closer to Earth. The satellite moves in GTO until it reaches the apogee. The **apogee** is the exact opposite of the perigee. When the apogee is reached, the satellite will use engines to reach its orbit.

The Graveyard Orbit is located 300 km behind the GEO. It is used as a warehouse for satellites that have completed their missions. If the mission is coming to an end, then the satellite is sent to the graveyard orbit. This prevents the generation of space debris, collisions, or fragments in the GEO.

In the picture 2.3, we can see the above-mentioned orbits. As we will see in chapter ??, there are many methods for the removal of space debris. Most of these methods depend on the type of orbit to be cleaned.

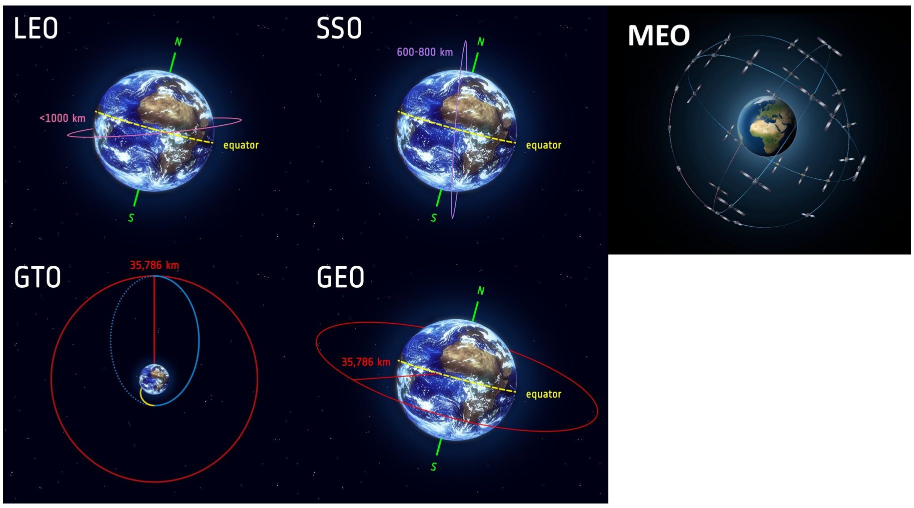


Figure 2.3. Types of orbits[10]

The Earth's orbit is much more dividing, such as the Inclined Geosynchronous Orbit (IGO), Extended Geostationary Orbit (EGO), Navigation Satellites Orbit (NSO), GEO-superGEO Crossing Orbits (GHO), Lagrange points, etc. A more detailed dividing of orbits is not necessary for this thesis. Due to space debris and its disposal, we are interested in LEO and GEO. Table 2.1 shows the pollution density of satellites in three typical orbits. As we can see, the MEO (in consideration of the orbit's size) is not very contaminated with debris. Conversely, LEO gets close to the limit that can trigger a cascading effect. The data in the table comes from a database containing current and past objects that have been launched into Earth orbit[11].

Type of orbit	Payload	Rocket body	Space debris
LEO	5596	938	11 023
MEO	579	933	2111
GEO	880	115	26

Table 2.1. Pollution density in individual orbits

2.5 Kessler syndrome

Donald Kessler was the first to point to a phenomenon with increasing debris in the space named after him. It is called **Kessler Syndrome** or **Kessler effect**. Over time, satellites, rockets, fragments, and debris have accumulated in space. This phenomenon may result in the impossibility of sending new satellites into orbit, e.g., LEO. Kessler's syndrome says that if the amount of debris reaches a certain limit, a cascading effect is triggered. This means that any collision of debris will trigger or result in further collisions, and the likelihood of collisions will continue to increase. The result is many small fragments that can damage active satellites and prevent the launch of new satellites[12]. In the table 2.2, we can see how many space objects are located in the space. Objects are divided into three categories by their size.

Size of the object	Number of objects
$\leq$ 1 cm	128 million
1 cm - 10 cm	900 000
$\geq$ 10 cm	34 000

Table 2.2. Number of objects currently in the space by their size[13]

2.6 Space Debris Mitigation

An important factor in removing retired satellites, rockets, or fragments is to prevent their generation. Therefore, the individual space agencies have come up with rules to be taken into account when planning, designing, and operating the satellite. The rules introduced by NASA are[14]:

- **„Limit debris released during normal operations"** - means to prevent or minimize payload/rocket mission related objects.
- **„Minimize the potential for break-ups during operational phases"** - avoid failures that could lead to satellite breakdown. If failure cannot be prevented, then minimize the impact on satellite decay.

- **„Limit the probability of accidental collision in orbit“** - prevention of large debris generation and malfunction of active satellites. Use avoidance maneuvers or postpone the start of the launch to avoid collisions.
- **„Avoid intentional destruction and other harmful activities“** - prevent ASAT tests or intentional destruction of spacecraft. If destruction is required, such destruction must take place at low altitudes so that the debris does not remain in orbit for long.
- **„Minimize potential for post-mission break-ups resulting from stored energy“** - Residual fuel in the tanks and charged batteries pose a high risk of damage to the inactive satellites. The solution is to passivate satellites.
- **„Limit the long-term presence of spacecraft and launch vehicle orbital stages in LEO region after the end of their mission“** - satellites or rockets that have completed their mission should not remain in orbit for long. Therefore, they should be deorbited. Deorbitation should not create a risk for people on earth. If it is not possible to perform deorbitation, they should be destroyed in space but without generating additional debris.
- **„Limit the long-term interference of spacecraft and launch vehicle orbital stages with GEO region after the end of their mission“** - In the case of GEO, the procedure is the opposite of the previous point. Satellites that have completed their mission should not remain in the GEO but should move into a graveyard orbit so as not to jeopardize operational satellites.

2.7 Debris Measurements and Tracking

One of the essential aspects of space debris is finding old satellites, rockets, or fragments and tracking them. Tracking is very important to avoid accidental collisions and to monitor the entry into the atmosphere of large satellites or parts of them. As we already know from the introduction of this chapter, accidental collisions can create enormous fragments and damage active satellites. With vast and heavy objects, there is a risk that they may not burn in the atmosphere and could fall into the inhabited part of the Earth.

We may want to measure several parameters such as size, composition, mass, or spatial distribution of space debris. There are several ways to track and measure object parameters, and these are[4]:

- Optical measurement
  - passive
  - active
- Radar
- In-Situ

An optical telescope helps us track objects 10 cm and more in size. Optical tracking is mainly used in GEO, where it is not possible to measure small parts of debris using radars[15]. Optical measurement can be either passive or active. Both require good weather conditions. Passive optical measurement is based on the collection of sunlight reflected from the surface of the object. In order to perform a passive measurement, several conditions must be met. In addition to good weather conditions, the observed object must be illuminated by the sun and be observed at night. From the observation, we can determine the rotational behavior and characterize the object and its surface. On the contrary to passive, active optical systems can also be used for measurements during the day. Active optical systems use a laser that emits photons that are then reflected from an object. The principle is similar to passive measurement, but in this case, the photons are emitted by a laser device[4]. The disadvantage of passive measurement is that it cannot determine the distance of the object from the observer but allows for the detection of new objects. The disadvantage of active measurement is that it cannot detect new objects but can determine the distance. Therefore, by combining both approaches, we can detect new objects and determine their altitude[16].

Ground-based radars are employed to measure objects in LEO. These radars enable the detection of objects approximately from 5 mm to 30 cm. They are not dependent on weather conditions, and the measurement can be performed all day[4]. The principle of radar measurement is based on echo. The radar sends a signal to the target object, which is then reflected from the object. Then the signal returns to the source. Using radar, we can measure the Doppler frequency and velocity of the target object. This is possible in the case of continuous signals. In the case of pulse signals, we can determine the distance between the object and the radar. Thanks to the obtained distance, it is possible to determine the position[17].

There is space-based in-situ measurement and tracking in addition to ground-based methods such as radars and optical telescopes. In-situ is suitable for detecting small particles with a size of 1 mm or less. The measurement can be performed in both polluted types of LEO and GEO orbits[18].

In Figure 2.4 we can see a graph that compares the above-mentioned methods in terms of size and altitude. The x-axis describes the particle size they can detect. The y-axis represents the altitude to which each method can measure and track. Space Surveillance Network is a combination of radars and optical telescopes. Haystack and Goldstone radars are the primary source for object monitoring at NASA. Using the Hubble Space Telescope (HST) and Space Shuttle (STS), the information helps model the orbit pollutation[19].

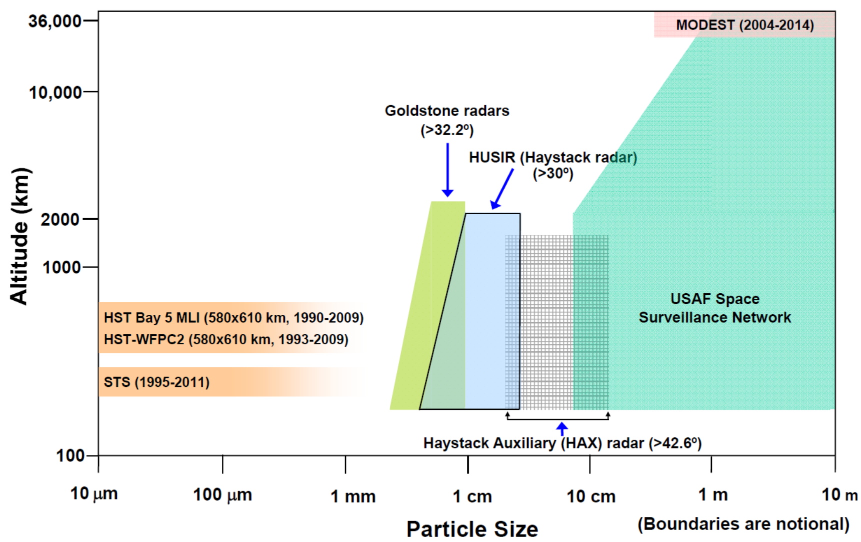


Figure 2.4. Debris Measurements[19]

2.8 Reorbitation and Deorbitation phases

We already know that deorbiting is one way we can deal with existing debris. This method of removal is used in LEO. The advantage of deorbitation is the destruction of an object in the atmosphere. In order for an object to burn, several phases must go through. As we will see in the third chapter, not every ADR (Active Debris Removal) method requires all phases. It depends on the method of disposal. In GEO, the altitude of the target object increases, which moves into the graveyard orbit. This method is called reorbitation.

Reorbitation and Deorbitation phases are:

- Launch - consists of launching a satellite into space and adjusting its velocity.
- Rendezvous - This is a maneuver when the chaser satellite approaches the target object. The chaser satellite is guided into the same orbit and adjusts the velocity relative to the target object.
- Capturing - After a close rendezvous of the chaser satellite and the target object, the capture phase follows. This is the penultimate phase in removing the remains. In this step, the object is captured. We distinguish between two types of capturing methods: contactless and contact.
- Removal - The last phase of reorbitation or deorbitation. At this phase, the altitude changes. In the case of deorbitation, the altitude decreases. Conversely, during reorbitation, the altitude increases. As we will see in Chapter 3, there are two ways of disposing of space debris: contact and contactless removal methods.

Chapter 3
Active Debris Removal Methods

The chapter consists of a guidepost and a description of ADR methods. Each method describes a principle of debris removal and a table with the advantages and disadvantages of the method. In addition, most methods include a table describing the technical parameters. The described technical parameters are indicative as the parameters depend on the properties of the target object.

- **Drag augmentation system (DAS)** - a group of methods where the area to mass ratio is increased, resulting in higher atmospheric drag. (3.1)
  - **Foam method** - uses foam, which is ejected by a chaser satellite. Subsequently, it covers the surface of the target object, which increases the area to mass ratio. (3.1.1)
  - **Inflated method** - increases the area to mass ratio using an inflated envelope. The method can serve to prevent the generation of debris but also dispose of it. (3.1.2)
  - **Fiber-based method** - uses fiber for deorbiting debris, which is extruded by a heat source. (3.1.3)
  - **Solar sail method** - a simple and lightweight way to remove debris with a large solar sail, which can control the velocity. (3.1.4)
- **Contactless removal methods** - a group of methods where is no direct contact between the chaser satellite and the target object. (3.2)
  - **Artificial atmosphere influence** - a pulse of atmospheric gases is fired in the path of the orbit of space debris. (3.2.1)
  - **Laser-based method** - uses a pulse laser beam that shoots onto the target object. As a result, the target object slows down and reduces altitude to the atmosphere, where burns up. We have two systems: a ground-based system and a space-based system. (3.2.2)
  - **Ion beam shepherd-based method (IBS)** - ejects a high-velocity and highly collimated neutralized ion beam into the space debris. IBS can be used in both LEO and GEO orbit types. (3.2.3)
  - **Electrostatic tractor** - a pulling method that emits an electron beam at a target object. Therefore, the target object is charged negatively, and the chaser satellite is positively charged. This will cause the objects to attract each other. (3.2.4)
- **Tether-based method** - a group of methods that uses tether for removal. (3.3)
  - **Electro-dynamic tether** - a method that uses Lorentz force to remove space debris. Requires a chaser satellite with a robotic arm or a harpoon. (3.3.1)
  - **Net** - a capture method that shoots a network at a target object. The network is connected to the chaser satellite via a tether. (3.3.2)
  - **Harpoon** - a capture method based on shooting a harpoon that is captured on a target object. Subsequently, reorbitation takes place using a tether placed on a harpoon. (3.3.3)

- **Contact removal methods** - direct contact between the chaser satellite and the target satellite. (??)
  - **Slingshot method** - a method that uses the power of throwing space debris to the Earth to move to the other debris. (3.4.1)
  - **Adhesive method** - consists of motherships and boys. The boy, which is equipped with a propulsion system, is attached to the target object using an adhesive compound. (3.4.2)
  - **Tentacles** - a stiff composite method that uses tentacles to capture the target object. (3.4.3)
  - **Single robotic arm** - a method of capture using a robotic arm that is attached to a chaser satellite. The chaser uses this arm to capture the target object. (3.4.4)
  - **Multiple robotic arms** - a method of capture using multiple robotic arms that allow better stabilization. (3.4.5)

3.1 Drag augmentation system

The main idea of the drag augmentation system (DAS) is to increase the area to mass ratio of the space debris. As a result, the atmospheric drag will increase, and the debris will decrease its altitude until it reenters the atmosphere, where it will burn. Such a solution has advantages of being independent of the size of the debris, not requiring docking, and the fact that the chaser satellite does not have to control the re-entry into the atmosphere. The DAS group includes the Foam, Inflated, Fiber-based, and Solar sail methods[20].

3.1.1 Foam method

The foam method requires a chaser satellite that is launched into the Earth's orbit. Subsequently, it is necessary to perform a close rendezvous when the chaser satellite approaches the target object. The chaser satellite has installed an ejection mechanism for ejecting foam onto the target object. As we can see in Figure 3.1, it is a mechanism consisting of containers for foam (Component A, Component B) and gas and a robotic arm with a controllable valve and a nozzle. The foam is covering the target in the shape of a sphere. The almost spheric shape does not require any special position control. The foam must also be stiff enough to withstand the impact of other space debris. The idea is the same as described above for DAS, to increase the area to mass ratio of debris to increase atmospheric drag, resulting in a deceleration of the object and a gradual reduction in altitude. There is no direct contact between the chaser satellite and the target object during capture. The foam structure is stiffer than tether, net-based, or sail structures described below in this chapter[21]. Drawbacks include complex foam formation, incomplete attachment, or incomplete vacuum expansion[22].

Pros	**Cons**
no position control	complex foam formation
no complex docking	incomplete vacuum expansion
no control required for reentry	specific storage requirements

Table 3.1. Pros and Cons of the foam method

Technical parameter	Value
dry mass	4 600 kg
power	12 kW
base components	gas foam tank foam ejection mechanism robotic arm sensors solar array propellant, battery

Table 3.2. Technical specification of the foam method[23]

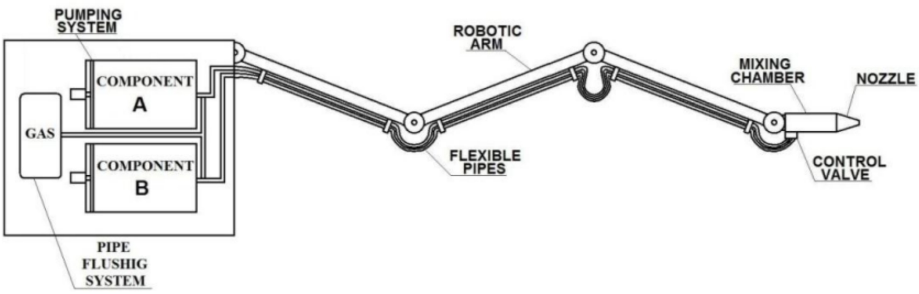


Figure 3.1. Example of the foam method[23]

3.1.2 Inflated method

The inflated method is similar to the previous foam method but instead of foam uses an inflated ball. The ball is a large and lightweight envelope. The inflated envelope has the task of reducing the ballistic coefficient by up to two orders of magnitude. Gossamer Orbit Lowering Device(GOLD) uses this approach. The stored GOLD system could be built directly into the newly launched satellites. The envelope would inflate at the end of the operating time, and the deorbitation process would start. Another option is to attach the envelope to the existing debris. This option requires a close rendezvous with a docking mechanism. The envelope is suitable for use at an altitude of 750 to 900 km. The enormous advantage is that it does not contribute to the generation of additional space debris in case of failure. The disadvantage of the method is that small fragments can damage the envelope[24].

Pros	Cons
no additional debris prevents the generation of debris	damage by small debris

Table 3.3. Pros and Cons of the inflated method

In Table ??, the target object located in the LEO with a mass of approximately 1200 kg is considered.

Technical parameter	Value
dry mass	500 kg
power	12 kW
gas	135 g
base components	gas tank stowed GOLD system sensors solar array propellant, battery

Table 3.4. Technical specification of the inflated method[24]

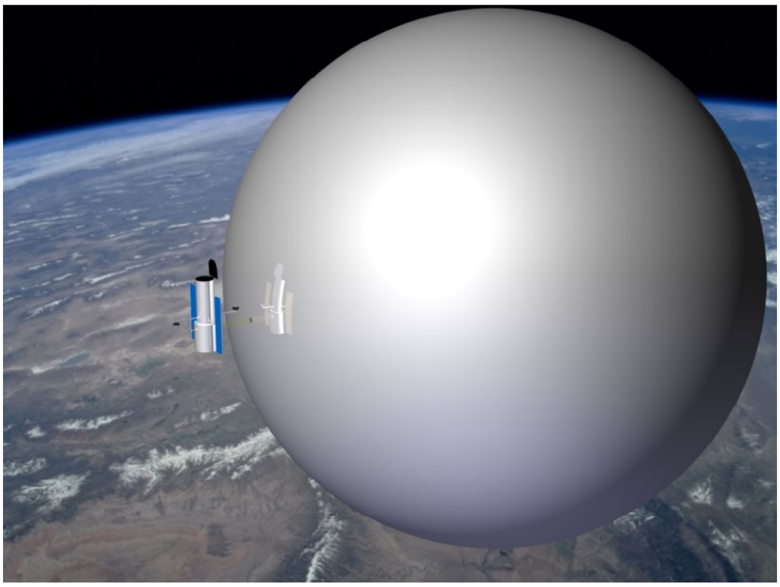


Figure 3.2. Example of the inflated method[24]

3.1.3 Fiber-based method

The fiber-based method, like the foam method, needs a chaser satellite with an ejection mechanism. The difference is that it will not eject a foam but fibers. The fiber is extruded by a heat source and wound around a target to intercept and expand it, thus increasing the area to mass ratio. The method does not need a docking mechanism because there is no direct contact between the chaser and the target object. The disadvantage of the method is the low Technology readiness level (TRL)[20].

