Active Space Debris Removal System

Gabriele GUERRA[1], Alexandru Camil MURESAN*[,1], Karl Gustav NORDQVIST[1], Antoine BRISSAUD[1], Naser NACIRI[1], Ling LUO[1]

*Corresponding author
[1]KTH Royal Institute of Technology,
Department of Aeronautical and Vehicle Engineering,
Brinellvägen 8, 114 28 Stockholm, Sweden,
gguerra@kth.se, muresan@kth.se*, gnordq@kth.se,
brissaud@kth.se, nacer@kth.se, lluo@kth.se

DOI: 10.13111/2066-8201.2017.9.2.8

Received: 09 April 2017/ Accepted: 30 April 2017/ Published: June 2017

Abstract: *Since the start of the space era, more than 5000 launches have been carried out, each carrying satellites for many disparate uses, such as Earth observation or communication. Thus, the space environment has become congested and the problem of space debris is now generating some concerns in the space community due to our long-lived belief that "space is big". In the last few years, solutions to this problem have been proposed, one of those is Active Space Debris Removal: this method will reduce the increasing debris growth and permit future sustainable space activities. The main idea of the method proposed below is a drag augmentation system: use a system capable of putting an expanded foam on a debris which will increase the area-to-mass ratio to increase the natural atmospheric drag and solar pressure. The drag augmentation system proposed here requires a docking system; the debris will be pushed to its release height and then, after un-docking, an uncontrolled re-entry takes place ending with a burn up of the object and the foam in the atmosphere within a given time frame. The method requires an efficient way to change the orbit between two debris. The present paper analyses such a system in combination with an Electric Propulsion system, and emphasizes the choice of using two satellites to remove five effective rockets bodies debris within a year.*

Key Words: *Space Debris, Satellite, Removal, Foam.*

1. INTRODUCTION

Orbital debris is any man-made object in orbit around the Earth which no longer serves a useful function. Such debris includes nonfunctional spacecraft, abandoned launch vehicle stages, mission-related debris and fragmentation debris [19]. Currently, there are more than 20,000 pieces of debris larger that a softball orbiting the Earth and 500,000 of the size of a marble. They all travel at speeds up to 7.5 km/s, fast enough also for a small piece of debris to damage a satellite or spacecraft. In Figure 1 we can see a sample picture illustrating the huge number of debris orbiting the Earth. Moreover, there are millions of smaller debris that are not tracked in their motion around the Earth. It is clearly seen how that huge number of objects can pose problems for the future of space exploration and exploitability hence the need to find a solution to this growing issue.

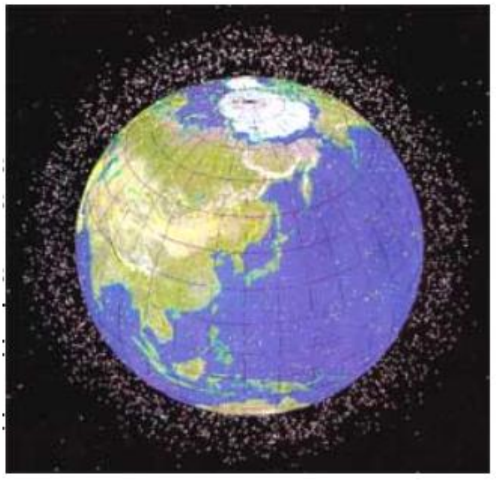

Fig. 1 – Tracked debris around Earth

A. Problem description

The increase of space debris is considered as one of the main threats for future sustainability of space activities and space access since the beginning of the space era in 1950s. The risk of debris collisions and the potential cascading effects due to their number and broad distribution, could prohibit future human and robotic space missions. Examples of the negative effect on the space environment caused by collision can be seen in Fig. 2 [35].

Hence engineers have to design space missions in order to decrease the population of space debris. Analyses show that in order to achieve a long-term decline of debris population, it would be enough to remove approximately 5 of the bigger debris in the 800 km to 1000 km altitude band every year, as can be seen in Figure 3 [36].

It is worth noting that currently there isn't any international regulatory system that states old satellites have to be removed, but recently, NASA [21] and ESA [10] have implemented a guideline for their satellites and it is likely that it will soon become a law.

