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Mission and System Architecture Design for Active Removal of Rocket Bodies in Low Earth Orbit

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Abstract

With the increasing risk caused by the population of Space junk in LEO and especially in SSO, an efficient active debris removal program must be developed. It has to take into account not only the technical challenges but also the econo-political ones implied by this complex task. This thesis presents three different ways to de-orbit the population of rocket bodies in SSO. The first one, the Picker, implies to send one spacecraft to remove one debris at a time. The second one, the Mothership, has to visit several debris to equip them with autonomous de-orbiting units. The last one, the Shuttle, travel back and forth between high and low orbits to catch the debris and bring them on short-lifetime graveyard orbits. The three scenarios are optimized in terms of number of debris per spacecraft, number of spacecraft per launch vehicle and propulsion system (chemical solid, liquid bi-propellant or electric). They are compared in terms of cost per mass removed. The results shown that the Picker is approximately ten times more efficient than the other ones, with normalized cost as low as 10,500 [$/kg]. In a second time, the comparison is made between a collaborative global program between the U.S., Russia and Europe and their three separated national program. The conclusion is that choosing the collaboration rather than the isolated program allows each government to save millions of dollars. A proposition of tax per launch is suggested to finance the ADR program.

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1 Introduction

1.1 Motivation

In 1957, the very first human-made satellite, Sputnik 1, is placed in orbit. Less than 60 years later, thousands of other payloads have been launched. Most of them are now inactive, drifting in Space and causing risk to the functional systems nearby. Indeed, due to relative velocities that can be as high as 15 [km/s], even the smallest object represents a huge amount of kinetic energy. It can thus be fatal to a working spacecraft in the case of a collision. The impact, in 2009, between a functioning american satellite, Irridium-33, and a decommissioned russian system, Cosmos-2251, confirmed the increasing risk caused by orbital debris. The most dramatic part of this incident is less the loss of a functioning payload than the creation of many small debris, which all represent a threat.

A spacecraft can be shielded against debris smaller than 1 [mm] without any critical increase of mass and cost. Objects bigger than 10 [cm] can be tracked from the ground and threatened spacecraft's operators can be warned and, if needed, move their payload to avoid any potential collision. These maneuvers cost money and propellant but allow to keep the satellite operational. Any object bigger than 1 [mm] and smaller than 10 [cm] is extremely dangerous: its energy is sufficient to damage or even destroy a satellite in the case of a collision and it cannot be tracked from the ground. The risk of collision and loss of payload increases with the number of these debris and hence they should be remove to ensure the safety of the orbits around Earth.

However, because they are so hard to find and because a feasible solution to capture them has yet to be developed, the general tendency focuses on avoiding that more of them are created. That is why it is important to remove the bigger debris as they represent huge reservoirs of potential smaller, invisible, high-energy objects. Among these big debris are the rocket bodies left in space after they delivered their payload. Their masses go from tens of kilograms to several tons and they obviously are in regions occupied by numerous working and decommissioned systems, where the probability of collision is high. One of these regions is the Sun-synchronous orbits (SSO), which are extremely useful for many types of satellites, commercial, governmental or scientific, from all the major Space countries.

Removing these objects is necessary to keep the near-Earth environment safe for new satellites and human exploration. However, the cost induced by such operations can be very high and, since the program is not expected to provide any type of return on investments except the increased safety, it is unlikely to have anyone willing to pay for it. It is therefore very important to define a fair methods for the funding of the program to take into account the responsibility of every stakeholder.

The goal of this Master thesis is to design spacecraft and plan missions to remove rocket bodies from SSO. They have to be cost-efficient and take into account the problems caused by the very international use of these regions of Space. The final output of the research is a proposition if funding plan that is as equitable as possible for the involved partners.

1.2 Research approach

The research for active debris removal is already advanced: subsystems are being developed for remote attitude determination, capture of non-compliant object, selection of the debris to be removed in priority, etc. This project assumes that the different technologies to reach, capture and de-orbit the debris will be ready to use in a near future and explores the next challenges. They include technical aspects such as the general characteristics of a removing spacecraft, the number of debris it can take care of, the number of spacecraft that are needed to de-orbit a given population of debris, the number and the type of launch vehicle required and the type of propulsion for the orbital transfers.

Once the technical aspect of the spacecraft are known, the cost of each architecture is estimated thanks to basic cost model at the subsystem level. This economic study allows to compare the performance of the different architectures for a given population of debris. The best system is the one able to remove them at a minimum cost per kilogram of mass removed. An additional tool used for the selection is the removing cadence, defined as the number of debris removed per year. NASA's recommendation of 5 major debris removed per year is used as a guideline rather than as a strict constraint.

Finally, political aspects are explored. Once the cost per debris and for the whole campaign is estimated, its distribution among the concerned stakeholders of this problem must be defined. Either each one works on its own program or a global collaborative program is developed. The goal is to have every involved partner paying a fair contribution to help solving a problem that can badly affect the use of Space in the future.

1.3 Structure of this thesis

Chapter 2 goes through the actual state of the research in the field of active debris removal. The actual orbital debris situation is presented along with the measures taken and the recommendations suggested by the different agencies for the observation and mitigation of the objects in Space. The different topics that have been or are explored are also reviewed. They include general mission planning, specific technological developments as well as the design of more advanced systems. Finally, launched demonstrators and end-to-end studies are presented. The gap analysis between the actual state of the research and this thesis is recalled.

Chapter 3 presents the targets that are studied in this project. Their orbital parameters and physical properties are shown to justify some of the assumptions and early choices that are made. The main areas of interests around Earth are also identified.

Chapter 4 explains the construction of the technological and economical model. It shows the general spacecraft design loop, the selection of the launch vehicle, the sizing of the propulsion and power subsystems and the cost models that are used. Three different high-level scenarios are presented: the Picker, the Mothership and the Shuttle.

Chapter 5 gives two different validations of the Model. Firstly, it is compared to a very generic and widely used cost model developed by NASA and, then, to an in-depth study of a very specific architecture designed at the University of Cranfield. This step allows to gain confidence in the results given by the Model and ensure that the design and the cost evaluation are relevant.

Chapter 6 presents the results of the technological study and the cost that are implied by each architecture. The three scenarios are studied separately, the Picker and the Mothership are detailed and the reasons why the Shuttle is discarded are given. The most efficient architecture in terms of cost and cadence is selected to proceed with the international policy study.

Chapter 7 compares the cost of the separated american, russian and european programs with a global one, in which the three countries collaborate to tackle the problem. The best solution is selected and the planning

of the full program is detailed. A fair distribution of cost between the different stakeholders in this problem is suggested.

Chapter 8 concludes this report with a summary of each part, a reminder of the optimal solution in terms of technology, economy and policy as well as suggestions for future work.

The general roadmap of the thesis is given in Figure 1.1.

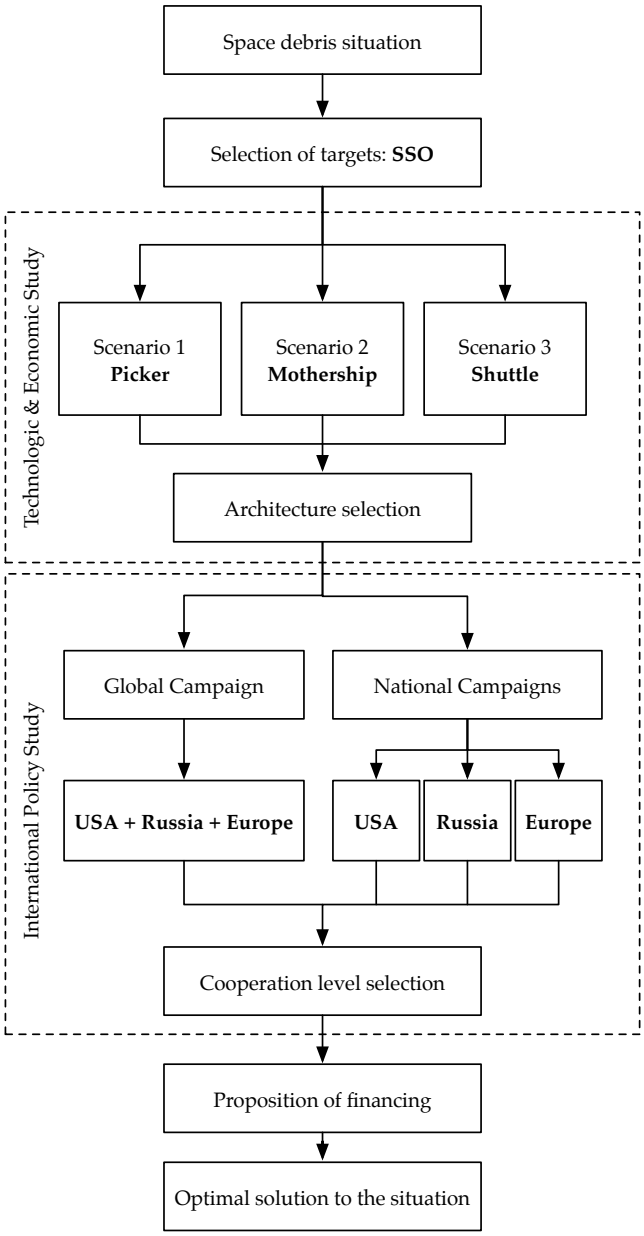

Figure 1.1: Thesis roadmap

All along the report, recommendations to improve the study or to use different technologies than the ones presented in the scope of this study are given. One goal of this research is to serve as a stepping stone in

terms of campaign planning and international collaboration to solve the orbital debris problem.

1.4 Terms and abbreviations

1.4.1 Mission, campaign and program

Three different terms are used to classify the activities needed to complete the full de-orbiting of a given population.

A mission represents all the tasks a single spacecraft has to perform. The mission of a Mothership spacecraft is to move from its injection orbit to the orbit of its first target, to rendezvous with it, to capture it, to equipped it with a Pilot Fish spacecraft and to move to the next debris. It mission ends when it reaches its final orbit with its final target.

A campaign is defined by all the activities from the first launch to the de-orbiting of the very last target of the selected population. The duration of the campaign is estimated as the time between the first and the last launch added to the average or maximum mission duration.

A program includes all the efforts required to remove a given population of debris. It basically consists in the campaign, the development process and the manufacturing of the functional spacecraft. The later overlap the development and campaign phases.

Figure 1.2 gives an example of a program constituted by a development phase and a campaign including 2 launches of 3 spacecraft each.

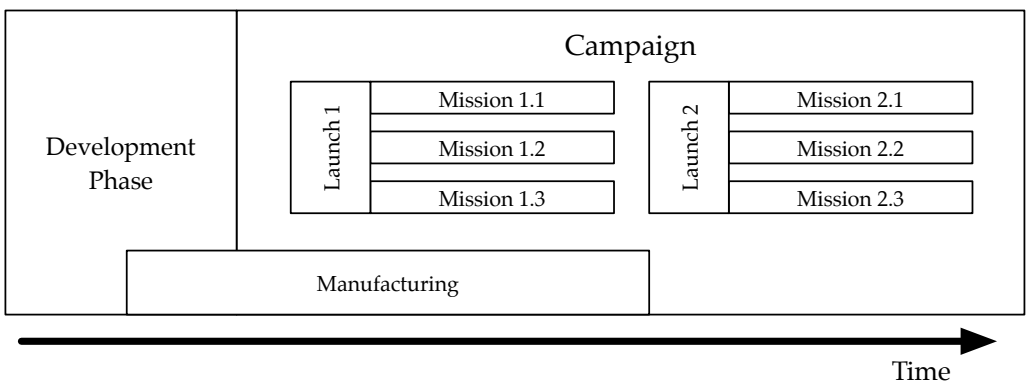

Figure 1.2: Program, campaign and mission definition

In this example, if the development phase lasts T_D, the time between both launches is T_L and the average mission duration is T_M, the campaign duration is $T_C = T_L + T_M$ and the program duration is $T_P = T_D + T_C$.

1.4.2 Debris and target

A debris is defined as any non-operational man-made system in orbit around Earth. In this thesis, they mostly refer to rocket bodies but the term is also used in its more generic sense. The debris are selected depending on different parameters that are specified later in this report. These parameters can be specific ranges of orbits, of physical properties, a limitation to certain countries, etc.

Once the debris is assigned to a campaign or, more specifically, to a mission, it is called a target. As explained earlier, a campaign is not complete before the very last target has been taking care of by one of the removing spacecraft.

1.4.3 Abbreviations

ADC	Attitude Determination and Control
ADCS	Attitude Determination and Control Subsystem
ADR	Active Debris Removal
ASI	Italian Space Agency
ASTRO	Autonomous Space Transport Robotic Operations
ATV	Automated Transfer Vehicle
BOL	Beginning Of Life
DARPA	Defense Advanced Research Projects Agency
DART	Demonstration of Autonomous Rendezvous Technology
DEOS	Deutsch Orbit Servicing
DLR	German Space Center
DR LEO	Debris Removal in Low Earth Orbit
EDT	Electro-Dynamic Tether
EOL	End Of Life
EPS	Electrical Power Subsystem
ESA	European Space Agency
ETS	Engineering Test Satellite
FY	Fiscal Year
GEO	Geo-stationary Orbit
GPS	Global Positioning System
GSO	Geo-Synchronous Orbit
GTO	Geo-synchronous Transfer Orbit
HTV	H-II Transfer Vehicle
IA&T	Integration, Assembly and Test
ID	Identification
Isp	Specific Impulse
ISS	International Space Station
JAXA	Japanese Aerospace Exploration Agency
JSC	Johnsson Space Center
LC	Learning Curve
LEO	Low Earth Orbit
L/V	Launch vehicle
MET	Momentum-Exchange Tether
MIT	Massachusetts Institute of Technology
MSSP	Mothership Spacecraft
NORAD	North American Aerospace Defense Command
NASA	National Aeronautics and Space Administration
PDR	Preliminary Design Review
PFSC	Pilot Fish Spacecraft
P/L	Payload
RAAN	Right Ascension of the Ascending Node
R/B	Rocket body
RDT&E	Research Development Test and Evaluation
S/C	Spacecraft
SDMR	Space Debris Micro Remover
SSN	Space Surveillance Network
SSO	Sun-Synchronous Orbit
SVLCM	Spacecraft/Vehicle Level Cost Model
TFU	Theoretical First Unit
TLE	Two-Line Elements
TRL	Technology Readiness Level
TSP	Travelling Salesman Problem
TTC&DH	Telemetry, Telecommand and Data Handling
USSR	Union of Soviet Socialist Republics

2 Litterature review

2.1 Space debris situation

With more than 50 years of Space activity, the situation in LEO is becoming very critical and is even already unstable according to recent reports [1] [2]. The actual measures taken by the different Space agencies are not sufficient to reduce the risk caused by the orbital junk. For example, many are the ones who think that lowering a satellite to a 25-year lifetime orbit at its EOL is not sufficient and even dangerous[3]. The situation is going to worsen as more systems are launched every year. However, these same agencies are aware of this problem and are working on their side and together to find solutions.

The U.S., thanks to the Space Surveillance Network (SSN), are able to track any object between LEO and GSO that is bigger than 10 [cm] [3]. This allows several agencies and research groups to observe and model the evolution of the debris around Earth and measures can be taken for collision avoidance [4][5]. However, much smaller debris also exist and cannot be seen by the SSN. If a pretty simple shielding can be sufficient against debris in the order of 1 [mm], every debris bigger than that and smaller than 10 [cm] (so invisible to the SSN) can be very dangerous for any spacecraft on a close orbit. According to the recent DARPA's Catcher's Mitt study [2], no valid system seems to be able to deal with these debris. However, while it is important to think about these smaller debris, the priority must be given to the bigger ones that have a bigger probability of collision and that are more likely to create multiple new debris. This was the case with several upper stages failures and more recently with the anti-sat missile launch by China in 2007 and the collision between Irridium-33 and Cosmos-2251 in 2009 [1]. Figure 2.1 shows the evolution of the trackable debris population since the beginning of Space exploration.

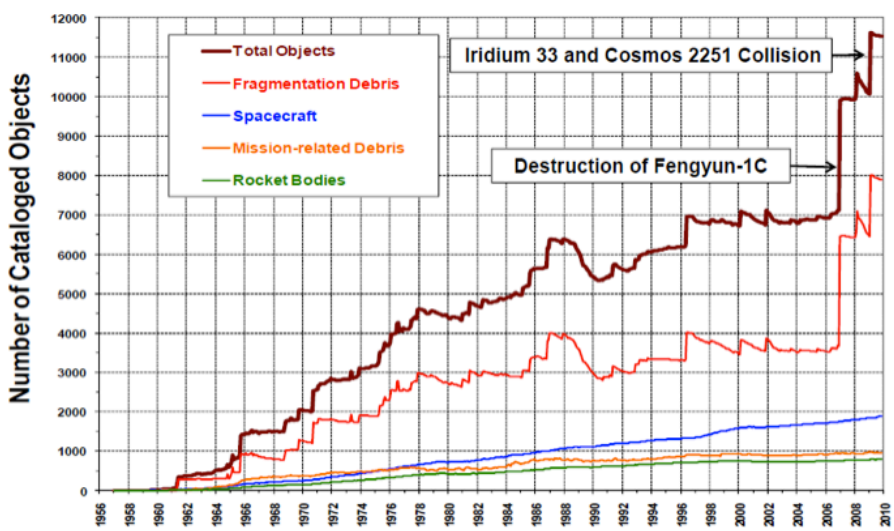

Figure 2.1: Evolution of the debris population [1]

According to NASA [1], the only way to stabilize LEO and make it a safer place is by removing at least 5 relevant debris per year, starting in 2020. However this is only in a very optimistic scenario where no new system is launched to Space. In reality, the number of debris that has to be removed per year can be as high as 15 depending on the studies.

2.2 Topics of research

2.2.1 General mission design

Although the architecture of the spacecraft can vary a lot depending on the de-orbiting method, the number of debris to capture and the orbits where it operates, the typical ADR mission scenario to remove a debris is usually similar to the one shown in Figure 2.2.

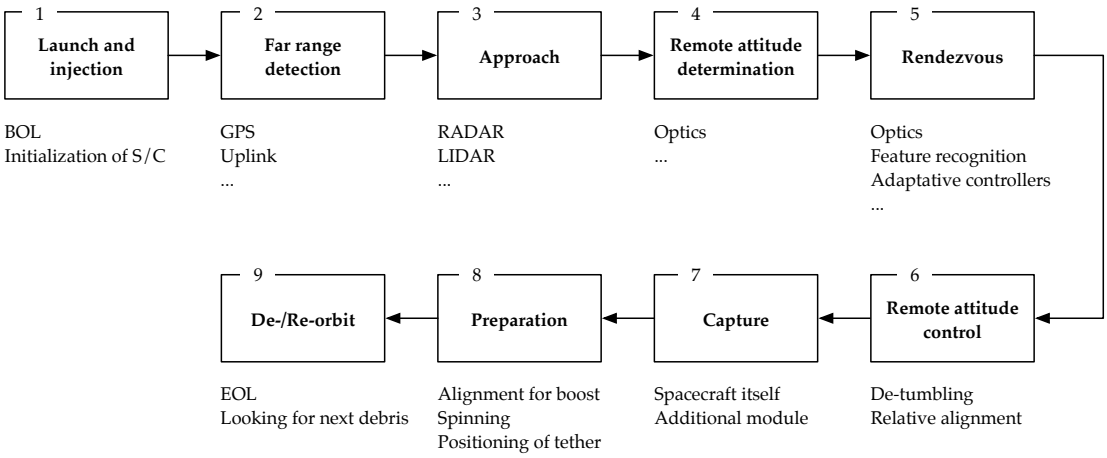

Figure 2.2: Generic ADR mission scenario

The notation below each step gives an idea of either the actions performed by the spacecraft or the sensors and actuators that can be used. Thus, the most active research fields for the Space debris problem are the ones related to these boxes.

2.2.2 Rendezvous techniques

The rendezvous in Space has a very long tradition: from the first Soyuz and Gemini missions to the Progress-M, ATV and HTV that are able to rendezvous and even dock autonomously to the ISS. The main difference between the ISS and a debris is that the first one is cooperative while the second one is only a floating piece of hardware lost in Space. However, conventional methods such as RADAR, LIDAR or similar [6] can be used to approach the debris.

2.2.3 Remote ADC and capture techniques

This is definitely one of the trickiest points of the problem since the debris can have big uncertainties in its position, attitude, mass and inertia. Progress in machine learning and image processing allows to use very simple optical systems to remotely determine the orientation and behavior of a free-moving object [14].

On the grabbing part, the Canadians, with their enormous experience in Space robotics, are working on a system able to deal with such a noncooperative object [13]. The arm needs to have enough compliance to avoid the transmission of torque to the removing spacecraft. JAXA is also working on a similar type of robotic grabber [9].

2.2.4 Removal methods

Big debates are ongoing about what the best de-orbit (or re-orbit) method is. The top 4 candidates commonly seen in the literature are [12] [15]:

1. Chemical propulsion
2. Electric propulsion
3. Electro-dynamic tether (EDT)
4. Momentum-exchange tether (MET)

The first ones have their advantages (very reliable, huge heritage, high thrust capability) but they are also very costly in terms of mass and volume . There are very good options for re-orbiting debris from GEO to a higher graveyard orbit, though [23]. The electric propulsion is already used on many satellites in GEO and present the advantage of having a very high Isp but the values of thrust are usually lower than 1 [N]. The EDT is very appealing due to its propellant-free ability and it seems to be the favorite choice of the researchers for ADR in LEO. However, it is hard to use in SSO due to the weakness of the Earth magnetic field at these high inclinations. The last candidate is less known. This type of tether offers the advantages of a propellant-free system but they lack some development and real heritage to be trusted. The EDT and the MET have a high risk of collision since the tether has to be deployed on several kilometers.