Pros	Cons
no additional debris no docking mechanism	low TRL

Table 3.5. Pros and Cons of the fiber-based method

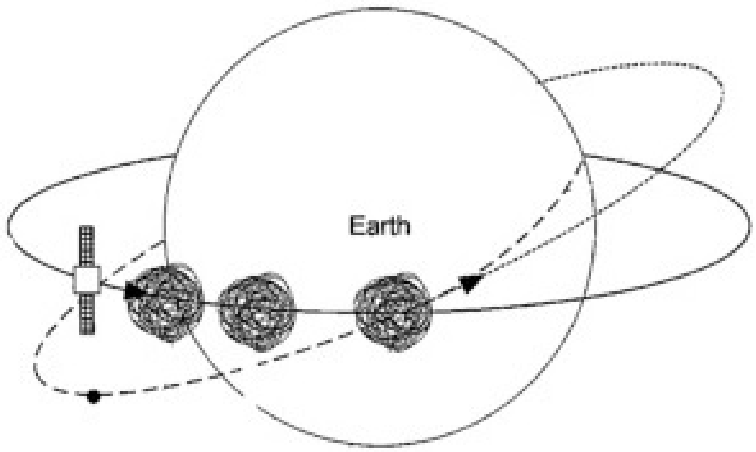


Figure 3.3. Example of the fiber-based method[20]

3.1.4 Solar sail method

The method consists of an aluminium-polyamide membrane attached to the satellite. This membrane is in the form of a sail whose size is much larger than the rest of the satellite. Thanks to the tilt of the sail, it is possible to control the velocity of movement and change the satellite's trajectory. This controlling motion by photons incident and reflected off a membrane generating solar pressure can replace today's fuel-dependent drive. A solar sail is a simple lightweight device that can help reduce further debris formation. The method is suitable for use in LEO but also in GEO. The advantage is the possibility of extending the operating time of the satellite. The drawback is the possible damage by small fragments[25].

Pros	Cons
no additional propellant	damage by small debris
lightweight	potential risk of collision
high TRL	
multipurpose method	

Table 3.6. Pros and Cons of the Solar sail method

Technical parameter	Value
chaser dry mass	300 kg
generate power	500 W
solar sail size	20 x 20 m
base components	booms membrane hub sensors bus and solar panel

Table 3.7. Technical specification of the solar sail method

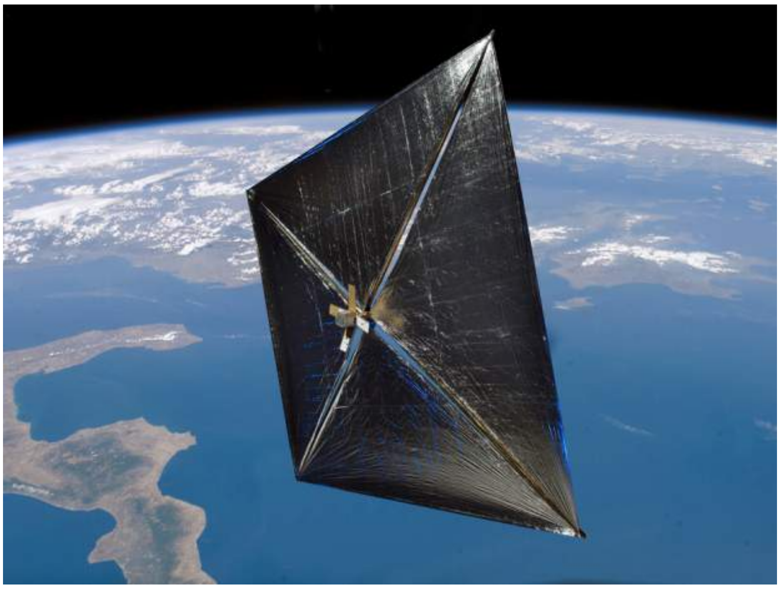


Figure 3.4. Example of the solar sail method[26]

3.2 Contactless removal methods

The princple of contactless methods lays in no direct contact between the chaser satellite and the target object during the removal of space debris, specifically in the capturing and removal phase. On the one hand, these methods have the advantages that it is not necessary to address the stability of the connection, it is a low risk of collisions, and almost no additional debris. On the other hand, methods require more time to remove space debris. Contactless removal methods include artificial atmospheric influence, laser-based method, ion beam shepherd method, and electrostatic tractor. All four methods try to slow down space debris and thus reduce the altitude of debris[20].

3.2.1 Artificial atmosphere influence

The artificial atmosphere influence consists of pulses of atmospheric gases fired into the orbit of space debris. The space debris that passes through this air pulse is slowed down, which results in it decreasing its altitude. An air pulse can affect multiple target objects if they are close to each other. This method generates no additional space debris because the gases are returned to the atmosphere and do not contain any solids. Thus, it is a safe green removal method for the mitigation of space debris. The drawback of this technique is that it needs a large amount of gasoline, about 500 galleons, to generate a single pulse, but scientists are currently trying to reduce gas volume[27].

Pros	Cons
green removal method	large amount of gasoline
no additional debris	less efficient
no docking	affecting multiple objects
no risk of collision	

Table 3.8. Pros and Cons of the artificial atmosphere influence

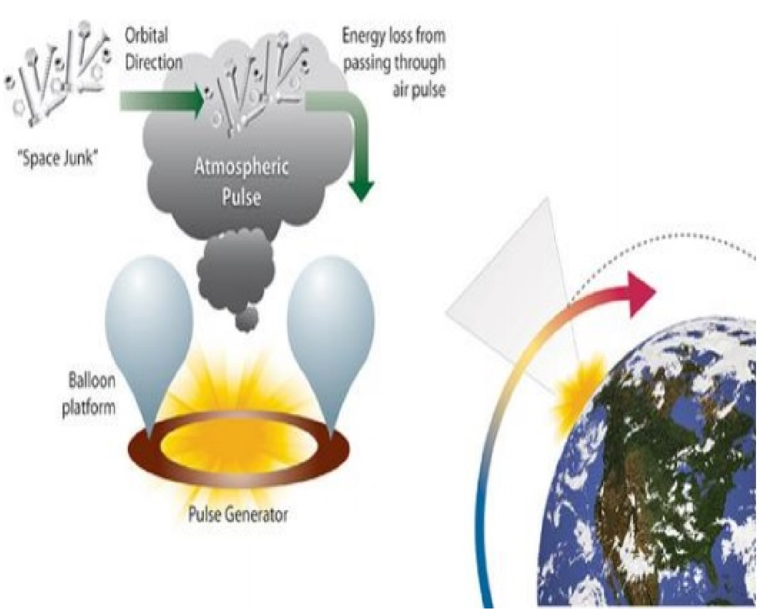


Figure 3.5. Example of the artificial atmosphere influence[20]

3.2.2 Laser-based method

The laser system consists of a pulse laser beam. The beam is shooting into space debris and slows down its velocity, and lowers its altitude. Like other methods, the laser system reduces an altitude in order to remove space debris, and after some time the space debris burns up in the atmosphere. The laser-based method is suitable for removing both large and small debris, but the main problem is objects with a size between 1 - 10 cm. On the one hand, the spacecraft can be shielded from objects of smaller sizes than 1 cm. On the other hand, the spacecraft can perform a maneuver and avoid objects larger than 10 cm. Orion is the first design of the laser-based system, which can remove space debris with a diameter of about 1 cm in a distance below 500 km. The next design could remove debris up to 1000 km in altitude by four years. Debris must be larger than 1 cm and mass less than 500 kg. The new laser-based system called Laser Orbital Debris Removal (LODR) can cause deceleration debris about 40 km every eight weeks[28].

The drawback of the laser method is that it can cause the generation of new space debris. Moreover, the readiness of this method is not too high. In spite of the drawbacks, it is a cost-effective method. The laser beam mechanism can be installed either on the ground (called a ground-based system) or on a satellite (called a space-based system). Both of these systems have advantages and disadvantages. A ground-based system can be easily constructed on the ground. In case of failure, it does not generate additional space debris. The problem with a ground-based system is that it has to cover long distances of 350-1000 km. Therefore, the device must be extremely accurate. A large laser beam director mirror is also required. Space-based systems try to eliminate these problems. This system can even be installed into large objects to avoid colliding satellites. An example is the ISS. In the picture below, we can see an example of a space-based system, where a chaser satellite chases a target object and shoots a pulse laser beam to slow down its velocity. As a result, the target satellite decelerates and reduces altitude down to the atmosphere where burns up[22].

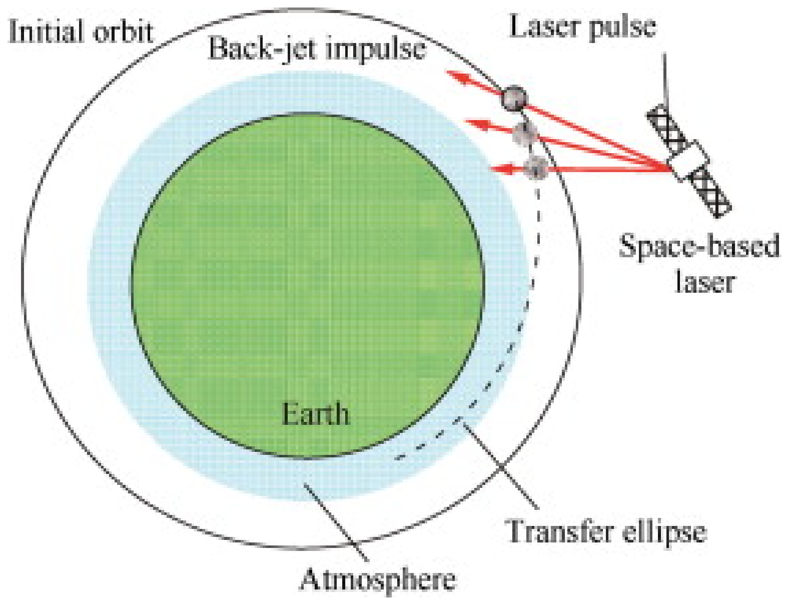


Figure 3.6. Example of the laser-based method[20]

Pros	Cons
large and small space debris cost-effective	additional space debris long distance (ground-based system) tracking of debris (space-based system)

Table 3.9. Pros and Cons of the laser-based method

Technical parameter	Value
chaser dry mass	2 300 kg
system power	64 kW
base components	fiber laser, heat sink telescope, solar array propellant, battery

Table 3.10. Technical specification of the laser-based method

3.2.3 Ion beam shepherd-based method

The Ion Beam Shepherd (IBS) method consists of a chaser satellite equipped with two propulsion systems (primary and secondary). Using the primary propulsion system, the chaser satellite ejects a high-velocity and highly collimated neutralized ion beam into the space debris. The secondary propulsion system is installed on the opposite side of the satellite body. The task of the secondary thruster is to eject the ion beam in the opposite direction to the ejected beam from the primary thruster. The result is a better stabilization of the chaser satellite. There is no plasma contamination as it falls into the atmosphere. The distance between the chaser satellite and the target object is from 10 to 20 meters. IBS is suitable for LEO but also for GEO. The disadvantage is that the shepherd satellite can be damaged by ions reflected from the target object, thus some parts can be contaminated such as solar panels. Another disadvantage is that the target object may spin if the ion beam deviates from the target's centre of gravity[22].

Pros	Cons
using in LEO and GEO no complex docking high de-orbiting efficiency due to use of electric propulsion	requires more power no controlled re-entry

Table 3.11. Pros and Cons of the ion beam shepherd

Technical parameter	Value
chaser dry mass	900 kg
system power	12 kW
base components	ion thrusters sensors, solar array propellant, battery

Table 3.12. Technical specification of the ion beam shepherd

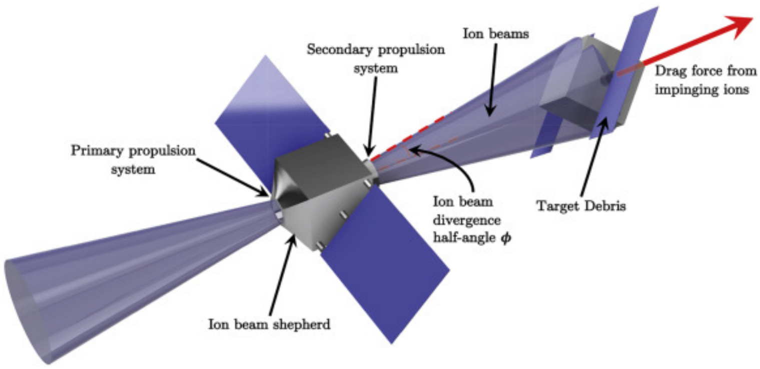


Figure 3.7. Ion beam shepherd[28]

3.2.4 Electrostatic tractor

An electrostatic tractor (ET) is a way to reorbit a retired satellite or rocket. This method is suitable for use in GEO. The technique uses Coulomb's law between the tug and the target object. In other words, an electrostatic force is used between the objects. The tug sends an electron beam to the target object, on which excess electrons accumulate. Therefore, the target object is charged negatively. In contrast, the tug is charged positively. We know that the positive and negative poles are attracted, which allows debris to be brought into the graveyard orbit. The tug uses inertial thrusters to move. The advantage is that the propellant is mainly needed only for propulsion. One removal of a retired satellite could take 2 to 4 months. Figure 3.8 shows the principle of operation of an electrostatic tractor[29].

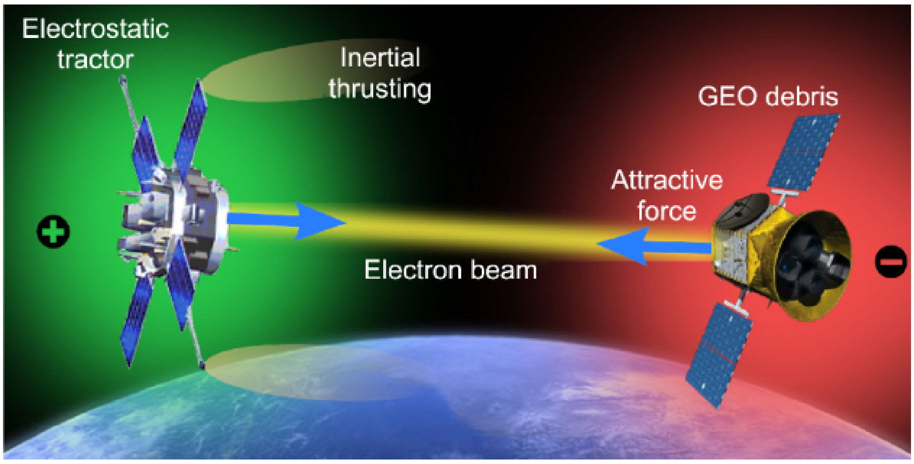


Figure 3.8. Electrostatic tractor[29]

ET is not a universal method in consideration of the type of the orbit. Its use in LEO is not possible due to the environment. The reason is the plasma produced by the Sun. In GEO, the plasma is relatively hotter and sparser. On the contrary, in LEO it is colder and denser. If we used an electrostatic tractor and started charging debris with an electron beam after a short time, debris would attract charged plasma particles. As a result, the target debris would become neutral[29].

Pros	Cons
no complex docking agnostic to the target which is steady or tumbling lower risk of collision no additional debris	can only be used in GEO

Table 3.13. Pros and Cons of the electrostatic tractor

3.3 Tether-based methods

The group of tether-based methods consists in the use of various pulling technologies to tow the target object. The tether's basis forms the link between the chaser satellite or kit and the target object. This group includes Electrodynamic tether, net, and harpoon methods.

3.3.1 Electro-dynamic tether

Electrodynamic tethers(EDT) use the Lorentz force to remove space debris. EDT needs a chaser satellite to place the EDT package on the debris. This can be done using a robotic arm or a harpoon that captures the target object, and then the chaser satellite can attach the EDT package3.9. The package contains one emitter that is attached to the debris using a conductive tether and collector. The emitter emits electrons at one end, and the collector collects electrons at the other end. As a result, a current in a closed circuit flows along the conductive tether. The current passes through the ambient plasma, which is located there. Subsequently, a Lorentz force is generated between the flowing current and the Earth's geomagnetic field. The Lorentz force acts against the movement of the target object. EDT can only be used in LEO due to a sufficiently strong geomagnetic field of the Earth 3.10.

The disadvantage of EDT is that a docking mechanism is needed to capture the target object and attach the EDT package[30].

Pros	Cons
no propulsion system	capture needed docking mechanism only in LEO

Table 3.14. Pros and Cons of the EDT

Technical parameter	Value
chaser dry mass	1 000 kg
system power	500 W
tether length	2.5 km
base components	gps receiver, star tracker stereo vision sensor robot arm solar array propellant, battery

Table 3.15. Technical specification of the EDT

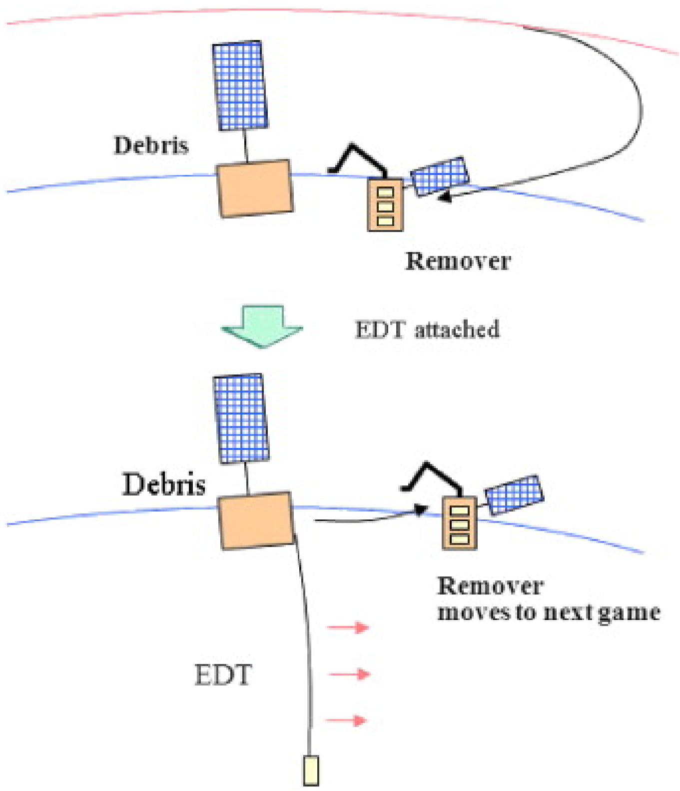


Figure 3.9. Attaching the EDT package[31]

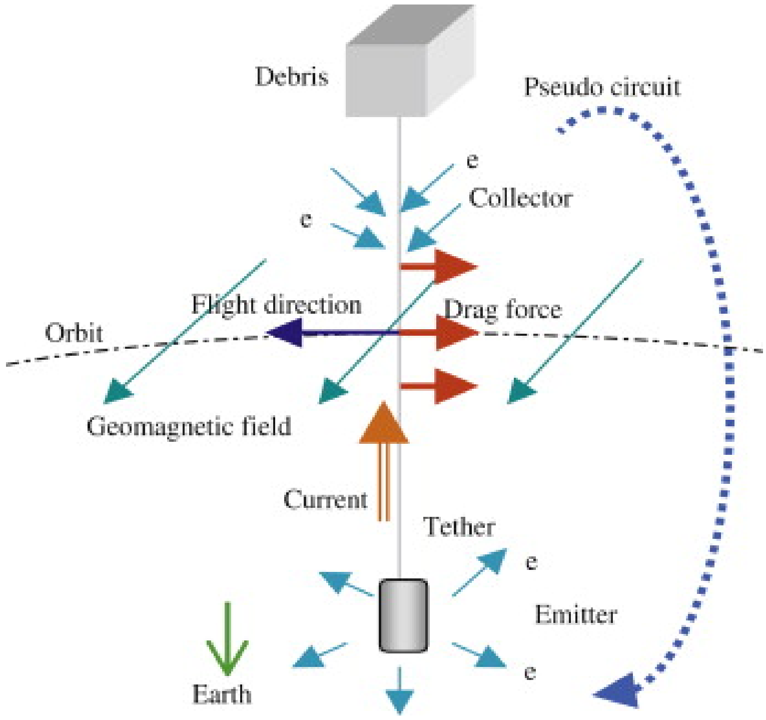


Figure 3.10. Principle of the EDT[30]

3.3.2 Net capturing

Net capturing is a flexible, simple, and cost-effective method for use in both LEO and GEO. The principle consists of an ejection net that is located in the chaser satellite. The net has bullets placed in the corners, which are used to expand the network and wrap the target object. The advantage of this mechanism is that it is not necessary to consider mass, inertia, and other physical parameters. Another advantage is that there is no direct contact between the chaser satellite and the target object but between the net and the target object. Thus, there is no need for a docking mechanism. A rope attached to the net and the satellite gives the distance between the chaser and the target. After capturing the debris, it is towed either to the atmosphere or into the graveyard orbit. The disadvantage of capture is its efficiency[20].