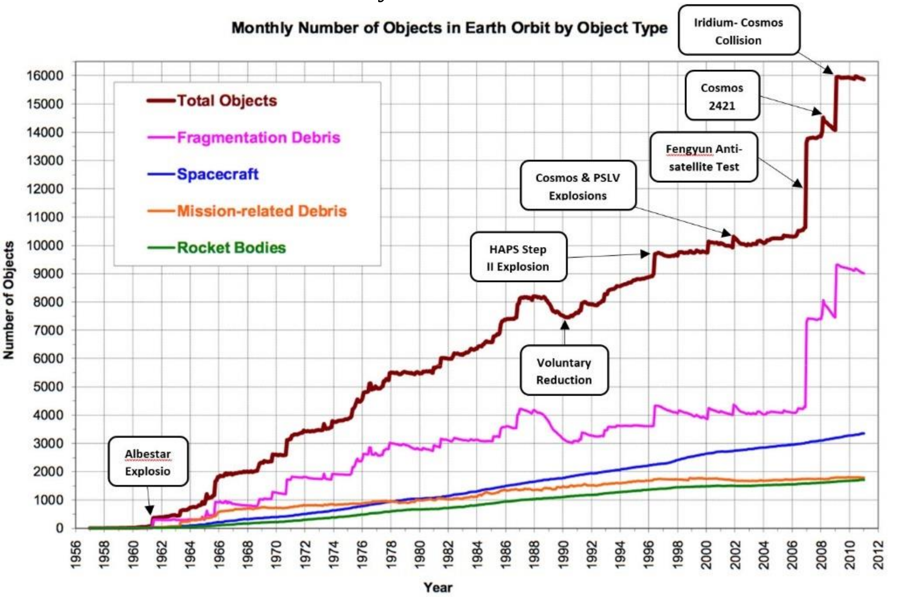

Fig. 2 – Debris population growth in the last 55 years

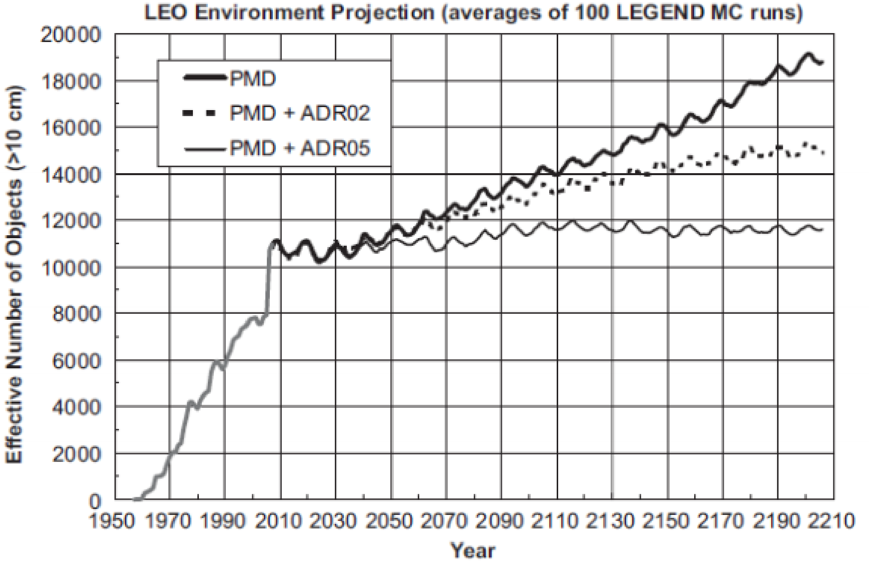

Fig. 3 – Debris population trend with and without ADR

B. Goals of the project

The main objectives of this project are to perform a preliminary mission analysis of a LEO satellite system, and do a feasibility study of an orbital debris removal mission. Our mission will include a study of a satellite system capable of descending 5 of the biggest space debris from their current orbits into a lower orbit with a maximum lifetime of 25 years. The biggest debris in orbit are rocket bodies. All this to be accomplished within a year.

C. Limitation of the work

Before starting the analysis of the proposed foam-based method, it is expected to explain which kind of assumptions were made since it is only a preliminary feasibility study.

- The mass of the spacecraft is considered constant during all phases of the mission and equal to the initial total mass of the satellite. This assumption was made because the propellant mass is less than 10\% of the total satellite mass.
- The motion of the satellite was considered independent from the effects of atmospheric drag perturbation, solar radiation, albedo, gravitational field of other celestial bodies in the solar system that are not the Earth and Earth oblateness effect.
- All the orbits are assumed to be circular due to their low eccentricity ($0.0011 \leq e \leq 0.0108$).
- The debris are considered as cooperative-targets in the meaning that they do not have angular momentum (they are not tumbling). This will simplify a lot the rendezvous phase between the deorbiting vehicle and the debris.
- The orbits of debris are considered unchanged during the time required to perform orbital transfer, so neither drag nor Earth oblateness effect perturbation were considered during the low thrust transfers.
- The argument of perigee change maneuver has been neglected.
- Optimal expansion of the foam is assumed. The foam structure resulting from the expansion won't bend under external forces.

2. MISSION PHASES AND STRATEGY

The method that will be analysed is made up of different phases that can be shortly described as below:

Launch

The platform in charge of targeting and deorbiting the debris has to be launched into an initial orbit. When launched into the desired orbit small adjustments are made by the satellite to correct for potential errors in the final position.

Catching the debris

After the satellite is launched its mission starts which consists in catching the first rocket body. The debris and the satellite orbits are in the same orbital plane and the only adjustment needed is within this plane. To catch up with the debris, the satellite will make a small altitude change in order to adjust its speed and later go back to the debris' altitude in order to continue with the next step.

Docking phase and release of the foaming device

Assuming that, in this phase, the debris has been reached by the designated vehicle, the docking phase will start. The vehicle will dock with the debris through a docking mechanism similar to that of the SMART-OLEV satellite [37] and then release the foaming device.

Debris De-orbiting and Foaming process

Once the foaming device has been attached to the debris, the first de-orbiting phase will start. The vehicle will pull the debris to the designed decay orbit. When the desired orbit height is reached, the vehicle will un-dock from the debris and an autonomous foaming process will take place. During this stage, the foam will expand its volume inside a bag, resulting in a sort of foam-parachute that will increase the drag of the system (composed by debris and foam). From that moment, a natural de-orbiting begins.

Targeting of next debris

The vehicle performs a set of orbital manoeuvres using its own electrical thrusting system with the aim to intercept the next target debris.

Platform self-disposal

Once the de-orbiting phase of the last debris had come to an end, the vehicle will remain docked to it. The foaming process will take place and after that, the vehicle itself will fall in the atmosphere with the debris.

A. Debris

Selection

As it can be seen from Fig. 4 the major number of debris is found in orbits that range between 600-1200 km of altitude.

All the debris that were chosen to be removed are found in this range, more accurately between 750 and 800 km.

Moreover, as it can be seen from [31] eq. (2), changing the inclination angle is the most expensive manoeuvre in terms of ΔV required.

For that reason, the chosen debris have almost the same inclination angles. In table 1 one can see all the debris chosen for the mission. All information about these debris was found in [9]. The mass of the debris selected is important because the mission is focused on removing debris with a significant size.

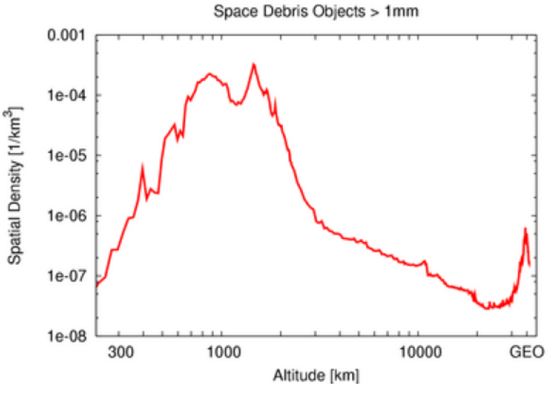

Fig. 4 – Location of debris

Table I. Selected Debris List with relative Orbit Parameters

Name	Mass (kg)	i (deg)	Ω (deg)	ω (deg)
ARIANE 40+ R/B	1764	98.49	13.2	1.49
ARIANE 40 R/B	1764	98.62	13.3	0.14
ARIANE 40 R/B1	1763	98.64	22.4	9.02
CZ-4B R/B	1000	98.32	28.6	6.17
ARIANE 40 R/B	1764	98.70	34.7	6.06

Orbital parameters

Parameters for the orbits chosen can be seen in Table I. The orbits are considered circular, as mentioned earlier. Figure 5 shows the debris orbits around the Earth.

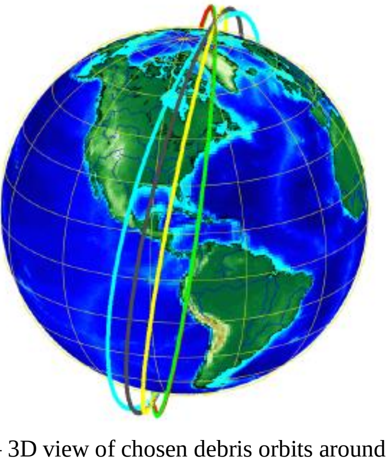

Fig. 5 – 3D view of chosen debris orbits around Earth

B. Satellite

The satellite chosen to perform the mission is derived from an existing Orbit Life Extension Vehicle (OLEV). The vehicle will dock with the debris and perform all the required operations. The OLEV system is based on the SMART-1 platform developed by Swedish Space Corporation, called SMART-OLEV (see Fig. 6). The mass of the satellite was assumed to be around 600kg with: 70kg of fuel, 100kg in structural mass, 130kg of other manoeuvring thrusters and their system, 20 kg for the grabbing arm, 50 kg of communication command and control systems, 50kg of the solar panels and batteries, 50kg for the electric motor and 30 kg per SDK, which is short for Sail Deorbiting Kit.