Other options such as LASER or ionic beam have been studied but the development is not as advanced as for the four main methods. The main limitations are the huge power needed as well as the fact that they can be compared to weapons, which are absolutely forbidden in Space. Aerobraking has also been studied: huge inflatable structures are used to increase the drag coefficient of the debris. The main risk with such a system is to have it impacted by multiple debris on its way and be destroyed before the end of the mission.

2.3 Example of launched systems and demonstrators

Several missions have already been launched to test some parts of the systems detailed in the previous section.

ETS-7

The Japanese were the first ones to launch a space robot able to dock with another satellite. The two satellites were launched together and several docking attempts (either remote-controlled or autonomous) have been performed between 1997 and 1999 [17]. The on-board software could be rewritten from the ground and thus several improvement were made within the mission time. The heritage of this mission has been implemented in the HTV.

The DART mishap

In 2005, DARPA launched the DART mission with the goal to perform a rendezvous from long distance (unlike ETS-7) with an already-in-orbit communication satellites. Due to bad software development, the DART spacecraft consumed more propellant than expected during the orbital transfer, misestimated the distance with the target and eventually impacted it. It decommissioned itself after only a couple of hours [18]. Although the mission can be considered as a failure, very precious lessons were learned from it in both domains of autonomous operations and space software development.

Orbital Express

In 2007, the Boeing company launched a demonstrator for autonomous rendezvous and capture. Two spacecraft were launched together and then separated. The ASTRO satellite performed an autonomous approach to the NextSat target and captured it with its robotic arm [19]. This was a huge achievement for the research on both ADR and On-Orbit Servicing.

2.4 Example of studied architectures

Several design solutions has been proposed since 2009 and the conference organized by NASA Johnson Space Center and DARPA about the Space debris issue [3]. Only a few of them are presented here as they are the most advanced.

DLR's DEOS

DEOS is the german version of Orbital Express, although it is a bit more than a demonstrator. It is now in Phase B (design phase before PDR) [16].

JAXA's SDMR

The japanese space agency is working on a small satellite able to de-orbit multiple debris. The satellite can perform orbital changes and rendezvous with a debris, attach an EDT module to it by the use of a compliant robotic arm, separate from the arm and go to the next debris [9]. Several hardware demonstrators have been built to verify the feasibility of the system.

MIT's Space Tug

In 2004, MIT was involved in the design of a family of spacecraft able to perform on-orbit servicing in both LEO and GEO. The new approach that was taken was to aim for a feasible and cost-effective design. [11] The present thesis will follow this path and adapt it to the ADR problem.

Cranfield University's DR LEO

As part of a class in aerospace engineering (2009-2010), students of Cranfield University in the U.K. designed a satellite able to collect 5 rocket upper-stages and de-orbit them by the use of conventional rocket engine. A risk analysis and a basic cost estimation were also conducted with a result of around $ 40,000 per kilogram of removed mass [20]. They defined their study as a benchmark for further research and this is exactly how this thesis is going to use their work.

ASI's Concept

This concept, proposed in 2011 by the italian Space agency, is about using a spacecraft in a region of Space to de-orbit all the small rocket upper-stages nearby. The spacecraft is equipped with two robotic arms (one to capture the debris and the other one to attach a de-orbiting module), a reserve of de-orbiting modules and a detachable propellant tank. Once its reserves are empty, the spacecraft is visited by a supply spacecraft that changes its tank and renew its stock of modules. The mission scenario is very interesting but no cost or risk analysis is found for this project [10]. It inspired the Mothership presented in the present thesis.

Star Inc.'s EDDE

Star, Inc. is a company with the goal to deliver the first LEO de-orbiter to the market. They use a dozen of EDT-based spacecraft able to travel up and down in LEO to clean the orbits from their debris [22]. Again, the idea sounds good but no accurate cost analysis is provided. Moreover, the technology gap seems to be big since their capture method with the use of a net is still yet to prove, The other very risky part of this project is the fact that multiple spacecraft will slowly cross the ISS orbit as well as many others for a long period of time. The efficiency of their tether system in SSO is also still obscure. However, the mission scenario is worth it and represents a interesting development opportunity for the Shuttle scenario presented in a later chapter.

2.5 Gap analysis

As explained in the previous sections, the research in the Space debris field is very dynamic and a lot of things are happening at all levels. Nevertheless, never an actual system was implemented and launched to start collecting the debris. There are three main reasons for the lack of such a system.

Firstly, the proposed systems are based on technologies that, for most, has yet to be tested if not developed. Since it would be very uncomfortable to have a debris remover crashing into its target and generating more debris, the system has to be even more reliable than any other Space system.

Then, the cost are usually a secondary problem in these types of researches. A space debris remover is very unlikely to be a profitable system. At best, the only profit would be to have safer near-Earth orbits, but nobody seems to want to pay for it. Some companies like RetroSpace are proposing an international "clean Space tax" [21] to sponsor their activities, but even if this fee would exist, the only way for a system to be launched is by keeping the cost as low as possible.

Finally, the level of collaboration is still quite low. Every national Space agency and most universities with a Space are proposing solutions for different aspects but it appears that a central decision-making unit to select the best of all suggestions is missing. Because it is very unlikely that any of these entity would be able to finance a full efficient program, it is important that the efforts are joined to provide a feasible solution, not only in terms of technical challenges but also for what concerns the economical and international constraints posed by this problem.

The next chapters present the high-level tradeoffs made to define a complete architecture able to address this problem in an cost-efficient and politically convenient way.

3 Catalog of targets

3.1 Rocket bodies

A rocket body, in this case, is a part of a launch vehicle that is used to place a payload, such as an Earth-orbiting satellite or a deep-Space probe, on its operational orbit or trajectory. The lower stages of a rocket are usually jettisoned at sub-orbital altitudes, they fall on the ground in a very short time after launch. The upper stages, in the opposite, are the ones used to give the final boost before the payload in separated and starts its operations. After the separation, most injection stages perform simple maneuvers to go away from the payload to avoid endangering it. However, it does not mean that they are able to place themselves on a full de-orbiting trajectory and most of the time, especially at high altitudes, where the atmospheric density is extremely low, they stay in orbit.

The reasons to treat rocket bodies as high-priority targets are [8]:

- They obviously are debris.
- They have a big probability of collision and can create many debris.
- They are easy to track.
- They have common features (cylindrical body, nozzle) that will simplify the capture process.
- Their movement is easier to predict than the one of a satellite due to their simple geometry.

Moreover, the legal aspect of their removing is usually less complicated than it is for a satellite. Indeed, a satellite is the responsibility not only of its operator but also of its manufacturer and of the country of the launcher. For instance: it is hard to define who has to pay to remove a satellite used by a british telecommunication company, built by a french manufacturer and launched with a russian rocket. For a rocket body, only the country of origin is responsible.

3.2 Sources of information

The catalog of debris that is used for this research is built from TLEs data taken from the website http://www.spacetrack.org. Every unclassified objects observed by NORAD can be downloaded in a conventional text format. The TLEs are then analyzed in Matlab to extract the objects defined as rocket bodies but not as debris, so only the ones that are still intact are taken into account. This extraction gives the names of the objects, the NORAD identifiers and the orbital elements: semi-major axis, inclination, RAAN, eccentricity, argument of perigee and anomaly. The catalog, as built in February 2012, contains 1629 rocket bodies.

3.3 Distribution around Earth

Observing the different characteristics of the targets allows to restrain the research to certain types and to make some assumptions. Figure 3.1 shows the distribution of the altitudes of perigees and eccentricities of all the rocket bodies in orbit around the Earth.

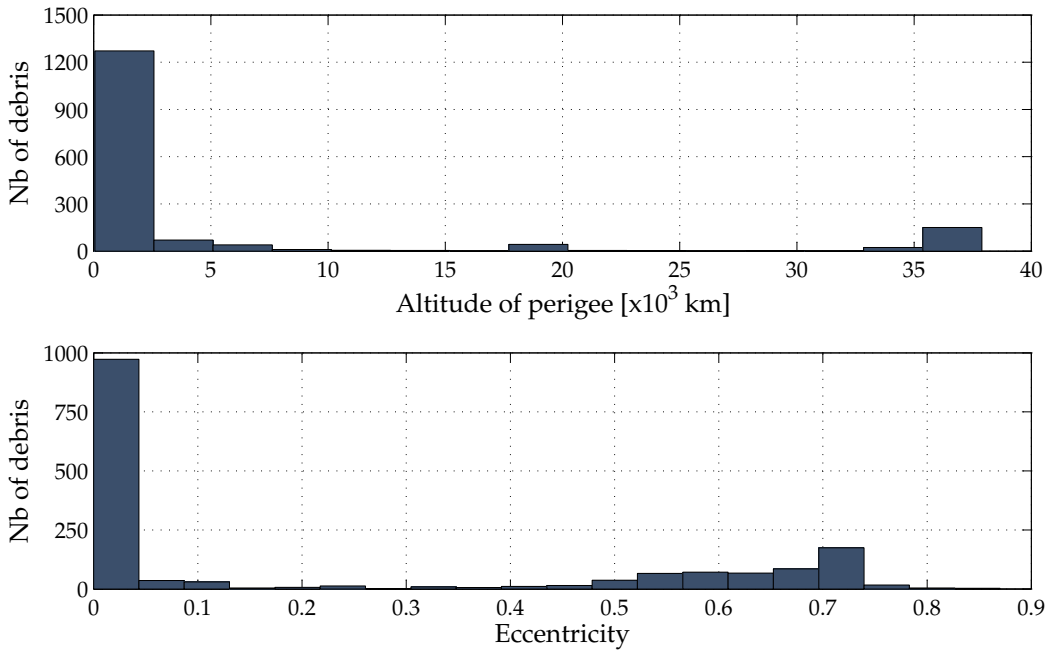

Figure 3.1: Orbital distribution of rocket bodies around Earth

It is clear that most of the upper stages are in LEO with only a few other clusters. The first one, situated around 20,000 [km] above the Earth is due to the injection of the GPS and GLONASS constellations. Another one, just above 35,000 [km], is constituted by the objects in GSO and GEO. Concerning the eccentricity, most orbits are circular. The higher eccentricities are due to rocket bodies on transfer orbits, such as GTO, or on particular orbits, such as Molniya. From these observations, it seems legit to restrain the catalog to object in LEO (altitude of perigee smaller than 2000 [km]) and on circular or quasi-circular orbits (eccentricity smaller than 0.05).

Figure 3.2 shows the distributions of inclinations and RAANs of the 809 targets satisfying these constraints.

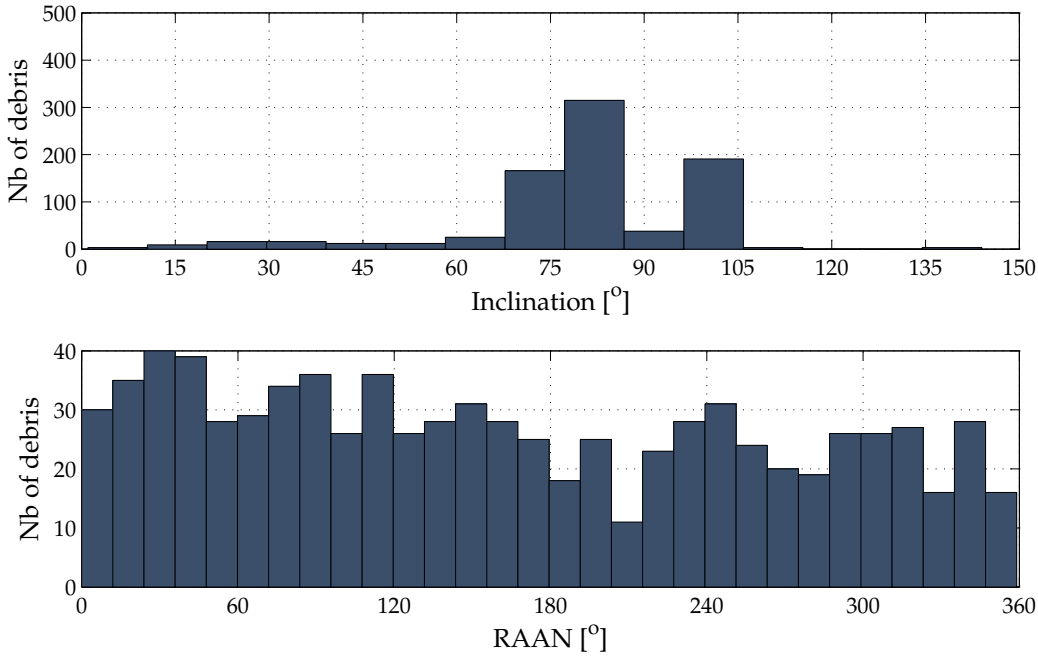

Figure 3.2: Orbital distribution of rocket bodies in LEO

The distribution of inclinations presents clear areas of interests such as the 70-80 [°] zone, mostly occupied by russian or soviet objects, or the 95-105 [°], which corresponds to SSO. On the other hand, the RAANs are very spread and big orbital changes are to be expected between two objects.

3.4 Physical properties

The information about the mass and dimensions of the potential targets are taken from the International Reference Guide to Space Launch Systems [25] and from the website http://www.astronautix.com. Figure 3.3 shows the distribution of mass and volume of the targets in LEO.

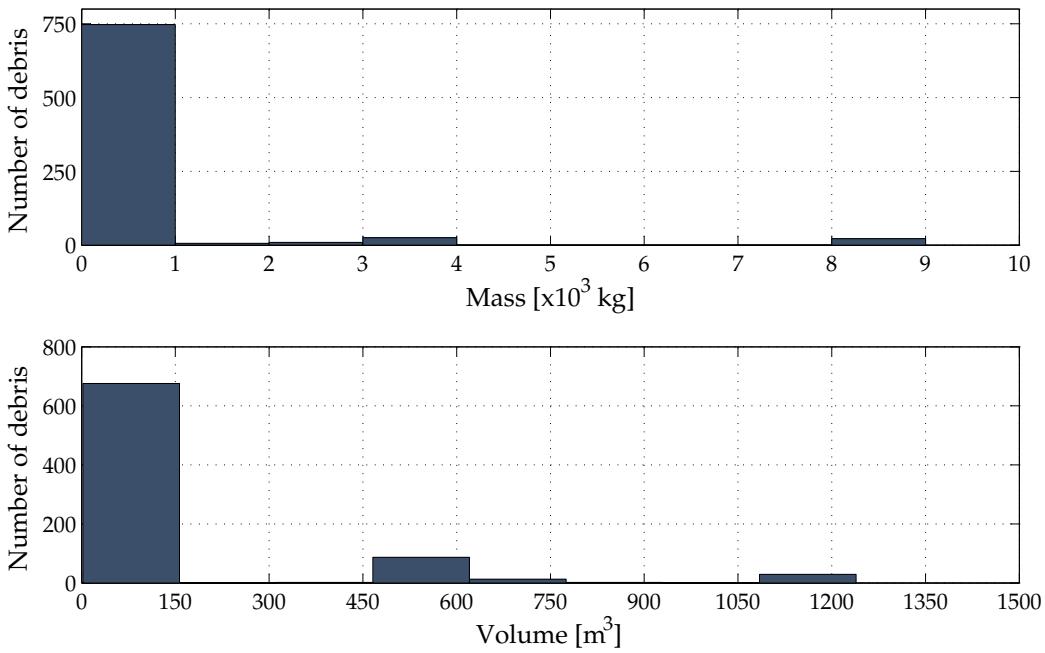

Figure 3.3: Mass and volume of rocket bodies in LEO

Here again, very specific characteristics are shared by most targets. However, the presence of exceptional objects, such as the 9-ton SL-16 rocket bodies, may require some flexibility in the design of the removing spacecraft.

3.5 National contribution

Figure 3.4 shows the participation of every major countries in the population of rocket bodies in LEO.

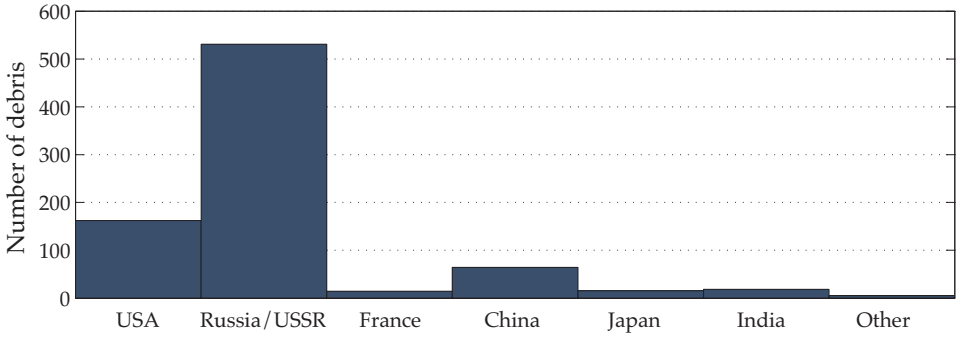

Figure 3.4: Contribution of rocket bodies in LEO per country

It is not a surprise that most of the upper stages come from Russia (or the former Soviet Union). The United States and China come next. These data are very important to determine the financial participation of each country in the case of a collaboration for debris removal.

3.6 Areas of interests

After discussion with J.-C. Liou and N. Johnson[1], it has been decided that the objects that have to be removed in priority are the ones situated in very crowded areas. Figure 3.5 shows the apogees and perigees of every rocket bodies passing in LEO.

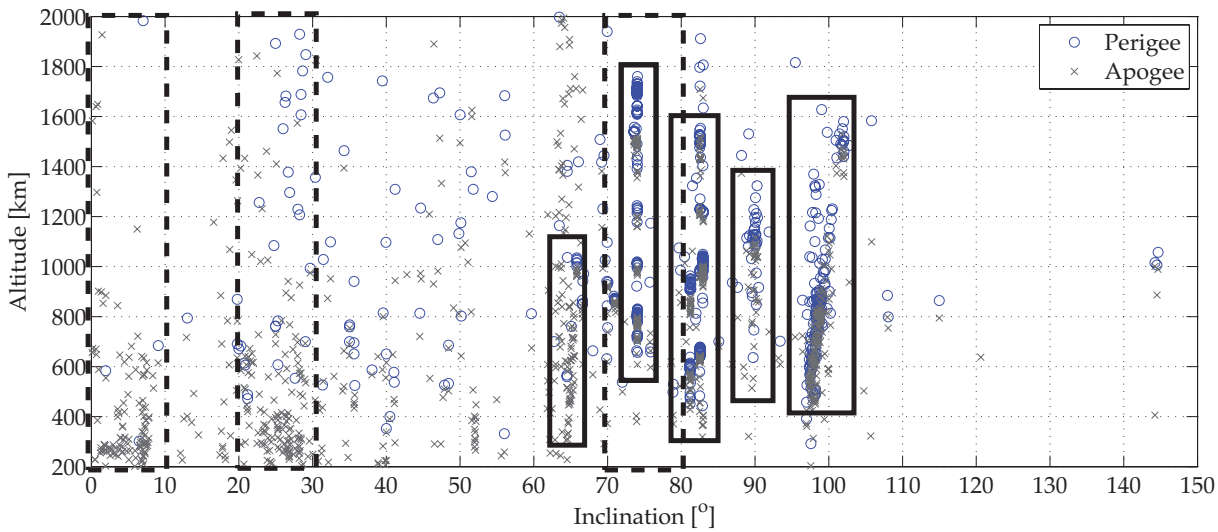

Figure 3.5: Areas of interest in LEO

The main areas of interests are marked by plain-line rectangle. Among these, the most interesting one for this thesis is the SSO (first from the right). The two main reasons for choosing this area as a case study are:

1. It is a very useful region for scientific and Earth-observing satellites.
2. It is very international, with equivalent numbers of payloads from the U.S., Russia and Europe.

The dotted rectangles highlight the areas crowded by objects with non-circular orbits that extend outside of LEO. The most important one is GTO (first from the left), due to its commercial use. However, as explained before, only the circular orbits in LEO are part of the scope of this project.

Now that the targets and their characteristics are known, the following sections will detail the construction of the model used to design the removing satellites.

[1] Orbital Debris Program Office at JSC.

4 Model construction

4.1 Scenarios

Three very high-level architectures (called scenarios) are studied in this research. Because of their differences, they are optimized separately and then compared to find the best one. These scenarios are:

Picker In this version, one removing spacecraft has to reach one and only one target. Once the capture maneuvers are executed, the chaser stays with its debris until the destruction of both.

Shuttle It is actually a multiple-target version of the Picker. In this case, the removing spacecraft visits a first target, places it on a short-lifetime orbit and move directly to the next debris. Thus, one chaser can remove multiple targets.

Mothership The main difference between this scenario and the Shuttle is the fact that the placement on a short-lifetime orbit is not done by the main chaser itself but rather by a removing unit which is sacrificed with the debris. Only the last debris is de-orbited by the main spacecraft.

Figure 4.1 shows the difference between the Picker, Mothership and Shuttle scenarios. The plain lines represents the maneuvers performed by the main removing spacecraft (Mothership spacecraft, MSSC) and the dotted ones the path followed by the de-orbiting units, called Pilot Fish Spacecraft (PFSC).

