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Systematic space debris collection using Cubesat constellation

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Abstract

Since recent space debris mitigation measures are not enforced at a global level among the community of both private and puclic satellite operators, Active space Debris Removal (ADR) will soon be needed to stabilize the amount of space junk in Low Earth Orbit (LEO),[3] in order both to ensure the safety of future space missions and protect the human populations on Earth against uncontrolled atmospheric reentries of large space debris.[2,9] Recently, many options were contemplated for ADR using various mission strategies and capture systems. Some of them are being tested by the European Union (EU) demonstration mission RemoveDebris.[7,11] However, demonstration costs mount up to hundreds of millions of euros in most cases.[5] Share My Space developed an incremental approach for systematically collecting small-sized catalogued space debris based on a constellation of 6U cubesats featuring a stereoscopic detection system combined with an integrated LIDAR,[16] and ion thruster operated with iodine.[20] The advantage of using a constellation is to target all the space debris around a given altitude while limiting expensive orbit inclination change maneuvers. Over its 2-year lifetime, each Cubesat can perform a rendezvous with up to 50 space debris and bring each of them to temporary storage tank in orbit for further recycling. Off-the-shelf Cubesat technologies make the demonstration process affordable. Moreover, flexible constellation management makes it easier to target the essential debris to be cleaned. Finally, SC design is scalable, allowing to capture larger space debris after technological maturation.

The aim of this study is to capitalize on recent improvements and mission design for space debris removal, particularly thanks to the European e.deorbit,,[1] CleanSpace,[19] and Remove debris[7] missions to assess how to remove small and medium-sized space debris using a constellation of nanosatellites.

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Nomenclature

Acronyms

ADR	Active Debris Removal
ATV	Automated Transfer Vehicle
GEO	Geostationary Earth Orbit
ISS	International Space Station
LEO	Low Earth Orbit
LIDAR	Light Detection and Ranging
PPS	Plasma Propulsion System
RAAN	Right Ascension of the Ascending Node
RADAR	Radio Detection and Ranging
SC	Spacecraft
SSO	Sun Synchronous Orbit

Constants

G	Gravitational constant	6.67×10^{-11} N m^2 kg^{-2}
M_T	Mass of the Earth	5.97×10^{24} kg
R_T	Earth radius	6 371 km
g	Gravitational acceleration on Earth surface	9.81 m/s^2

SC propulsion

T	Thrust of the spacecraft (SC)
P_{elec}	Electric power dedicated to the propulsion system
m_0	SC dry mass (without the propellant)
$\dot{m}$	Thruster mass flow rate
Δm	Propellant mass required to perform a full maneuver
m_p	On-board propellant mass at launch
m	Instant mass of the SC
δm	Propellant mass spent during one orbit period
I_{sp}	Specific impulse of the propulsion system
v_{ex}	Exhaust gas velocity of the propulsion
Δp	Pulse required to perform elementary maneuver
E_m	Mechanical energy of the orbiting SC

Orbital parameters

T_{OR}	Time for orbit raising
h	Orbit altitude
r	Orbit radius
v	Orbit velocity of the SC
ω	Orbit angular velocity of the SC
T_{orbit}	Orbit period
θ	Orbit inclination
ϕ	Orbit right ascension of the ascending node (RAAN)
$\delta\alpha$	Elementary angular displacement of an orbit
α	Angular path in a change of orbit. Sum of $\delta\alpha$.

Constellation management

p	Number of satellites in the constellations
n	Target number of collected debris per satellite
N	Number of target debris for the constelation
Ω	Solid angle where the debris are spread, in (θ, ϕ) coordinates

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Introduction

The problem of space debris has begun to strike the awareness of some space experts since the late 1970's, as the space industry was expanding rapidly. Some simulations were conducted in the past to assess the evolution of the quantity of space debris over the next century. There is a very large discrepancy in those predictions but the trend is always found to be exponential, with a characteristic time ranging from 10 years to 100 years depending on the scenarii and the assumptions made in the models.[10] Nowadays, the US Air Force tracks over 18 000 space debris larger than 10 cm orbiting at altitudes between 100 km and 36 000 km and with various inclination angles. Some plots showing the distribution of space debris are presented in Figure 1. The concentration of space debris is the highest for inclination angles between 80° and 100° and at an altitude between 700 km and 900 km. This roughly corresponds to sun synchronous orbit (SSO), which is a class of orbits that will always be of high interest for Earth observation. SSOs may exist at higher altitudes, but just shifting the orbits to higher altitudes may feature several drawbacks and limitations both on short and long terms:

- Communication time increases as the orbit altitude increases
- If the debris remain in place at around 800 km, they may jeopardize the new satellites at higher orbits during their orbit transfer phase.
- There is another peak in the space debris density at about 1 400 km

Therefore, just increasing the altitude of space operation is not a satisfactory solution both economically speaking and for sustainability reasons. Mega-constellation operators such as OneWeb plan to drive their satellites at altitudes around 1 200 km, where there is a minimum in the density of orbiting objects at the moment. However, since they plan to operate up to 3 000 satellites with a 3 to 5-year life-time, they will need to deorbit approximately 600 satellites every year, out of which at least 5% may fail before deorbitation which leads to an increase of 30 debris every year. Comparison with Figure 2 clearly shows that they would make their 10 km spherical slice of operation as saturated as the most populated zone around 800 km in just 10 years.