Fig. 6 – SMART-OLEV Satellite

Data - variables

The first data needed to perform the mission is the mass of the two satellites, which also include the fuel that each of them require to be able to remove all the designed debris. Two satellites are used since the results showed that one satellite is not enough to complete the mission in the time constraint.

Propulsion system

The SMART-OLEV satellite uses a Low Thrust Ion Propulsion system. The most important parameters are the provided maximum thrust T and the specific impulse I_{sp}. Knowing these values, it is possible to calculate the ΔV required to perform all the orbital transfers as shown in the equations below:

$$\Delta V_r = \left| \sqrt{\frac{\mu}{r_0}} - \sqrt{\frac{\mu}{r_1}} \right| \tag{1}$$

$$\Delta V_i = V \sqrt{2 - 2 \cdot \cos {}^{\pi}/_{2} \cdot \Delta i} \tag{2}$$

$$\Delta V_{\Omega} = \frac{\pi}{2} \sqrt{\frac{\mu}{a}} |\Delta \Omega| \sin(i) \tag{3}$$

In order to make a change, with a low thrust engine, in altitude (a), inclination (i) or change in the reference angle RAAN (Ω), equations (1 - 3) from [25] are used. When an engine with a Specific Impulse (I_{sp}) and thrust (T) is picked the time required for each change can be calculated [18], [24].

$$\Delta t = \frac{\Delta V}{a}, \{a = \frac{T}{m_{sat}}\} \tag{4}$$

Where $m_{sat} = m_{dry} + m_{prop}$. The calculation of the propellant mass requires ΔV, I_{sp}, g_0 and m_{sat}, resulting in the following equation:

$$m_{prop} = m_{sat} \cdot \left(e^{\frac{\Delta V}{I_{sp}\, g_0}} - 1 \right) \tag{5}$$

To catch up with the debris as the satellite reach its orbit, a worst-case scenario is calculated by using eq. (1) and assuming a transfer to a lower orbit if the debris is ahead or a higher orbit if the debris is behind. With the given equations above and the catch-up method, time and fuel consumption can be calculated in order to see if the mission requirements and satellite properties are met.

Catch and grab dynamics

As assumed during all the mission, the debris are supposed to not have any angular momentum so they are not tumbling. Rendezvous with the debris can take few days and can't occur 24 hours a day due to specific illumination needed by the sun.

- Approach: The satellite will join the "parking position" from the "rendezvous point" according to a pre-defined speed profile. Distance between the satellite and the debris client shall be calculated on ground using stereo image [16].
- Insertion: The boom will be deployed on ground command with constant speed. Specific sensors shall be used to notice the ground operator of the situation [16].
- Capturing: These sensors will also give data from which operators on ground can calculate the nozzle profile of the client debris and the penetration depth inside the nozzle. The boom speed shall be calculated from the previous analysis [16].
- Coupling: Because of some perturbations due to the capturing phase, operations shall wait until these perturbations are settled. Then, the ground operator can command the boom to retract [16].

As shown in the Figure 7, the capturing tool docks inside the debris' nozzle [37].

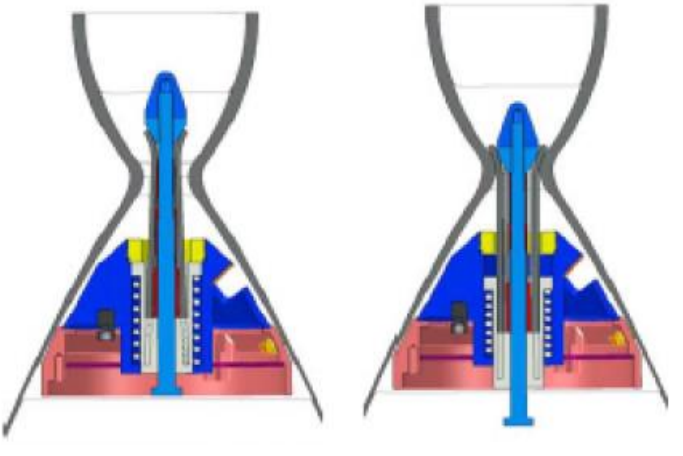

Fig. 7 – Dockings tools of the satellite

C. De-orbit

The de-orbiting system is developed in order to satisfy the cost and time constraints of the mission. With the intention to reduce them, an advanced method was used to increase the release height of the rocket bodies. The chosen method consists of attaching an inflatable structure in which a two-component foam would combine, inflate the structure, solidify it and ensure its rigidity. The structure shall be designed as a disc to increase the drag coefficient, with a main radius as maximum as the technology allows. The attachment point shall be the rocket body nozzle with a crab claw that would lock to the nozzle. The inflatable structure would only inflate after the separation from the main satellite. The structure shall also be designed to burn during re-entry.

This would be a better approach than the second method of just spraying expandable foam on the rocket because the foam might not stick to all structures and therefore this method might need more tests to qualify for space use. Inflatable structures are getting used more often (e.g. Bigelow Expandable Activity Module on the ISS) and their Technology

Readiness Level is high. Foams are known to have high expanding ratios of up to 1:1000 at standard atmosphere (1 bar) [12]. The mass of the attached inflatable sail can be neglected since the medium density of the foam is around 10 kg/m^3 [4].

Orbital decay is a known effect that happens to the Low Earth orbiting satellites that gives a physical lifetime to all spacecraft in that region and is determined by the interaction with the atmosphere, neutral particles, plasma and space radiation. For our project, it is crucial to predict the lifetime of the rocket bodies to compute their necessary release height for a 25 years lifetime. Although it is very hard to predict, the model used has a precision of around ±10% therefore we are taking that into account. The uncertainty is mainly due to the most influential components of the solar activity that affects LEO spacecraft: geomagnetic index known as A_p index and the solar radio flux on the 10.7cm wavelength called $F_{10.7}$ index [14].

The atmospheric model used to predict the orbital decay time of our rocket bodies has a maximum usable height of around 500km but further examination has shown that it can also be used up to 600km. The air density used is computed from a density exponent and from height. As stated above, all the important Coronal Mass Ejections that affect our rocket bodies plus the Earth's atmosphere give the following equations:

$$T = 900 + 2.5(F_{10.7} - 70) + 1.5\,A_p$$

$$m = 27 - 0.012 \cdot (h - 200) \quad \text{if} \quad 180 < h < 600\ (\text{km})$$

$$H = \frac{T}{m} \tag{6}$$

$$\rho = 6 \cdot 10^{-\frac{h-175}{H}}$$

Although the rocket bodies travel at a height where the atmosphere is almost inexistent we can still use the drag equation:

$$D = \frac{1}{2}\rho A v^2 C_d \tag{7}$$

where: D is the drag force, ρ is the density of the fluid derived from eq. (6), v is the speed of the object relative to the fluid, A is the cross-section area, C_d is the drag coefficient.