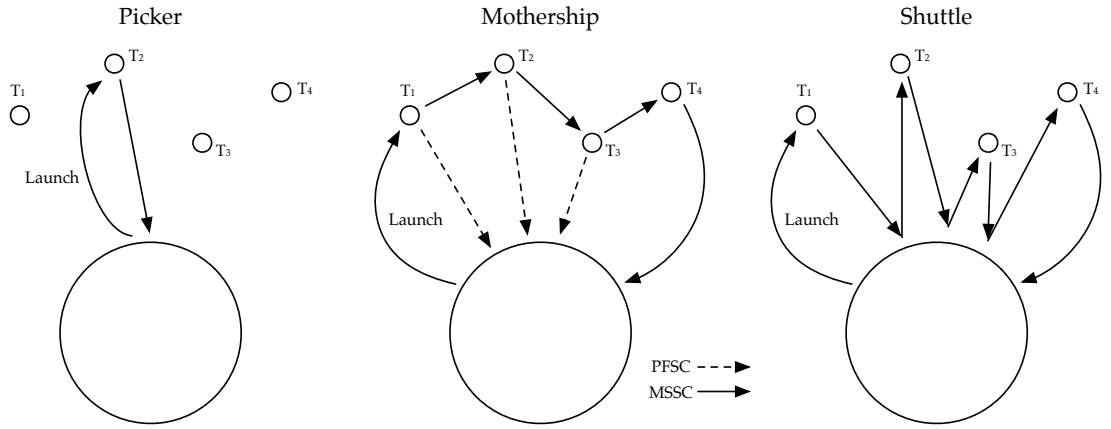

Figure 4.1: Picker, Mothership and Shuttle scenarios

In the case where one removing spacecraft is launched at a time, the injection orbit is simply the orbit of the first debris. However, if two or more spacecraft are launched together, a parking orbit must be used. The reason for this is that nonetheless the altitudes and inclinations of the different targets to be removed are different but so are the RAANs. As explained later in this report, the altitude and inclination changes are performed thanks to the propulsion system but the RAAN changes are done by the use of drifting orbits.

Therefore, if two or more spacecraft are launched together, they will be injected on an orbit on which they will wait for the synchronization of their RAAN with the ones of the debris they have to reach. These operations are shown in Figure 4.2.

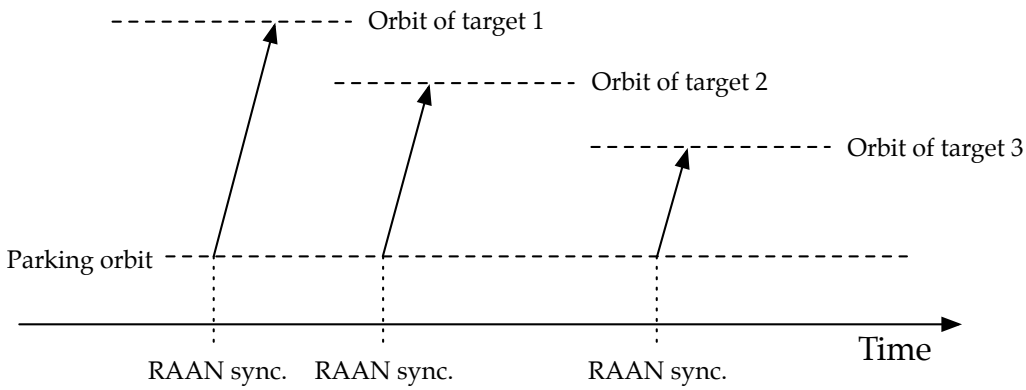

Figure 4.2: Use of a parking orbit in case of multiple spacecraft per launch

4.2 Design variables

Four main design variables are used to optimize each scenario. They are:

1. Maximum number of spacecraft per launch vehicle
2. Number of target per spacecraft (fixed to 1 in the case of the Picker)
3. Specific impulse of the propulsion system
4. Thrust of the propulsion system

The three propulsion types that are tested are:

1. Solid propellant
2. Liquid bi-propellant
3. Electric with loss of mass

The following sections present the construction of the Model used to evaluate each architecture for each scenario and to study the influence of these variables on the objective which is the cost per mass removed in [$/kg].

4.3 Path optimization

In the case where several removing spacecraft are used and each has to visit several targets, the paths that is the most efficient in terms of ΔV has to be found. This can be treated as an adaptation of the traveling salesman problem (TSP). The original TSP is when a salesman has to visit N cities once, and only once,

and come back to his initial city. In the case adapted for this study, several salesmen must visit N cities (each city being visited by one, and only one, salesman) and all terminate their journey in the same specified point, the EOL orbit, which is not one of the city. Figure 4.3 shows the differences between the common TSP and the adapted one.

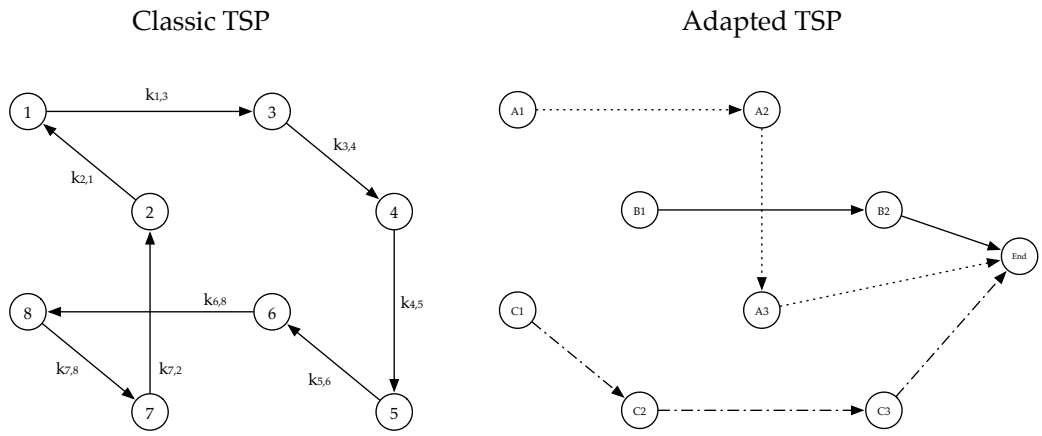

Figure 4.3: Classic TSP and adapted version for debris removal

A genetic algorithm is used to quickly find the shortest way in terms of ΔV for each spacecraft. The main needed inputs are the matrix of ΔV between the different points to be visited and the minimum number of targets that must be visited (Mothership) or de-orbited (Shuttle) by each spacecraft. The matrix of ΔV does not take the evolution of the environment into account because this strongly depends on the system. A future work might take this next step and add the order of visit as a full design variables to have a path that is optimal not only in terms of ΔV but directly in terms of cost.

Figure 4.4 shows a typical output of this algorithm.

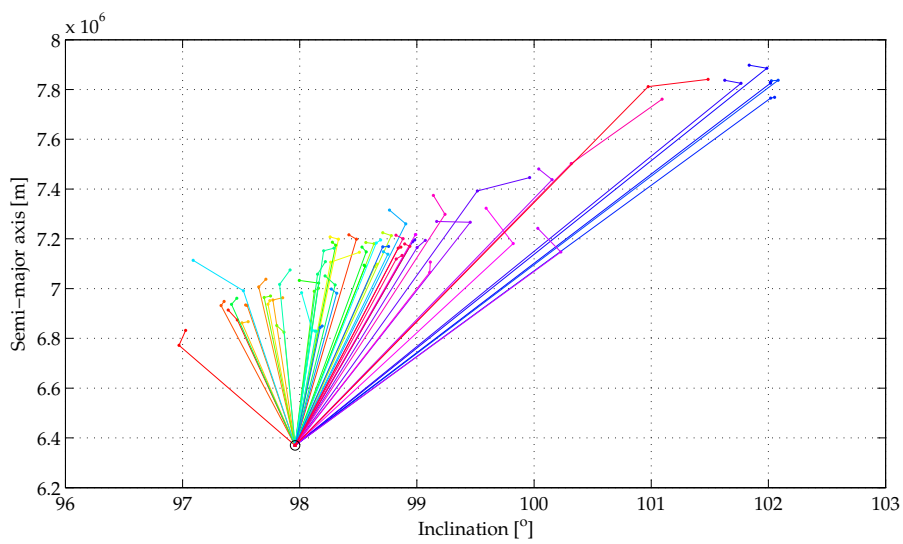

Figure 4.4: Path optimization between targets

In this case, a minimum of 2 targets per spacecraft is given as an input. The targets are placed on the graph depending on their semi-major axis and their inclination but the end point is arbitrary placed below the

targets. The Model computes the ΔV between the last target and the end point as if the latest was placed at the same inclination that the one of the target so the last city is extended to multiple inclination.

4.4 Spacecraft design

4.4.1 Typical removing spacecraft

The usual subsystems found in a removing spacecraft, regardless of the specific scenario are the ones presented in Figure 4.5.

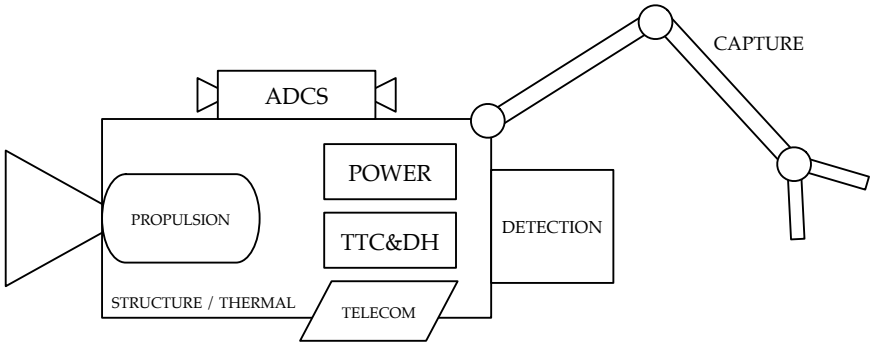

Figure 4.5: Generic removing spacecraft

The capture mechanism is able to grab and maintain the target and eventually to position it for its de-orbiting. In the case of the Mothership, a second one can be added to help with the installation of the PFSC. The detection subsystems is used to find the target in both long and short ranges and to determine its attitude. It can operate in the visible wavelengths and/or using other methods. This subsystems and the software it depends on are one of the biggest challenges posed by the ADR problem. Both these subsystems constituted the payload of the removing spacecraft.

The propulsion subsystem is used to perform the orbital transfers The ADCS stabilizes and adapt the attitude of the spacecraft, especially during the rendezvous and capture maneuvers. The TTC&DH subsystem manages the on-board data and handles the up- and downlink flows received and sent by the telecommunication subsystem. The power subsystem guarantees the distribution of energy among the different subsystems. The thermal system ensures that every part of the spacecraft stays within its operational temperature range and the structure holds everything together.

Some of these subsystems, such as the payload or the TTC&DH, are taken from similar architectures to simplify the model while others, like the propulsion subsystem, are the center of the architecture selection. A very special case is the structure which is simply defined as having a mass that is 30% of the satellite mass.

4.4.2 Overview of the design loop

An overall loop is used to generate all the required spacecraft to de-orbit a given population of targets. It is used to define the technical characteristics of the removing system, the development and manufacturing cost of each of them, the number of launches, the types of launch vehicles and their cost and finally the overall cost of the program and the cost per mass removed. Figure 4.6 gives a simplified overview of this main loop.

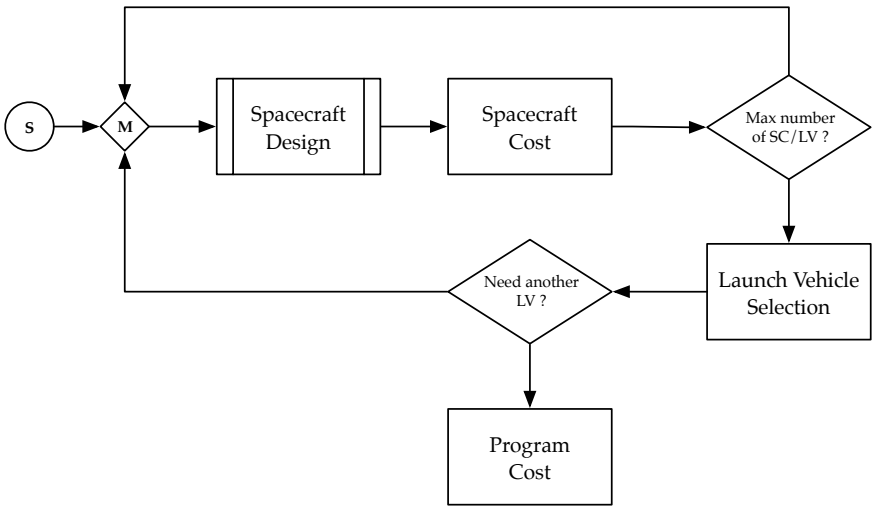

Figure 4.6: Overall design loop

The total number of spacecraft to design depends on the number of targets to catch and the number of targets for each spacecraft. The first one is given as a parameter and the second one is one of the design variables. The total number of launch vehicles is defined by the number of spacecraft and the maximum number of spacecraft per rocket, which is another design variable. Figure 4.7 gives a simplified flow chart of the loop used to design one spacecraft.

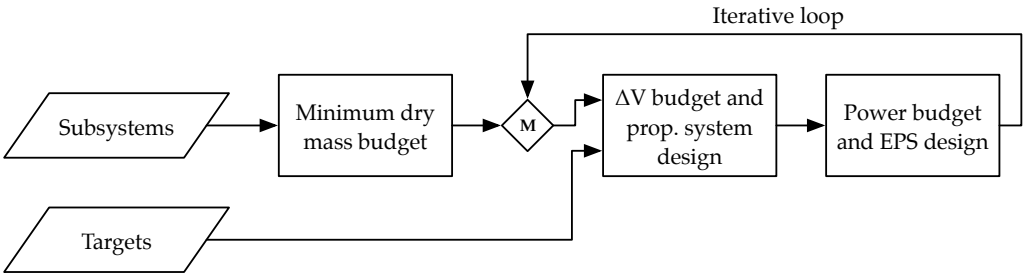

Figure 4.7: Spacecraft design loop

All subsystems, except the propulsion and the EPS, are assumed to be mission-independent and are defined only by their mass and their power consumption. They are taken from similar architectures that flown or presented in more in-depth studies. Each spacecraft is assigned a certain number of targets either as a result of the path optimization, in the case of the Mothership and the Shuttle, or one by one, in the case of the Picker. The orbital parameters and the mass of these targets are used as inputs to the propulsion subsystem design module. Depending on the orbits to be reached, the lifetime of the spacecraft and the power requirements of each subsystems, the EPS is then designed. The design process is iterative to ensure the convergence of the characteristics of the systems, especially the dry and wet masses. Moreover, margins of at least 30% are added to the different mass and power budgets to ensure that design uncertainties are taken into account.

4.4.3 Orbital maneuvers module

Assumptions

The following assumptions are made to simplify the Model:

1. Hohmann's equations are used for high thrust orbital change and Edelbaum's for low thrust.
2. All the orbits are assumed to be circular as shown in the presentation of the debris population.
3. The atmospheric drag is not taken into account for what concerns the target population.
4. High thrust maneuvers are executed quickly enough so the orbital parameters of the target population do not change.
5. Only the J_2 element is assumed to have an effect on the orbital parameters.

High-thrust propulsion transfer

To execute a complete orbit change (altitude, inclination and RAAN) with a high thrust propulsion, the following strategy is used (see Figure 4.8):

1. A first combined maneuver is performed to place the spacecraft on a waiting orbit. This can implies a change of both altitude and inclination. A first boost (ΔV_A) is given in A to match the altitude of apogee with the altitude of the waiting orbit. A second boost (ΔV_B) is given in B to circularize the orbit and change its inclination.
2. The spacecraft stays on the waiting orbit until the natural orbit drifting synchronizes the RAAN of its actual orbit with the one of the next orbit to reach. During this time, the propulsion subsystem must provide a boost (ΔV_W) to compensate atmospheric drag and maintain the orbit.
3. A second combined maneuver is performed to inject the spacecraft on its final orbit. The first boost (ΔV_C) is given in C and the second one (ΔV_D) in D to circularize and match the inclination of the final orbit.

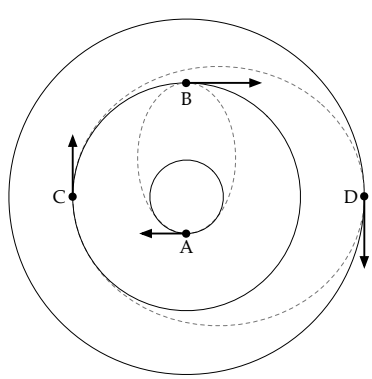

Figure 4.8: Use of a waiting orbit for RAAN synchronization

The ΔVs required for the maneuvers are thus given, for high thrust systems, by the equations

$$V_{A,1} = \sqrt{\frac{\mu}{Ri}} \tag{4.4.1}$$

$$V_{A,2} = \sqrt{\frac{2\mu}{R_i} - \frac{2\mu}{R_i + R_w}} \tag{4.4.2}$$

$$\Delta V_A = |V_{A,2} - V_{A,1}| \tag{4.4.3}$$

$$V_{B,1} = \sqrt{\frac{2\mu}{R_w} - \frac{2\mu}{R_i + R_w}} \tag{4.4.4}$$

$$V_{B,2} = \sqrt{\frac{\mu}{Rw}} \tag{4.4.5}$$

$$\Delta V_B = \sqrt{V_{B,1}^2 + V_{B,2}^2 - 2 \cdot V_{B,1}^2 V_{B,2}^2 \cdot cos(|I_w - I_1|)} \tag{4.4.6}$$

ΔV_C and ΔV_D are found by simply replacing R_1 by R_w, R_w by R_2, I_1 by I_w and I_w by I_2 in Equations (4.4.1) to (4.4.6).

In the case of chemical propulsion (high thrust), the RAANs are assumed to stay unchanged during the transfers. Therefore, the time to spend on the waiting orbit is simply given by

$$T_w = \frac{\Delta\Omega}{\Delta\dot{\Omega}} \tag{4.4.7}$$

where $\Delta\Omega$ may take different values as shown in Table 4.1.

Table 4.1: Values of $\Delta\Omega$

	$\dot{\Omega}_1 > \dot{\Omega}_2$	$\dot{\Omega}_1 < \dot{\Omega}_2$
$\Omega_1 > \Omega_2$	$\Delta\Omega = 360 - \|(\Omega_1 - \Omega_2)\|$	$\Delta\Omega = \|\Omega_1 - \Omega_2\|$
$\Omega_1 < \Omega_2$	$\Delta\Omega = \|\Omega_1 - \Omega_2\|$	$\Delta\Omega = 360 - \|(\Omega_1 - \Omega_2)\|$

As explained earlier, only the J_2 element is taken into account and the RAAN change rate is therefore given by (for a circular orbit)

$$\dot{\Omega}(R, I) = -1.5 \cdot n \cdot J_2 \cdot \frac{R_E}{R}^2 \cdot cos(I) \tag{4.4.8}$$

Low-thrust propulsion transfer

In the case of electric propulsion (low thrust), a constant thrust is given until the final orbit is reached. Changes of altitude and inclination are done in the same time. The strategy is:

1. A constant thrust is given to move in a spiral motion from the initial orbit to the waiting orbit.
2. The required amount of time is spent on the waiting orbit for the RAAN to be synchronized.
3. Another constant thrust is given to move from the waiting orbit to the final one.

The ΔV required to perform such a low thrust maneuvers is given by the Edelbaum equation:

$$\Delta V = \sqrt{V_1^2 + V_2^2 - 2 \cdot V_1 V_2 \cdot cos(\frac{\pi}{2}\Delta i)} \tag{4.4.9}$$

where V_1 (respectively V_2) is the velocity on the initial (respectively final) orbit and Δi is the difference of inclination between the two orbits.

The time needed to perform the transfers is much bigger than with a high thrust propulsion system and it is very likely that the difference of RAAN between the initial and final orbits will change during the maneuvers. To solve this, the evolution of the difference of RAAN during the transfers is taken into account. The transfer time is given by

$$T_t = \frac{\Delta V}{F_{th}} \cdot (m_0 - \frac{1}{2} m_p) \tag{4.4.10}$$

where F_{th} is the thrust of the engine, m_0 is the mass of the spacecraft before the transfer and m_p is the propellant mass required to perform the transfer.

As a first order approximation, the RAAN drift is computed as if the spacecraft spent the whole transfer time on an orbit half way between the initial and the final ones. This is accurate enough because both the altitude and inclination changes are small (maximum 1000 [km] and around 5 [°] respectively). The semi-major axis and inclination of such a equivalent orbit are simply given by

$$S_e = \frac{R_i + R_f}{2} \tag{4.4.11}$$

$$I_e = \frac{I_i + I_f}{2} \tag{4.4.12}$$

The RAAN of the spacecraft orbit after the transfer is given by

$$\Omega_t = (\dot{\Omega}(S_e, I_e) \cdot T_t + \Omega_0) \text{ modulo } 360 \text{ [°]} \tag{4.4.13}$$

where Ω_0 is the RAAN of the spacecraft orbit before the transfer. From now on, the modulo is applied but not written.

By the time the spacecraft reaches its waiting orbit, its orbit's RAAN will be

$$\Omega_s(1) = \Omega_i + T_{t,1} \cdot \dot{\Omega}_{e,1} \tag{4.4.14}$$

and the final orbit's will be

$$\Omega_f(1) = \Omega_f + T_{t,1} \cdot \dot{\Omega}_f \tag{4.4.15}$$

After the second transfer, and without taking the time spent on the waiting orbit, these RAANs will become

$$\Omega_s(2) = \Omega_i + T_{t,1} \cdot \dot{\Omega}_{e,1} + T_{t,2} \cdot \dot{\Omega}_{e,2} \tag{4.4.16}$$

$$\Omega_f(2) = \Omega_f + (T_{t,1} + T_{t,2}) \cdot \dot{\Omega}_f \tag{4.4.17}$$

The difference of RAAN between the two orbits is given by similar relations than in table 4.1 and the time that has to be spent on the waiting orbit is given by

$$T_w = \frac{\Delta\Omega}{|\dot{\Omega}_f - \dot{\Omega}_w|} \tag{4.4.18}$$

where $\dot{\Omega}_w = \dot{\Omega}_w(R_w, I_w)$ is the RAAN change rate of the waiting orbit.

Figure 4.9 shows the evolution of RAAN during such a transfer. The advantage of this intermediate orbit is obviously a shorter transfer time.