Pros	Cons
no complex docking	heat resistant materials required as the thruster plume is directed towards the net
can handle any shape, attitude or spin rate	potential fragmentation
allows a large capturing distance	hard to control
reduced requirements on precision	risk of critical oscillations
compatible for different size of debris	hard to test on ground

Table 3.16. Pros and Cons of the net capturing

Technical parameter	Value
chaser dry mass	1 300 kg
power	12 kW
net size	1 x 1 m - 30 x 30 m
bullet mass	$\leq$ 4 kg
wire dimension	1 mm
mesh length	1 x 1 m
tether length	100 m
base components	net ejection mechanism (gas generator, pressure sensor, gas tank) sensors, solar array propellant, battery

Table 3.17. Technical specification of the net

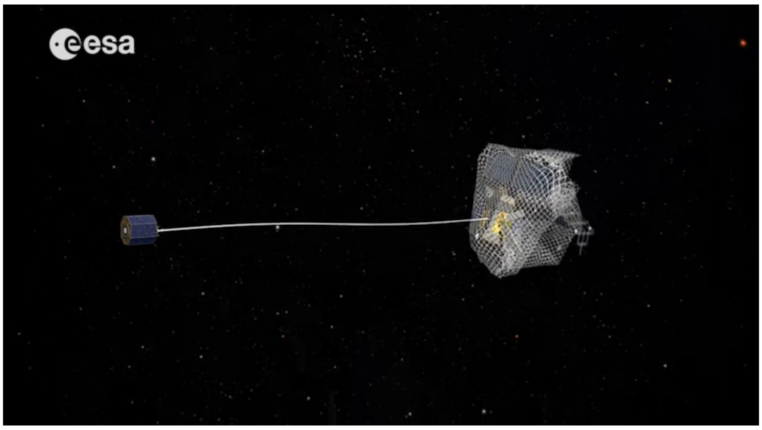


Figure 3.11. Example of net capturing

3.3.3 Harpoon capturing

The principle of the method is based on a harpoon, which is placed on a chaser satellite. The harpoon consists of a set of barbs for attaching. This attachment must be strong and stable enough to prevent unintentional release during gripping. Another part is the crushable section to absorb the energy that is generated during the crash. The last important part is the tether for the chaser and target connection[32]. When the chaser approaches the target object to a suitable distance, the chaser fires the harpoon that hits the target and grabs it. Then the last phase can occur, either deorbitation or reorbitation depending on the type of orbit[20].

Pros	Cons
cost efficiency	high risk of creating additional debris
higher TRL	problem with high tumbling rate
easily testable on the ground	penetration into the target object
various target object shapes	keeping the harpoon after penetration
no complex docking	

Table 3.18. Pros and Cons of the harpoon capturing

Technical parameter	Value
chaser dry mass	150 kg
system power	20 W
harpoon size	585 x 40 mm
harpoon system mass	6.7 kg
tether length	1 km
base components	harpoon firing system tether sensors, solar array propellant, battery

Table 3.19. Technical specification of the harpoon

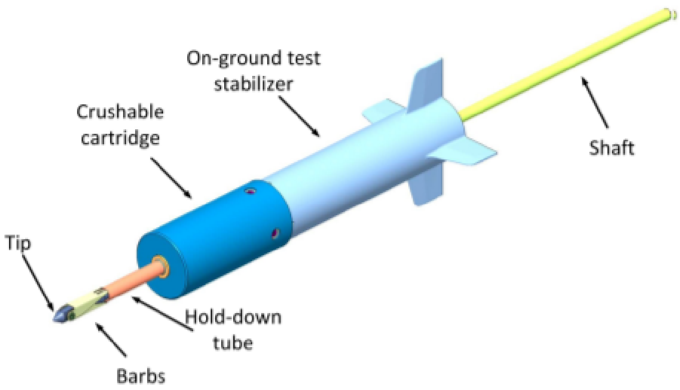


Figure 3.12. Example of the harpoon capturing[33]

3.4 Contact removal methods

Contact removal methods use direct contact between the chaser satellite and the space debris. Representatives of this method are slingshot, adhesive, tentacle, single robotic arm and multiple robotic arms methods.

3.4.1 Slingshot method

The Slingshot method consists of a satellite that can remove multiple space debris in a single launch. The advantage is that it saves energy. The University of Texas developed this satellite called Sling-Sat Space Sweeper (4S), which we can see in the picture 3.13a. The satellite first captures the debris and then sends it to the ground. It then uses the moment from sending debris to the Earth, thus generating a move to the next debris. Sling-Sat consists of two collectors connected by two extendable masts, which have the form of scissors. The method goes through four phases: capture, spin-up, ejection, and return. Contact occurs when one of the collectors plastically captures an object[20].

Pros	Cons
multiple targets removed	rendezvous needed
save energy	complex control system

Table 3.20. Pros and Cons of the slingshot method

3.4.2 Adhesive method

The second method is the adhesive method. It was designed by Astro Scale in Singapore. The technique consists of Mothership, which contains several boys. Boys are deorbiting kits, which are placed on the target debris. The advantage of the adhesive method is that Mothership includes several deorbiting kits. It means that it can remove multiple debris in one flight, similar to the slingshot. Each boy is equipped with a propulsion system. On the front, there is a surface with a silicone adhesive compound. The adhesive compound is installed using a joint with a 20-degree tolerance. The boy and target debris must be synchronized before installation. Subsequently, the boy can approach the target debris either along the tumbling axis or orthogonal to the tumbling axis. The picture 3.13b shows an example of an approaching boy to target debris in both directions[20].

Pros	Cons
multiple targets removed	synchronization
short working period	rendezvous needed
	complex control system

Table 3.21. Pros and Cons of the adhesive method

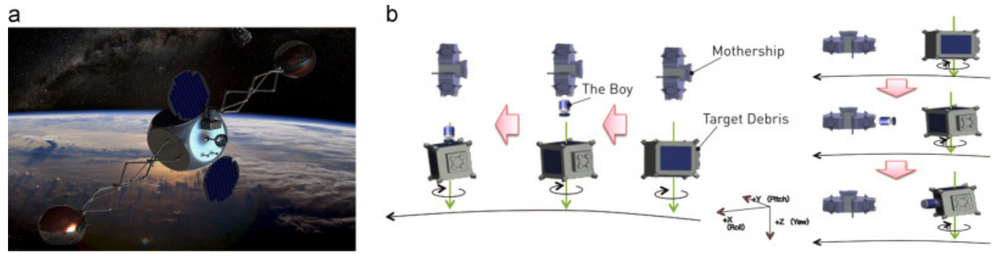


Figure 3.13. Example of the slingshot(a) and adhesive methods(b)[20]

3.4.3 Tentacle capturing

In ESAs e.Deorbit project, capturing using tentacles, can be performed either with or without a robotic arm. With a robotic arm used, tentacle capturing embraces the space debris with a clamping mechanism after holding a point on the target by the robotic arm. Finally, a velocity increment by the chaser will deorbit the combined object. A trade-off shows that tentacle capturing with a robotic arm leads to a higher cost, mass, volume, hazardousness, and complexity of design than one without a robotic arm. Tentacle capturing without a robotic arm follows the capturing before touching strategy, i.e., the tentacles should ideally embrace the target before performing physical contact. In this way, the bouncing of the chaser satellite is avoided, and the attitude control system is allowed to stand by during capturing. The clamping mechanism is then locked, and the composite (chaser and target) essentially turns stiff after capturing[34].

Pros	Cons
stiff composite	complicated rendezvous phase
easy to test on the ground	possible to be bounced
	accurate relative positioning and velocity needed

Table 3.22. Pros and Cons of the tentacle capturing

Technical parameter	Value
chaser dry mass	1435 kg
base components	booms sensors, solar array propellant, battery

Table 3.23. Technical specification of the tentacle

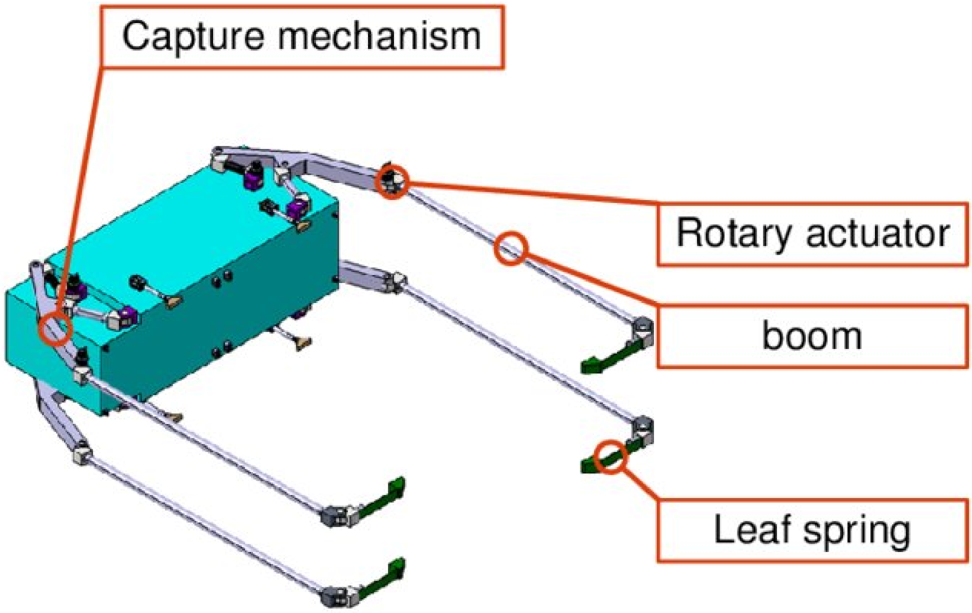


Figure 3.14. Example of the tentacle mechanism[34]

3.4.4 Single robot arm capturing

The method of capturing with a robotic arm is based on a chaser satellite equipped with a robotic arm(lightweight manipulator) that has pliers at its end to hold a target object. This concept is not new in the field of capturing. For example, the first satellite with a robotic arm was the ETS-VII. Space debris removal is more complex than missions, where the satellite was equipped with a robotic arm to capture the object. The target object is often uncontrollable, does not communicate, and tumbling can also occur[20]. In Figure 3.15 we can see the service (chaser) satellite and the client (target) satellite and the three phases that this method goes through. The image comes from the Deutsche Orbitale Servicing Mission (DEOS), which took place in 2018 and demonstrated debris disposal using a manipulator. Finally, this mission did not take place and was canceled[28].

The most dangerous phase is capture. At this phase, the two satellites may collide, or the robotic arm may be damaged. Other issues that need to be addressed with this method are: Minimizing the impact influence, De-tumbling, and Attitude synchronization. The method's pros over the tether-based methods is the ability to re-capture the target object if the capture failed on the first attempt and no damage occurred. Table 3.24 shows other pros and cons[20].

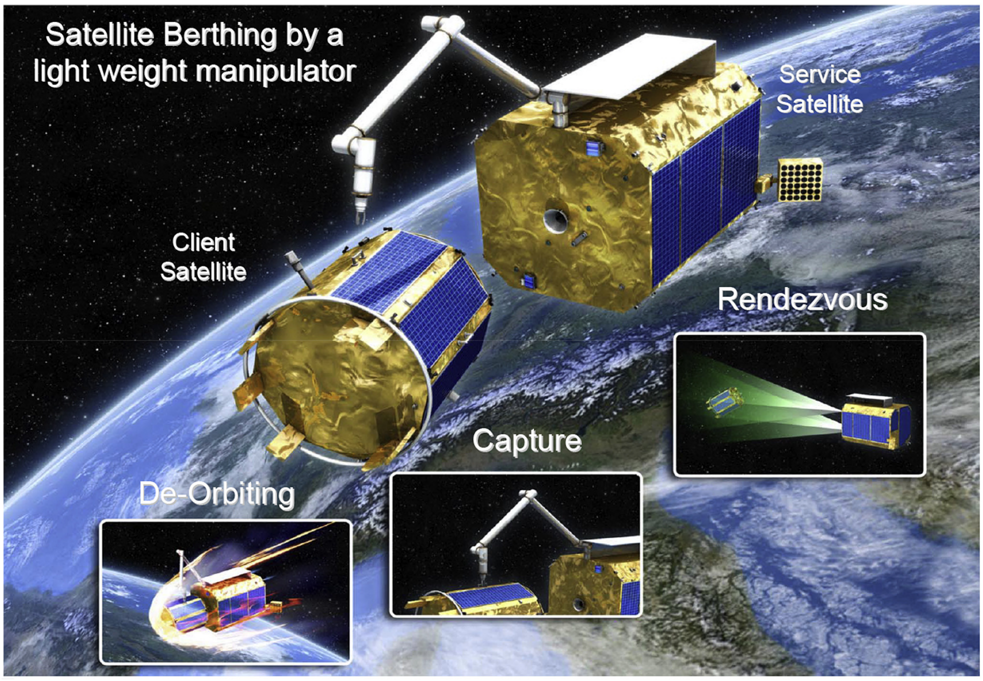


Figure 3.15. Example of the single robotic arm[28]

Pros	Cons
possible re-capture easy to test on ground stiff composite	additional space debris rendezvous needed docking needed higher probability of collision grappling point required

Table 3.24. Pros and Cons of the single robotic arm

Technical parameter	Value
chaser dry mass	732 kg
power	280 - 700 W
arm length	1.5 m
base components	robotic arm sensors, solar array propellant, battery

Table 3.25. Technical specification of the single robotic arm

3.4.5 Multiple arms capturing

Conceptually, it is similar to the previous method, but unlike a single robotic arm, two or more arms are used. The use of other arms is suitable, for example, for stabilizing the target object. In the picture 3.16, we can see an example of a chaser satellite equipped with two robotic arms. The idea comes from the Advanced Telerobotic Actuation System (ATLAS)[20].

Pros	Cons
possible re-capture	additional space debris
easy to test on ground	rendezvous needed
stiff composite	docking needed
flexible capturing	higher probability of collision
	grappling point required
	higher mass and cost
	complex control system

Table 3.26. Pros and Cons of the multiple robotic arms

The use of multiple arms increases the overall mass and cost. It is also necessary to use a more complex control system when handling the individual arms so that they can work together. The table 3.26 shows the pros and cons of this method.

Figure 3.16. Example of the multiple robotic arms [20]

Chapter 4
Comparison of ADR methods

4.1 Technology Readiness Level

Technology readiness level (TRL) describes the level at which a particular technology occurs. TRL is used to define the level of hardware as well as software. The TRL has the following nine levels[35]:

1. 1„Basic principles observed and reported“
2. 1„Technology concept and/or application formulated“
3. 1„Analytical and experimental critical function and/or characteristic proof of concept“
4. 1„Component and/or breadboard validation in laboratory environment“
5. 1„Component and/or breadboard validation in relevant environment“
6. 1„System/sub-system model or prototype demonstration in an operational environment“
7. 1„System prototype demonstration in an operational environment“
8. 1„Actual system completed and flight qualified̈through test and demonstration“
9. 1„Actual system flight proven through successful mission operations“

Figure 4.1 is a graph showing the individual ADR methods described in Chapter 3. Solar sails, harpoon, IBS, single and multiple robotic arms appear to be the most promising methods from a TRL perspective.

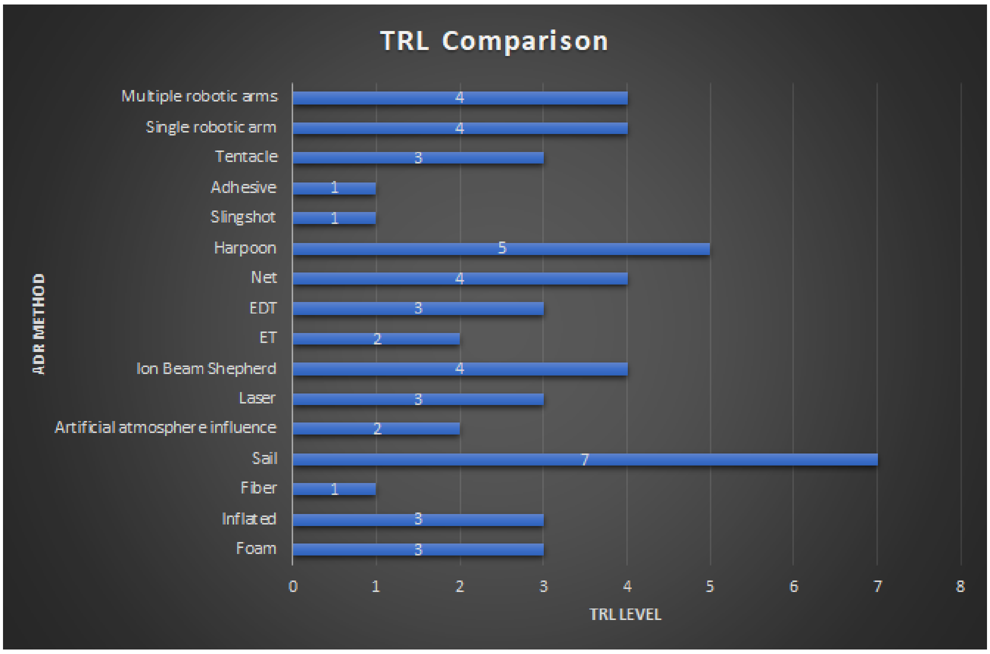


Figure 4.1. TRL Comparison

4.2 Orbit suitability

The ideal method should be cost-effective, not contribute to the generation of new debris, and remove more than one debris in a single launch. Construction and subsequent operation should not be complicated. However, neither method is ideal, each has its advantages and disadvantages. They use different ways to mitigate the population of debris in space.

Each of the methods described in Chapter 3 can be used in LEO except for the electrostatic tractor. Methods such as EDT and artificial atmosphere influence can only be used in LEO because they use physical properties that can only be performed in LEO. Other methods such as net, harpoon, IBS are more flexible and can be used in both types of orbits. The suitability of using particular methods in a specific orbit can be seen in the table 4.1.