For our purposes, we shall use a cross-sectional area for the final equation $A_e = A \cdot C_d$. From Newton's second Law and the conservation of Momentum we can extract the following relation between the period P and the semi-major axis a:

$$P^2 \cdot G \cdot M_E = 4 \cdot \pi^2 \cdot a^3 \tag{8}$$

where: G is the universal gravitational constant, M_E is the mass of Earth.

Therefore, we can derive the difference on period for a specific amount of time with:

$$\frac{dP}{dt} = -3 \cdot \pi \cdot a \cdot \rho \cdot \frac{A_e}{m} \tag{9}$$

The last two equations are iterated with the release height as starting point. The end point is chosen to be at 180km, since an unpowered spacecraft at that altitude has a lifetime of maximum one day [1].

Time constraint and removal of our own satellite

Accordingly to [15], spacecraft's or orbital stages that terminate their operational life in the LEO region should be de-orbited (i.e. direct re-entry) or, if possible, maneuvered into an orbit with a reduced lifetime. The lifetime limit for satellite whose deorbit phase is determined mainly by atmospheric drag was found to be 25 years. In the following calculation, this value will be assumed as the upper limit on the lifetime of the disposed debris. Concerning the disposal of the two vehicles, as stated above they will be de-orbited together with their last respective debris (the system vehicle-debris must also comply with the time constraint).

D. Rocket

Requirements and rocket choice

The satellite with a given mass must be launched to the Low-Earth Orbit (LEO). To do so, an appropriate rocket should be used while minimizing the overall cost of the mission.
Existent rockets have been searched, and prices from different launchers have been compared.
The Falcon 9 v1.1 rocket has been chosen because it has the lowest cost per kg per launch, with an overall cost of around 62 M$ [7]. The payload will be shared on the rocket with other organizations that want to launch their own satellites, resulting in a launch cost of around 4109$ per kg. Usually the price is around 20200$ per kg [38]. Specifications for the Falcon 9 v1.1 rocket are shown in table 2.

Table II. Falcon 9 v1.1 specifications

Type	Value
Total mass	528 520 kg
Total payload mass	13 150 kg
Structural mass of the first stage	23 100 kg
Fuel mass of the first stage	395 700 kg
Structural mass of the second stage	3 900 kg
Fuel mass of the second stage	92 670 kg
Average thrust of the first stage	6 165 kN
Average specific impulse of the first stage	296.5 s
Average thrust of the second stage	801 kN
Average specific impulse of the second stage	340 s

The trajectory can now be simulated from the launch pad until the desired orbit, which has been chosen to be the one of the first debris orbit in order not to have to apply an additional ΔV for the transfer to the debris' orbit.

The models

To simulate the launch of the rocket, several models have been used, mainly for the atmosphere and gravity. Since the rocket will reach high altitudes, the atmospheric density and the gravity acceleration can't be considered constant and models must be used to calculate them.

The atmosphere

The Earth's atmosphere is made of layers which have properties that vary with altitude, especially the density which concerns the mission since the drag has to be calculated at each point of the simulation.
To model the atmosphere's density, the hydro-static equation (see eq. (10)) has been used and integrated to get the density. Eq. (10) has been calculated by supposing that the temperature is constant, which isn't the case in the atmosphere, but it's a reasonable local description of it.

$$\rho = \rho_0 \cdot \exp\left(-\frac{H}{H_0}\right), H_0 = \frac{RT}{g} \tag{10}$$

Gravity

The gravity acceleration has been calculated at each altitude by using the equivalence between Newton's Law of universal gravitation and the gravitational force. The following expression for the gravity acceleration was used:

$$g = \left(\frac{g_0}{1 + {}^{h}/_{Re}}\right)^2 \tag{11}$$

where: g_0 is the acceleration at the surface of earth, h is the altitude, Re is the radius of the Earth.

Steps of the rocket launch

In order to reach the orbit, the rocket goes through three main steps. First, it goes straight up as a sounding rocket until it reaches the altitude where the gravity turn starts. At some point during this gravity turn, the fuel of the first stage burns out, and the stage tears off from the rocket. The same thing happens for the second stage, and after its separation from the payload, we reach the desired orbit.

Sounding rocket

In the first step of the launch the rocket operates as a sounding rocket. It follows a vertical path and is subjected to gravity, thrust and drag. The equations of motion can be written as follows [31]:

$$\begin{cases} m\dfrac{dV}{dt} = T - D - W \\ \dfrac{dH}{dt} = V \end{cases} \tag{12}$$

where $T = -V_e\frac{dm}{dt}$ is the thrust, $D = \frac{1}{2}\rho A v^2 C_d$ is the drag and $W = mg$ is the weight.

Gravity turn: After some time, the gravity turn starts by inducing a small pitch angle which causes a small part of the gravitational force to be directed perpendicular to the longitudinal axis. The time at which this turn starts is chosen by plotting the kinetic pressure that is applied on the rocket, and choosing a time when that pressure isn't high. The reason behind this is that rockets are not strong enough in the transverse direction, and therefore a small kinetic pressure should be kept when the angle of attack is not exactly zero in order not to break them. By writing the equations of motion in the local-horizon frame, one can get a set of coupled non-linear ordinary differential equations [31]:

$$\begin{cases} \dot{V} = \dfrac{T}{m} - \dfrac{D}{m} - (g - \dfrac{\dot{X}^2}{R_e + H}) \sin\gamma \\ \dot{\gamma} = -\dfrac{1}{V}(g - \dfrac{\dot{X}^2}{R_e + H}) \cos\gamma \\ X = \dot{V}\cos\gamma \\ \dot{Y} = \dot{V}\sin\gamma \\ \dot{m} = -\beta(t) \end{cases} \tag{13}$$

During the time between the burnout of the first stage and the start of the second stage, there's no thrust since the two stages are being separated. In that case, the same equations as in the gravity turn can be used and the thrust should be set to 0. The same thing applies after the burnout of the second stage when the satellite reaches orbit.

Stage Optimization

In addition to simulating the launch process using a SpaceX Falcon 9 v1.1 rocket, designing a new rocket with optimized stage masses in order to launch only the satellite concerned by the mission would have been possible. To do so, by using the values for the number of stages, the structural ratios $\in_i = \frac{m_{si}}{m_{si}+m_{fi}}$ as well as the effective speed from the Falcon 9 v1.1 rocket, one can use the Lagrange method to maximize the overall payload ratio. The mathematical problem can be written as follows:

$$\begin{gathered} Maximize: \pi_* = \prod_{k=1}^{N} \pi_k \\ Knowing\ that: V_* = -\sum_{k=1}^{N} V_{ek} \ln[\in_k + (1 - \in_k)\pi_k \end{gathered} \tag{14}$$

The Lagrange equation for this problem is:

$$\ln \pi_* = \sum_{k=1}^{N} (\ln \pi_k + \lambda \left\{ \frac{V_*}{N} + V_{ek} \ln[\in_k + (1 - \in_k)\pi_k] \right\}) \tag{15}$$

Where λ is the Lagrange multiplier. Calculating the N partial derivatives of this equation give us an equation for all the payload ratios:

$$\pi_k = -\frac{\in_k}{(1 - \in_k)(1 + \lambda V_{ek})} \tag{16}$$

As λ is unknown, another equation is needed to calculate it and deduce the π_k. The equation is obtained by replacing equation 14 in the condition of the problem:

$$V_* = -\sum_{k=1}^{N} V_{ek} \ln(\in_k - \frac{\in_k}{1 + \lambda V_{ek}}) \tag{17}$$

Computing this method on Matlab gives the results on Table III, assuming $\in_1 = 0.0552$, $\in_2 = 0.0404$, $V_{ef1} = 2.9087$ km/s and $V_{ef2} = 3.3354$ km/s.