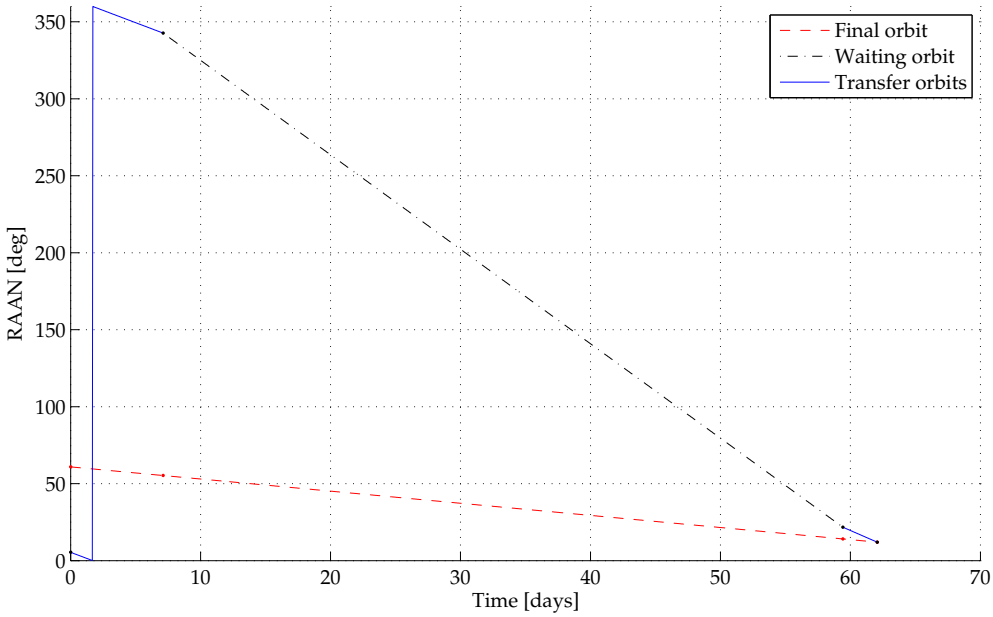

Figure 4.9: RAAN drift during transfer

Although having an intermediate orbit at an altitude and inclination very different from both the initial and final orbit drastically reduces the waiting time, it also increases the required ΔV. In general, the best waiting orbit is simply the initial one. This saves both time and propellant.

Orbit maintenance

The atmospheric drag is neglected for what concerns the targets. It means that the natural decay during the campaign duration is not taken into account, which is a very conservative assumption. Because of this, the active de-orbiting maneuver is not accelerated as the altitude is lowered. In other words: the spacecraft and the targets are assumed to evolve in a perfect vacuum environment. However, it happens that a removing spacecraft has to stay for a long time on a given orbit, especially when a drifting orbit is used to perform a RAAN synchronization. In these cases, and especially at low altitudes, neglecting the atmospheric drag is too optimistic. Indeed, even in a low-density atmosphere, the atmospheric drag slows the spacecraft down. If it has to stay on its orbit, a certain ΔV has to be given to cancel the effect of the drag.

To quickly estimate this ΔV, approximated values taken from [26] are used to build a very simple station keeping model. Figure 4.10 shows the required ΔV per year to maintain the altitude of the orbit.

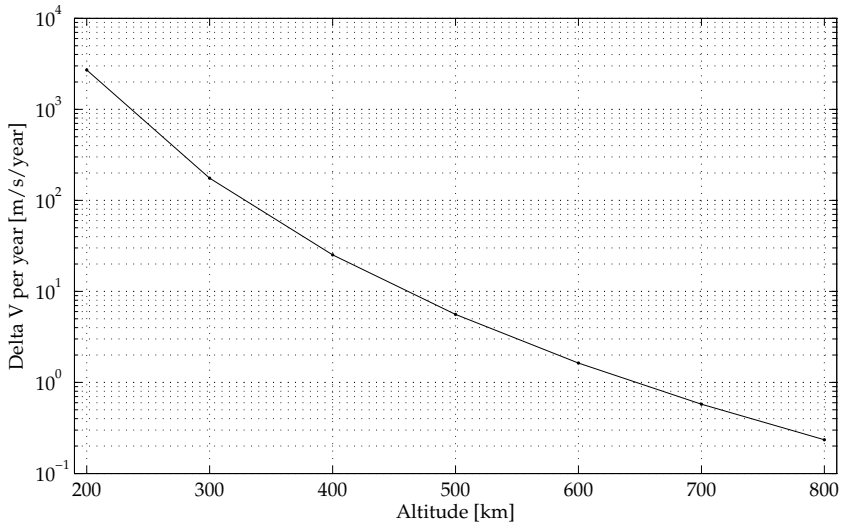

Figure 4.10: ΔV required per year for station keeping

These values are averaged data taken from actual systems during a solar maximum, when the drag is the most important.

4.4.4 Propulsion module

Types of propulsion

Three common types of propulsion are studied in the Model: solid propellant, liquid bi-propellant and electric. Table 4.2 compares the three methods.

Table 4.2: Comparison of propulsion type

	Solid	Bi-propellant	Electric
Thrust	Very high	Medium	Very low
Isp	Medium	Medium	Very high
Complexity	Low	High	Medium
Power	Low	Medium	Very high
TRL	9	9	6-9

Solid propulsion

This is probably the simplest and oldest type of rocket propulsion. The propellant is formed of grains and stored in a combustion chamber. The ignition is generally made by making an electric current passing in a resistive wire which warms up and transfer its energy to the grains that burn. The pressure increase inside the chamber expels mass through the exhaust nozzle and generate thrust. Typical Isp of such an engine are usually about 200 [s] with thrust in the order of 1 [kN] or higher. They present the advantage of being very simple systems which require a power input only until the ignition happens. The inconvenient is that building an engine able to provide different boosts at different time is harder.

Liquid bi-propellant

This type of propulsion is extremely popular and well developed. It needs a more complex valving system to carry both the oxidizer and the fuel to the combustion chamber but has the advantage of being restartable and having an Isp around 300 [s]. The thrust range is very wide but in the case of a main propulsion subsystem, a value of 200 [N] is reasonable.

Electric propulsion

Two of the most common electric propulsion are the ion and Hall effect engine. In both cases, charged particle are accelerated and ejected out of the engine to produce a thrust. Such engines have very high Isp with values in the order of 1,000 [s]. As the research advances in the field of the power management, higher Isp can be achieved with values as high as 10,000 [s]. However, the thrust are usually very low, typically in the order of 1 or 10 [mN].

Propellant mass and tank sizing

The propellant mass m_p for given initial mass m_0 and ΔV is determined thanks to the rocket equation

$$m_p = m_0 \cdot (1 - e^{\frac{\Delta V}{Isp \cdot g_0}}) \tag{4.4.19}$$

The tanks are assumed to weight 10% of the mass of propellant. This dry mass is added to the mass of the thrusters to be applied as an input to the cost model for the propulsion system defined later in this report.

Space tethers

Space tethers are long wire extended from a spacecraft. In the case of the EDT, the interaction between an electric current passing through the conducting tether and the Earth magnetic field, creates a force that is used to move the spacecraft. If the system is active, i.e. power is given from a power supply, the force for a tethered system in LEO is collinear with and in the same direction than the velocity vector and the altitude can be increased. On the opposite, if the system is passive, the force is opposed to the velocity vector and the altitude is decreased. This last application is very useful in the case of ADR since it allows a de-orbiting without the need for a power supply. Typical tether are about 10 [km] long, 1 [mm] of diameter and produce very low thrust. However, the performances strongly depend on the ability of the wire to exchange electric charges with the environment. In the case if the MET, the wire is much longer and allows to have significant differences in the forces applied at both ends due to the difference of energy of the orbits. Great tensions are observed in the tether and the momentum transfer that appears when the cable is broken can be used to give an impulse to both ends. Because it was not possible to get any information to build a cost model for the Space tethers, these are not directly included in the Model. It is important to understand that due to their low level of readiness and the absence of a specific cost model, the comparison with more conventional propulsion is extremely delicate.

4.4.5 Power module

The EPS generate, store and distribute the electric power among the spacecraft. It is very dependent on the propulsion system as the different types have different requirements. Typical values are used for the solid an bi-propellant engines. Specifically, 30 [W] for the solid propellant which is given as a peak to start each ignition, once the propellant in the chamber is burning, power input is not required anymore. In the case of

the liquid propellant, the power requirement is around 80 [W]. Unlike the solid propulsion, the bi-propellant systems require power to actuate the valves and keep the engine burning.

In the case of the electric propulsion, it is not possible to use any averaged values since the power requirement is strongly dependent on the thrust and the Isp of the thruster. The relation between the power needed from the power supply and the characteristics of the engine is given by

$$P = \frac{F_{th} \cdot Isp \cdot g_0}{2\eta} \tag{4.4.20}$$

where F_{th} is the thrust in [N], the Isp is in [s], $g_0 = 9.81$ [m/s^2] is the standard gravitational acceleration and η is the efficiency of the thruster. A typical electric propulsion system, using either ionic propellant or hall effect, has an efficiency between 60 and 70%. In the Model, the power requirement is computed before the design loop, as soon as the propulsion type, the Isp and the thrust are defined.

The design is also influenced by different mission profiles as the eclipse time and the mission duration vary. The Model only considers architectures using solar panels and secondary batteries. This choice is made because it is the most typical configuration for LEO satellites without excessive power requirements.

4.5 Cost estimation module

4.5.1 Cost models at the subsystem level

To clearly see the influence of the choice of a specific subsystem, especially for the propulsion subsystem, the cost are studied at the subsystem level. Simple parameters, usually the mass or overall size of the subsystem, are used to estimate the cost. These cost models are taken from [26], Table 4.3 shows the models used for propulsion and power. The other ones are given in the appendix.

Table 4.3: Subsystem cost models

Subsystem	Parameters	Model for RDT&E [FY2000$K]	Model for TFU [FY2000$K]
Propulsion	Mass [kg] M	$17.8 \cdot M^{0.75}$	$4.97 \cdot M^{0.823}$
Power	Mass [kg] M, Power [W] P	$2.63 \cdot (M \cdot P)^{0.712}$	$112 \cdot M^{0.763}$

All these models are based on existing hardware and have an error between 30 and 50%. This maximum error is used as a margin in the conclusions of this report to estimate the worst case. The output of the models is given in U.S. dollars from fiscal year 2000. To have a more recent cost estimation and to allow the comparison with other models in the chapter about validation, a conversion factor of 1.225 is used to get the value in FY2010$.

4.5.2 Cost model for small spacecraft

Because of their small size and their simplified subsystems, the Pilot Fish Spacecraft, used in the Mothership scenario, do not fit in the models detailed earlier. For this reason, the following cost model is used

$$1.4 \cdot (781 + 26.1 \cdot M^{1.261}) \tag{4.5.1}$$

In this case, only the dry mass M is used as an input and it is therefore not possible to see the influence of a specific choice of subsystems on the mission cost. Moreover, this model includes both development (RDT&E) and manufacturing (TFU) cost.

4.5.3 Multiplying factors for TRL

The risks due to new or under qualified technologies are translated in terms of cost by using the Technology Readiness Levels (TRL) that are defined in Table 4.4.

Table 4.4: Integration of TRLs

TRL	
1	Basic principles observed and reported
2	Conceptual design formulated
3	Conceptual design tested analytically or experimentally
4	Critical function/characteristics demonstration
5	Component/brassboard tested in relevant environment
6	Prototype/engineering model tested in relevant environment
7	Engineering model flight tested
8	Flight-qualified system
9	Flight-proven system

Following the advice found in [24], the multiplying factors detailed in Table 4.5 are applied directly to the cost models at the subsystem level.

Table 4.5: Cost factors used to represent TRLs

TRL	Factor
3-4	1.75
5-6	1.32
7	1.00
8	0.82
9	0.68

Each subsystem must be given a TRL to allow its cost to be estimated by the Model.

4.5.4 Launch vehicles

The launch vehicles are selected once the mass of each removing spacecraft has been computed. If the mass to be launched by a single launcher exceeds its maximum capacity, the next biggest one is selected. If there is not a rocket able to launch a specific architecture, the launch cost is set as infinite and the architecture is discarded.

Because of the economic and policy model detailed further in this report, only launchers from the United States, Europe and Russia are included in the Model. Table 4.6 details these vehicles, their cost and their capacity to SSO.

Table 4.6: Launch vehicles used in the Model

Name	Capacity [kg]	Cost [M$]	Country
Kosmos	775	12	RU
Athena	1,165	50	US
Vega	1,395	20	EU
Delta II	3,186	65	US
Soyuz	4,300	50	RU
Delta IV Medium(+)	6,832	100	US
Ariane 5	9,500	155	EU
Delta IV Heavy	19,665	125	US

These information were taken from the International Reference Guide to Space Launch Systems [25].

4.5.5 Program cost

The global cost of the program includes the development cost, the manufacturing cost of all the required spacecraft and the launch cost. The operations and ground segment cost are already included in the cost model. The total launch cost is simply the sum of the cost of every launch vehicle used during the campaign. The development cost is the maximum of the RDT&E cost given by the Model for a specific architecture. Ideally, only one type of spacecraft is designed and built to simplify the fabrication process. Because more knowledge is gained and errors are corrected every time a spacecraft is built, the manufacturing cost are expected to decrease with the number of produced unit. This is expressed thanks to the learning curve. If the architecture to be developed has a cost C_{TFU} for the TFU, the Nth to be build will cost

$$C(N) = C_{TFU} \cdot LC^{N-1} \tag{4.5.2}$$

where LC is a value between 0 and 1, typically around 0.9, representing the decrease of the cost per unit with an increasing number of produced units.

5 Validation

5.1 Validity with respect to NASA's SVLCM

The System/Vehicle Level Cost Model (SVLCM) is a very high level cost model developed by NASA and available on the JSC website[1]. The inputs needed are the type of spacecraft (unmanned Earth-orbiting in this case), the dry mass, the quantity of spacecraft to build and the learning curve.

To compare both models, the mass of the different subsystems are changed and the TFU and development cost are compared with the corresponding cost. The results are shown in Figure 5.1.

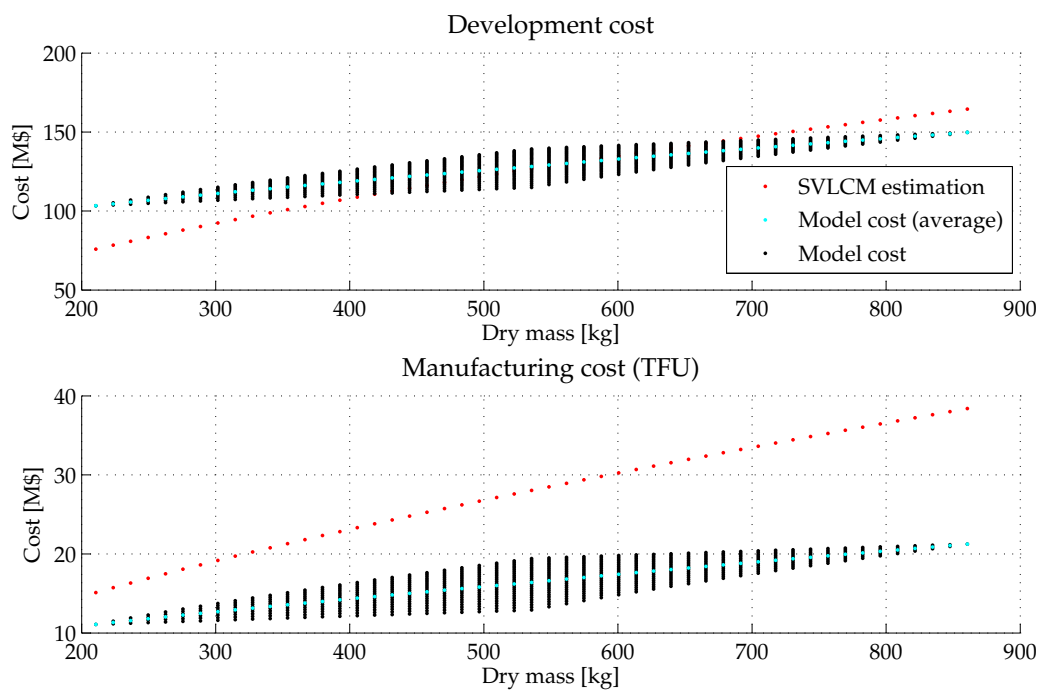

Figure 5.1: Comparison between the Model and the SVLCM

Big absolute differences are observed between both curves, especially for what concerns the manufacturing cost of the TFU. It is also possible to see how the cost spreads around the average when the mass of different subsystems is changed. However, the cost given by both methods are in the same order of magnitude. The differences are due mostly to the fact that the SVLCM must represent all types of unmanned Earth-orbiting satellites, which include a very wide variety of system. The spacecraft may be used for Earth or deep Space observations, telecommunication, for commercial or scientific use, and the mass range goes from 76 to 8,849

[1] http://cost.jsc.nasa.gov/SVLCM.html

[kg]. Moreover, only 35 data points were used to build this model. These differences are however not a problem as the Model is used to compare architecture and not to accurately evaluate the cost of a given architecture.

Figure 5.2 gives the plot of the error between the SVLCM and the Model.

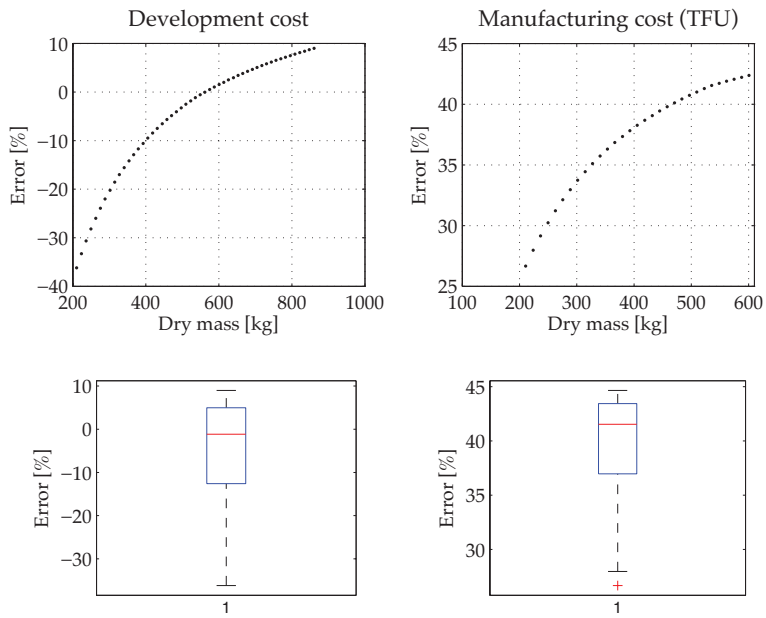

Figure 5.2: Error between the Model and the SVLCM

These data confirm that the order of magnitude given by the Model is good enough. These values, around 30 % are about the same as the ones expected from the subsystem level cost model. However, if an absolute cost value is needed for a specific architecture, nothing will replace an in-depth study as the one presented in the next section.

5.2 Architecture reproduction: DR LEO

The best way to verify the accuracy and the relevance of the Model is to compare it with a previous study. Not many were done in the field of ADR that also include a cost study but among these in DR LEO, completed by Master students from the University of Cranfield in 2010. It studies the feasibility of an architecture able to remove 5 rocket bodies from SSO. These are the Ariane 4 upper stages presented in Table 5.1.

Table 5.1: Targets for DR LEO

NORAD ID	Type	Mass [kg]	Altitude	Inclination
20443	Ariane 40 R/B	1600	790.62	98.71
21610	Ariane 40 R/B	1600	780.84	98.75
22830	Ariane 40 R/B	1600	810.27	98.70
23561	Ariane 40 R/B	1600	790.35	98.56
25979	Ariane 40 R/B	1600	631.98	98.12

The model is used to reproduce an architecture able to capture these targets. Most technical and mission

data are taken directly from the DR LEO study, the goal being to validate the models used to represents the propulsion system, the electrical power system and, most of all, the cost. Thus, the mass and power consumption of the following subsystems are common to DR LEO and to the architecture studied by the model: ADCS, TTC&DH, Telecom, Thermal and Payload (vision system and robotic arm).

The comparison of the mass budgets of both architecture is presented in Table 5.2.

Table 5.2: Mass budget

Subsystem	Mass [kg] (DR LEO)	Mass [kg] (Model)	Error [%]
ADCS	24.30	24.30	0.00
TTC&DH	12.00	12.00	0.00
Telecom	3.00	3.00	0.00
Thermal	20.00	20.00	0.00
Payload	59.00	59.00	0.00
Power	62.08	31.10	99.61
Propulsion	72.62	76.30	4.82
Propellant	156.22	200.00	21.89
Structure	110.87	97.20	14.06
Margin	26.20	INCLUDED	-
Dry mass	363.87	349.10	4.23
Wet mass	520.09	549.10	5.28

Even though the overall error is quite small, it is important to notice that it is the results of the compensation of two main differences. Indeed, the propellant mass is smaller for the architecture designed by the Model. This is due to the fact that the epoch of the targets are not the same which means that the RAANs of the orbits differ. Moreover, it appears that the Model underestimates the propellant required for the rendezvous phase. The other big difference comes from the power subsystem. The reasons for such a bigger mass is the fact that all the systems are supposed to work at their average power consumption (plus a 30-% margin) at all time. This assumption was made to simplify the design process. The DR LEO team did not need this assumption and where able to analyze more in depth the power requirements along the mission and came up with a more efficient design.

Concerning the cost study, DR LEO includes not only the TFU, Development and launch cost but also an insurance fee. Because the Model does not compute this fee, it is simply added to the overall campaign cost. The comparison of cost between the two studies is shown in Table 5.3. The results are given in euros (FY2010) to match the data taken from the DR LEO report [20]. The cost given by the Model are in FY2010$ and have been converted by using a factor 0.7 [euros/$] which is an average of the conversion rates in 2010.