Methods	Orbit suitability
Foam method	LEO
Inflated method	LEO
Fiber-based method	LEO
Sail method	LEO/GEO
Artificial atmosphere influence	LEO
Laser-based method	LEO/GEO
Ion beam shepherd	LEO/GEO
ET	GEO
EDT	LEO
Net	LEO/GEO
Harpoon	LEO/GEO
Slingshot method	LEO
Adhesive method	LEO
Tentacles	LEO/GEO
Single robotic arm	LEO/GEO
Multiple robotic arms	LEO/GEO

Table 4.1. Comparison of the use of individual methods on different types of orbits.

4.3 Effectiveness of ADR methods

The effectiveness of ADR methods can be compared in a number of respects, such as the amount of fuel required per mission, the number of debris that a particular method can remove per launch, the time required to remove single debris. Table 4.2 shows the time required to remove one debris. Removing the debris may take shorter or longer for methods where it has not been possible to determine or estimate the approximate time. Shorter means less than a year. Longer means a year or more. It is not easy at this time to determine how much a particular method could remove debris per launch. For example, a chaser satellite with a harpoon mechanism could contain one but also six harpoons. It depends on the design of the chaser satellite. It is similar to other methods

Methods	Duration
Foam method	1 year
Inflated method	1 year
Fiber-based method	1 year
Sail method	Longer
Artificial atmosphere influence	Longer
Laser-based method	121 - 165 days
Ion beam shepherd	200 - 360 days
Electrostatic tractor	2 - 4 months
EDT	232 - 572 days
Net	217 - 345 days
Harpoon	Shorter
Slingshot method	Longer
Adhesive method	Longer
Tentacle	282 - 406 days
Single robotic arm	282 - 406 days

Table 4.2. Comparison of the duration of ADR methods per deorbitation or reorbitation

4.4 Cost comparison

Table 4.3 compares the individual methods by cost. For laser, IBS, EDT, net, and single robotic arm methods, the price includes design, construction, launch, and operation[28]. For other methods, only the cost of constructing the method is given. The price was based on the assumption that one 1kg costs $25k[36]. For methods where the mass of the satellite is not known, the price cannot be estimated. Therefore, they are not included in the comparison.

Methods	Cost
Foam method	$115 million
Inflated method	$12.5 million
Fiber-based method	$115 million
Sail method	$7.5 million
Laser-based method	$473 million
Ion beam shepherd	$425 million
EDT	$788 million
Net	$374 million
Harpoon	$3.8 million
Tentacle	$35.9 million
Single robotic arm	$353 million

Table 4.3. Cost comparison of ADR methods

4.5 Pros and Cons comparison

This section contains four tables to compare the features of each method in groups.

Blue square sign means that the feature in the first column of the table is included in the method.

The character - (minus) means the exact opposite. The feature is not included in the method.

Feature	Foam	Inflated	Fiber	Sail
docking	-	-	-	-
risk for collision	-	-	-	■
additional debris	■	-	-	-
test on ground	easy	easy	easy	hard
rendezvous	■	■	■	-
size of debris	small and large	small and large	small and large	small and large
control reentry	no needed	no needed	no needed	possible
reliability	lower	lower	lower	good
distance from the TO	a few meters	-	a few meters	-
lower mass	■	■	■	■
various target object shape	■	■	■	■
tumbling rate	-	problem with high	-	-
damage by small debris	■	■	■	■
tracking of debris	■	■	■	-
multiple targets removed	■	■	■	-
multipurpose method	-	-	-	■
control system	no needed	no needed	no needed	no complex
additional propellants	■	■	■	-
TRL	1	1	1	7

Table 4.4. Comparison of pros and cons of Drag augmentation system.

Feature	AAI	Laser	IBS	ET
docking	-	-	-	-
risk for collision	-	higher	possible	lower
additional debris	-	■	possible	-
test on ground	easy	easy	harder	hard
rendezvous	-	partially	partially	partially
size of debris	small and large	small and large	small and large	small and large
control reentry	no control	no control	no control	no control
reliability	good	good	good	good
distance from the TO	100 meters	100m - 1000km	10 - 20m	a few meters
capture operation time	-	-	-	-
object shape	-	-	-	-
tumbling rate	-	-	-	agnostic
damage by small debris	-	-	-	-
tracking of debris	-	■	■	■
multiple targets removed	■	■	■	■
control system	no complex	no complex	no complex	no complex
additional propellants	-	■	■	■
TRL	1	3	4	1

Table 4.5. Comparison of pros and cons of Contactless removal methods.

Feature	Net	Harpoon	EDT
docking	no complex	no complex	■
risk for collision	-	possible	higher
additional debris	■	■	-
test on ground	hard	easy	hard
rendezvous	■	■	■
size of debris	small and large	large	large
control reentry	■	■	no control
critical oscillations	■	-	-
reliability	lower	lower	good
distance from the TO	100 m	1 km	-
lower mass	■	■	■
capture operation time	ahort	short	short
various target object shapes	■	■	■
tumbling rate	-	problem with high	problem with high
penetration problem	-	■	-
damage by small debris	-	-	-
tracking of debris	■	■	■
multiple targets removed	■	■	■
control system	no complex	no complex	no complex
stiff composite	no stiff	■	■
additional propellants	■	■	-
TRL	4	5	3

Table 4.6. Comparison of pros and cons of Tether based methods.

Feature	Tentacle	Single arm	Multiple arms	Slingshot	Adhesive
docking	■	■	■	■	■
risk for collision	higher	higher	higher	higher	higher
additional debris	■	■	■	■	■
test on ground	easy	easy	easy	hard	hard
rendezvous	■	■	■	■	■
size of debris	large	small and large	small and large	large	large
control reentry	■	■	■	-	■
reliability	lower	lower	lower	lower	lower
chaser mass	higher	higher	high	higher	high
capture operation time	higher	higher	higher	higher	higher
tumbling rate	-	$\leq 30°/s$	$\leq 30°/s$	$\leq 30°/s$	$\leq 1\text{-}2°/s$
damage by small debris	-	-	-	-	-
capture needed	■	■	■	■	■
multiple targets removed	■	■	■	■	■
control system	no complex	no complex	complex	complex	complex
stiff composite	■	■	■	■	-
additional propellants	■	■	■	■	-
TRL	3	4	4	1	1

Table 4.7. Comparison of pros and cons of Contact removal methods.

4.5.1 Chosen method

Initially, the Ion Beam Shepherd method was chosen to demonstrate the simulation based on the above comparison of ADR methods. The reasons for the selection were as follows:

1. Orbit flexibility
2. Lower risk of collision
3. No capture phase
4. No additional debris
5. Higher TRL (Ion thruster in 4. level)

Due to the development of the Czech Republic's own laser, the IBS method was replaced by a laser space-based method for simulation.

Chapter 5
Laser space-based method

This chapter describes the theoretical knowledge of the disposal of space debris using the laser space-based method.

The laser method aims to reduce the altitude of the debris by using a laser beam. After emitting the beam that impinges on the surface of the debris, a thin layer of the surface of the debris is ablated. Due to ablation, the recoil occurs in the opposite direction, and thus the debris slows down. In this way, it is possible to send the debris to an elliptical orbit and make its perigee as close as possible to the Earth's surface (for example, perigee P = 100 km).

The first important parameter in the laser method is the **mechanical coupling coefficient (C_m)**. The coupling coefficient is related to the energy of the beam incident on the surface of the debris. Its value depends on the material type of the target object[28].

$$m_{debris}\Delta v = C_m E \tag{5.1}$$

where,
m_{debris} - mass of debris
Δv - change of velocity
E - laser pulse energy

The **ablation rate (μ)** determines the change in the mass of the debris after the i-th beam is ejected.

$$m_i = m_{debris} - \sum_i \mu E \tag{5.2}$$

where,
m_i - mass of debris after the ith pulse

It is possible to calculate the **velocity change (Δv_i)** from the newly calculated mass. This means how much the debris will slow down[37].

$$\Delta v_i = \frac{C_m E}{m_i} \tag{5.3}$$

Table 5.1 shows the coupling coefficient and ablation rate values for different materials. The ablation rate is unknown for polyethylene and kevlar[37].

Material	Coupling coefficient [μN/W]	Ablation rate [μg/J]
Aluminium	20	80
Carbon	14	10.2
Polyethylene	50	-
Kevlar	160	-

Table 5.1. Values of the mechanical coupling coefficient for different materials

Currently, the changed velocity of the target is known, but we do not know its transfer orbit. This can be calculated using the Hohmann transfer, explained in section 6.4.

Another parameter related to the laser method is the **duration of one laser pulse (τ)**. The value usually ranges from a few picoseconds to nanoseconds. **Optimal laser fluence (F_{opt})** is related to the duration of the beam. If the laser pulse duration value is less than or equal to 100 ps, then the optimal laser fluence is equal to one. If the laser pulse duration value is greater than 100 ps, then the optimal laser fluence is equal to $\sqrt{\tau}$. Knowledge of laser fluence enables to determine the **energy of one laser pulse (E_{opt})**. The laser pulse's energy depends on the beam's **spot size (d_L)**, which can be calculated using a formula (5.5)[38].

$$E_{opt} = \pi \left(\frac{d_L}{2} \right)^2 F_{opt} \tag{5.4}$$

$$\frac{d_L}{2} = \frac{M^2 \lambda L}{D} \tag{5.5}$$

where,
M^2 - beam quality ($M^2 \approx 1$ for the space-based method)
λ - wavelength
L - the distance between the chaser satellite and the target object
D - mirror diameter

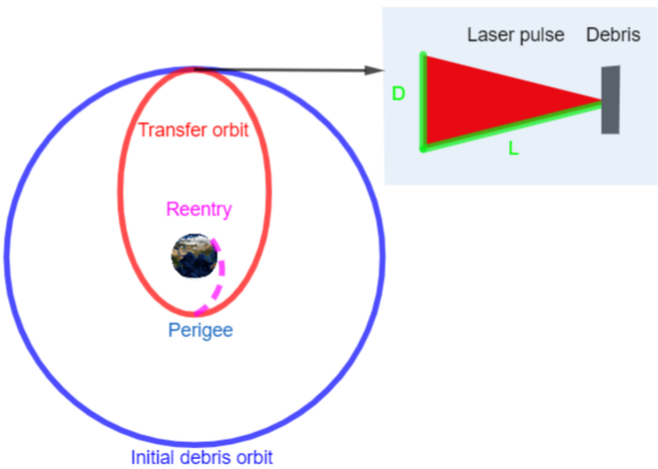


Figure 5.1. Laser deorbitation

5.1 Construction of the chaser satellite

The construction of the chaser satellite consists of[38]:

- 4 solar panels to deliver sufficient laser power,
- Telescope for focusing and sweeping the beam,
- The primary and secondary mirrors for emitting the beam and its recapture after reflection from the surface of the target debris. This allows us to find out the parameters of the debris in shooting mode (for example, its velocity by comparing two reflected laser beams). The beam can be tilted up to 5 degrees from the optical

axis of the telescope. It is also possible to move the secondary mirror about 5 cm transversely.

- Rigid body containing amplified and phase combined laser fibers to create an array where each phase can be precisely controlled.

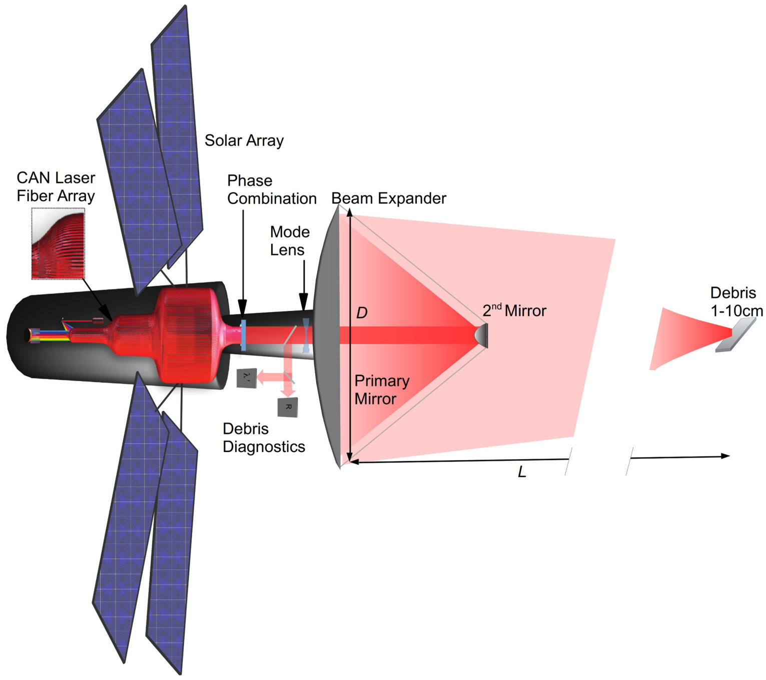


Figure 5.2. The design of the chaser satellite[38]

5.2 Removal process

Laser removal consists of several steps:

- Launch
- Rendezvous
- Scanning debris - the phase when the laser beam is swept and scans the space about size πr_{scan}^2.
- Tracking target object - the phase when the beam is focused specifically at the target object and determines its movement and velocity.
- Shooting beam - laser beams are emitted to ablate the surface and change the trajectory of the target object.

Table 5.2 contains values for three different numbers of fibers used, assuming that the size of the primary mirror is $D = 3$ m. L_{scan} is the maximum beam distance for scanning. In the tracking zone from L_{scan} to L_{max} space debris is detected and tracked. In the shooting zone from L_{min} to L_{max}, the debris surface is ablated.

Number of fibers	L_{scan} [km]	r_{scan} [m]	L_{max} [km]	L_{min} [km]	r_{min} [km]
1000	110	30	20	10	0.88
10000	180	60	60	20	1.8
100000	300	400	170	70	6.1

Table 5.2. Values describing zones for three different numbers of fibers[38]

Chapter 6
Basic principles of orbits and satellites

6.1 Orbital elements

If we want to describe the orbit on which the object is located and the position of this object in the orbit, then it needs to know six orbital elements[39]:

- a - semi-major axis
- e - eccentricity
- i - inclination
- Ω - right ascension of the ascending node(RAAN)
- ω - argument of perigee
- ν - true anomaly

The **semi-major axis** value defines the size of the orbit. The semi-minor axis is often not used to define an orbit. Instead, **eccentricity** is used to define the shape of the orbit. In Table 6.1 we can see the value of eccentricity and the corresponding shape of the orbit along which the objects move. Only two orbit shapes are considered in the thesis, namely the circle and the ellipse, which are typical for the Earth's orbit[39].

Eccentricity value	Shape
$e = 0$	Circle
$0 < e < 1$	Ellipse
$e = 1$	Parabola
$e > 1$	Hyperbola

Table 6.1. Shape of orbits

The eccentricity can be calculated either by the ratio of the distance between the focus and the centre of the ellipse and the semi-major axis or by the semi-major axis and semi-minor axis as shown by the following formula[39].

$$e = \frac{c}{a} = \sqrt{1 - \frac{b^2}{a^2}} \tag{6.1}$$

where,
e - eccentricity
c - distance between the focus and the centre of the ellipse
a - semi-major axis
b - semi-minor axis

In Figure 6.1 we can see both shapes of orbits. The elliptical orbit shows the semi-major axis (red), the semi-minor axis (orange), the distance between the centre of the orbit and the earth centre (green), the **apogee** which is the farthest orbit's point from the Earth's surface and the **perigee** which is the closest orbit's point to the Earth's surface, and the focal point (F_2). In a circular orbit, only the semi-major axis (called the radius) is shown because the semi-major axis equals the semi-minor axis, and the eccentricity is zero. Hence, the Earth is the centre of the circular orbit[39].

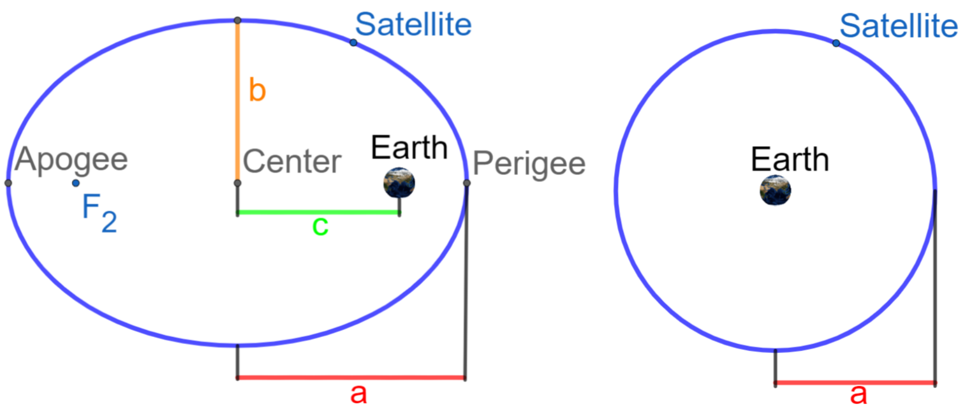


Figure 6.1. Elliptical and circular orbit

The values of apogee and perigee can be calculated using the equations (6.2) and (6.3).

$$A = a(1 + e) \tag{6.2}$$

$$P = a(1 - e) \tag{6.3}$$

The following orbital element is related to the orientation of the orbital plane. It is an **inclination** that describes the tilt of the orbital plane with respect to the equatorial plane. The inclination value is in the range of 0 to 180 degrees. If the inclination value is 0 degrees or 180 degrees, it is an equatorial orbit. If the inclination value is approximately 90 degrees, it is a polar orbit. The inclination value also defines the direction of the object's movement in orbit. If the value is from 0 to 90 degrees, then the object moves in the direction of the Earth's rotation called **prograde**. If the value is from 90 to 180 degrees, then the object moves in the opposite direction of the Earth's rotation, also called **retrograde**. Figure 6.2 shows an example of the inclination of the orbital plane with respect to the equatorial plane[39].

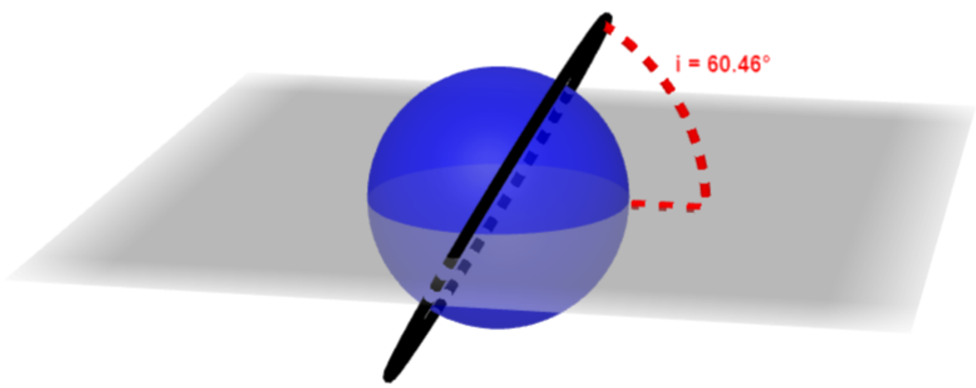


Figure 6.2. Inclination of the orbit

The **right ascension of the ascending node** is another element that is related to the orientation of the orbital plane. The **right ascension** is the angle between the vernal equinox and the intersection of the declination circle. If the inclination is not 0 or 180 degrees, then there are two points that result from the intersection of the inclined plane and the equatorial plane. The **ascending node** is one of these two points, the point when an object moving in an inclined orbit passes from the bottom to top, intersecting the equatorial plane. The right ascension of the ascending node is the angle measured from the vernal equinox to the ascending node along the equator. The value ranges from 0 to 360 degrees[39].