Table III. Mass ratios

Ratios	Falcon 9 v1.1	Optimized rocket
π_1	0.2076	0.1502
π_2	0.1199	0.5073
π_{tot}	0.0249	0.0762

E. Launch

Place

Because the SpaceX Falcon 9 rocket was chosen, the launching place that SpaceX is actually using shall be used. The launching site which is actually used is at Cape Canaveral Air Force Station (CCAFS) [7]. [17] gives the exact coordinates of the launch site, which are 28.396837, -80.605659.

Trajectory

Table 1 shows the inclination angle *i*. The average of the inclination angles is almost 98°. Therefore, the launch of the rocket shall be done in this direction.

Because the Earth rotates eastward on its axis with one complete turn each day, and the Equator's surface is rotating at 1675 kilometers per hour [20], the launch of the rocket shall be done in the eastward direction to take benefit of the energy of the Earth's rotation to minimize the fuel used. [31] gives the formula of the azimuth angle A which is $\sin(A) = \frac{\cos(i)}{\cos(l)}$, where i is the orbital inclination of the orbit and l is the latitude of the Cape Canaveral launch site, hence is A = -9.10°. Since an optimal launch shall take benefit of the energy of the Earth's rotation, the launch will be in the opposite angle which is A' = 170.9°. The direction of the launch will be South East (note the definition of the azimuth angle, which is different from the inclination angle).

Window

In order for a satellite to be successfully put into an orbit, it has to be launched during a specific time during which the launch site on the surface of Earth intersects through the orbital plane. This interval of time is called launch window. However, this launch window does not necessarily exist: one has to compare the latitude of the launch site with the inclination angle of the orbit. If the inclination angle is smaller than the latitude of the launch site, this means that the satellite never goes over the launch site, which means no launch window: the satellite has to perform orbital maneuver to reach the orbit.

However, if the latitude of the launch site is equal to the orbit's inclination angle, the satellite goes exactly above the launch site once per day. And, consequently, if the latitude is smaller than the inclination angle, there are two launch windows per day. In the case of this project, the latitude for the Cape Canaveral launch site is around 28.4° and the inclination angle of the first orbit is around 98° and therefore there are two launch windows per day. As stated above, the window depends also on the rotation of the Earth and the position of the orbit compared to the inertial frame. Since this study is only a preliminary study, no estimation of the launch date was done.

F. Cost optimization

The reusable design of the Falcon9 was meant to save money, since the boosters and the external tank could be used again post-launch. Refurbishment costs won't come anywhere close to the cost of manufacturing an entirely new rocket. Most of the refurbishment will be dedicated to inspections and making minor adjustments so that the vehicle meets the standards required for spaceflight. Those costs would be about half a million dollars which compared to the $60 million needed to build a first stage represents a pretty significant cost reduction [30].

G. Risk Analysis

After having explained all the steps of the mission from the launch to the deorbit phase, all hazards arising this approach will be presented [3]. This risk analysis shall briefly present risks which have to be considered for the reliability of the mission.

Launch

During the launch phase, mechanical and thermal constraints have to be considered. Because the temperature inside the combustion chamber is very high, materials which compose the nozzle have to be designed to support these kind of temperatures. Breach on tanks or on the body of the rocket could appear because of these constraints.

The tank of the foam shall be also reinforced since the mechanical stresses are very important during the launch phase [3]. If one break on the foam tank or on the fuel tank

appears it will result in catastrophic damages to the launcher and the mission could be immediately canceled.
Since the Falcon9 rocket was chosen for the mission, one shall take in consideration the risk of a crash of the rocket. In fact, one Falcon9 rocket crashed on 2016 September 01 in the Launch Complex40 (LC40) at Cape Canaveral Air Force Station, and caused 200 Millions of dollars lost in the process [28]. SpaceX explained [28] that one evidence suggests that a “large breach in the cryogenic helium system of the second stage liquid oxygen tank took place”. Although the technology of reusable launchers is just new, the first re-usable rocket has been launched by SpaceX few weeks ago, confirming the increased reliability of this technology.

The approach of the satellite to the debris
As assumed in the mission, the debris are considered as cooperative-targets in the meaning that they do not have an angular momentum. If they do have an angular momentum so that they are tumbling, the grab phase may be impossible.

Foam expansion
The foam ejection and expansion is maybe the most critical part of the mission because the technology used in the mission is new and not so many experiments took place in vacuum environment. When the foam expansion takes place, it is mandatory to control the viscosity and density of the components used in the chemical reaction, otherwise it will result a flow outside the satellite which can represent hazards. One possibility is that the flow outside the satellite will make a blockage of the foam expansion device. Therefore an inflatable structure would solve a lot of possible issues.

Re-entry behavior
The foam/debris part may collide with other debris or active satellites during the re-entry phase. Because of its large area-to-mass ratio, the foam/debris has got a higher possibility to hit other debris and take damage. One solution to avoid collisions with active satellites is to track the foam/debris part on ground and it will be the care of the active satellite to avoid the impact.

3. RESULTS

A. Satellite

The propulsion system of the SMART-OLEV satellite mentioned in Sec. 2-B does not provide enough thrust to even remotely satisfy the time constrain of the mission. It was decided to use a different type of Low-Thrust Ion Propulsion system called “NASA Solar Technology Application Readiness” (NSTAR), the same that was used for the Deep Space 1 mission. This thruster operates over a 0.5 kW to 2.3 kW input power range providing thrust from 19 mN to 92 mN. The specific impulse ranges from 1900 s at 0.5 kW to 3100 s at 2.3 kW; the propellant used is Xenon gas.
Because of the ion engine's power requirements, the satellite requires high-power solar arrays. Multi-junction solar cells were chosen as the best option for their high efficiency (between 20% and 25%).
The specific power of this type of solar cells is P_0=301 W/m^2 [34]. Knowing the thruster power requirement of 2.3 kW and adding 0.7 kW for the satellite sub-systems, a total power requirement P_r = 3 kW can be estimated.

From this value and the specific power of the chosen solar panel, the total required solar-array area is calculated:

$$A_{sa} = \frac{P_r}{P_0} = 10m^2 \tag{18}$$

The total area will be split in two solar wings each of which is composed of five panels measuring about 1 m^2. At launch, the wings will be folded up so that the spacecraft would fit into the launch vehicle's fairing.