Table 5.3: Cost model validation

Subsystem	Cost [Mios euros] (DR LEO)	Cost [Mios euros] (Model)	Error [%]
TFU	136.84	133.7	2.35
R&D	59.68	50	19.36
Insurance	15.2	15.2	0
Launch	50	50	0
Global cost	261.72	248.9	5.15

The cost per mass removed is 32,720 [euros/kg] for the results given by the model and 31,110 [euros/kg] for DR LEO. As for the mass budgets, the results are very close and help in gaining confidence in the developed model. It is important to keep in mind that both systems are not completely identical and that a bigger

error on the cost might be possible. However, the estimation of the cost is excellent and, according to this comparison, the Model can be trusted.

5.3 Summary

Comparisons are made with a very generic and widely adopted cost model as well as with a very specific study. In both cases, the results given by the Model are sufficiently accurate to be trusted for the following analyses. It is still crucial to remember that the main purpose of the Model is the comparison of different architectures and not the determination of the absolute cost of a specific system.

6 Results

6.1 Picker

6.1.1 Trade-space exploration

One mission variable (the number of spacecraft to be launched together) and three technical variables (the type of propulsion, the Isp and the thrust) are studied. Only discrete values are taken to keep the problem at a sufficiently high level and to reduce the simulation time. However, the influence of all these variables on the objective (cost per mass to be removed) is studied at once to ensure an optimal design.

Concerning the number of spacecraft per launch vehicle, the values are kept between 1 (to favor very cheap and light launcher) and 10 (to reduce the number of launches). The higher bound was arbitrary fixed to limit the complexity of the cluster of removing spacecraft. The results for this variable are shown if Figure 6.1.

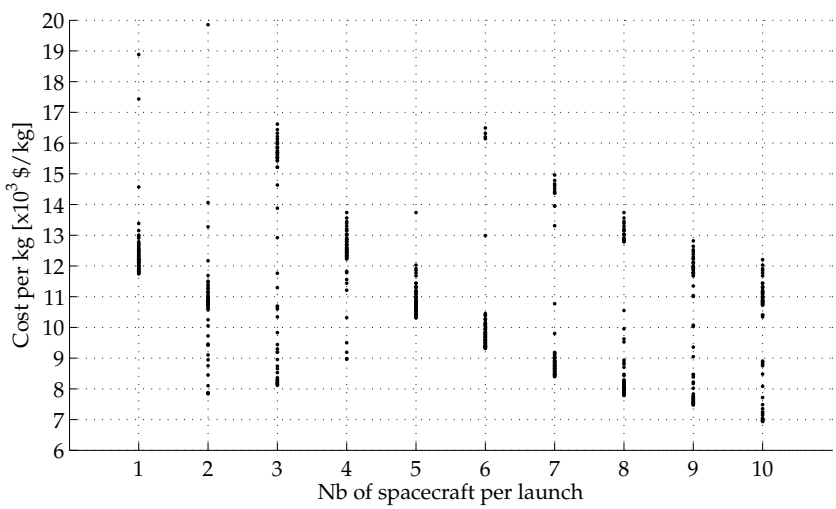

Figure 6.1: Number of spacecraft per launch and cost per mass

Each dot on the graph represents a feasible architecture. The only constraint that is not applied is the time constraint (at least 5 targets to be removed by year) as it is studied further in this section. As expected, the cost per mass removed decreases when the number of spacecraft per launch increases, with a minima on the upper bound.

The propulsion type (electric, chemical solid or chemical liquid) is then studied and the results are presented in Figure 6.2.

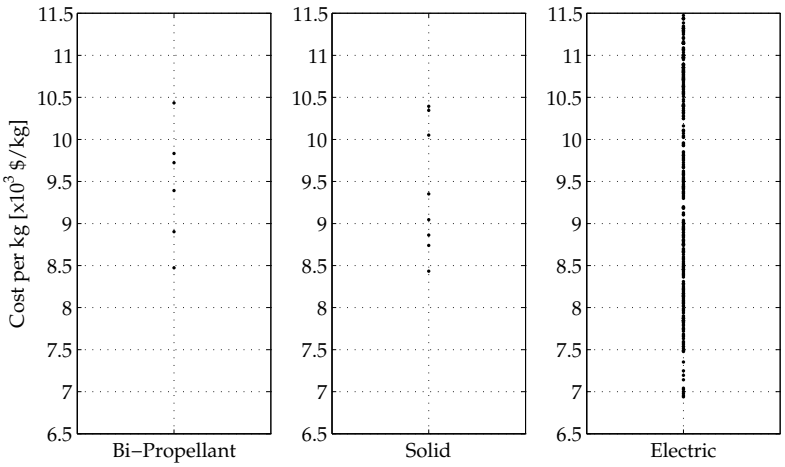

Figure 6.2: Propulsion type and cost per mass

The electric propulsion is usually better than any type of chemical system. This is due to the fact that the mass to launch is usually smaller (due to the higher Isp) and allows cheaper launchers. However, this cost depends a lot on the specific characteristics (thrust and Isp) of the engine. The results for the Isp are presented in Figure 6.3.

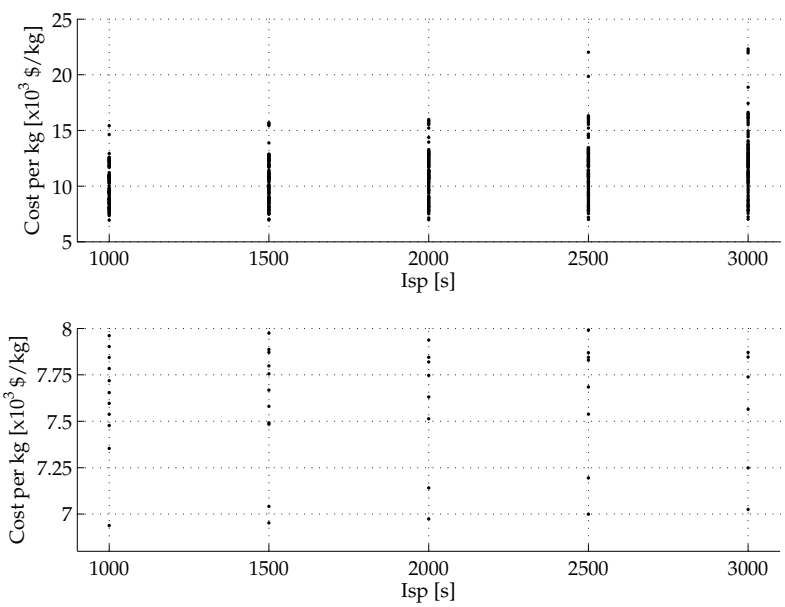

Figure 6.3: High Isp and cost per mass

A counter-intuitive results is observed in this graph: the cost increases with the Isp. It would be expected to be the opposite. Indeed, with a more efficient engine, the spacecraft would be lighter and would cost less. However, because the power required by an electric propulsion system is proportional to its thrust and Isp, a higher Isp requires more power and therefore larger solar panels and batteries. Even though the propellant

mass decreases, the power subsystem is heavier. Moreover, because the cost study is done at the subsystem level, this increase in power is reflected on the cost. A more detailed explanation is given in the Appendix.

Not only a lower Isp reduces the cost because of a lighter power subsystem, it also ensures the use of proven technologies. Indeed, ion and hall engines with Isp of 3000 [s] or lower have already flown and are therefore already qualified (TRL 9). Isp as high as 8000 [s] are expected in the future but the research is still ongoing and the development cost are much higher.

The results for the other characteristic of the propulsion, the thrust, are given in Figure 6.4.

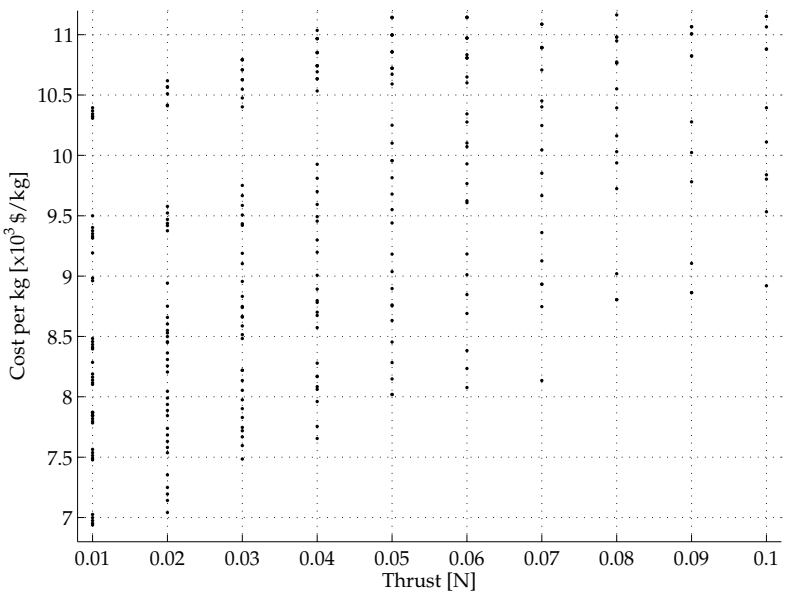

Figure 6.4: Low thrust and cost per mass

Here again, the same type of results is observed: the cost decreased with the thrust. However, in this case there is a bad consequence: with lower thrust, the orbital transfers may take more time. The right value of the thrust has then to be selected to aim for an average of 5 debris per year are de-orbited.

6.1.2 Campaign duration and launch planning

To try to remove 5 rocket bodies per year from SSO, not only the selected architecture must be able to perform the mission in a reasonable time but the availability of the selected launch vehicles must be taken into account. For instance, assuming that the solution of sending 1 spacecraft at a time was the most cost efficient, if it takes 10 launches per year with Vega, the solution cannot be implemented. Firstly, because the production line may not be able to provide so many rockets and secondly because Arianespace has other customers who also need this vehicle during the time the ADR campaign takes place.

Because increasing the number of spacecraft per launch reduces the number of required rockets while also reducing the cost, the solution of sending 10 spacecraft per launch is kept. The time required for each group of 10 removing spacecraft to perform its mission is then studied. This time is given by the lifetime of the slowest spacecraft in one group. When this slow remover re-enters the upper atmosphere with its target, all the spacecraft of the group have completed their mission. For instance, the first group of 10 spacecraft using electric propulsion (Isp of 1000 [s] and thrust of 10 [mN]) have the lifetimes shown in table 6.1.

Table 6.1: Lifetimes of removing spacecraft (Group 1)

S/C ID	Lifetime [days]
1	116.0
2	355.4
3	202.9
4	107.9
5	523.8
6	202.2
7	271.9
8	153.8
9	292.4
10	316.3

The slowest spacecraft of the group takes 523.8 [days] to remove its target, this represents the maximum mission duration, $T_{M,max}$. Figure 6.5 shows the maximum, minimum and average values of $T_{M,max}$ as a function of the thrust for bi-propellant and electric propulsion. For the later, an Isp of 1000 [s] is used as it is the less expensive in all cases.

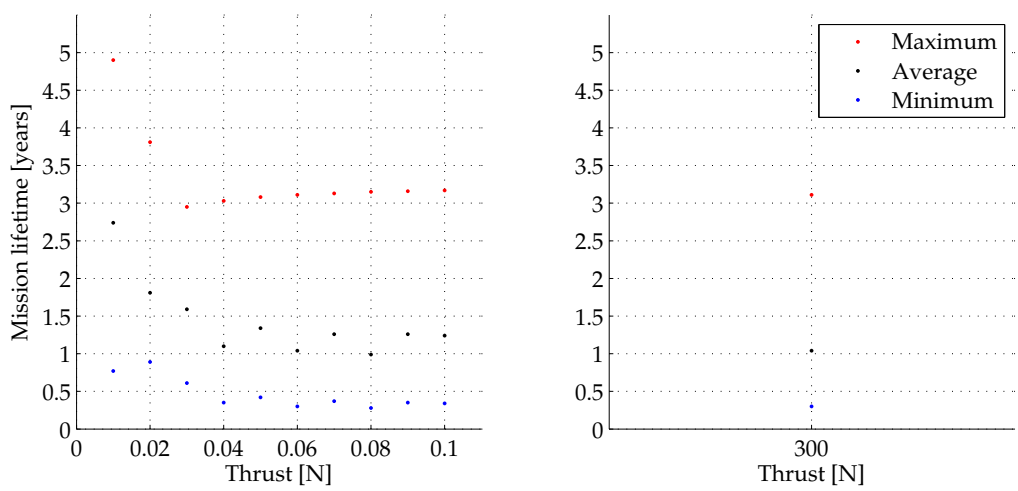

Figure 6.5: Low thrust and mission duration

The difference in lifetime between the chemical and electric propulsion is small because, with both methods, the most time-consuming part of the mission is the wait for the RAAN synchronization. Because the date of launch is not a part of this model, the situation in terms of RAAN is not optimal. Selecting this date to minimize the RAAN differences would reduce the time spent on the waiting orbits and eventually show that the chemical propulsion presents much shorter mission times than the electric. Nevertheless, the lifetimes presented here show that even a low thrust electric propulsion is sufficient to respect the constraint.

Indeed, the 0.01-[N] propulsion has an average mission time of 2.74 [years]. With 1 launch per year, the campaign lasts 13.74 years to remove the 112 objects in SSO (11 years of launch added to 2.74 until the last mission ends). This makes an average of 8.15 debris per year which is more than the 5 debris per year recommended by NASA. This brings to a cost per unit of mass removed of 6,938.5 [$/kg], which is the minimum achievable within the actual design space.

To take a more conservative point of view, the same calculation is done with the maximum mission time for this architecture: 4.9 [years]. In this case, the 0.01-[N] architecture can achieve 7.04 debris per year which

is still higher than the recommendation. The cost per debris stays the same as before.

To improve the cadence, 2 launchers per years can be used, one every 6 months. The campaign now lasts 8.24 [years] in the average case and 10.4 [years] in the worst one. This respectively gives respectively 13.59 [debris/year] and 10.76 [debris/year], adding a consequent margin to NASA's minimum objective and keeping the cost as low as possible.

Another possibility is to develop the 0.02-[N] architecture. It has an average mission time of 1.81 [years] with a maximum of 3.81 [years]. The cost per mass is of 7,353.4 [$/kg]. In this case the achieved performance with one launch per year is of 8.74 [debris/year] (average) or 7.56 [debris/year] (worst case). Figure 6.6 shows the dry and wet masses of the 0.01-[N] Pickers.

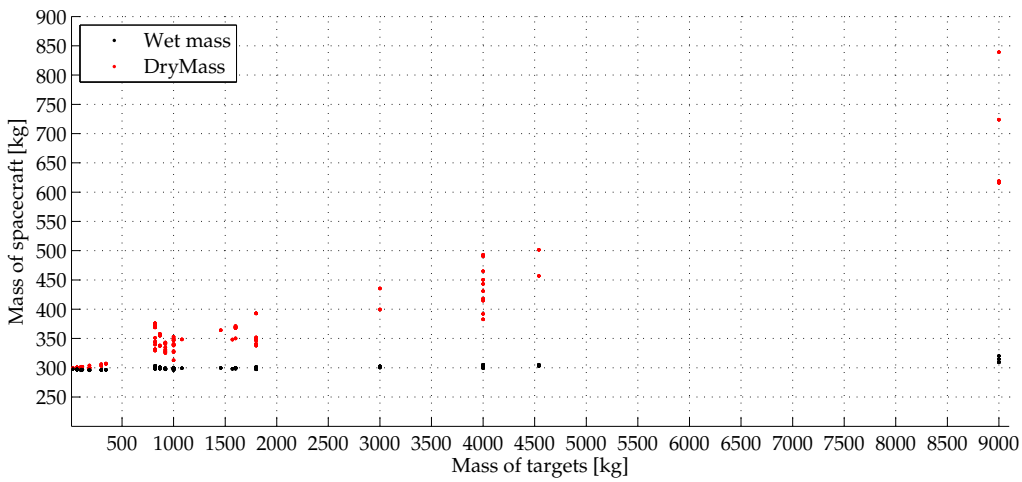

Figure 6.6: Masses of spacecraft and targets (Thrust: 0.01 [N])

A first observation is that the difference of dry mass between the heaviest and the lightest spacecraft is only of 25 [kg]. This is an excellent news in terms of manufacturing: all the spacecraft can all be exactly the same without having too much wasted mass. This allows to rationalize the manufacturing process by building hundreds of spacecraft that all have the exact same characteristics and parts. However, due to the bigger differences in terms of wet mass, not all the spacecraft are launched with the same amount of propellant. Only what is required to perform the mission, plus a security margin, is put in the tanks and helium is used to ensure the proper pressure all along the mission lifetime. 11 Soyuz and 1 Kosmos are needed to launch the 112 spacecraft that take care of the targets in SSO.

The masses for the 0.02-[N] architecture are shown in Figure 6.7.

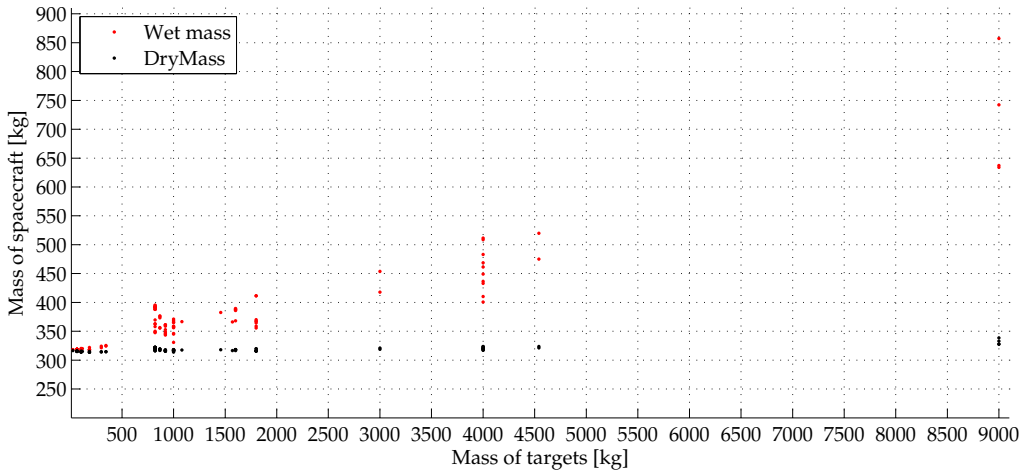

Figure 6.7: Masses of spacecraft and targets (Thrust: 0.02 [N])

In this case also, the variation of dry mass is sufficiently small to allow the reproduction of the same spacecraft. The campaign with the 0.02-[N] engine requires 10 Soyuz, 1 Delta IV medium and 1 Vega. The change of launch vehicle is due to the slightly higher wet masses. To avoid the use of too many different types of rockets, the solution with the 0.01-[N] engine is selected for now.

6.2 Mothership

6.2.1 Design of Pilot Fish Spacecraft

The small de-orbiting modules that are delivered to the different targets by the Mothership are called Pilot Fish Spacecraft (PFSC). They are designed first in order to minimize the mass to be carried by the main spacecraft and the cost added to the overall program budget. Two different types of propulsion are tested, electric and solid. The bi-propellant systems are excluded due to their complexity, the goal is to have either a very simple system (solid propellant) or very efficient (electric). The cost are estimated using the model adapted for small spacecraft explained earlier.

Solid propulsion

The results in terms of mass for a solid propulsion system with an Isp of 200 [s] are shown in Figure 6.8.

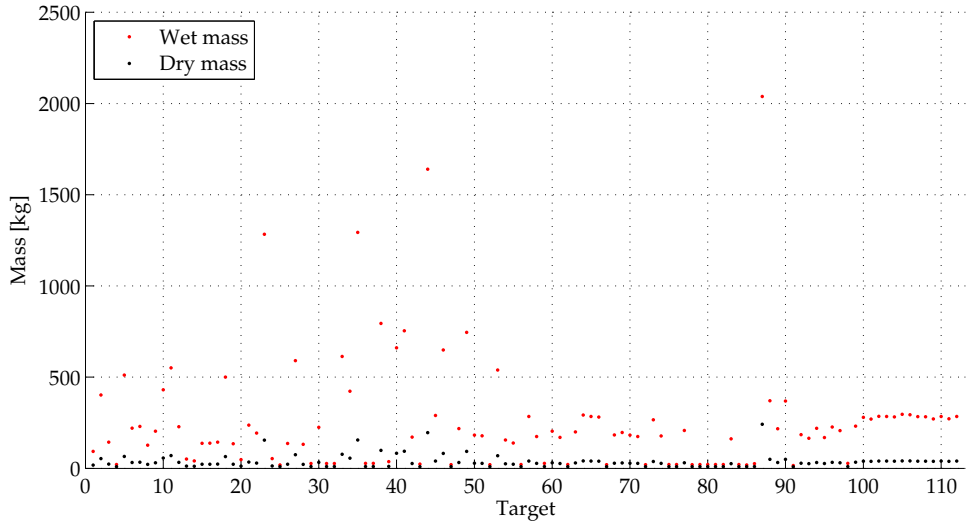

Figure 6.8: Mass of solid propellant PFSC

The mean cost for one of these units is $4.89 millions with a maximum of $38.01 millions. Developing and building all of them would cost $547.94 millions. The wet masses are very spread with an average of 256.90 [kg] and a maximum of 2,038.18 [kg]. If more than one of these have to be carried by a Mothership spacecraft, the total mass to launch would be extremely high and might even be impossible to fit in any existing launcher.

The advantage of the solid propulsion, however, is it very high thrust and therefore its very short lifetime. De-orbiting a debris would take an average of 46.46 [min] and a maximum of 50.16 [min].

Electric propulsion

As for the Picker scenario, the electric propulsion subsystem needs to be optimized in terms of Isp and thrust. Figure 6.9 shows the results of these simulations.