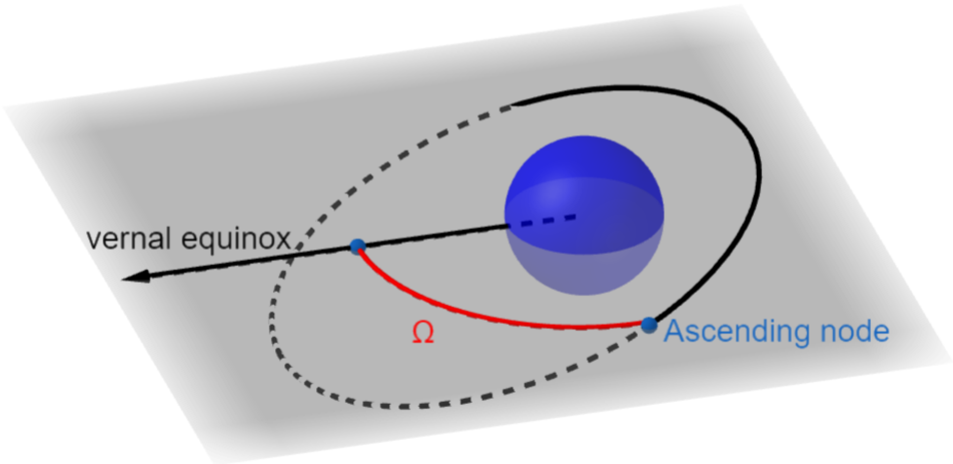


Figure 6.3. Right ascension of the ascending node

The fifth element is the **argument of perigee**. It is the angle between the ascending node and the perigee. Its value can range from 0 to 360 degrees. The argument of perigee is used to determine the position of the perigee, whether it is located in the northern or southern part of the hemisphere[39].

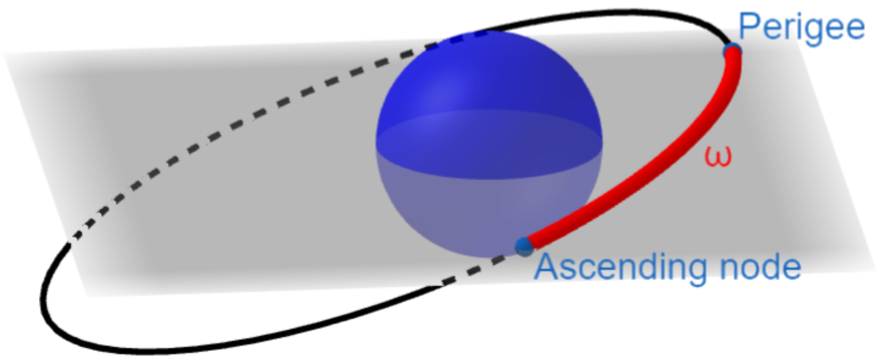


Figure 6.4. Argument of perigee

The last element is a **true anomaly**. It is the only value that changes over time. True anomaly is the angle between the perigee and the current position of the satellite. The angle value is in the range of 0 to 360 degrees. It is used to determine the exact position of a satellite in the orbit[39].

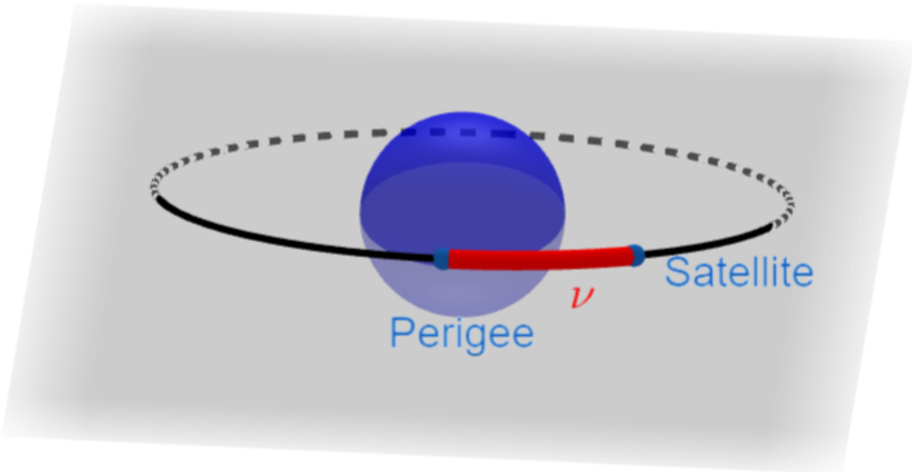


Figure 6.5. True anomaly

If the orbit is circular (i.e. $e = 0$), then there is no perigee and apogee, and it is not possible to use the argument of perigee and true anomaly. Instead, the **argument of latitude (u)** is used, which is the angle between the ascending node and the position of the satellite. In the case of an equatorial orbit (i.e. $i = 0°$ or $i = 180°$), there is no ascending node, and therefore, we cannot use RAAN and the argument of perigee. Instead, it is possible to use the **longitude of perigee (Π)**, which is the angle between the vernal equinox and the perigee. If the orbit is circular and at the same equatorial orbit (i.e. $e = 0$, $i = 0°$ or $i = 180°$), it is impossible to use RAAN, the argument of perigee and true anomaly due to the absence of ascending node and perigee. In this case, it is possible to use the **true longitude (l)** to measure the exact position of the satellite. True longitude is the angle between the vernal equinox and the position of the satellite. Table 6.2 summarizes all the above-mentioned orbital elements[39].

Orbital Element	Symbol	Value	Constraint
Semi-Major axis	a	$a > 0$	no restriction
Eccentricity	e	$e \geq 0$	no restriction
Inclination	i	$0° \leq i \leq 180°$	no restriction
RAAN	Ω	$0° \leq \Omega \leq 360°$	existing ascending node
Argument of perigee	ω	$0° \leq \omega \leq 360°$	existing ascending node existing perigee
True anomaly	ν	$0° \leq \nu \leq 360°$	existing perigee
Argument of latitude	u	$0° \leq u \leq 360°$	circular orbit
Longitude of perigee	Π	$0° \leq \Pi \leq 360°$	equatorial orbit
True longitude	l	$0° \leq l \leq 360°$	circular and equatorial orbit

Table 6.2. Summary of orbital elements

6.2 Satellite trajectory

The orbit or trajectory of a satellite can be calculated using the parametric equation of the ellipse where we get two points x and y at angle t. However, we still need to determine the third coordinate of the Cartesian coordinate system. We set the coordinate z equal to zero. This makes inclination impossible, but thanks to the framework-specific function used in the implementation, it is possible to tilt the plane, i.e. perform the orbit inclination. In the parametric equation of the ellipse (6.4), we can see an additional variable c when calculating the point x. In the case of a circular orbit, its value is zero. In the case of an elliptical orbit, it can be calculated using the formula (6.1). Adding this value places the centre of the Earth at the right focus of the elliptical orbit.

$$
\begin{aligned}
x &= a\cos(t) + c \\
y &= b\sin(t)
\end{aligned} \tag{6.4}
$$

where,
t - eccentric anomaly $(0 \leq t \leq 2\pi)$

6.3 Satellite Motion

The motion of satellites in Earth's orbit is influenced by a single force, and that is the gravitational force that controls the motion. For this reason, it is necessary to set the velocity of the satellites correctly so that they can orbit the Earth. Satellites that are closer to Earth need a higher velocity than satellites that are farther from the Earth because the gravitational force is greater the closer satellites are to Earth. The **velocity (v)** of the satellite can be calculated using a formula (6.5). Since the mass of the satellite is negligible relative to the mass of the earth, it is not included in the velocity formula. If we multiply the gravitational constant and the mass of the Earth we get the gravitational parameter (μ)[38].

$$
v = \sqrt{\frac{G \cdot M_{earth}}{R}} = \sqrt{\frac{\mu}{R}} \tag{6.5}
$$

where,
v - velocity
G - gravitational constant $(6.674 \times 10^{-11}\ m^3kg^{-1}s^{-2})$
M_{earth} - mass of the Earth $(5.972 \times 10^{24}\ kg)$
R - radius of the orbit

The second parameter that interests us when the satellite moves is the **period (T)**. The period is one orbit of the satellite around the earth. The period can be calculated using a formula (6.6)[40].

$$
T = \sqrt{\frac{4 \cdot \pi^2 \cdot R^3}{G \cdot M_{earth}}} \tag{6.6}
$$

where,
T - period of the satellite

6.4 Orbital Change

An orbital change is moving a satellite from one orbit to another. The satellite moves to another orbit along an elliptical orbit using the **Hohmann transfer**. It is assumed that the current orbit and the target orbit have a circular shape and are coplanar. Coplanarity is not generally required, but we have limited ourselves to orbits on the same plane in the thesis.

During the deorbitation of space debris using the laser method, we want to reduce the altitude of the debris. The change in altitude can be calculated using the Hohmann transfer. In addition to changing the altitude of the debris, we may require a change in the orbit of the chaser satellite, which will move from one debris to another. The Hohmann transfer consists of two steps. The first step is to change the current orbit to a transfer orbit. In the case of debris, only the first step is needed. The debris will pass into an elliptical orbit and will be closest to the Earth's surface in the perigee. This will result in entry into the atmosphere and subsequent burning in the atmosphere. The second step consists in changing the transfer orbit to the target orbit. To calculate the orbital change, we need to calculate ΔV_1 for the first step. ΔV_1 is the velocity change from the current orbit to the transfer orbit[38].

$$\Delta V_1 = |V_{transfer1} - V_{orbit1}| \tag{6.7}$$

where,
ΔV_1 - change of velocity from initial orbit to transfer orbit
$V_{transfer1}$ - velocity in the transfer orbit at initial orbit
V_{orbit1} - velocity in the initial orbit

For the second step, it is necessary to calculate ΔV_2 which is the change in velocity from the transfer orbit to the target orbit. All velocities can be calculated according to the formula (6.5) from the previous section[40].

$$\Delta V_2 = |V_{orbit2} - V_{transfer2}| \tag{6.8}$$

where,
ΔV_2 - change of velocity from transfer orbit to target orbit
V_{orbit2} - velocity in the target orbit
$V_{transfer2}$ - velocity in the transfer orbit at target orbit

To calculate the **total velocity change (ΔV_{total})**, it is enough to add ΔV_1 and ΔV_2.

$$\Delta V_{total} = \Delta V_1 + \Delta V_2 \tag{6.9}$$

Another parameter in the orbital change is the **mechanical energy (ε)** of the satellite. Mechanical energy can be calculated using a formula (6.10). Thanks to mechanical energy, it is possible, for example, to estimate the amount of fuel to perform a maneuver[40].

$$\varepsilon = \frac{\mu}{2a} \tag{6.10}$$

where,
ε - mechanical energy of the satellite
μ - gravitational parameter

The last parameter that interests us is the **time of flight (TOF)**. The time of flight can be calculated as a half of the period because the flight lasts exactly 180 degrees[40].

$$TOF = \frac{T}{2} = \pi\sqrt{\frac{a_{transfer}^3}{\mu}} \tag{6.11}$$

where,
T - period of the satellite
$a_{transfer}$ - semi-major axis of the transfer orbit

In equation (6.11) we can see the unknown variable $a_{transfer}$ which is the semi-major axis of the transfer orbit. $a_{transfer}$ can be calculated as the sum of the radii of both orbits (current orbit and target orbit) divided by two[40].

$$a_{transfer} = \frac{R_{current} + R_{target}}{2} \tag{6.12}$$

To visualize the transfer orbit during an orbital change, it is necessary to replace the parametric equation of the ellipse described in Section 6.2 with the modified parametric equation of the ellipse(6.13).

It is necessary to replace the parametric equation of the ellipse described in Section 6.2 with a modified parametric equation of the ellipse (6.13) to visualize the transfer orbit during an orbital change. Modification is necessary because an object moving in a circular orbit changes its position constantly. This means that it is necessary to rotate the ellipse of the transfer orbit concerning the object's position.

$$\begin{aligned} x &= a\cos(t)\cos(\theta) + \sin(t)\cos(\theta) + c \\ y &= b\sin(t)\cos(\theta) - \cos(t)\sin(\theta) \end{aligned} \tag{6.13}$$

where,
θ - current angle ($0 \leq \theta \leq 2\pi$)

6.5 Rendezvous

The **rendezvous** phase occurs with many of the ADR methods described in Chapter ??. The laser method that was chosen to simulate space debris is one of them. Specifically, it is far-rendezvous because the distance between the chaser satellite and the target object is approximately 100 km. Simulation of the approach of a chaser satellite to a target object uses only a **coplanar rendezvous**. This means that the orientation (inclination and RAAN) of both orbits is the same. Timing is important for rendezvous using the Hohmann transfer explained in the previous section. This means that it is necessary to know when to start the engines, the initial position of both objects, and the waiting time for the start of the Hohmann transfer. First, we need to determine the **angular velocity (ω)** of both objects. This can be calculated according to the formula (6.14). We need the angular velocities of both objects because their velocities are changeable[40].

$$\omega = \sqrt{\frac{\mu}{a^3}} \tag{6.14}$$

Next, we need to calculate the duration of the Hohmann transfer (time of flight). The formula is the same as when calculating the TOF for an orbital change. **Lead angle (α_{lead})** is another parameter in the rendezvous calculation. The lead angle is the angle that the target object encloses with the end point of the orbital change of the chaser satellite. For simplicity, it is the distance (expressed in angle) that the target object must travel until the chaser and target objects meet from the start of the Hohmann transfer. The lead angle is calculated by multiplying the angular velocity of the target object and the TOF[40].

$$\alpha_{lead} = \omega_{target} TOF \tag{6.15}$$

where,
α_{lead} - lead angle
ω_{target} - angular velocity of the target object
TOF - time of flight

Phase angle (ϕ_{final}) is the initial angle between the chaser satellite and the target object. In essence, it is complementary to the lead angle, because the angle that the chaser satellite must pass is 180 degrees, and if we know the lead angle, it is enough to subtract it from 180° or π because the lead angle is expressed in radians[40].

$$\phi_{final} = \pi - \alpha_{lead} \tag{6.16}$$

We currently know the angular velocity of both objects, the duration of the orbital change, the phase angle between the chaser and the target object, and the lead angle of the target object. This is all that is needed to perform a rendezvous if both objects are in the correct position. However, the probability that both objects are in the correct position is low. Therefore, it is necessary to calculate the **wait time**, after which both objects will reach the initial position. Wait time is calculated as ratio of the subtraction between the final phase angle and the initial phase angle and the subtraction between the angular velocities of the target object and the chaser satellite. The initial phase angle is the angle at the current position of both objects. The final phase angle is the angle at the start of the Hohmann transfer[40].

$$\text{wait time} = \frac{\phi_{final} - \phi_{initial}}{\omega_{target} - \omega_{chaser}} \tag{6.17}$$

where,
$waittime$ - time to start the Hohmann transfer
$\phi_{initial}$ - initial phase angle
ω_{chaser} - angular velocity of the chaser satellite

The wait time can be negative due to the position of both objects, in which case it is necessary to add or subtract 2π to the $\phi_{initial}$, depending on which of the two options returns a positive wait time. We can add or subtract 2π because we have circular orbits and after adding or subtracting 2π (360°) we find the chaser in the same position.

In Figure 6.6 we can see an example of rendezvous, which was described above. We can see that the chaser satellite wants to move from a lower altitude to a higher one. The chaser satellite is in the phase of initiating an orbital change. Thus, the wait time is equal to zero.

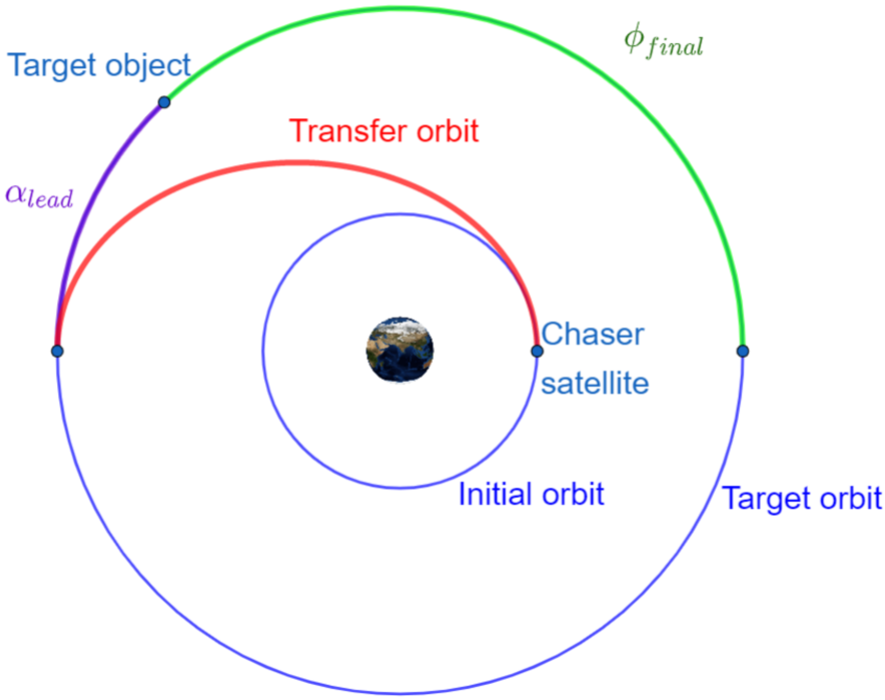


Figure 6.6. Rendezvous

Chapter 7
Models

Several models had to be designed to simulate a disposal of space debris using the laser method. The first created model is a chaser satellite with four solar panels, two mirrors for beam expansion, and its recapture. The model's appearance was created based on the description from Chapter 5. The Galileo satellite is the second created model. This model is designed to simulate an orbital change or rendezvous. Other models created are target object models designed to simulate deorbitation. Thus, fragments with different shapes have been created, resembling fragments formed during collisions of objects. The last created model was a screwdriver because space debris is not only debris formed from the accident but also tools thrown away, for example, from the ISS station. In Figure 7.1 we can see the created fragment models.

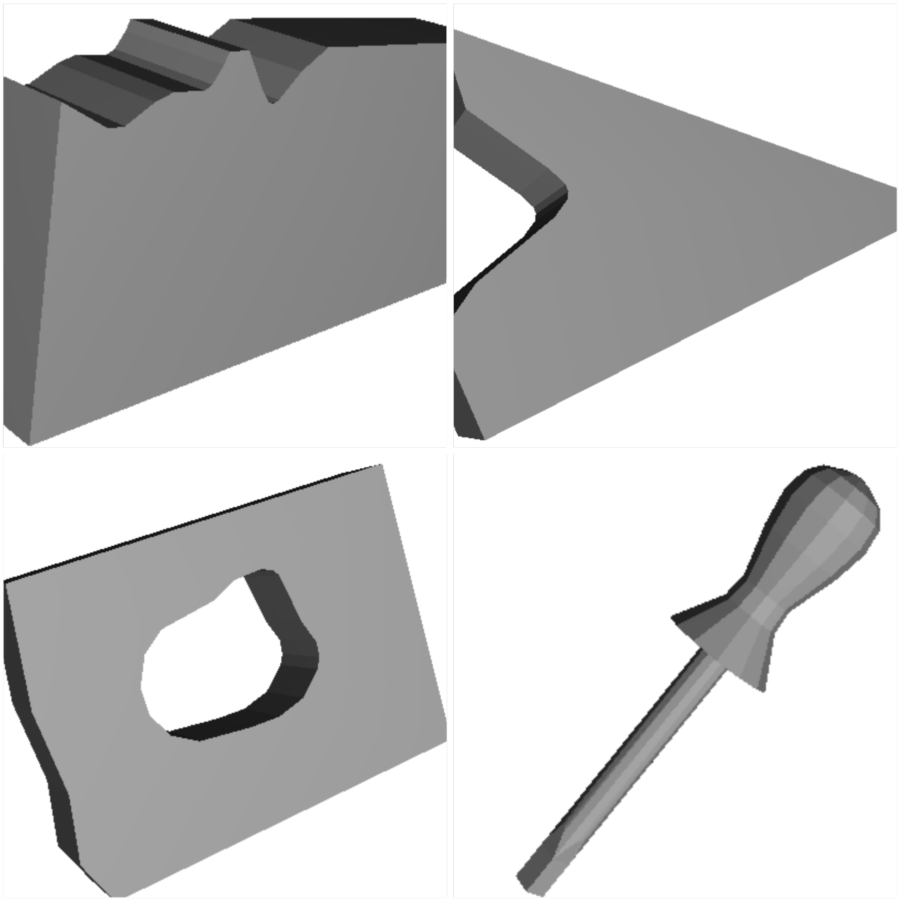


Figure 7.1. Fragment models

Models were created in Solid Edge. Solid Edge is CAD software used for 3D model design, simulation, 3D printing, data management, electrical equipment design, and so on. Its advantage is that it is easy to use and allows a quick design of simple models[41].