In order to estimate the time, propellant and ΔV required equations [1][2][3] and [5] were used along with the assumption that the satellite needs to travel the furthest possible distance (half an orbit) to catch the debris for every rocket body. Full thrust is assumed during all thrusting phases, i.e. T = 92 mN, I_{sp} = 3100 s. The result is shown in Table 4 and due to the time requirement that is not met the mission requires two satellites.

Table IV. Requirements one satellite

$\delta V_{req}[km/s]$	$\delta t_{req}[days]$	$m_{prop}[kg]$
5.42	512	141

The result from using two satellites is summed up in Table Va, Vb, Bc along with the Figures 8a, 8b, 8c where the figures show the resources used as the satellites travel from rocket body to rocket body as two satellites instead of one. The legend in each plot describe which part of the mission each color defines.

The resources required for the inclination change is small and it is therefore hard distinguish. In order to interpret the figures think of time passing in the positive y-direction and the starting point being debris nr. 3 for both satellites.

Table V.

(a) Required ΔV for each step of the mission

Satellite nr.	ΔV_r	ΔV_i	ΔV_Ω	ΔV_{catch}	ΔV_{tot}
1	0.534	0.0307	1.860	0.0787	2.854
2	0.359	0.143	2.492	0.0526	3.397

(b) Required Δt [days]

Satellite no.	Δt_r	Δt_i	Δt_Ω	Δt_{catch}	Δt_{tot}
1	111	2.32	140	14.9	254
2	59.4	10.82	188	9.9	258

(c) Required Δm_{prop} [kg]

Satellite no.	Δm_r	Δm_i	Δm_Ω	Δm_{catch}	Δm_{tot}
1	29.06	0.61	37.77	3.11	69.44
2	15.54	2.83	50.13	2.07	68.50

B. Debris Release Height and De-orbit Phase

To compute the orbital decay time, it is first needed to know all the input values. The medium solar radio flux $F_{10.7} = 80 \cdot 10^{22} \left[\frac{J}{s \cdot m^2 \cdot Hz}\right]$ as been taken from [6].

The medium geomagnetic disturbance A_p = 80 taken into account is from [2]. The drag coefficient of the rocket bodies alone is that of a cylinder therefore C_d = 1 and with the drag sail it will increase to around C_d = 1.2 [13].
According to [4] the orbital decay algorithm has 10% error therefore the error has been taken into account by reducing the maximum 25 years allowed period by 10%.
Given the five debris chosen in Tab. 1 (all of them are exhausted rocket bodies) and the equations written in Sec. 2 – C, we have computed the Release Height (RH) of the rocket bodies with and without the drag sail (ds).
Note that the ARIANE rocket bodies are all identical. The results are presented for the two types of debris in the Tab. 6.

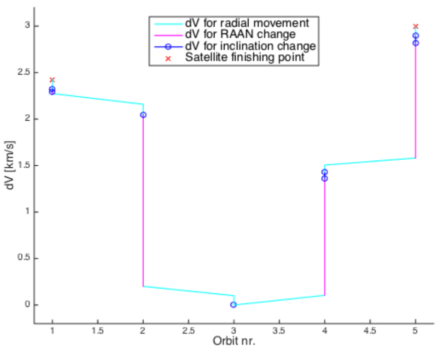

Fig. 8a – ΔV Required

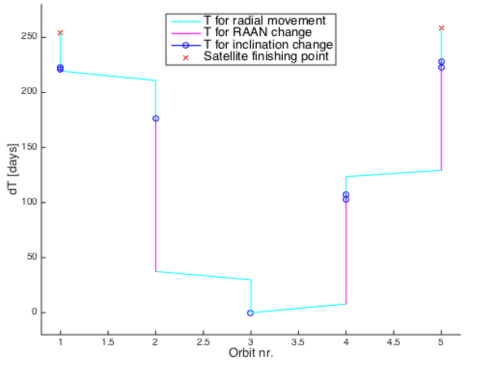

Fig. 8b – Time Required

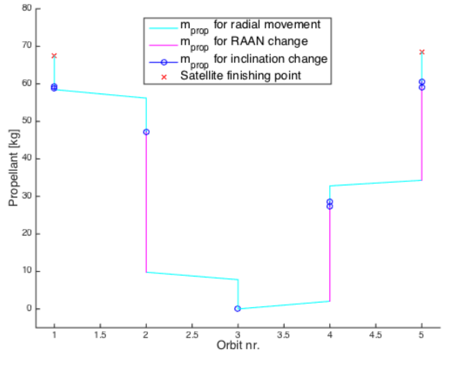

Fig. 8c – Satellite Projection

Table VI. Release Height [km]

Debris Name	RH w/ SDK	RH w/out SDK	ΔRH
ARIANE 40+ R/B	607	495	112
CZ-4B R/B	643	539	104

C. Rocket

The equations for the launch were implemented in *Matlab* in order to have the trajectory and adjust the different parameters to reach the altitude of the first object. To do so, some assumptions and choices had been made in order to reach the desired altitude with a flight path angle equal to zero.

Assumptions and choices

First, the rocket flies straight up as a sounding rocket before its flight path angle goes from 90° to an angle slightly lower. Then, gravity turns the rocket thanks to the induced acceleration. At some point after tilting the rocket, the first stage burns out. It separates from the rest of the rocket and the second stage's boosters start generating thrust. This means that there is a delay between when the first stage burns out and when the second stage starts producing thrust, a delay that needs to be determined.
Again like for the first stage, the second stage keeps producing thrust until it burns out. It separates from the payload, and the payload flies into the orbit. From this description of the trajectory, one can see that there are at least three parameters that need to be determined and chosen: the initial flight path angle, the time or altitude at which the gravity turn starts and the delay time between the burnout of the first stage and the start of the second one. In this simulation, these values were chosen:

- ϕ_{ini} = 89.4°: This value has been chosen manually by testing different values in order to reach the desired altitude.
- t_{delay} = 31s: Value chosen manually as well.
- t_{turn} = 32s: Because the gravity turn should start when the dynamic pressure on the rocket is still low, we had to plot the evolution of the pressure on time and choose a value that is quite low (see Fig.9).

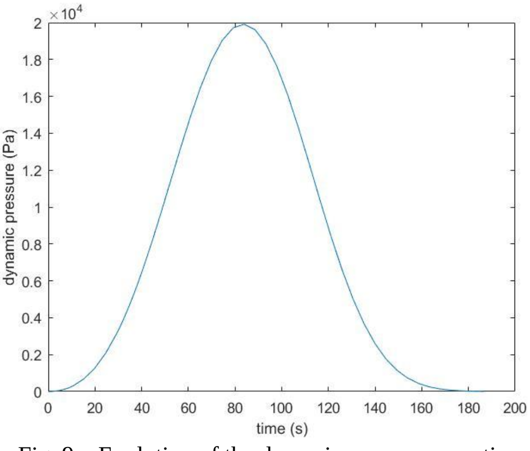

Fig. 9 – Evolution of the dynamic pressure over time

In addition to this, and because some of the parameters for the rocket were estimated (no data provided from SpaceX), the burnout time had to be deduced from the masses, the specific impulse and the thrust using the following formula:

$$T_{bo} = \frac{M_{fuel} I_{sp} g_0}{T_{thrust}} \tag{19}$$

In addition to that, the drag coefficient was assumed to be 0.3 as it's the typical value for a long cylinder terminated by a cone nose, which is quite close to what the falcon 9 looks like.