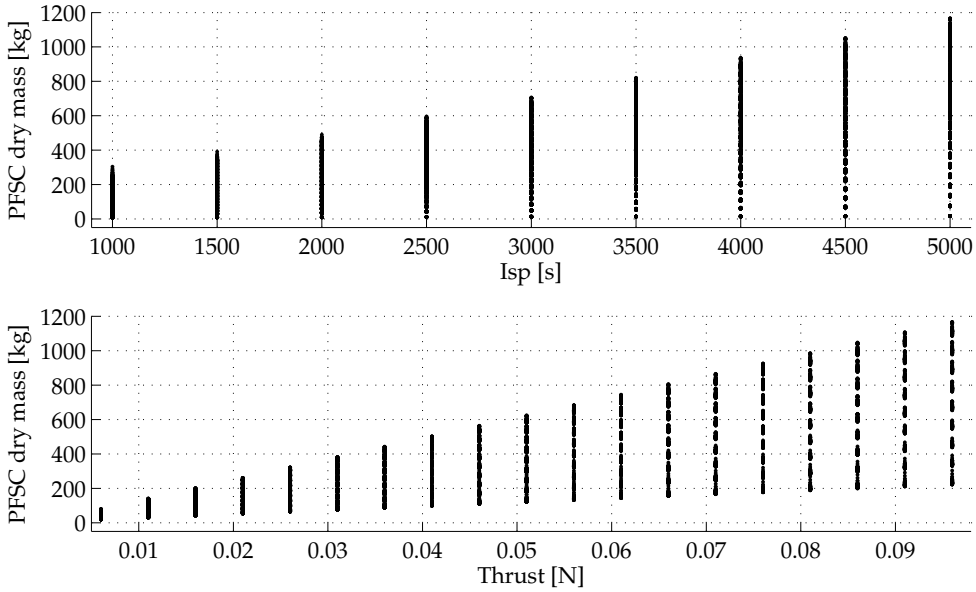

Figure 6.9: PFSC's propulsion and cost per mass

Lower values of Isp and thrust result in lower cost. However, the very low acceleration provided by the thrusters leads to lifetime longer than 20 years for the heaviest target. The idea is then to have modules that are optimized for the class of debris to remove. Figure 6.10 shows the thrust that are needed for every PFSC to ensure a lifetime of 5 [years] or less while minimizing the cost per mass removed.

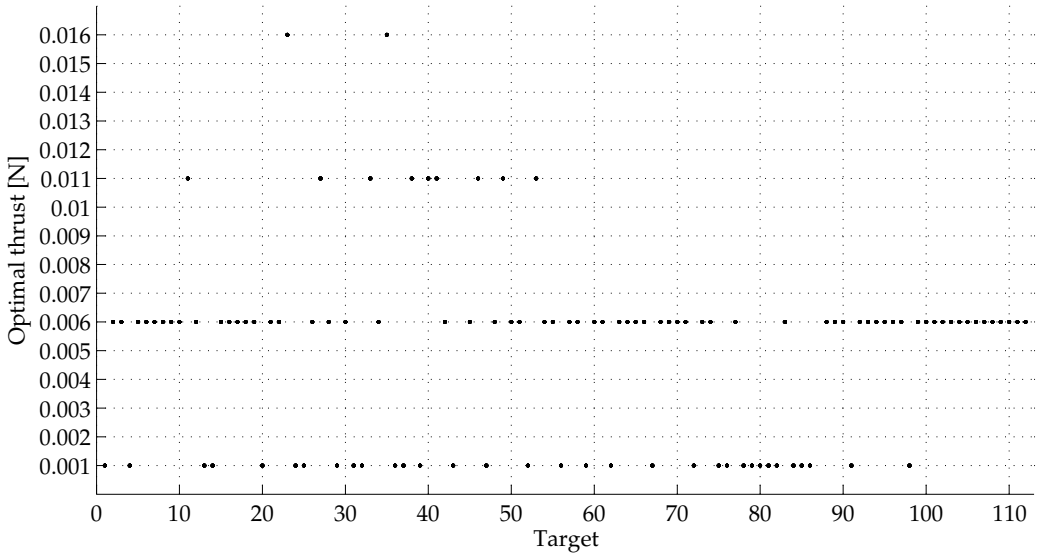

Figure 6.10: Optimal thrust for each debris

Different classes of targets require different value of thrust to be de-orbited in a acceptable time. Two approaches can be taken: either the most common 6-[mN] architecture is used for every debris or the

optimized versions are developed and built. The first solution leads to sub-optimal spacecraft but with similar dry mass and therefore easier to manufacture and the second one, as shown in Figure 6.11, leads to the development of 4 different classes of PFSC but with a considerable gain in terms of lifetime.

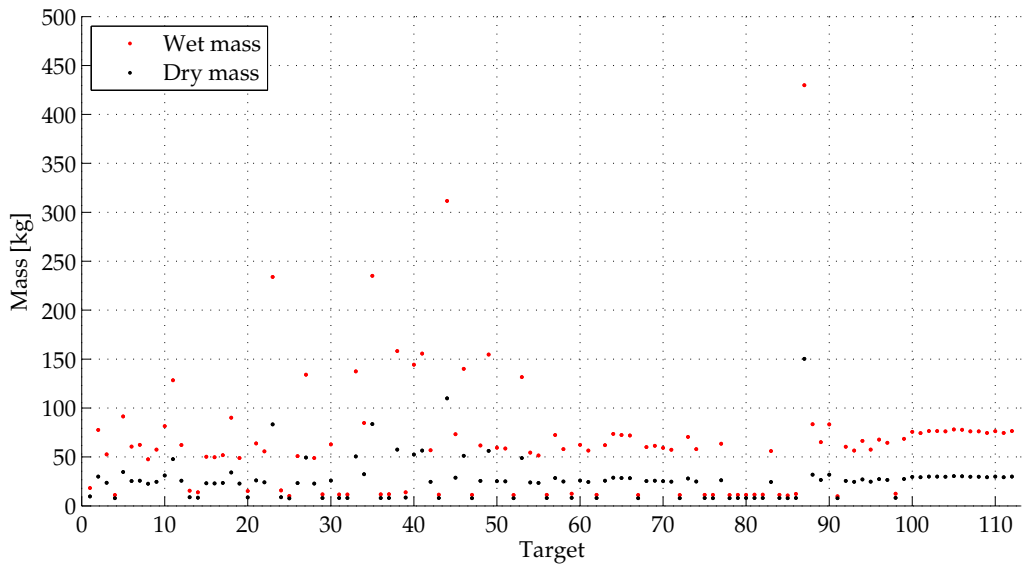

Figure 6.11: Mass of electric PFSC

Table 6.2 compares the use of an optimized versus a standardized PFSC architecture.

Table 6.2: Comparison between optimized and standardized propulsion

	Optimized	Standardized
Mean cost [M$]	3.52	3.53
Max cost [M$]	21.39	11.94
Total cost [M$]	394.97	395.36
Mean wet mass [kg]	64.62	92.69
Max wet mass [kg]	429.78	646.76
Total wet mass [kg]	7237.74	10,382.14
Mean lifetime [years]	2.21	2.58
Max lifetime [years]	4.91	22

The gain in cost is very small but the maximum lifetime and the wet masses are reduced with the 4 optimized architectures. This solution is therefore selected to be installed in the Mothership.

6.2.2 Design of Mothership spacecraft

Once the PFSC are optimized, the main vehicle, the Mothership Spacecraft (MMSC) is defined. The design variables are similar to the ones used for the Picker. One major difference is the addition of the number of targets visited by one MSSC, which goes from 2 to 8. The second one is the fact that it is not possible to launch more than one Mothership spacecraft at a time. The reason for this is that the combined mass of the spacecraft is too high to be launched by any existing launch vehicle.

Figure 6.12 shows how the objective is influenced by the minimum number of targets visited by each MSSC.

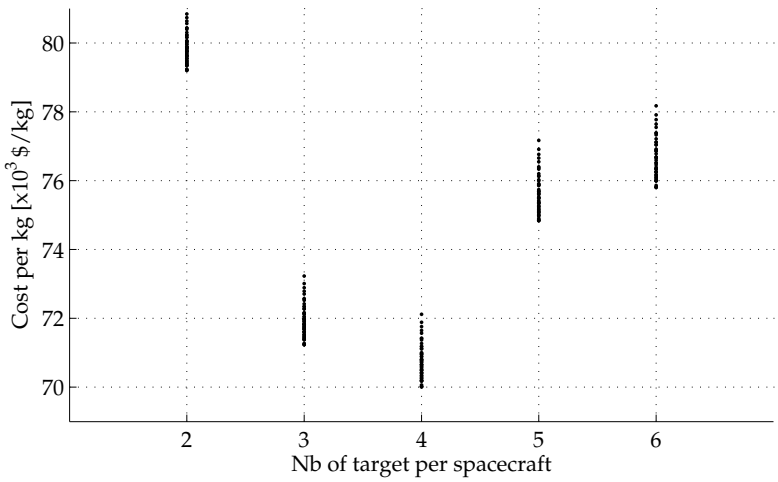

Figure 6.12: Number of targets per spacecraft and cost per mass

Only the feasible solutions are shown. The optimum is found for a minimum of 4 targets visited by one single spacecraft.

The same variables used in the Picker scenario are applied for the Mothership's propulsion system. Figure 6.13 shows the relation between the type of propulsion and the cost per mass.

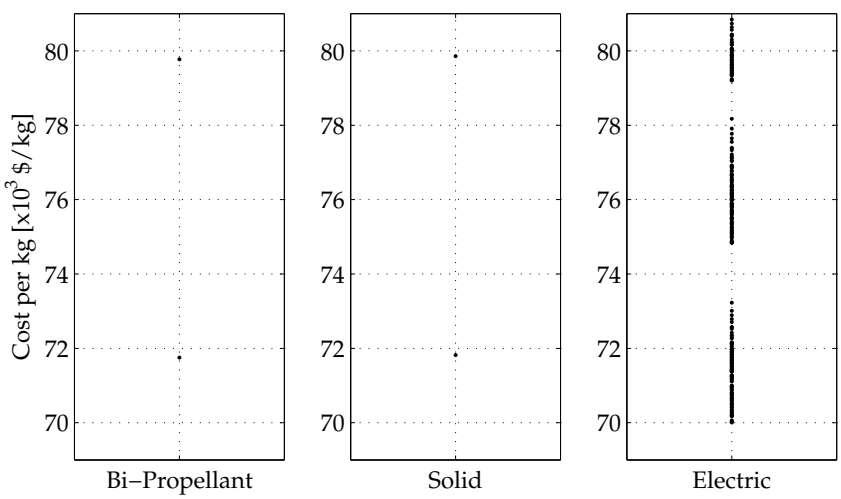

Figure 6.13: Propulsion type and cost per mass

The electric propulsion clearly appears to be more efficient than both types of chemical solutions. It is then studied more in depth to define the optimal Isp (Figure 6.14) and thrust (Figure 6.15).

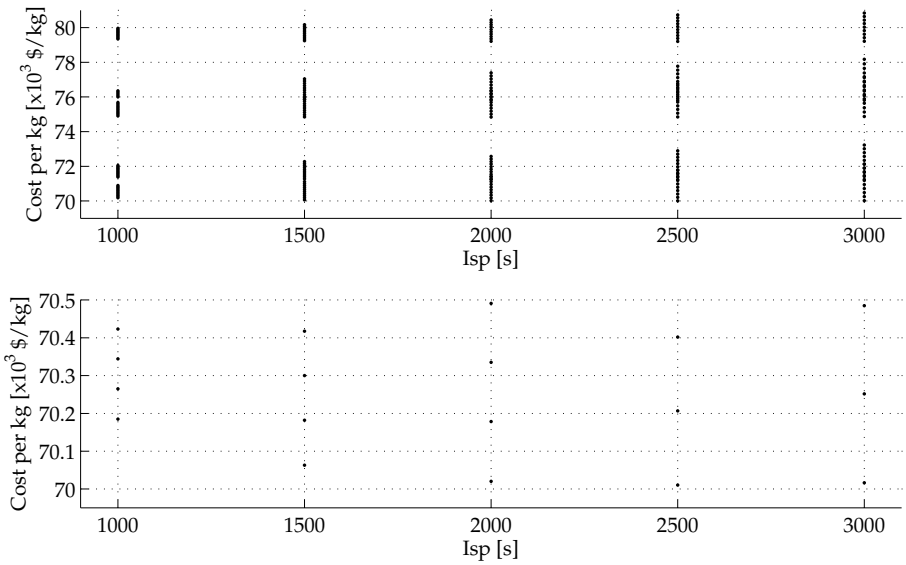

Figure 6.14: High Isp and cost per mass

As explained earlier and in the Appendix, the optimal Isp is not the highest one. In this case, the best solution depends on the selected thrust: 2500 [s] for 0.01 [N], 2000 [s] for 0.02, 1500 [s] for 0.03 [N] and 1000 [s] for higher thrust.

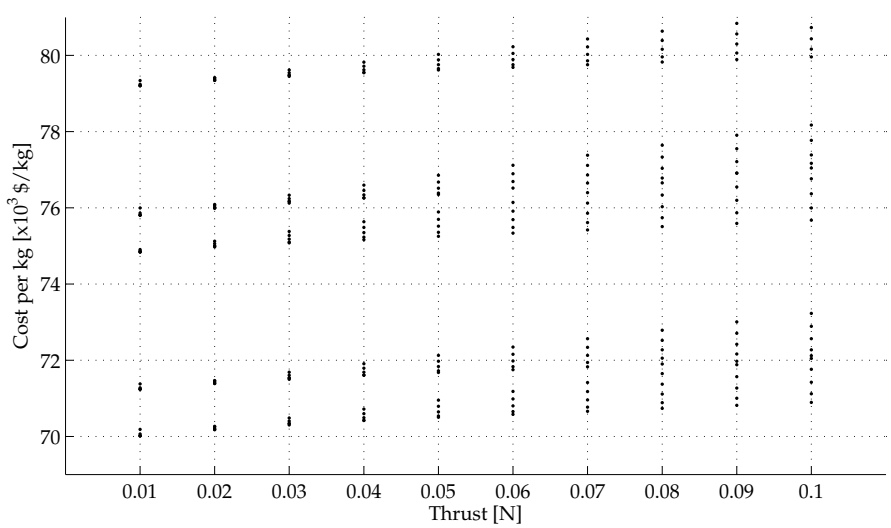

Figure 6.15: Low thrust and cost per mass

The minimal cost is, once again, achieved for the lower values of thrust. As explained in the section about the Picker scenario, the optimum is found after studying the mission durations.

6.2.3 Campaign duration and launch planning

As for the Picker scenario, the final architecture and the launch planning are driven by the 5-debris-per-year objective. Here again, a tradeoff is done between the cadence and the cost, the second one being more important. Figure 6.16 shows the maximum, average and minimum lifetimes for the different values of thrust.

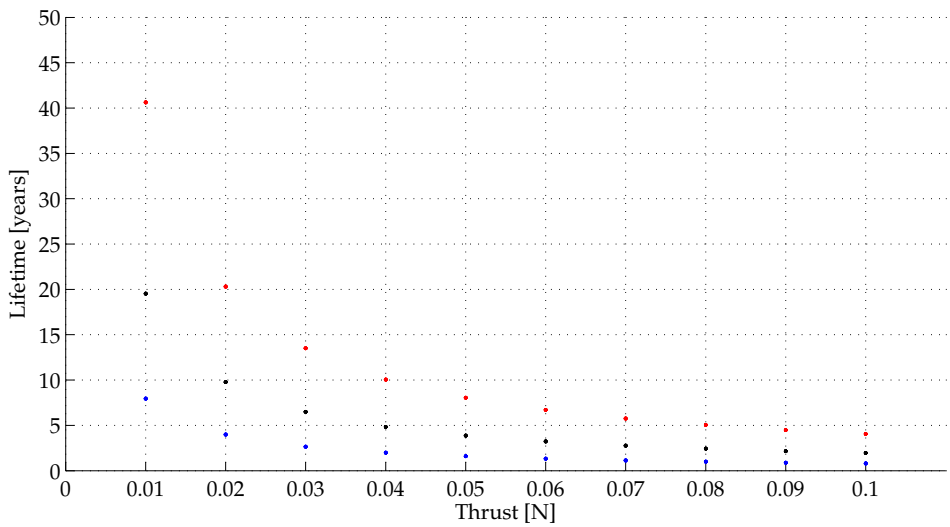

Figure 6.16: Lifetime and low thrust

Assuming one launch per year and selecting the 0.04-[N] thruster, with an average lifetime of 5 [years], all the 112 objects in SSO are de-orbited in 32 [years]. The cadence is 3.50 [debris/year] for a cost of 70,408 [$/kg]. Using the 0.05-[N] thruster allows a cadence of 3.54 [debris/year] but for a cost of 70,486 [$/kg]. Due to the cost increase and the very small improved in terms of cadence, the first one is used.

6.2.4 Overview of the selected architecture

As presented in the previous sections, the optimal architecture for the Mothership scenario removes the 112 targets from SSO in a maximum of 14.7 [years]. The removing cadence is at least of 7.6 [debris/year]. The MSSC are launched one by one and each of them is equipped with 3 PFSC. A total of 28 MSSC and 84 PFSC must be designed, tested and launched. The launch vehicle used is the Delta IV heavy.

The total campaign cost, as given directly by the Model, is of $11.49 billion, for a normalized value of 70,408 [$/kg]. As explained in the cost model section, this cost may be underestimated and a margin of 50% is added to be on the safer side. This lead to a campaign cost of $17.235 billion and a cost per unit of removed mass of 105,612 [$/kg].

Figure 6.17 shows the dry and wet masses of the MSSC used to accomplish the missions. The PFSC's masses are included in the dry mass of the spacecraft.

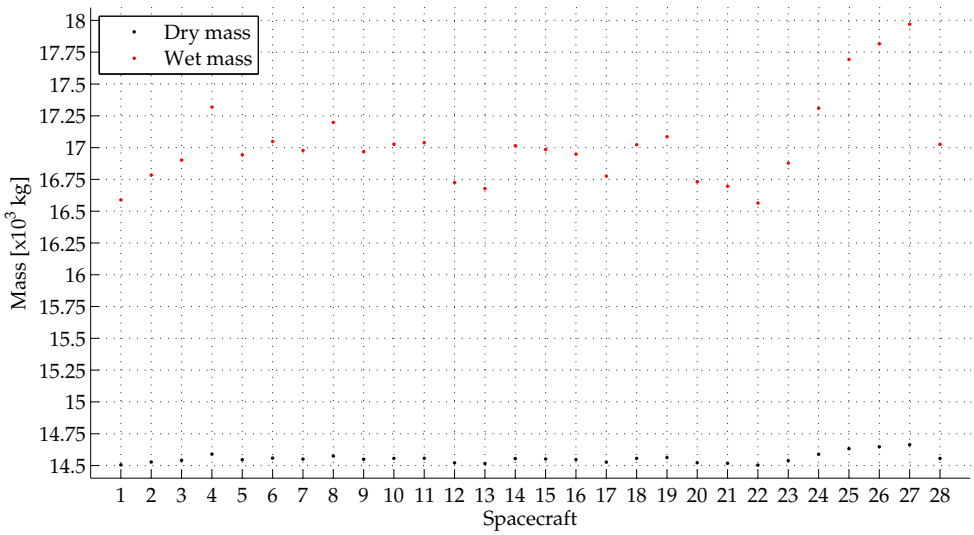

Figure 6.17: Wet and dry mass of MSSC

The variation of dry mass is sufficiently small to allow to simply reproduce the heaviest one and use it for every mission. Like this, 28 identical spacecraft are developed and built. The only difference is that the tanks of most satellites are fill only with the propellant required to perform the mission, and the necessary margin. Concerning the PFSC, optimized versions are used depending on the characteristics of the targets, as explained previously.

6.3 Shuttle scenario

Following the results for the Mothership scenario, the Shuttle is very unlikely to be an efficient solution with the conventional propulsion methods that are studied here. Indeed, while the Mothership leaves a part of its dry mass every time a PFSC is separated, the Shuttle has to carry its whole structure back and forth between SSO and the re-entry orbit, around 200 [km], where the target is released. Not only is this architecture very heavy and therefore costly, but it is also extremely slow. This because a debris must be carried all the way down to the re-entry orbit before the next one can be taken care of.

The average ΔV a Mothership Spacecraft has to achieve is 397.04 [m/s] and the maximum is 826.69 [m/s]. The average Pilot Fish Spacecraft produces $\overline{\Delta V} = 342.80$ [m/s] and the one with the longest travel has to provide 691.65 [m/s]. Because the Shuttle should be designed to catch the same targets, the overall ΔV for the campaign is, at least

$$(2N_t - Ns) \cdot \overline{\Delta V} \tag{6.3.1}$$

where N_t is the number of target and N_s is the number of shuttles. The minimum ΔV is achieved when a maximum of spacecraft is used. This means 56 shuttles for the 112 objects in SSO. The average ΔV per shuttle is then 1,028.4 [m/s]. This is less than the 1,425.5 [m/s] for a MSSC and its 3 PFSCs but the Shuttle has to carry its whole dry mass and the rocket bodies while this task is taken care of by the much lighter PFSC in the case of the Mothership.

Despite being less efficient with conventional propulsion, the Shuttle would greatly take advantage of the space tethers. This is actually the architecture suggested by Tether Inc. with EDDE.

6.4 Architecture selection

The results presented in the previous sections clearly leads to the selection of the Picker scenarios, at least within the studied design space. Not only is this scenario the cheapest and the fastest one but it is also massively parallel and takes advantage of a very conventional and available launch vehicle, the russian Soyuz. A summary of these architecture and the comparison with the discarded Mothership are presented in Table 6.3.