7.1 Model creation

Creating a 3D model consists of the following steps:

1. **Sketch creation** - is the application of basic 2D models (for example, line, circle, rectangle, etc.) to create the final form of a 3D model. The shape created in the sketch represents a cross section of the 3D model. Figure 7.2 is a sketch of a circle for which we have defined its radius.

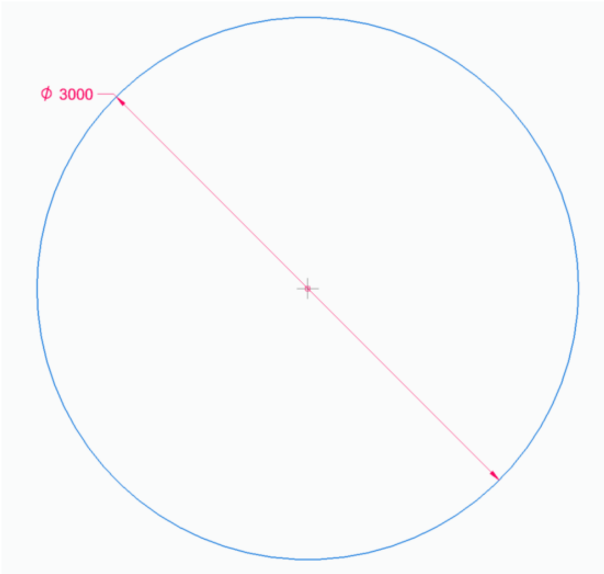


Figure 7.2. Sketch of a primary mirror

2. **Creating a model from a sketch** - in this part, the 3D model is shaped using basic solid edge functions such as extrude, cut, revolve, revolve cut and others. In Figure 7.3 we can see the created model from the sketch by applying the extrude function, whose task is to create a 3D model from a 2D object by simply holding and moving the mouse.

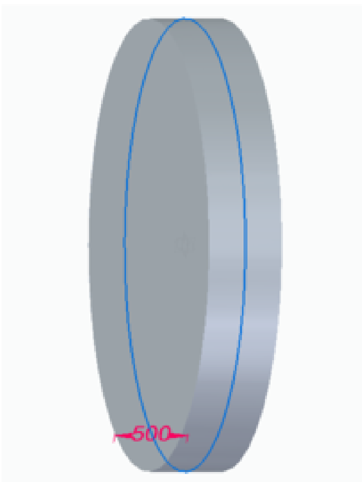


Figure 7.3. Primary mirror model

3. **Final part** - a set of models created in point 2. In Figure 7.4 we can see the final part and the tree of operations from which the part is composed. In addition to the sketches and the models created from them, the tree structure also contains two rounds for the primary mirror and rigid body. The rounds were created using the round function in the solid edge, where it was necessary to define the round angle.

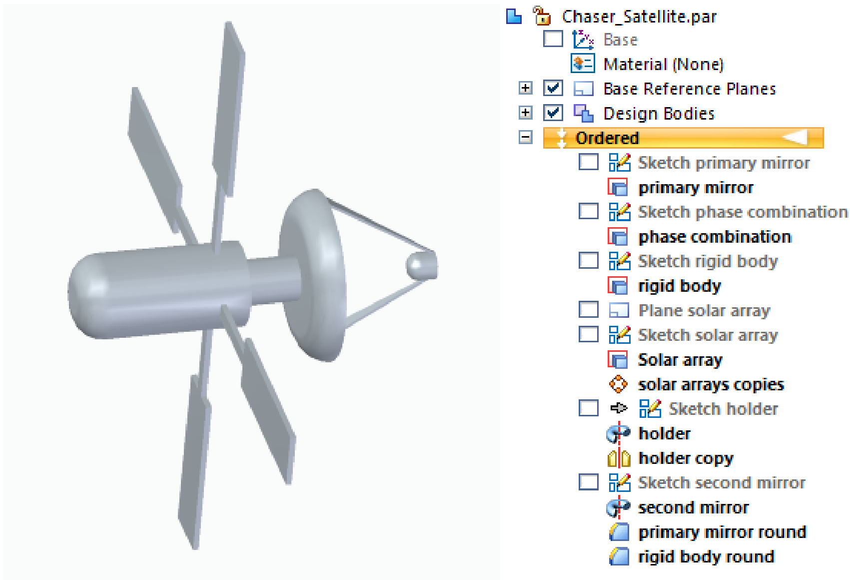


Figure 7.4. Final part of satellite chaser and tree operation

4. **Final set of parts** - connecting several parts and creating links between them.

Chapter 8
Application

8.1 Application Features

- 3D visualization of Earth orbit
- possibility to change the velocity of objects and rotation of the Earth (1 - 2048 times faster than the real velocity of objects and rotation of the Earth)
- possibility of zooming in and out
- possibility of rotation in all three axes
- allows us to add target objects from predefined models
- allows us to view the trajectory of the selected object or all currently orbiting objects
- orbital change simulation
- rendezvous simulation
- simulation of fragment deorbitation using the laser method
- allows us to create a record of orbiting objects. A new record is created whenever one of the three offered simulations is started.
- data from objects are summarized in a table where we can view them.
- view statistics (e.g., orbit pollution by orbit types)

8.2 Application Description

The application consists of several panels:

- The menu bar contains only one help item that contains information about the application.
- The top toolbar contains:
  - Reset button to reset all three x, y, z axes and return the zoom value to the initial 100 percent,
  - Toggle button to display all three auxiliary axes,
  - Toggle button to display all orbits,
  - Slider to change the speed of moving objects.
- The left panel contains:
  - List of stored simulations / orbital changes / rendezvous,
  - Three sliders for change in individual axes X, Y, Z,
  - Slider to change the zoom value.
- The main panel contains 3D objects and their motion simulation. In addition, it is possible to display the values of all six orbit elements.
- The right/simulation panel consists of several titled panes for:
  - Creating an object,

- Setting simulation parameters and running it,
- Co-planar orbital change setting,
- Settings for far rendezvous,
- View statistics.

- The bottom/information panel contains:
  - A table that displays information about currently occurring objects in the main panel
  - Possibility to search by object name
  - Option to delete the selected object

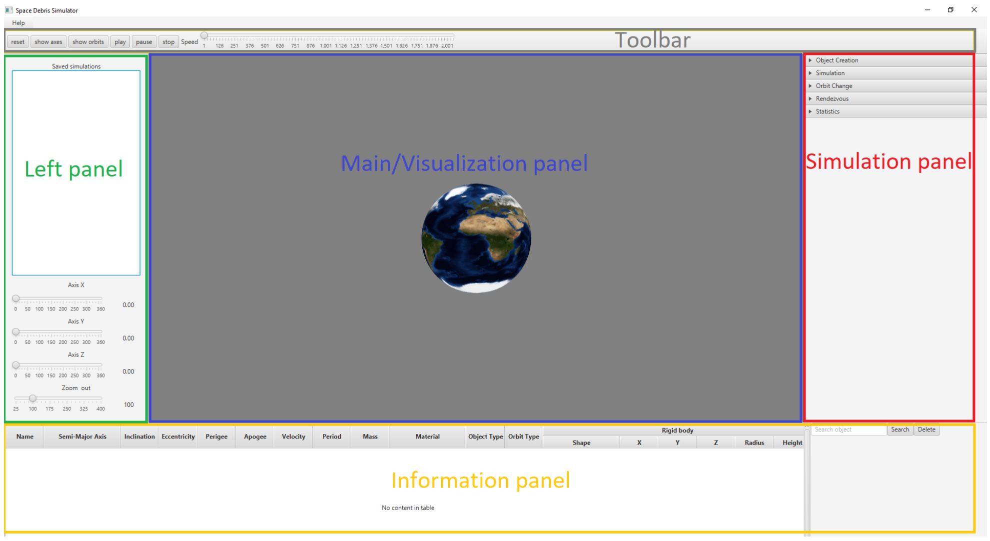


Figure 8.1. Application panels

8.2.1 Visualization panel

The visualization or main panel is used to display individual objects. In the middle of the panel is a model of the Earth around which are orbits the individual object's. It is possible to display the trajectory of individual objects, see the current information about the objects (six elements of the orbit), and progress information about the simulation.

It is possible to view all trajectories at once by pressing the toggle button `show orbits`. Pressing the toggle button again will hide all trajectories. The trajectory of a specific object can be displayed by pointing the mouse cursor at the object, but it is displayed only while holding the cursor on the object. If we click on the object, this view is permanent (even after removing the cursor from the object). Clicking on the object again will hide the orbit. Moreover, if the trajectory of one object is already displayed and the user clicks on another object, the trajectory of the first object will be hidden and the trajectory of the second object will be displayed.

By pressing the right mouse button and dragging, it is possible to rotate in all three x, y, z axes and see the visualization from a different perspective. We can use the scroll wheel to zoom in and out on objects.

The main panel contains two hidden groups of information. The first group can be shown by clicking on an object or selecting a row from the table (selecting a specific object) from the information panel. Then, information about the object is displayed in the lower right corner of the main panel. Specifically, these are the values of the six elements of the orbit. The first five (semi-major axis, eccentricity, inclination, RAAN, argument of perigee) are fixed values entered or calculated when creating the object. The last value of true anomaly is regularly updated depending on the position as explained in Chapter 6. The second group of information is displayed in the lower-left corner of the panel when the simulation starts. The group contains information on which phase the simulation is in (Far rendezvous phase, scanning mode, tracking mode, shooting mode, de-orbit mode). As we move through the stages, the individual texts are highlighted.

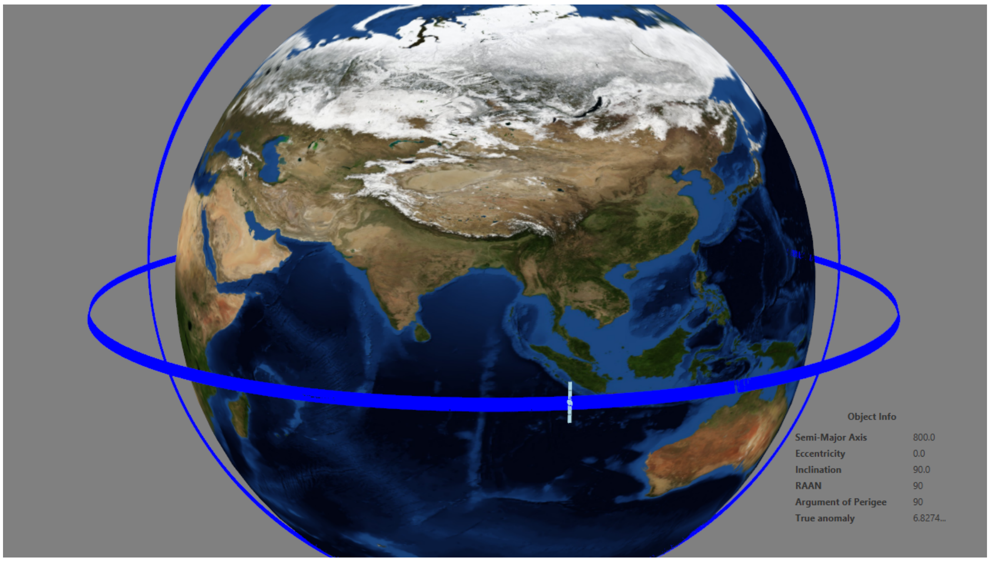


Figure 8.2. Visualization panel

8.2.2 Object Creation

Creating an object consists of entering the name of the object we want to create. This name is also a unique identifier. This is followed by entering several elements of the orbit: size, eccentricity, and inclination of the orbit. After entering the orbit size and eccentricity values, values describing additional information related to the orbit and motion of the object are displayed: Perigee, Apogee, Velocity, and Period. The eccentricity value can be entered in the range from 0 to 0.95. The inclination value is allowed in the range of 0 to 180 degrees, as mentioned in Chapter 6. During writing eccentricity or inclination, the application warns when the values are out of range. The next part of the object creation panel is occupied with defining the type of the object. We can select one of three object types:

- Satellite
- Fragment
- Rocket

Next, we need to enter the mass and material of the object. We can select one of these three material types:

- Polyethylene
- Aluminium alloys
- Kevlar

The mass and type of material are important information for performing the simulation. The last part of defining an object is choosing the appearance and size of the object. The application offers predefined objects for each type of object (satellite, fragment, rocket). Each type offers three basic types:

- Box
- Sphere
- Cylinder

In addition, the appearance of the Galileo can be selected for the satellite. The fragment type offers three basic appearances as well as three specific fragment shapes and one working tool (screwdriver). Each of the appearances contains default values for the object size, but it is possible to change them. For the sphere, it is possible to change the radius, for the cylinder, it is possible to change the radius and height, and for other appearances of objects, it is possible to scale sizes in all three x, y, z axes. After pressing the `Create` button, the entered parameters will be validated, and if the parameters match, the satellite will be created and will visualized. If any of the parameters have not been entered or its value does not match, an error message and a description are displayed as shown in figure 8.3.

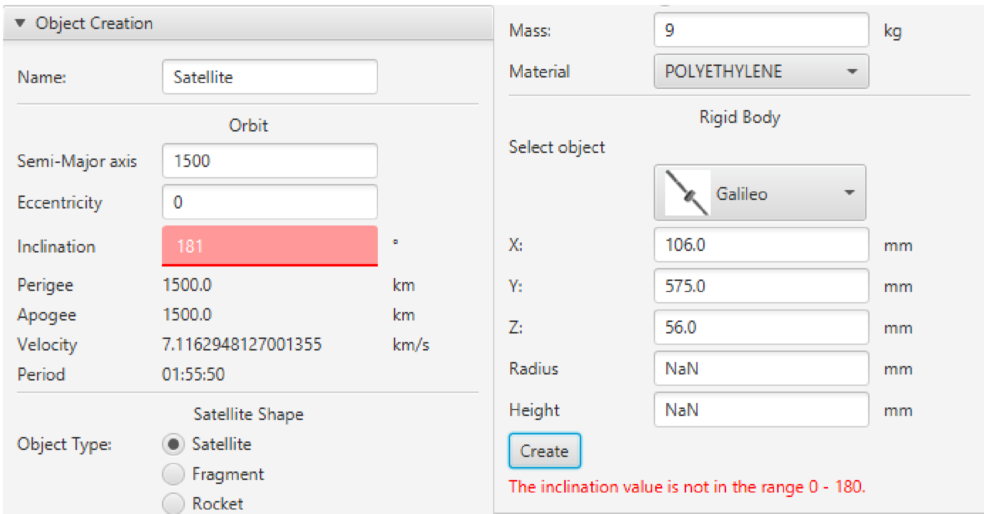


Figure 8.3. Object creation titled pane

8.2.3 Simulation

The simulation of deorbitation using the laser method consists of selecting the target object and entering the laser parameters. After selecting the target object, the user will see the calculated values for the total change velocity needed to perform the deorbitation and the total energy. After selecting the target object, the user will see the calculated values for the total change velocity and the total energy. The user can change the parameters:

- Optimal fluence - the value is related to the duration parameter. This means that after changing the optimal fluence, the duration value will also change,
- Wavelength - the coupling coefficient and ablation rate depend on this value. This means that if the user changes the wavelength, he must enter values for the coupling coefficient and ablation rate,
- Beam Quality - the value needed to calculate the spot size,
- Laser aperture - the value related to the size of the beam,
- Distance - the distance between the chaser satellite and the target object after performing rendezvous,

- Coupling coefficient - this value is known for four types of materials as described in Chapter 5,
- Ablation rate - this value is known for two types of materials. If the target object has a material type for which the ablation rate is unknown, the user must enter its value.

After entering the parameters mentioned above, the values for the energy of one pulse and the spot size will be calculated and displayed. Next, the user must define the radius (altitude of the chaser satellite). The radius value of the chaser satellite must be different from the radius value of the target object. After clicking the `Start` button, the entered parameters will be validated. If the entered parameters are valid, the simulation will start. If the specified parameters are invalid, the user will be informed by the red text under the `Start` button which parameter/s is invalid.

After starting the simulation, the rendezvous process begins. When the chaser satellite approaches the target object, the target object is ablated. This means that the chaser satellite ejects a laser beam to the target object. After a very short time, the trajectory of the target object will change. The perigee of the transfer orbit of the target object is set to 100 km. After reaching this perigee, the target object will be deleted.

It is possible to see the phase in which the simulation is located in the left corner of the visualization panel during the simulation. The phases are as follows:

- rendezvous phase,
- scanning phase,
- tracking phase,
- shooting phase
- deorbitation phase.

8.2.4 Orbital Change

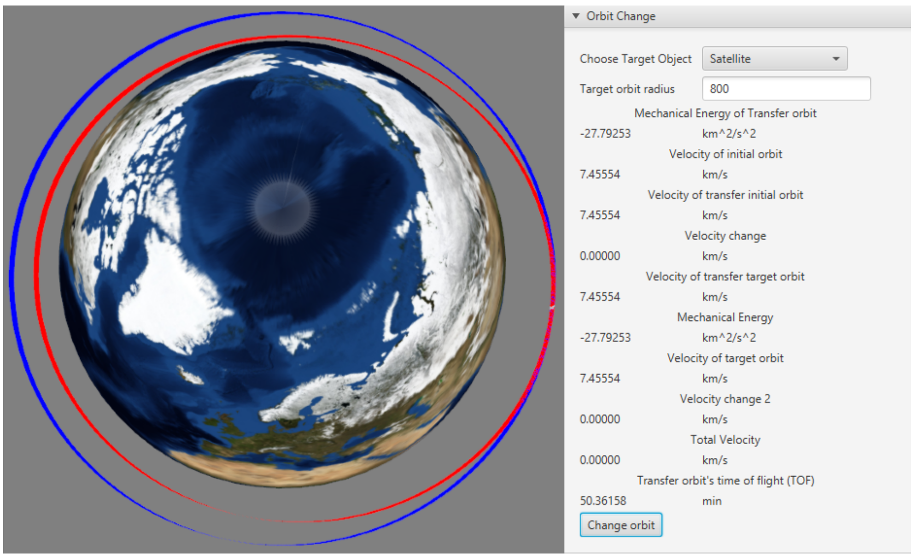


Figure 8.4. Orbit change visualization and titled pane

The simulation of a co-planar orbital change consist in selecting the object for which we want to change the altitude. It is still necessary to write the radius of the target orbit for the selected object. Suppose the object is selected and the radius of the target orbit is entered. In that case, the individual values related to the orbit change, which is described in section 6.4, are calculated. After pressing the `start` button, the simulation is started. If the trajectory of the specified object is displayed, it is possible to see two orbits, where the blue circle is the initial orbit and the red ellipse is the transfer orbit. After passing 180 degrees in the transfer orbit, only the new orbit with the required radius is displayed.