Results

Figure 10 represents the evolution of the different parameters (altitude, velocity and flight path angle) over time. The final values (Table 7) are then obtained by reading the values at the end of the simulation.

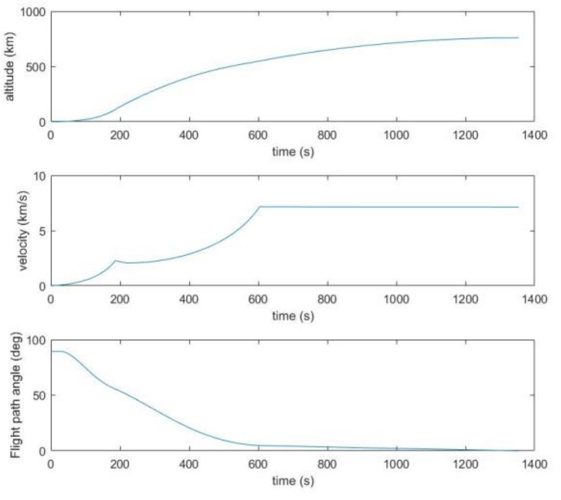

Fig. 10 – Evolution of the parameters over time

Table VII. Final values for the launch

Altitude	758.93km
Velocity	7.48km/s
Flight path angle	0.007°

In order to stay in the circular orbit, a certain velocity $Vc = \sqrt{\frac{\mu}{Re+|h|}} = 7.48\ km/s$ is needed. This value is compatible with the final value we get from our simulation. This means that we reach the right orbit with the right velocity and altitude. However, we can see that the flight path angle is not exactly equal to zero. Since it's only a feasibility study, we can accept that value, as long as it's close enough to zero. Another way of seeing the results is by plotting the altitude as a function of the downrange distance, which represents the trajectory the rocket takes (See Figure 11).

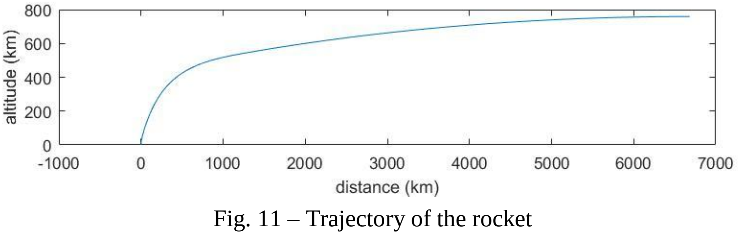

Fig. 11 – Trajectory of the rocket

D. Cost estimation

In October 2015, the Falcon9 commercial launch price was 61.2 million US$ (up from 56.5 million US$ in October 2013 competing for commercial launches in an increasingly competitive market). The cost of the rocket can be approximated to 63 million US$ [32]. We consider the cost of manufacturing two satellites as affordable when put into comparison to the launch cost, staff cost and the cost of office space and other necessary facilities. For the rocket, many hazardous systems must be taken in to consideration, such as among else ordnance operations, pressurized systems that operate below a 4-to-1 safety factor, lifting operations or systems that include toxic or hazardous materials, high-power RF systems and a variety of other systems and operations. Typically, additional precaution is required for operating systems that are considered hazardous, such as a redundant valving between pressurant and propellant. Additional precautions will be determined during the safety approval process with SpaceX and the launch range. Ordnance operations in particular, require coordination to provide reduced RF environments, cleared areas, safety support and other requirements [11].

4. DISCUSSIONS

A. Reliability

In order to assess the reliability of the overall system, it is needed to individually assess its components: launcher, satellite, mission. The SpaceX Falcon 9 has been chosen because of its low cost but the reliability must be taken into account. An analysis of launch failure of the Falcon 9 launcher history denotes a reliability factor of 97%, at the time of this paper, meaning that one of the 29 launches was a total failure with the loss of the payload. The analysis also showed that 91% of known failures can be attributed to three causes: engine failures, stage separation failures and, to a much lesser degree, avionics failures. With nine Merlin engines clustered together to make up the first stage, the vehicle is capable of sustaining an engine failure and still successfully completing its mission [23], [26]. Moreover, Falcon 9 uses a hold-before-release system, a capability required by commercial airplanes, but does not implemented on many launch vehicles. After the first-stage engine ignites, the Falcon9 is held down and not released for flight until all propulsion and vehicle systems are confirmed to be operating normally. An automatic safe shut-down occurs and propellant is unloaded if any issues are detected. From the satellite's point of view the launcher has the only objective of taking it to the transfer orbit after which its plasma engine starts guiding it to the rendezvous point with the first rocket body. There are known ways to deal with the problems one satellite and its systems encounter e.g. multiple navigation units, a bigger solar panel area than needed. The main concern for our own satellite are the new systems: grabbing arm and sail de-orbiting kit. Robotic arms like Canadarm have been used with great success [8], but a smaller one that has the capacity to grab random structures has yet to be developed and used in space. The second new system would be the SDK that again has not been designed and used, but inflatable habitable structures are already in the test phases e.g. Bigelow Expandable Activity Module currently on the ISS [5].The mission also has some key moments that can lead to failure: rendezvous, catch, attach, deployment and inflation of the SDK's. To mitigate the possible reliability issues a series of tests, inspections, analysis and reviews can be devised.

B. Future re-usability

Re-usable rockets were a dream for decades, and even after the first success in December 2015, the development of a reusable rocket is extremely challenging due to the small percentage of a rocket's mass that can make it to orbit. Typically, a rocket's payload is only about 3% of the mass of the rocket which is also roughly the amount of mass in fuel that is required for the vehicle's re-entry. The return, vertical landing and recovery was possible because the SpaceX manufacturing methodologies result in a rocket efficiency exceeding the typical 3% margin. A SpaceX rocket operating in the reusable configuration has approximately 30% less payload lift capacity than the same rocket in an expendable configuration [33].

C. Others mitigation standards practices

Since the growing population of space debris is an issue for the safety of future missions, there are many possible ways to deal with this problem. Tethers Unlimited developed among else a technology called Grapple, Retrieve and Secure Payload or GRSP [29]. It is a deployable net technology which enables a small satellite capture and manipulation of debris. In September 2004, one demonstrated the GRASP mechanism in micro gravity. This de-orbit module can be put on cubesats, this project is actually in TRL-5, which means in components and prototypes work [27]. Busek Co. is actually in a development of a debris-capturing called Satellite on an Umbilical Line, or SOUL [22]. The satellite SOUL will be equipped with a 100-meter cord. SOUL will go to the rendezvous with the debris, capture it, and attach a smaller de-orbit satellite to the debris, then drag the smallsat-combination to the desired location. The larger satellite would then throw the debris/smallsat combination to de-orbit.