Table 6.3: Selected architecture after the technical and economic study

Scenario	Picker	Mothership
Target to remove	112	112
Dry mass	310 [kg]	14,660 [kg]
S/C per launch	10	1
Launch per year	1	1
Launch vehicles	11 Soyuz, 1 Kosmos	28 Delta IV heavy
Isp	1000 [s]	1000 [s]
Thrust	0.01 [N]	0.04 [N]
Cost per mass removed	10,407 [$/kg]	105,612 [$/kg]
Nominal campaign duration	13.74 [years]	32 [years]
Nominal cadence	8.15 [debris/year]	3.50 [debris/years]
Maximum campaign duration	15.90 [years]	42 [years]
Minimal cadence	7.04 [debris/year]	2.66 [debris/year]

6.5 Debris due to the ADR missions

The three scenarios presented in this report make the same assumption: the launch they required and their in-orbit operations do not create any new debris. However it is theoretically true for the later[1], it does not take into account that the upper stage used to launch the removing spacecraft. In the case of the Picker, because the parking orbit is very low in this case (around 250 [km] above Earth), the decay time of the rocket body and of the structure used to maintain the spacecraft together is quite short. However, if the Mothership or the Shuttle are to be used[2], the upper stage will release its payload on a much higher orbit and may take many tens of years to re-enter the Earth atmosphere.

Fortunately, the Pilot Fish Spacecraft presented in the section about the Mothership can be used to de-orbit future upper stages at a reasonable cost. In this case, there is not the need for any Mothership spacecraft: the PFSC can be fixed on the rocket stage before the launch. Once the payload is injected on its first orbit, the remaining propellant in the attitude control system of the upper stage can stabilize and orient it for a retro-boost. When the body is in position, the PFSC is ignited and the debris is de-orbited. This system can be used not only in SSO but at every inclination in LEO. With typical cost around $4 millions, they increase the cost of a launch of about 10% and because of the additional mass that has to be added to the rocket, they reduce the maximum capacity of the launcher. However, this is considered to be a very efficient way to de-orbit future upper stages only a couple of years after they injected their payload. In any case it is much better than the 25-year rule suggested by the agencies and it can be applied to any rocket body used in LEO.

[1] Because a robotic arm is used, the target is stays intact until its re-entry if the mission is performed nominally.
[2] For example, if the development of the tether technologies allows their use in the future.

7 Economic and political model

7.1 Motivation

Once the general characteristics of the removing spacecraft are known a new problem appears: who will pay (and how much) for this expensive and, at first glance, not profitable campaign? In a first time, the main actors are detailed. They obviously include the countries responsible for the biggest contribution in the debris population. The present and future users of the near-Earth orbits must also have some contribution in the funding. In a second time, a comparison is made between two scenarios:

Global program

In this case, all the nations included in the study collaborate for the development, launch and operation of the removing spacecraft. The cost are spread among the actors in according to their participation to the orbital debris population.

National programs

On the other side, the cost study and the financing model are done in the case where the different nations develop and fund their own system.

The targets in SSO are taken into account to restrain the problem while having a representative population in terms of mass in orbit and plurality of the nations.

7.2 The main actors

For this study, three nations, or group of nations, only are taken into account: the United States, Europe and Russia. They already have a long history in terms of collaboration with the ISS being the most important example. They are the biggest contributors to the population in SSO with 80 targets out of a total of 112 upper stages. Only these 80 targets are taken into account.

Other actors are all the companies, governments and universities developing payloads and spacecraft and launching them with the rockets from the three countries. Because they are the reason why material was, is and will be sent up to Space, it seems fair that they should pay to clean it. Paying for this will allow these users to reduce the risk of seeing their payloads being destroyed by a collision and keep possible the use of all the orbits around the Earth.

The insurance companies that are covering the spacecraft in the case of a malfunction could also have a role to play in this. However, for the time being, the amounts they ask for for the risk caused by orbital debris is not significant when compared any other ones (malfunction of the payload itself or the launch vehicle, etc.)

7.3 Global and national programs

7.3.1 Optimal scenario and spacecraft

The technical characteristics of the removing spacecraft are taken directly from the simulation presented earlier in this report. The Picker is obviously the most efficient way to remove targets in SSO. The solution with an Isp of 1000 [s] and a thrust of 0.01 [N] is also the best in terms of cost for all the subsets studied in the following sections. The cadence of de-orbiting is tested for each program.

If the scenario and technical variables are taken directly from the simulation previously presented, the number of spacecraft per launch is re-optimized as the available rockets depends on the program. The international program uses Kosmos, Vega, Soyuz, Delta IV medium(+) and Delta IV heavy. The U.S. program is limited to Athena, Delta II, Delta IV medium(+) and Delta IV heavy. The russian program allows only Kosmos and Soyuz. The european program takes advantage of Vega and Ariane 5.

7.3.2 Global solution

Return on investments

In the real world, the collaboration would include concrete benefices for the countries participating in the program. These can be the use of a fair percentage of launch vehicles from each partner, the distribution of the development, production and operations tasks among companies from each country, etc. This is done to ensure that every government participating in the campaign receive a fair return on its investment, namely: knowledge, industrial contracts, academic projects, etc. However, to avoid increasing too much the complexity of this study, this is not taken into account. The distribution of the tasks among the collaborators and the restriction in the use of specific launchers and launch sites are not considered as constraints. In the same idea, the order of removal is not influenced by any political rule.

Campaign cost and planning

In SSO, the american, russian and european targets represents a total of 80 objects for a combined mass of 97,943 [kg]. Figure 7.1 shows the optimization of the number of removing spacecraft per launch.

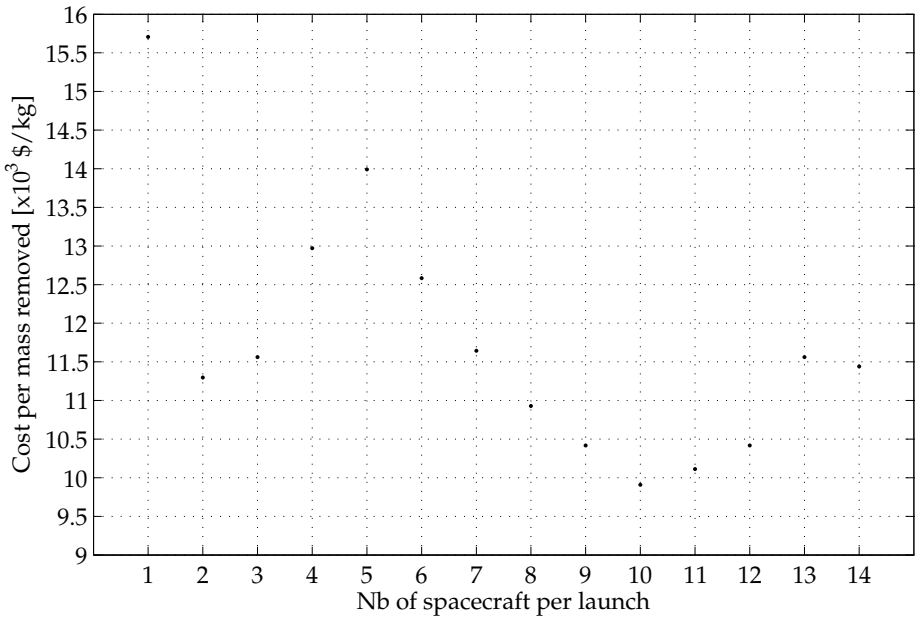

Figure 7.1: Spacecraft per launch (global)

The optimum is found when 10 spacecraft are launched together. 8 Soyuz are required for the campaign and the cost per mass removed is as low as 9,907.8 [$/kg]. The average and maximum lifetimes of the spacecraft used in this program are respectively 2.3 [years] and 6.3 [years]. Assuming 2 launches per year, the maximum campaign duration is of 9.8 [years] with a cadence of 8.16 [debris/year]. This is achievable thanks to the excellent availability and reliability of the Soyuz: 15 of them where launched in 2011 from Baikonur and Kourou.

This global program presents performances very close to what is demonstrated in the previous chapter with a total cost of $970.402 millions.

7.3.3 National programs

The same types of spacecraft are used by the different countries but the information about the development and the manufacturing processes are assumed to not be shared. Thus, the development cost are fully paid by each country and the learning curve reduction is not applied across the countries. Here again, the number of spacecraft per launch is re-evaluated.

USA

The american targets are the most present in SSO: there are 49 objects for a total mass of 23,046 [kg]. The optimization of the number of spacecraft is shown in Figure 7.2.

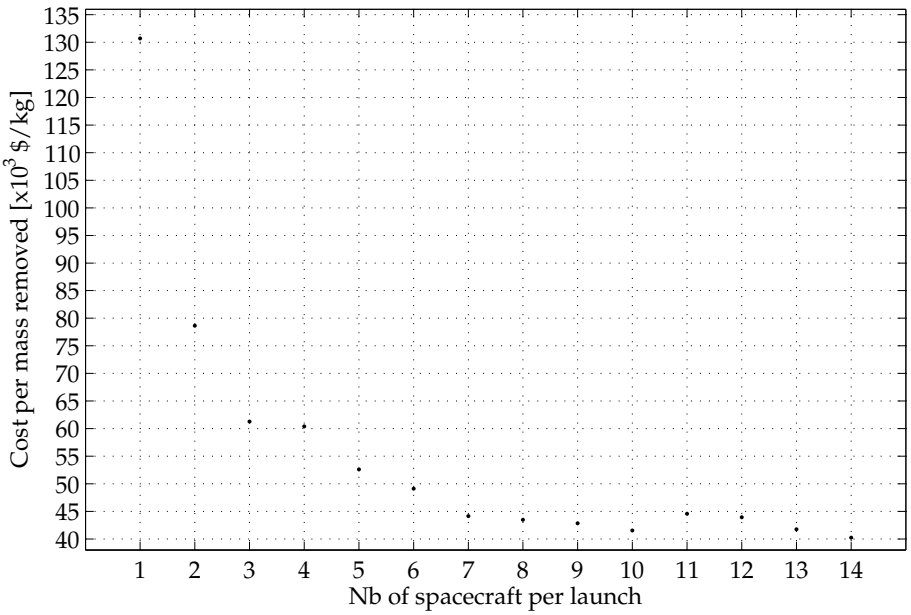

Figure 7.2: Spacecraft per launch (USA)

The optimal solution is achieved with a maximum of 14 spacecraft per launch and required the use of 3 Delta IV medium(+) (14 pickers per launcher) and 1 Delta II (7 pickers). The average spacecraft needs 1.3 [years] to de-orbits its target and the slowest one takes 6.0 [years]. If one launcher is sent per year, the campaign lasts 8 [years] at most and allows a cadence of 5.44 [debris/year]. This seems to lack a reasonable margin compared to the 5 [debris/year] asked by NASA but the advantage of the national program is its parallelism. Indeed, while the U.S. launch their de-orbiter, Russia and Europe are assumed to do the same.

This program costs $927,144 millions, which represents 40,230 [$/kg]. This is 95.5% of the cost of the global mission for only 23.5% of the mass removed.

Russia

Russia is the biggest contributors in SSO in terms of mass with 53,190 [kg] but it is condensed in only 21 objects. The cost per mass as a function of the number of spacecraft launched at once is shown in Figure 7.3.

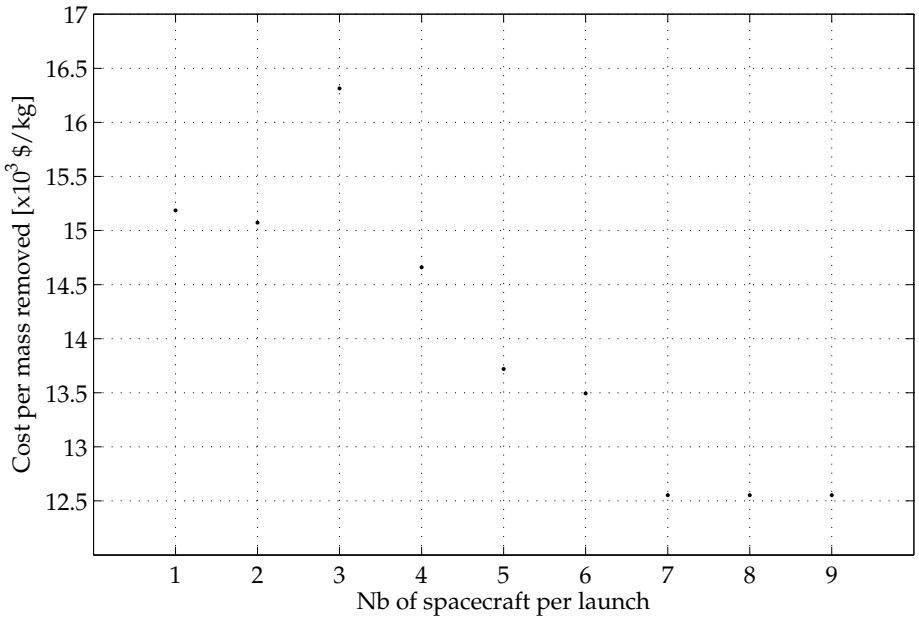

Figure 7.3: Spacecraft per launch (Russia)

The optimal campaign requires the launches of 3 Soyuz: 2 with 8 picker spacecraft and 1 with only 5. The average mission time is 1.6 [years] and the longest one is 6.8 [years]. As for the global program, 2 Soyuz can safely be launched per year to achieve a cadence of 2.69 [debris/year]. This is far below the performance hoped by NASA. However, only the 3 very heavy SL-16 rocket bodies take more than 2 [years] to be removed. Moreover, as explained earlier, the american and european activities are expected in ADR to reach a higher global cadence.

The cost for the russian campaign is $667,300 millions with a cost per mass of 12,554 [$/kg]. If not the fastest one, the russian program is the most efficient in terms of cost per kilogram, although not as good as the global campaign.

Europe

Europe is the fourth contributor in SSO, after the U.S., Russia and China, with 10 Ariane upper stages for a combined mass of 21,707 [kg].

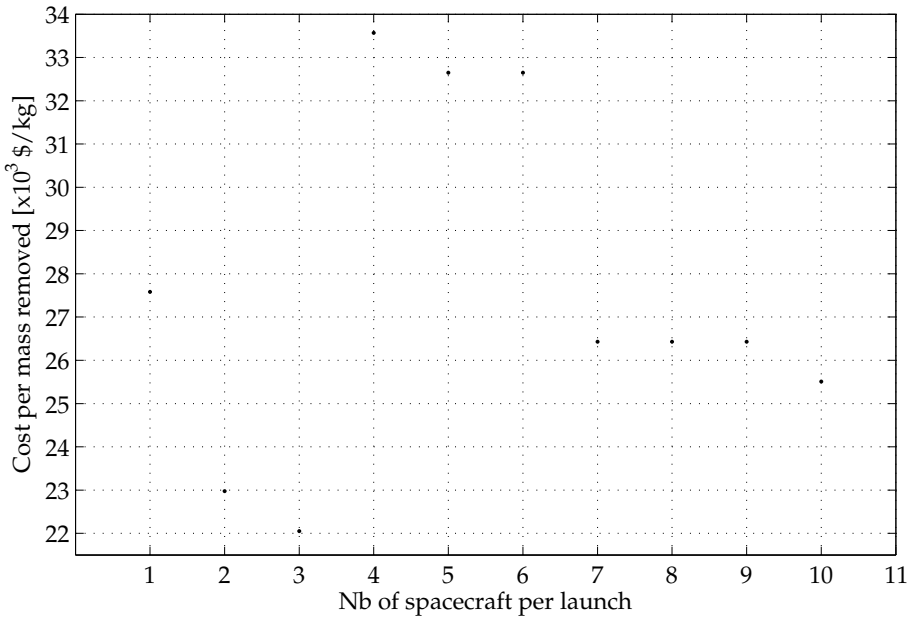

Figure 7.4: Spacecraft per launch (Europe)

Although the 10 pickers could fit in one single Ariane 5, using 4 Vega with 3 times 3 spacecraft and a single picker sent alone is the best options. The european missions are the fastest with an average duration of 1.6 [years] and a maximum of 2.6 [years]. With one launch per year, the total for the campaign is of 5.6 [years] and allows a cadence of 1.78 [debris/years]. However, assuming that all three campaigns start together, the 80 targets selected in SSO are removed in a maximum of 8 [years] (duration of the american campaign) for an overall cadence of 10 [debris/year].

The cost for this campaign is $478.699 millions, hence a cost per mass removed of 22,503 [$/kg].

7.3.4 Comparison

Adding the cost of the separated campaigns in the United States, Russia and Europe is $2,073.1531 million. The collaborative solution leads to a campaign cost of $970.402 million. The difference is then of $1,102.751 million and it appears that the global campaign is more advantageous.

To determine the separation of the cost among the governments, two ways come to mind: either the cost is distributed as a function of the number of debris or based on the mass to remove. Both these methods are detailed in Tables 7.1 and 7.2.

Table 7.1: Distribution of cost based on number of debris

Country	Targets	Percentage	Cost (global) [M$]	Cost (national) [M$]	Difference [M$]
USA	49	61.25	594.37	927.14	-332.77
Russia	21	26.25	254.73	667.30	-412.57
Europe	10	12.5	121.30	478.69	-357.39
TOTAL	80	100	970.40	2,073.15	-1,102.75

Table 7.2: Distribution of cost based on mass of debris

Country	Mass [kg]	Percentage	Cost (global) [M$]	Cost (national) [M$]	Difference [M$]
USA	23046	23.53	228.33	927.14	-698.80
Russia	53190	54.31	526.99	667.30	-140.31
Europe	21707	22.16	215.06	478.69	-263.63
TOTAL	97943	100	970.40	2,073.15	-1,102.75

Both methods allow every partner to save hundred of millions dollars however the choice of the methods can turn to very aggressive debates. Indeed, the U.S. will definitely prefer the second one where they have much less to pay while Russia and Europe will clearly take advantage of the first one. These solutions look at the problem from two different points of view.

The first one considers that the countries have to pay for the fraction of the cost represented by the number of removing spacecraft to build. Indeed, because one target is removed by one picker, a russian target should be removed by a system built mostly with russian money.

The second one is based on the fact that the ones who polluted Space the most have to pay more. In this case, even though Russia has less than half the number of debris the U.S. have, it causes more risk as if one of its huge upper stages is broken it will create more dangerous mass than an american one.

To try to settle this, and because both points of view are valuable, a parameter, α, is introduced to define the participation of the cost for each country. The fraction of the global cost for the country i is now define as

$$F_i = \alpha \cdot \frac{M_i}{M_{tot}} + (1-\alpha) \cdot \frac{N_i}{N_{tot}} \tag{7.3.1}$$

$$F_{\text{USA}} + F_{\text{Russia}} + F_{\text{Europe}} = 1 \tag{7.3.2}$$

where M_i is the mass of debris generated by the country i, M_{tot} is the total mass of the targets, N_i is the number of targets due to the country i and N_{tot} is the total number of targets.

Figure 7.5 shows the amount every country has to pay as a function of this α-parameter.

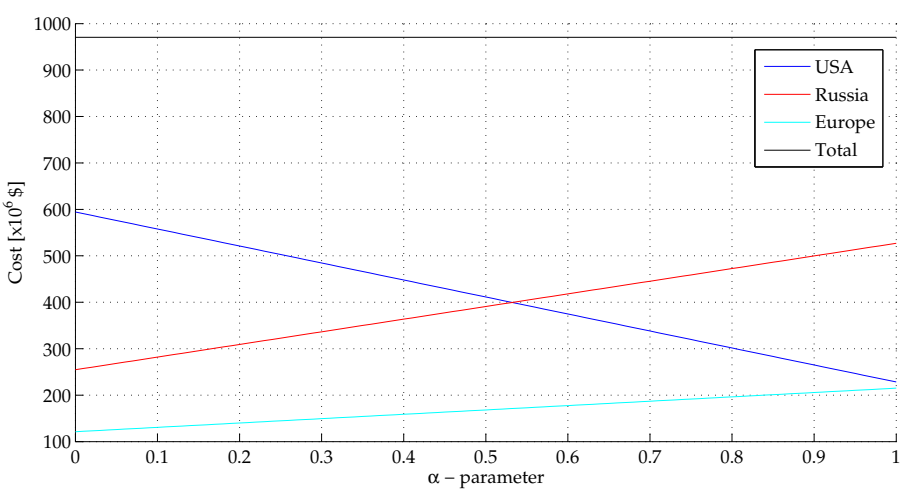

Figure 7.5: α-parameter and cost per country

The optimum is assumed to be $\alpha_0 = 0.532$ where the U.S. and Russia have both to pay the same amount. Europe having much lower cost, it is not seen as an active variable in this problem. With the two most important Space countries having to contribute in an identical way, the problem can be seen as equitable. In this case, America pays for its numerous rocket bodies while Russia contributes for the high mass. The campaign cost for each country, with a margin of 50% are presented in Table 7.3.

Table 7.3: Cost contribution per country

Country	Cost [M$]	Contribution [%]
USA	599.40	41.18
Russia	599.40	41.18
Europe	256.79	17.64
TOTAL	1,455.40	100

However, the U.S. and russian governments and the European Union do not have to cover the whole cost with public money. That is why a fee should be paid by any government, company or university wanted to access to Space. This is defined in the next section.

7.4 Propositions of financing model

7.4.1 Sources of funding

The financing of this campaign may come from two different sources

1. The users of the near-Earth environment.
2. The countries themselves.

The contribution of the first source will ensure the safety of Space in the future and help reducing the risk of a collision for their own payloads. This money can be taken in the form of a fee added to the launch cost. Because it wouldn't be fair of having the present and new users paying for the mistakes made in the past, the governments should also participating in the cost. The contribution of this second source could be either included in the national Space agencies budgets and used directly to help with the development, manufacturing and operations (Figure 7.6) or taken from this budget and given to an entity responsible for the removing campaign (Figure 7.7). These structures are very similar to the ones presented by RetroSpace in 2010 [21].