8.2.5 Rendezvous

In the rendezvous titled pane, it is possible to simulate the rendezvous of two different existing objects. In the first combo box, it is necessary to select the chaser satellite to approach the target object selected in the second combo box. Next, it is necessary to enter the distance in kilometers, which is expected to be between the two objects at the end of the simulation. After entering these three parameters, the values related to the rendezvous are calculated immediately. The individual values are described in section 6.5. When we press the `start` button, the simulation starts, and the values are recalculated because it is important to know the starting angle between the two objects, which is constantly changing.

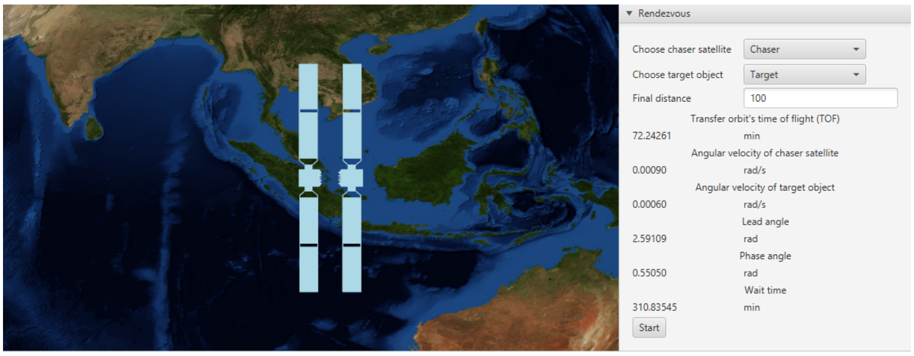


Figure 8.5. Rendezvous visualization and titled pane

8.2.6 Statistics

Statistics titled pane contains two charts. The first chart shows the pollution by orbit type. The second chart shows the pollution by orbit type and object type. The application distinguishes three types of objects: operational satellites, fragments, and rockets. Both graphs are updated as soon as there is a change in the rendering of objects. The change can occur when adding, deleting, or running a saved simulation.

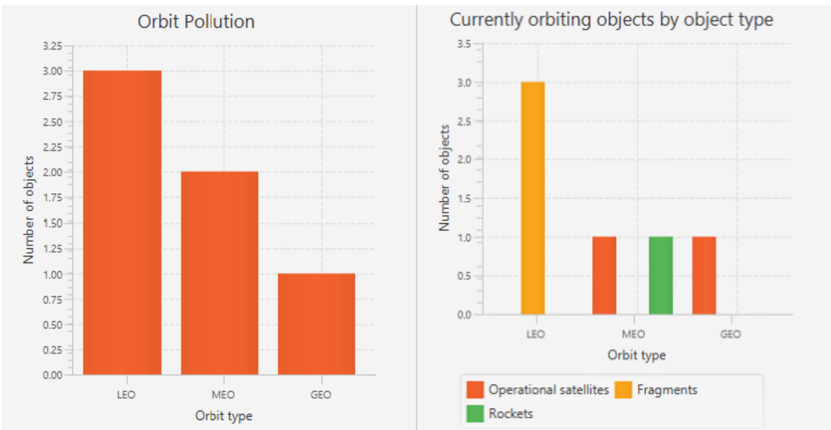


Figure 8.6. Statistics panel

8.2.7 Tool panel

In the toolbar, we can find individual buttons whose actions are closely related to the visualization. The first button is the `reset` button, which resets the camera view, which means that the slider values located in the left panel are set to the initial position. Specifically, these are the sliders for the x, y, z axes (set to 0) and zoom (set to 100). The second button is used to display the auxiliary axes. The next button is `show orbits`, which is used to show and hide orbits. The last tool in the toolbar is a slider for setting the speed of moving objects. It is possible to set the value from 1 to 2048. If the speed is set to 1, the velocity and rotation of the Earth are directly proportional to the velocity of the objects in the real Earth orbit and the speed of rotation of the real Earth. For values greater than one and less than or equal to 2048, this is a multiple of the velocity and rotation of the actual values.

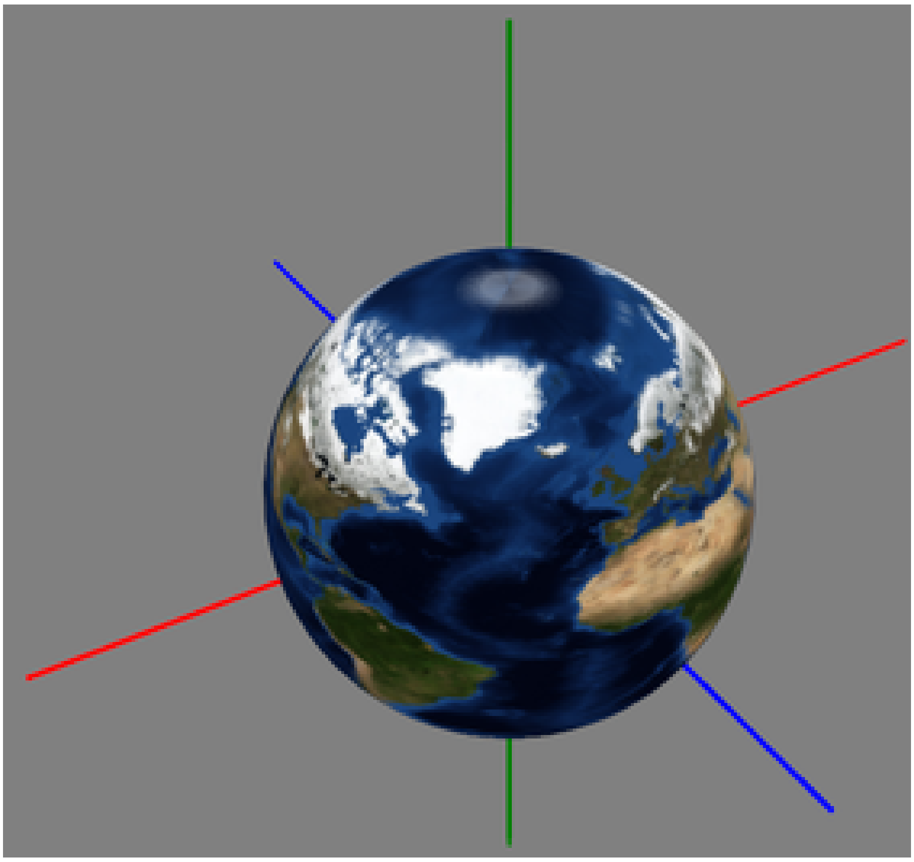


Figure 8.7. The auxiliary axes

8.2.8 Left panel

The left panel consists of two parts. The first section shows the saved simulations, which can be played by selecting a specific simulation from the list view and pressing the `start` button. Subsequently, currently moving objects are stopped in the main panel and are removed. Next, the objects from the uploaded simulation will be displayed, and the table will be overwritten in the information panel. It is impossible to add a new object or perform any of the orbital change, rendezvous, and object deorbit simulation actions in running a saved simulation. When the simulation ends, all objects stop. At the end of the simulation recording, it is possible to return to the point before starting the simulation using the `current` button. The second part consists of four sliders, where the first three are used to adjust the camera's rotation in the x, y, z axes, and for each axis, it is possible to set the angle from 0 to 360 degrees. The fourth slider sets the zoom value from 25% to 400%.

8.2.9 Information panel

In the information panel, we can find a table that contains information about the currently occurring objects in the main panel. In addition to the values that the user enters in the object creation, it contains information about the type of orbit, where the value can be one of three types of orbits according to altitude:

- LEO
- MEO
- GEO

There is a text field in the right part of the panel for entering the name of the object we want to find in the table. If any line is selected, six orbit elements will be displayed in the main panel. We can use the `delete` button to delete the currently selected row in the table.

Chapter 9
Implementation

9.1 Languages, Frameworks and Libraries

- **Java** a programming language in which the application is written. Java 11 was used to create the application.
- **Maven** a build tool for application management, control, and automation.
- **JavaFX** a modern framework for creating desktop applications. Provides a wide range of controls, layouts, and support for 2D and 3D graphics.
- **ORM** a way to ensure data conversion between an object in an object-oriented programming language and a relational database.
- **JPA** represents the interface for ORM. It simplifies the storage of entities in the database and their retrieving.
- **Hibernate** is an implementation of JPA.
- **MySQL** a relational database for persistent data storage.
- **FXyz** a library that expands the basic offer of 3D shapes in JavaFX with other shapes such as cone, torus, etc. It also allows the user to draw a line in 3D. The library was used to draw the trajectory along which objects move in the application[42].
- **InteractiveMesh** a library that allows the user to load 3D models from various formats such as `stl.` `3ds.` etc. The library was used to load models from `stl.` files[43].

9.2 Design

There are many architectural patterns for GUI application design. Model-View-Controller (MVC), Model-View-Presenter (MVP), and Model-View-Model (MVVM) are the best known. All three patterns separate the user interface from the data. Choosing the most suitable pattern is not easy because everyone has their advantages and disadvantages. When choosing a specific pattern, we must consider several factors such as performance, testability, maintainability, adherence to the single responsibility principle, loose coupling, and more. In terms of performance, MVVM is worst due to the overhead of data binding. On the contrary, MVVM is the most suitable for test-driven development. MVVM also dominates in the other mentioned factors. It adheres to the SOLID principle. Loose coupling is between the view and view model and is well maintained because one view/view model does not affect the other if the layout of the controls changes. MVC enables faster development. In MVP, it is possible to swap views and manage a code easily. The MVVM pattern was finally selected for the application. MVVM is an architectural pattern that separates data, application state, and user interface.

MVVM consists of three layers:

- **Model** - describes the data it works with. The model solves data storage in the database, business logic, parsing, and data validation.
- **View** - represents the user interface. This is the display of the window and the components located in the window. Its task is to display data to the user and take user input.
- **ViewModel** - connects Model and View and holds the application state. The controls are connected to the ViewModel using a binding and take content from it.

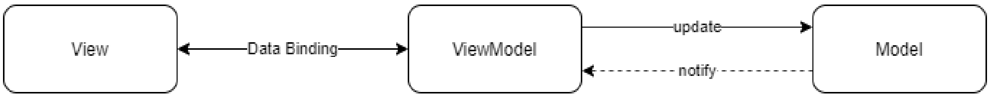


Figure 9.1. Model-View-ViewModel

JavaFX supports application design using MVVM. JavaFX brings the ability to create FXML files. FXML is XML, where it is possible to define the application's appearance separately from the rest of the code. JavaFX also supports data binding and provides FXCollections. FXCollections are collections for listening and tracking changes as they occur.

9.3 Application structure

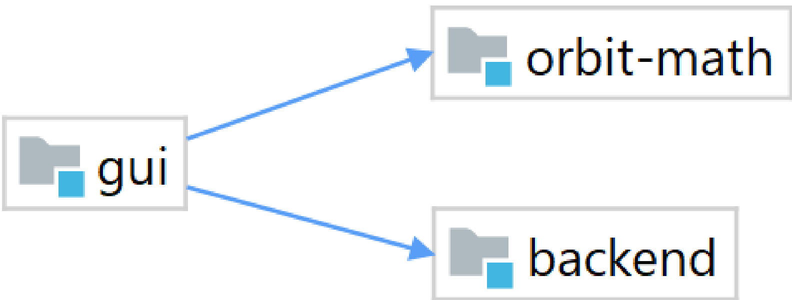


Figure 9.2. Application modules

The application contains the following modules:

- `backend`
- `gui`
- `orbit-math`

The `backend` module is responsible for the model part in MVVM. The module contains DAO, service and facade layers.

The `gui` module is responsible for the view and viewmodel parts in MVVM. In addition, the module contains configuration files to:

- run application (`config` package),

- connect to the database (`META-INF/persistence.xml`),

and other files needed to run the application:

- FXML files,
- `stl.` files for model visualization,
- texture for the Earth[44],
- png files for displaying images in comboBox and tableView

The `orbit-math` module contains the implemented mathematical formulas described in Chapters 5 and 6. The module provides five interfaces. `ElipticOrbit` and `CircularOrbit` are responsible for calculating a period, apogee, perigee, velocity, and for obtaining coordinates to visualize the orbit and position of the satellite. `OrbitManipulation` provides an interface for calculating orbital change. `CoplanarRendezvous` is an interface for calculating rendezvous. `LaserMethod` is an interface for calculating the properties of the laser method based on data entered by the user.

9.4 Functional and Non-functional requirements

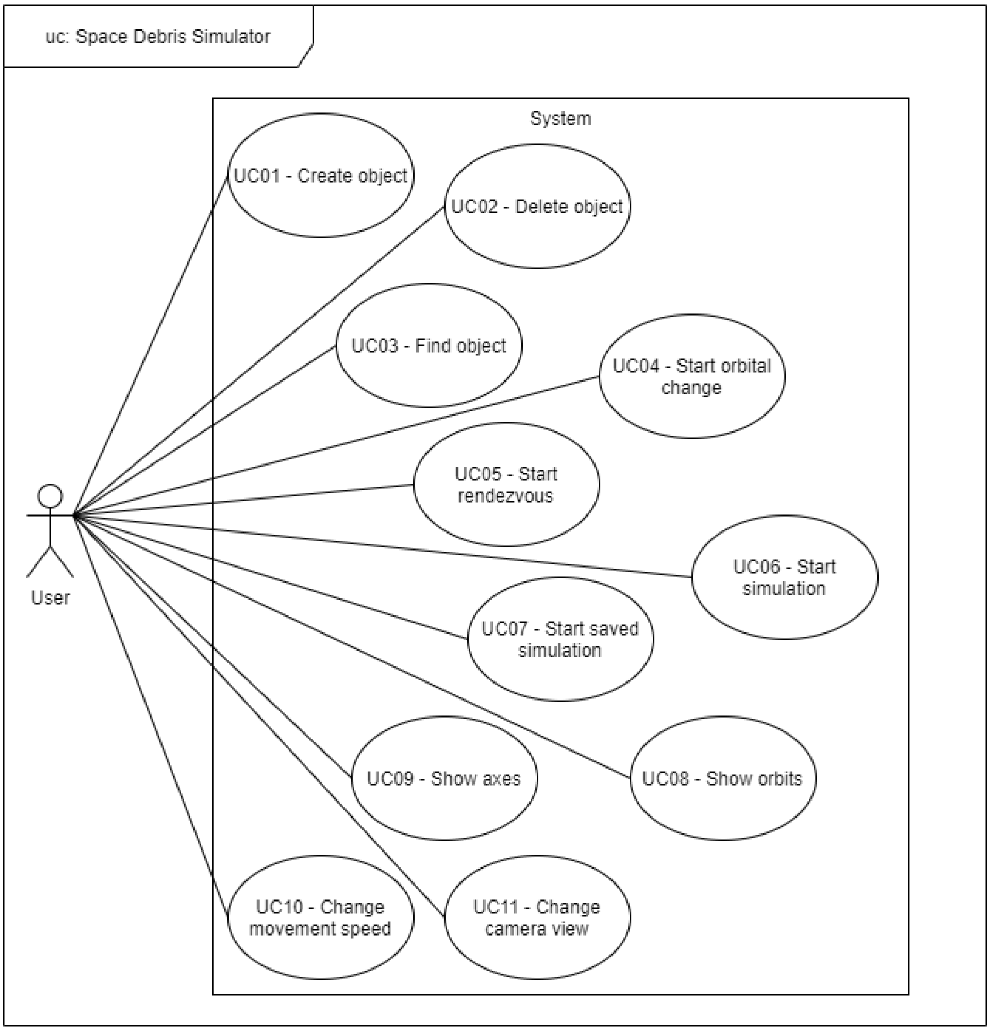


Figure 9.3. Use case diagram

List of functional system requirements:

- F-01 The system allows the creation of an object separately for satellites and separately for fragments,
- F-02 The user can delete any currently occurring object in orbit,
- F-03 The user can search for a specific object among the current objects in orbit,
- F-04 The system allows the simulation of orbital change,
- F-05 The system allows rendezvous simulation,
- F-06 The system allows to perform a deorbitation simulation,
- F-07 The user can view the reference (auxiliary) axes,
- F-08 The user can view the orbit of a specific object or all orbits,
- F-09 The user can change the speed of movement of objects,
- F-10 The user can change the view of the camera,
- F-11 The user can zoom in and out of objects,
- F-12 The system allows the display of current objects in orbit,
- F-13 The system allows the display of charts for orbit pollution.

List of non-functional system requirements:

- N-01 The system will use a persistent database,
- N-02 The system will run in a Java virtual machine (JVM).

Closely related to the functional requirements are the use cases shown in the figure 9.3.

9.5 Logical view

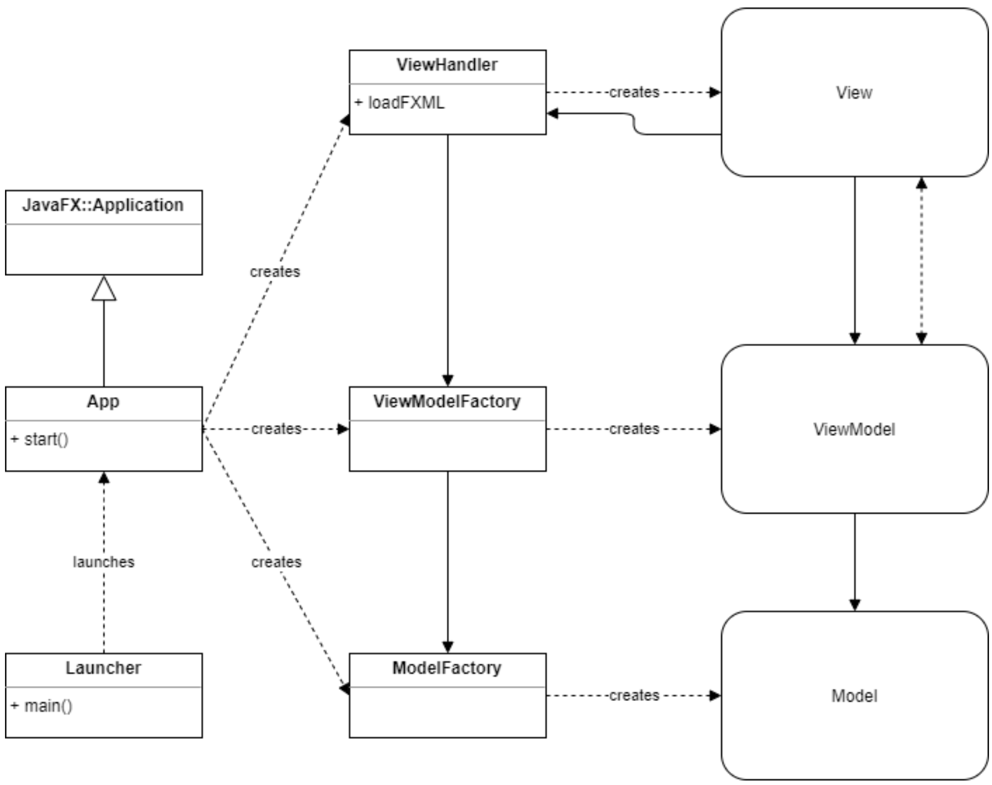


Figure 9.4. Class structure

Figure 9.4 shows a class structure that consists of classes:

- `Launcher` - launches the application
- `App` - creates `ViewHandler`, `ViewModelFactory` and `ModelFactory`
- `ViewHandler` - loads all FXML files and creates ViewControllers.
- `ViewModelFactory` - uses the factory method pattern to create all view-models. At the same time, it registers view-models in `ViewModelMediator` and attaches observers to facades.
- `ModelFactory` - uses the factory method pattern to instantiate all the objects needed for the Model. Creates instances of `EntityManager`, all DAOs, services, and facades.