5. CONCLUSIONS

The concept proposed for space debris removal with satellites propelled with electric Hall effect engines and accelerated drag using an expandable foam sail can be implemented with a low cost, high reliability and high impact. The proposed satellite should have a mass of around 600kg and shall be able to move up to three large rocket bodies to the 25 year decay orbit with the help of individual sail de-orbiting kits attached to each rocket body. The launcher can leave the satellite at a transfer orbit of around 550km. The de-orbiting kits have 3 internal tanks for the bi-component foam and pressure vessel, and a static mixing nozzle. Foam shall mix inside the expendable sail thus inflating and solidifying it. The drag sail is able to reduce with more than 100km the release height of the rocket bodies thus minimizing the time and ΔV required to complete the mission for a very low amount of mass. To keep the mission within the time constraint of one year, two satellites are being used because the electric Hall effect engines cannot provide enough thrust to finish the whole mission within one year.

APPENDIX. DIVISION OF WORK

L. LUO and N. NACIRI were in charge of the simulation of the rocket and the stages optimization;
G. NORDQVIST and A. MURESAN were in charge of the satellite *Matlab* code;
G. GUERRA and A. BRISSAUD were in charge of the satellite dynamics and the foam solution.
However the lines were not drawn by a permanent marker. In other words, we've been working a lot together and helping each other out.

REFERENCES

[1] * * * Air Force Geophysics Laboratory, *Handbook of Geophysics and the Space Environment*, 1985.
[2] J. H. Allen, *The Ap* Index of Maximum 24-Hour Disturbance for Storm Events*, USA, 2004.
[3] M. Andrenucci, P. Pergola, A. Ruggiero, *Active Removal of Space Debris, Expanding foam application for active debris removal*, http://www.esa.int/act, 2011.
[4] Australian IPS Radio and Space Services, *Satellite Orbital Decay Calculation*, The Australian Space Weather Agency, Sydney, 199.
[5] * * * SpaceRef, NASA Contract to Bigelow Aerospace, USA, 2013.
[6] E. A. Bruevich, V. V. Bruevich, G. V. Yakunina, *Correlation Study Of Some Solar Activity Indices In The Cycles 21 - 23*, Russia, 2015.
[7] * * * Capabilities and Services — SpaceX, http://www.spacex.com/about/capabilities.

[8] L. Dotto, *A Heritage of Excellence: 25 years at Spar Aerospace Limited*, Spar Aerospace Limited, Canada, 1992.
[9] M. M. Castronuovo, Active space debris removal - A preliminary mission analysis and design, *Acta Astronautica*, Volume **69**, Issue 9-10, p. 848-859, ASI - Agenzia Spaziale Italiana, Rome, Italy, 2011.
[10] * * * esa.int, http://www.esa.int/Our Activities/Operations/Space Debris/International cooperation
[11] * * * Falcon 9 Launch Vehicle Payload User's Guide, Rev 2, 2015 http://www.spacex.com/sites/spacex/files/falcon 9 users guide rev 2.0.pdf.
[12] * * * Fire Fighting and Fire Protection Systems Standards, *Engineering Standard for Foam*, Generating and Proportioning Systems, IPS-E-SF-140, 1993.
[13] * * * Fluid Mechanics Demystified, https://www.amazon.co.uk/dp/0071626816?tag=roymeccouk-21&camp=1406&creative=6394&linkCode=as1&creativeASIN=0071626816&adid=0SD6R7PR6WAYT91K02MF.
[14] J. K. Hargreaves, *The Upper Atmosphere and Solar-Terrestrial Relations*, Van Nostrand Reinhold, New York, 1979.
[15] * * * Inter-Agency Space Debris Coordination Committee, IADC Space Debris Mitigation Guidelines, Rev. 1, 2007.
[16] C. Kaiser, F. Sjberg, J. M. Delcura, B. Eilertsen, SMART-OLEV – An orbital Life Extension vehicle for servicing commercial spacecrafts in GEO, *Acta Astronautica*, **63**(1-4):400-410, July 2008.
[17] * * * latlong.net, http://www.latlong.net/place/cape-canaveral-florida-usa-9780.html
[18] * * * Low Thrust Transfer to GEO: Comparison of Electric and Chemical Propulsion, ASEN 5050, December 10, 2009.
[19] * * * Nasa.gov, http://www.nasa.gov/mission pages/station/news/orbital debris.html.
[20] * * * Nasa.gov, http://spaceplace.nasa.gov/launch-windows/en/.
[21] * * * Nasa.gov, https://www.orbitaldebris.jsc.nasa.gov/library/usg od standard practices.pdf.
[22] * * * Nasa Techport, SOUL System Maturation, Phase I Project SBIR/STTR Programs, techport.nasa.gov.
[23] * * * Next Big Future, http://www.nextbigfuture.com/2013/03/upgraded-spacex-falcon-911-will-launch.html.
[24] P. Pergola, A. Ruggiero, M. Andrenucci, L. Summerer, *Low-thrust Missions for Expanding Foam Space Debris Removal*, The 32nd International Electric Propulsion Conference, Wiesbaden, Germany, 2011.
[25] A. Ruggiero, *Low-Thrust Maneuvers for the Efficient Correction of Orbital Elements*, 2011.
[26] * * * spaceflightnow.com, http://www.spaceflightnow.com/falcon9/004/falcon9.html.
[27] * * * Spacenews.com http://spacenews.com/nasas-interest-in-removal-of-orbital-debris-limited-to-tech-demos/
[28] * * * SpaceX.com http://www.spacex.com/news/2016/09/01/anomaly-updates.com
[29] * * * Theaters Unlimited, GRASP, http://www.tethers.com/GRASP.html
[30] * * * theverge.com http://www.theverge.com/2015/12/24/10661544/spacex-reusable-rocket-refurbishment-repair-design-cost-falcon-9.
[31] W. E. Wiesel, *Spaceflight Dynamics*, 3rd ed., Aphelio Press, USA, 1999.
[32] * * * Wikipedia.org https://en.m.wikipedia.org/wiki/Falcon 9.
[33] * * * wikipedia.org https://en.m.wikipedia.org/wiki/SpaceX reusable launch system development program.
[34] J. L. Wiley, J. R.Wertz, *Space Mission Analysis and Design*, 3rd ed., Microcosm Press, USA, 1999.
[35] * * * Colorado Center for Aerodynamic Research http://ccar.colorado.edu/asen5050/projects/projects 2010/borowski/Space Debris Environment.html
[36] * * * Cranfield University Space Research http://www.crantalk.com/cranspace/projects/projects-sustainable-space/
[37] C. Kaiser, P. Rank, R. Krenn, K. Landzettel http://robotics.estec.esa.int/i-SAIRAS/isairas2008/Proceedings/SESSION%2012/m117-Kaiser.pdf
[38] * * * businessinsider.com http://www.businessinsider.com/spacex-rocket-cargo-price-by-weight-2016-6?r=US&IR=T&IR=T