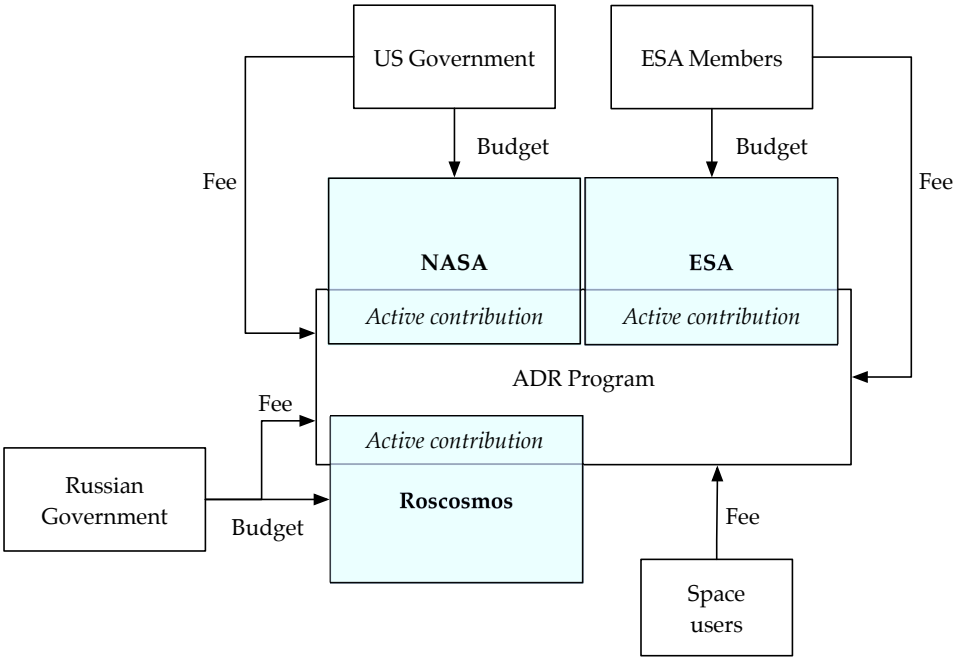

Figure 7.6: Active participation of the Space agencies in the ADR program

Having the national Space agencies directly involved in the process can be a good thing since they can bring their own knowledge and people at every levels. However, this implies an enormous amount of work to organize and distribute the work load. NASA and ESA projects are known to take a very long time due not only to the technical challenges but also because of the difficulty of managing a very distributed design process.

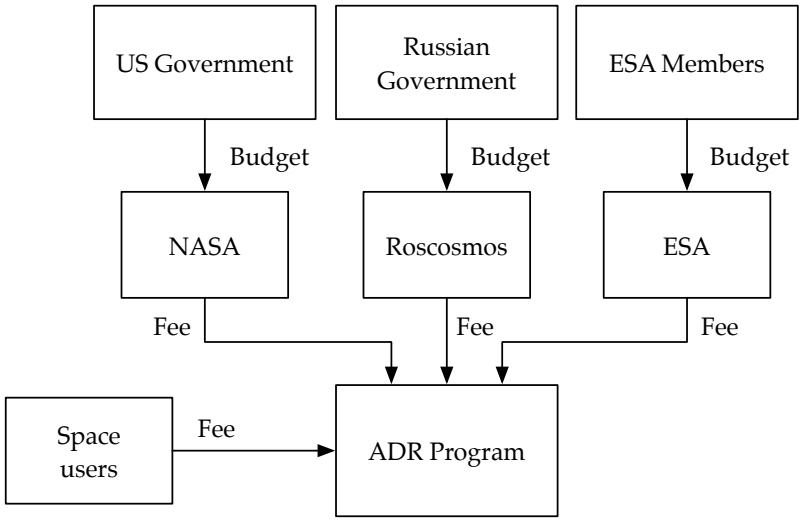

Figure 7.7: Financial participation of the Space agencies

The recent results achieved by SpaceX with their Falcon 9 launcher and Dragon capsule show the advantages of regrouping all the aspects of the development not only within one entity but also in one physical place. This would be very hard to achieve with the Space agencies and all the industrial partners sharing the work load and wanting to work in their usual ways and facilities. Due to the short timeline of this project, this part of the problem is not studied any further but should definitely be part of a future work.

7.4.2 Contribution from each source

Evaluation approach

The best way to make the users and the Space agencies paying for the ADR program is to tax them when they use a launch vehicle. To estimate the launches in the future, the data from 2011 [!REF] are taken as a reference and assumed to stay the same. Table 7.4 gives the use of the different launch vehicle for the U.S., Russia and Europe between October 2010 and September 2011.

Table 7.4: Launches in 2010-2011

Country	Launcher	Number
USA	Atlas V	4
USA	Delta II	3
USA	Delta IV	4
USA	Falcon 9	2
USA	Minotaur	4
USA	Taurus	1
TOTAL		18
Russia	Dnepr	1
Russia	Kosmos	1
Russia	Proton	9
Russia	Rockot	1
Russia	Soyuz	15
Russia	Zenit	2
TOTAL		29
Europe	Vega	1
Europe	Ariane	8
TOTAL		9

Cost per launch

The full campaign duration, as shown earlier, is about 10 [years]. If the development starts in 2013 and the first launch is assumed to happen in 2020, the full program lasts 17 [years]. The ideal would be to have all the actors paying during the development period to make sure the budget is equilibrated when the first launch occurs. However, this would require a huge amount of money in quite a short time. It is more reasonable to aim for a full return on investment at the end of the campaign. In this case, $1,455.59 millions dollars have to be paid in 17 [years], which means a cost per year of $85.65 millions. This budget has to be balanced between the U.S., Russia and Europe as shown in Table 7.3. All the launch vehicles presented in Table 7.4 are assumed to be used at their full capacity (for LEO) and the cost is weighted according to the capacity of the rocket. Table 7.5 shows the estimated total mass to be launched per each country, the distribution of cost among these launches and the cost per launch for each country. The expected number of launch is the one that happened in 2011 multiplied by the 17 years the full program is assumed to last.

Table 7.5: Distribution of cost per launch and per country

Launcher	Capacity [kg]	Launches	Mass [kg]	Fraction	Cost [M$]	Cost per launch [M$]
Atlas V	20,520	68	1,395,360	0.381	228.47	3.36
Delta II	5,102	51	260,202	0.071	42.60	0.84
Delta IV	23,260	68	1,581,680	0.432	258.97	3.81
Falcon 9	10,450	34	355,300	0.097	58.17	1.71
Minotaur	607	68	41,276	0.011	6.76	0.10
Taurus	1,590	17	27,030	0.007	4.43	0.26
TOTAL		306	3,660,848	1.000	599.40	
Dnepr	300	17	5,100	0.001	0.55	0.03
Kosmos	1,500	17	25,500	0.005	2.76	0.16
Proton	21,000	153	3,213,000	0.580	347.94	2.27
Rockot	1,950	17	33,150	0.006	3.59	0.21
Soyuz	7,000	255	1,785,000	0.322	193.30	0.76
Zenit	13,920	34	473,280	0.086	51.25	1.51
TOTAL		493	5,535,030	1.000	599.40	
Vega	1,395	17	23,715	0.018	4.63	0.27
Ariane	9,500	136	1,292,000	0.982	252.16	1.85
TOTAL		153	1,315,715	1.000	256.79	

The cost per launch presented in Table 7.3 can be seen as an "ADR tax". Because the countries are assumed to be responsible for the rocket upper stages they launched in the past, they have to pay this tax whenever they send a new launcher. However, when the launcher is used to place a payload in SSO, the situation is a bit different. In this case, the payload operator would benefit from the ADR program. Because the money of the tax is used to remove debris in SSO, it makes the area safer and then decreases the risk of collision. The operator is then asked to pay the tax corresponding to the mass it places in orbit. This allows to have the Space users to participate fairly in the ADR program and to reduce the amount of money asked from the governments.

The case of Soyuz is given as a concrete example. If one of the russian rocket is used to deliver a 7000-[kg] payload to LEO at an inclination of 50°, the satellite operator would not benefit from the ADR program happening in SSO. The $760,000 of the ADR tax are then entirely paid by the russian government. If now a second Soyuz is launched to deliver a 4300-[kg] payload to SSO, the operator would benefit from the ADR program so he should participate and pay the fraction of the fee corresponding to the mass of his payload with respect to the full capacity of the Soyuz. Specifically in the case of this second Soyuz, the operator would pay $466,857 and the russian government would cover the remaining $293,143.

In the general case, the tax that should be paid by the user is given by

$$T_{user} = \frac{M_{SSO}}{M_{CAP}} \cdot T_{LV} \tag{7.4.1}$$

where M_{SSO} is the mass of the payload injected in SSO and M_{CAP} is the maximum capacity of the launcher. This last mass is usually the one that can be injected at an altitude between 185 and 200 [km] and at an inclination that corresponds to the latitude of the launch site. T_{LV} is the nominal tax as shown in the las column of Table 7.3. In the case when the launch vehicle is not used to inject mass in SSO, the tax is fully paid by the country of the launcher.

8 Conclusions

8.1 Summary of the technical study

The first simulations for the whole target population in SSO show a clear advantage when the Picker scenario is used. Its very simple configuration and the small orbital transfers required to accomplish a mission allow the fabrication of light spacecraft. The small difference of dry mass between the removers needed in the campaign allows a very effective industrial production. The best propulsion methods is by using electric thrusters with minimum Isp (1000 [s]) and thrust (0.01 [N]) to avoid the need for a heavy and costly power management subsystem. Even with the lowest thrust tested during the simulations, the mission and campaign durations stay relatively short and permit cadences that are compliant with the guidelines given by NASA. With less than 10,500 [$/kg], the selected architecture for this scenario is very efficient in terms of cost and the cheapest of all the ones that were tested during this project.

The Mothership scenario is clearly less efficient with much bigger spacecraft that cannot be launched except with the heaviest launch vehicles. The best architectures are about 10 times more expensive than the ones taking advantage of the Picker scenario. Both these results lead to the conclusion that this scenario should be discarded for the time being. However, further studies may make it more attractive. Firstly, the path between the targets should be optimized by taking the propellant mass into account, not only the required ΔV. This implies a much more complicated optimization problem but it could allows to reduce the mass and the cost. Secondly, ways to reduce the mass of the PFSC must be found. The best one could be to use Space tethers to avoid the need for propellant and therefore having a PFSC's mass independent from the mass of the target. Thirdly, electric propulsion could still be used but the heaviest debris must be removed from the target population. For example, this scenario could be used to remove any type of debris with a mass between 10 and 100 [kg].

Although the Mothership scenario was discarded, the analyze was not worthless. Indeed, it shows that the PFSC, if installed prior to launch can be a very efficient and cheap way to de-orbit future upper-stages. This will also help in reducing the mass of junk in orbit around Earth in the next years. These de-orbiting units could be installed on existing rocket bodies or being fully integrated in new stages to be even more efficient.

The Shuttle was not studied in details due to the huge ΔVs it has to achieve and the mass that has to be carried by one single spacecraft. As for the Mothership it would have a lot to gain from the tethered system and if used to remove smaller debris.

The selected architecture, without taking the international policy constraints into account is given in Table 8.1.

Table 8.1: Selected architecture after the technical and economic study

Scenario	Picker
Targets to remove	112
Dry mass of one spacecraft	310 [kg]
S/C per launch	10
Launches per year	1
Launch vehicles	11 Soyuz, 1 Kosmos
Isp	1000 [s]
Thrust	0.01 [N]
Cost per mass removed	10,407 [$/kg]
Nominal campaign duration	13.74 [years]
Nominal cadence	8.15 [debris/year]
Maximum campaign duration	15.90 [years]
Minimal cadence	7.04 [debris/year]

All the spacecraft are the same to allow a simplified manufacturing process.

8.2 Summary of the policy study

After the Picker was selected as the best candidate for ADR in SSO, it was found that the characteristics of the propulsion subsystem were not changed if the full target catalog was replaced by one of its subsets. However, due to the change in the availability of the launch vehicle, the number of spacecraft per rocket is different for the global and national programs. Once an optimum was found for this variable, the study clearly shown that the U.S., Russia and Europe can avoid spending several millions of dollars by developing a collaborative ADR program.

The optimal solution in the case of a global program is given in Table 8.2.

Table 8.2: Selected architecture for the global ADR program

Scenario	Picker
Targets to remove	80
Dry mass	310 [kg]
S/C per launch	10
Launch per year	2
Launch vehicles	8 Soyuz
Isp	1000 [s]
Thrust	0.01 [N]
Cost per mass removed	14,859 [$/kg]
Maximum campaign duration	9.8 [years]
Minimal cadence	8.16 [debris/year]

The cost of this program have to be distributed between the countries as a function not only of the number of targets but also of the mass of debris. The total cost to be paid in total by each country is only a fraction of the annual budget of their national agencies and it can also be shared with the satellites manufacturers and operators who want to send their payload in SSO. The fee each of this user has to pay in only a fraction of the regular launch cost and will help to ensure the safety of their systems from BOL to EOL.

8.3 Future work

Technologic model

The first improvement could be to develop each subsystem module more in depth to extend the design space and analyze more architectures. The ADCS is a very important part of the problem due to the necessity to rendezvous with a non-compliant object. It does not mean that the full control laws must be implemented in this high-level model, but it could be a good idea to test different types of sensors and actuators to design a more detailed spacecraft. Similarly, the level of on-board autonomy could be implemented to see its influence on the cost.

Adding specific models for the Space tethers would allow to really study the state-of-the-art technologies and finalize the link between the subsystem development and a full architecture study. It could also help making the Mothership and Shuttle scenarios more appealing. The development of the model would imply a better knowledge of the electro-magnetic environment around Earth and the typical performances in terms of charges exchange.

The optimization of these multi-target spacecraft should also be brought one step further with a path optimization with respect to the mass, the cost and the lifetime, rather than only the ΔV. It implies a very complex model and a much wider design space in which the order of visit and the distribution of the targets among the removing spacecraft are design variables.

Other orbits should be studied, such as GTO or even GEO that presents both an considerable population of debris but also are very important commercially speaking.

Economic model

If subsystems are to be defined more in depth, it would be important to verify that the cost models still give coherent results. In general, it would be critical to develop cost models that are specific to the ADR problem. This is especially true for what concerns the vision and capture subsystems.

A cost model for the Space tethers should be developed if they are to be fully integrated within the module. This would ensure that the comparison are made on safe bases.

International policy and funding

The concrete applications of these rules and fees must absolutely be detailed. This thesis simply assumes that the U.S., Russia and Europe will be willing to pay millions of dollars to finance the ADR program but it is far from being put in place. A more politically-oriented research should consider the tasks to go from the actual situation to the ideal one presented in this report.

In the same idea, the distribution of cost among the countries could be improved and other countries such as China or India could also be added to the Model.

Use of Pilot Fish

As briefly explained in this report, the PFSC from the Mothership scenario could be used on its own to de-orbit future rocket bodies. A research fully dedicated on this application could be done to better select the best design options and define actual ways to fix it on the rocket. It could also be good to compare a system developed by a third company and sold to the launch operator and a version fully integrated within the rocket and developed by the rocket builder.

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Benoit Chamot

Cambridge, MA, August 17, 2012

A Appendix

A.1 Pre-determined subsystems

Apart from the propulsion and the power subsystems that are fully sized by the design loop, the structure mass which is defined as a fraction of the dry mass, and the propellant used by the ADCS to perform the rendezvous maneuvers, the other subsystems of the removing spacecraft are assumed to be mission-independent. Their masses and power consumptions are taken directly from similar architecture such as DEOS [16], Orbital Express [19] or DR LEO [20]. Their cost are determined thanks to the models presented in [26].

Table A.1 presents the values used as parameters in the Model.

Table A.1: Pre-defined mission-independent subsystems

Subsystem	Mass [kg]	Power consumption [W]	Cost RDT&E [FY2012k$]	Cost TFU [FY2012k$]
ADCS	30	22.8	10,850	5,045
TTC&DH	15	40	4500	3200
Thermal	20	40	3,300	520
Payload	100	140	36,000	53,000

The telecom subsystem is included in the TTC&DH subsystem.

A.2 Other cost models

The subsystems are not the only aspect of a mission that requires money. The integration, assembly and testing, the ground segment, the launch and in-orbit operations and the project management participate to the overall cost. Table A.2 gives the models, taken from [26], used to evaluate these cost.

Table A.2: Pre-defined mission-independent subsystems

	Parameters	Cost RDT&E [FY2012k$]	Cost TFU [FY2012k$]
IA&T	Subsystems cost C_s, Wet mass M_w	$989 + 0.215 \cdot C_s$	$10.4 \cdot M_w$
Program level	Subsystems cost C_s, TFU cost C_t	$1.963 \cdot C_s^{0.841}$	$0.341 * C_t$
Ground segment	Subsystems cost C_s	$9.262 \cdot C_s^{0.642}$	-
Operations	Wet mass M_w	-	$4.9 \cdot M_w$

All these cost models have outputs in FY2000k$. The conversion to FY2010k$ is done my multiplying by a factor 1.225.

A.3 Relation between Isp and power subsystem

A tendency shown in the results may be counter-intuitive: the fact that the cost increase with the Isp. The common sense would say that because of a higher Isp, the propellant mass is smaller thus lowering the overall cost. But because the cost are studied at the subsystem level two things happen when the Isp increases:

1. The mass of propellant does decrease thus decreasing the mass of tanking and the cost of the propulsion subsystem.
2. In the case of electric propulsion, the power demand increases with the Isp, thus increasing the cost of the power subsystem.

Therefore the changes of mass and cost of both these subsystems have to be studied to understand the increase of cost with the Isp. A simple simulation is done in which the Isp is increased from 1000 to 4000 [s] while the following parameters are used

- A constant ΔV of 5000 [m/s] has to be given to the spacecraft.
- The thrust is fixed at 0.01 [N].
- The initial mass is 1000 [kg].

Only the power and mass for the propulsion and power subsystems are taken into account, the other subsystems are supposed to not consume any energy and to not weight anything. Figure A.1 shows the evolution of these masses with the Isp.

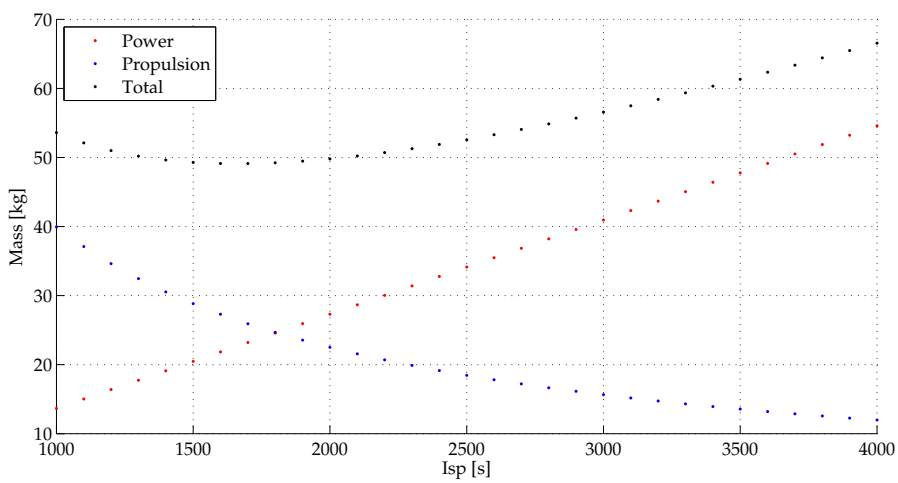

Figure A.1: Evolution of mass with Isp

Although, an minimum exists around the point Isp = 1750 [s], the tendency of the overall mass is to increase with the Isp beyond this point. Thus, the idea that a very high Isp will reduce the mass of the system is wrong for the electric propulsion and with this method for sizing the power subsystem.

Figure A.2 shows the evolution of the cost (TFU and development cost) with the Isp.

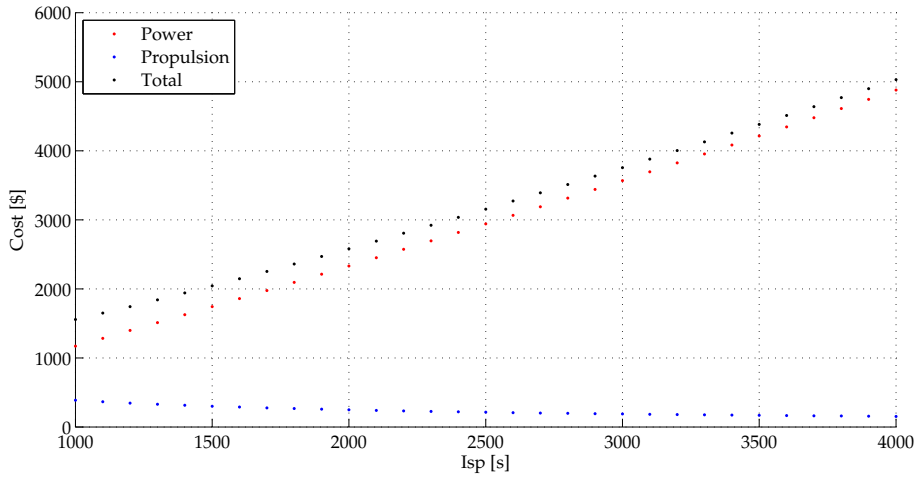

Figure A.2: Evolution of cost with Isp

It is now possible to see the influence of the Isp on the cost of both subsystems. Although the cost of the propulsion system does decrease, the increase of the cost due to the bigger power system is much bigger and therefore so is the overall cost.

This simple example uses parameters that clearly show the tendency of the cost to increase with the Isp and justify to choose an electric propulsion subsystem with an Isp that is not very high. It is also important to note that this example does not take the TRL into account (a factor of 1 is applied in the cost model). If it was the case, the cost increase would be even bigger as the systems with very high Isp are still not completely qualified.

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