The internal connectivity in View, ViewModel, and Model can be seen in the component diagram 9.6 in the following section 9.6.

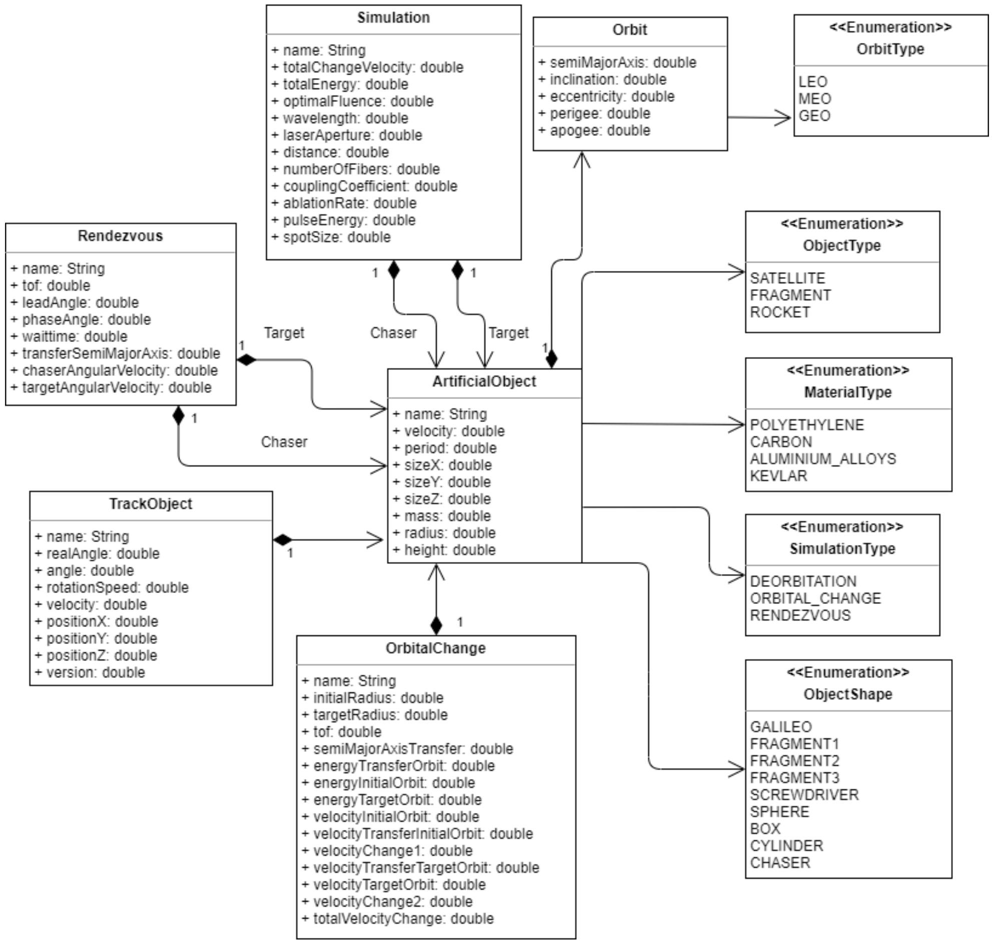


Figure 9.5. Class diagram

Figure 9.5 shows a class diagram, which consists of classes:

- `ArtificialObject` - contains information about the object (satellite, fragment) such as its velocity, mass, period and size,
- `Orbit` - contains information about the size, shape, and inclination of the orbit.
- `OrbitalChange` - collects orbital change information that was calculated before the orbital change,
- `Rendezvous` - contains the information needed to bring the chaser satellite closer to the target object,

- `Simulation` - contains information related to fragment deorbitation as user-entered parameters such as optimalFluence, wavelength, laserAperture, distance, coupling-Coefficient, ablationRate,
- `TrackObject` - is a snapshot before performing a deorbitation, orbital change, or rendezvous. `TrackObject` contains the current position of the object, angle, and velocity.

9.6 Development view

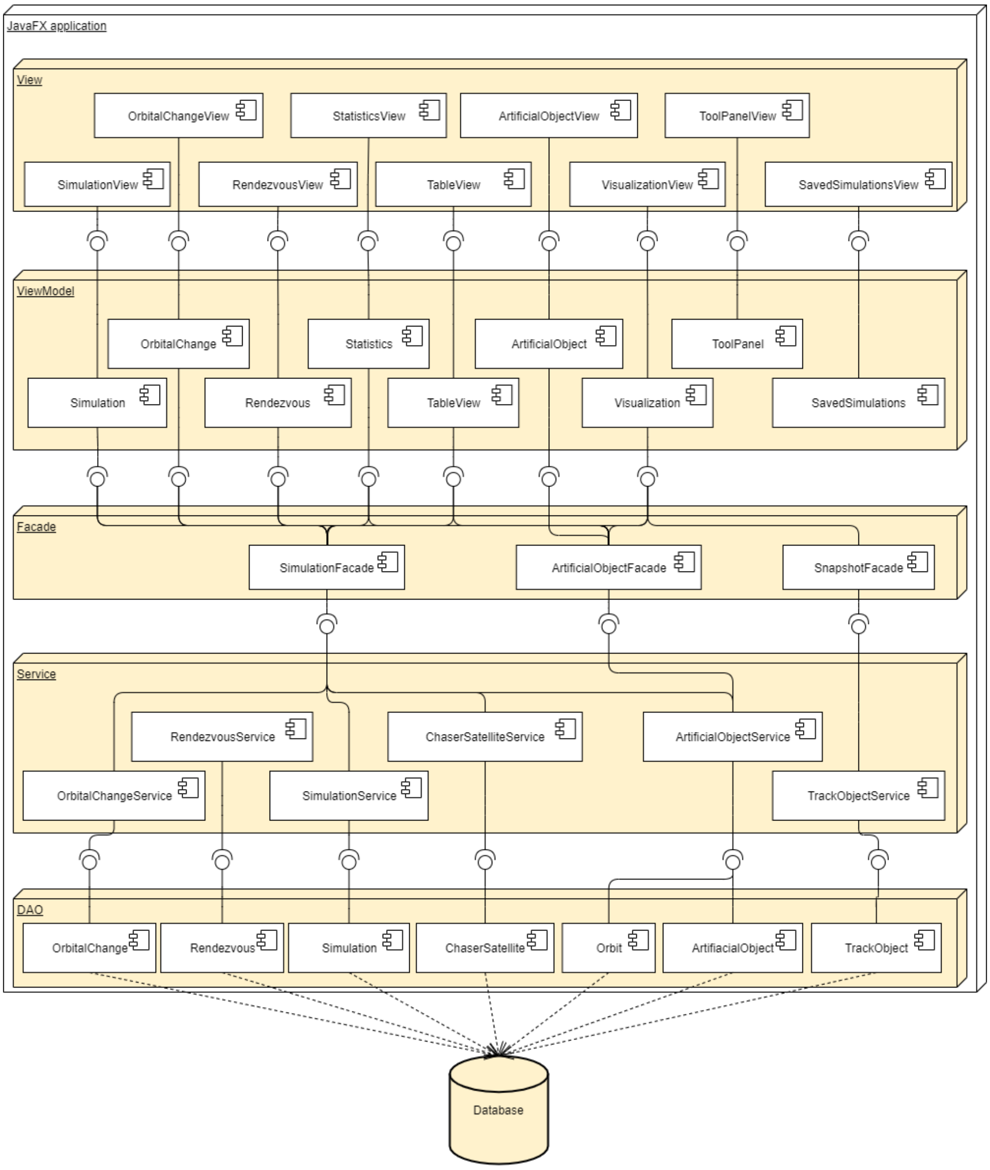


Figure 9.6. Component diagram

Figure 9.6 shows the component diagram with the individual application layers and components. The first layer of the application is the DAO layer, whose task is to access the data stored in the database. It also stores data in a database. Service is the second layer, which serves to mediate the communication between DAO and the facade. The layer serves the information for storing and retrieving data. The next layer is the facade. The task of the facade is to provide an interface for ViewModel for data manipulation.

The layer contains three facades:

- `ArtificialObjectFacade` - used to store, update, search and delete objects (satellites or fragments),
- `SimulationFacade` - used to store and access individual simulations (deorbitation, orbital change, rendezvous),
- `SnapshotFacade` - provides an interface for creating a snapshot that is created when the simulation is run.

Each facade uses an **observer pattern** to notify of changes, for example, saving an `ArtificialObject`, where some view models are notified to update their list of objects. The DAO, service, and facade layers represent the model part in MVVM.

ViewModel is the next layer in the diagram component. The view model layer does not work directly with the domain model but uses a transfer object (classes with the suffix TO). A **mediator pattern** is used for mutual communication between view models.

The last layer is the view. Each controller has its own FXML file, which prescribes the application's appearance and, at the same time, separates the appearance from the code. The task of controllers is to listen to and respond to user actions and display data, which it obtains from the view model layer thanks to data binding. It is worth mentioning the `VisualizationViewController`, which creates an `AnimationTimer` when creating an object. `AnimationTimer` triggers an implemented handle method for each frame. JavaFX tries to adhere to 60 fps, but this is not guaranteed. `AnimationTimer` is used to visualize the motion of objects and to rotate the Earth.

9.7 Process view

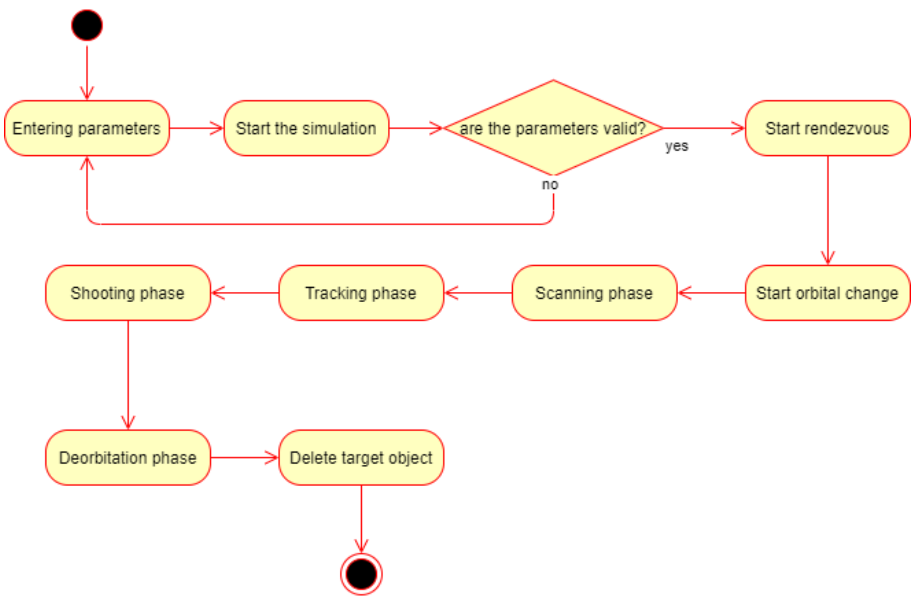


Figure 9.7. Deorbitation simulation activity diagram

The figure 9.7 shows the activity diagram for the deorbitation simulation. The individual activities correspond to the description in Section 8.2.3.

9.8 Physical view

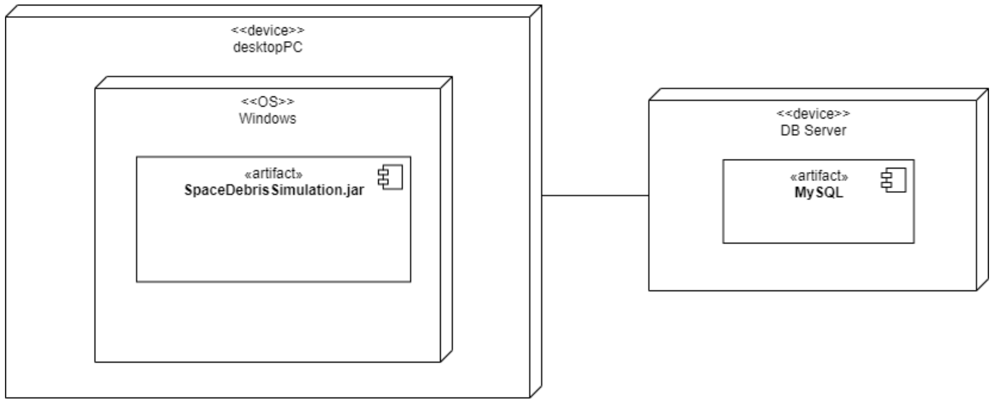


Figure 9.8. Deployment diagram

9.9 Testing

Tests are written in Groovy. Groovy is an object-oriented programming language designed for the Java platform. Groovy can run Java files. It has a simpler syntax than Java. The Spock framework was used for unit and integration tests. The great advantage of the Spock framework is that the tests are easy to read. Separates part of data preparation, execution, and assertion. The in-memory database H2 was used for integration tests.

Chapter 10
Installation

10.1 User guide

Running the application requires having Java and MySQL Database Server installed. The following sections in this chapter describe the individual setup steps.

10.1.1 Java setup

Setting up Java consists of these steps:

1. Download Java 11 from the official oracle website `https://www.oracle.com/java/technologies/javase-jdk11-downloads.html`. Select the appropriate file according to the platform you are using,
2. Install Java 11,
3. Set `JAVA_HOME` environment variable.

10.1.2 MySQL setup

Setting up a MySQL database server consists of these steps:

1. Download MySQL Installer from the official MySQL website `https://dev.mysql.com/downloads/mysql/`. Select the operating system on which you want to install MySQL and select the type of file you want to download,
2. Install MySQL server,
3. Create user:
   - login name: `root`
   - password: `root`
4. Create a database using the following script:

```
CREATE DATABASE simulation;
```

10.1.3 Run application

The following steps are required to start the application:

1. Insert a CD into the drive,
2. Unzip the Application.zip archive to your disk,
3. Run `SpaceDebrisSimulator.jar` (It may take approximately 10-15 seconds to load the application).

Note: The `SpaceDebrisSimulation.jar` file has been tested on Windows 10.

10.2 Developer guide

10.2.1 JavaFX setup

1. Download SDK according to your platform from the official JavaFX website `https://gluonhq.com/products/javafx/`. The application uses JavaFX 16. In case of downloading another version, it is necessary to change the version in maven `pom.xml` in the `gui` module,
2. To run the main class `App.java`, it is necessary to set the path to the JavaFX lib file. In IntelliJ IDEA, it is possible to do in `Run -> Edit Configurations -> VM options` where you enter the following text with your path to the downloaded JavaFX SDK.

```
--module-path
path-to-your-javafx-sdk\javafx-sdk-16\lib
--add-modules
javafx.controls,javafx.fxml
```

10.2.2 Database setup

You can change the connection settings to the database server in the `resource/META-INF/persistence.xml` file located in the gui module.

Chapter 11
Conclusion

The aim of the master's thesis was to get acquainted with the issue of space debris, find out the possibilities and methods of how space debris can be removed, choose a suitable method for subsequent simulation, and implement a desktop application.

The thesis is divided into three parts. The first part explains the reasons for the generation of debris, its risk, but also the prevention of its generation and disposal methods. Sixteen ADR methods have been described. For each method, the principle of removal is explained. At the same time, each method contains two tables, the first of which contains a description of the advantages and disadvantages, and the second one describes the technical parameters of the method. The methods were divided into four groups according to their common properties: drag augmentation systems, contactless methods, tether-based methods, and contact removal methods. The last chapter of the first part describes the mutual comparison of methods from different perspectives, namely, TRL, price, suitability for the orbit, the duration of removal of one debris, and the advantages and disadvantages. Based on the comparison, it is impossible to choose only one method, which would be universal and remove all debris in a relatively shorter unit of time, without the risk of additional debris and with the lowest possible cost. The laser method was chosen for the simulation due to the possibility of using this method in the Czech Republic.

The second part describes the space laser method in more detail. Describes the suitability of using a laser on different sizes and masses of the target object.The laser method considers common types of materials used in fragments, such as aluminium or carbon. The next chapter of the second part deals with the description of orbital elements, plotting of trajectories, and movement of satellites. Subsequently, the findings were taken into account in the design of the simulation.

An application for the simulation of the laser method was designed. The application was developed in Java using the JavaFX framework, which was used to visualize 3D objects and components. The MVVM architectural pattern was used for the GUI design, allowing easy addition of other components without affecting existing ones. The user can add satellites, fragments according to the modeled models or existing 3D objects that are part of the JavaFX framework in the application. Based on the second part, the application offers three types of simulations: deorbitation, orbital change, and rendezvous. Each type of simulation offers the ability to set simulation parameters, see related calculations, and record a simulation that can be replayed.

11.1 Future work

The work can be further expanded, and other constructs related to space debris can be added. Possible extensions include:

- Implementing of other methods that have been described in this work.
- Implementing a laser ground-based method for comparison with a satellite-based method.

- Improving the behavior of the target object for the possibility of testing the disposal of debris under different conditions. An example is tumbling.
- Connecting the application with an existing database of artificial satellites and objects.
- Implementing coorbital change and rendezvous.
- Improving the graphical user interface.
- Exporting data to CSV format.
- Possibility to add user models.
- Optimizing the number of queries to the database and use the cache instead.

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Appendix A
Symbols

ADR	Active Debris Removal
DAO	Data access object
DAS	Drag Augmentation system
EDT	Electrodynamic Tether
ET	Electrostatic Tractor
GEO	Geostationary Earth Orbit
GTO	Geostationary Transfer Orbit
GUI	Graphical User Interface
IBS	Ion Beam Shepherd
JPA	Java Persistence API
ISS	International Space Station
LEO	Low Earth Orbit
LODR	Laser Orbital Debris Removal
MVC	Model-view-controller
MVP	Model-view-presenter
MEO	Medium Earth Orbit
MVVM	Model-view-viewmodel
ORM	Object-Relational Mapping
RAAN	Right ascension of the ascending mode
TO	Target Object
TRL	Technology Readiness Level

Appendix B
Nomenclature

C_m	Mechanical Coupling Coefficient
m_{debris}	Mass of debris
E	Laser pulse energy
Δv	Change of velocity
μ	Ablation rate
a	Semi-Major axis
b	Semi-Minor axis
c	Distance between the focus and the centre of the ellipse
e	Eccentricity
i	Inclination
Ω	Right ascension of the ascending node (RAAN)
ω	Argument of perigee
ν	True anomaly
A	Apogee
P	Perigee
u	Argument of latitude
Π	Longitude of the perigee
l	True longitude
t	Eccentric anomaly
G	Gravitational constant
M_{earth}	Mass of the Earth
R	Radius
T	Period of the satellite
ε	Mechanical energy of the satellite
TOF	Time of Flight
$a_{transfer}$	Semi-Major axis of transfer orbit
ω	Angular velocity
μ	Gravitational parameter
α_{lead}	Lead angle
ϕ_{final}	Phase angle

Appendix C
Content of attached CD

The content of the CD is organized into the following files and directories:

- `masters_thesis` - the directory containing the master's thesis in pdf format, source tex files, and images.
    - `masters_thesis.pdf` - master's thesis in pdf format
    - `tex` - directory of tex files
    - `teximages` - images used in the master's thesis
- `application.zip` - application source files
- `SpaceDebrisSimulation.jar` - jar archive to run the application