In this study, we attempted to make the best possible use of the space-track[6] data base of unclassified objects orbiting in space. This data base is built up over a daily surveillance of over 18 000 space objects using military RADARs. The orbital parameters are derived and the trajectories are predicted until the next detection. This technique combined with the use of RADAR signature of the objects allows to track all the well identified objects, with a precision of less than 100 m. Complementary information can also be found on data bases such as N2YO[13] or NSSDCA.[15]

The space industry needs a systematic, affordable, and scalable solution to tackle the problem of space debris management. Share My Space is developping a technology based on Cubesat components – at least for demonstration – and is assessing how to manage space debris removal at a constellation level. The second section of this paper gathers some elements that justify the choices that were made for spacecraft (SC) design. Third part deals with mission design and assesses the maneuver capabilities of a Cubesats propelled with a thruster fueled with iodine. Fourth section describes in more details what can be achieved by a Cubesat constellation in SSO. It was found that a well-distributed constellation of 38 6U cubesats would be able to remove up to 1 000 debris ranging from 1 cm to 1 m, in most concentrated areas, and bring them to a temporary on-orbit storage tank.

SMS-001, a 6U cubesat demonstrator

Mission requirements

As top level requirements for the Cubesat design, the SC should (i) have the ability to travel from one debris to an on-orbit storage vessel several times with its own propulsion system, and (ii) have the ability to catch and release a debris. The former point implies that the Cubesat has enough Δv and the latter that the propulsion system can provide relatively high thrust for the final approach phase. In the case where the debris has a rotation speed and the de-tumbling is not sufficient, the SC should be synchronized with the main axis of the rotation.

Operation

The nano-satellite is launched in LEO and captures the debris one by one. The plasma thrust enables the satellite to slightly deviate from its course in inclination, right ascension of the ascending node (RAAN), and altitude, to track the

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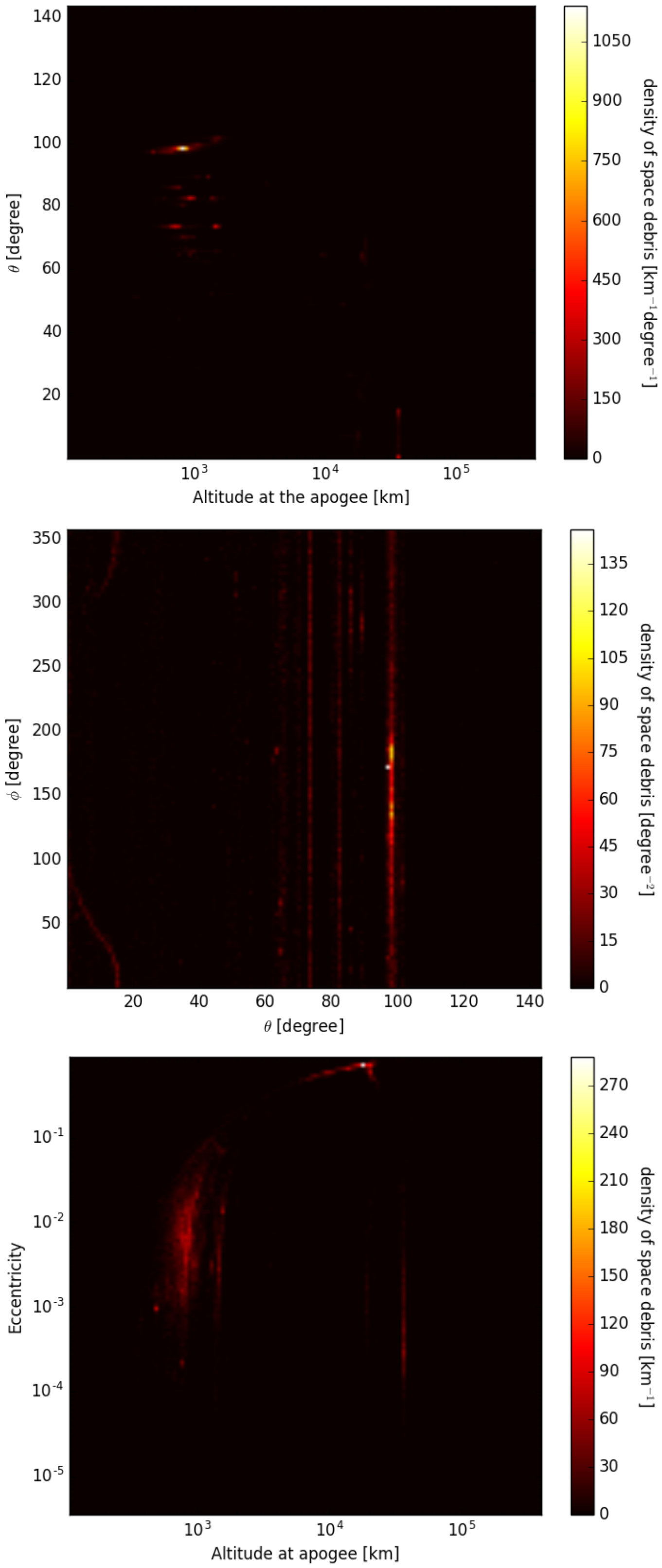

Figure 1: Distribution of tracked space debris in terms of apogee, inclination angle θ, right ascension of the ascending node (RAAN) ϕ, and eccentricity. Unclassified data form space-track.org.

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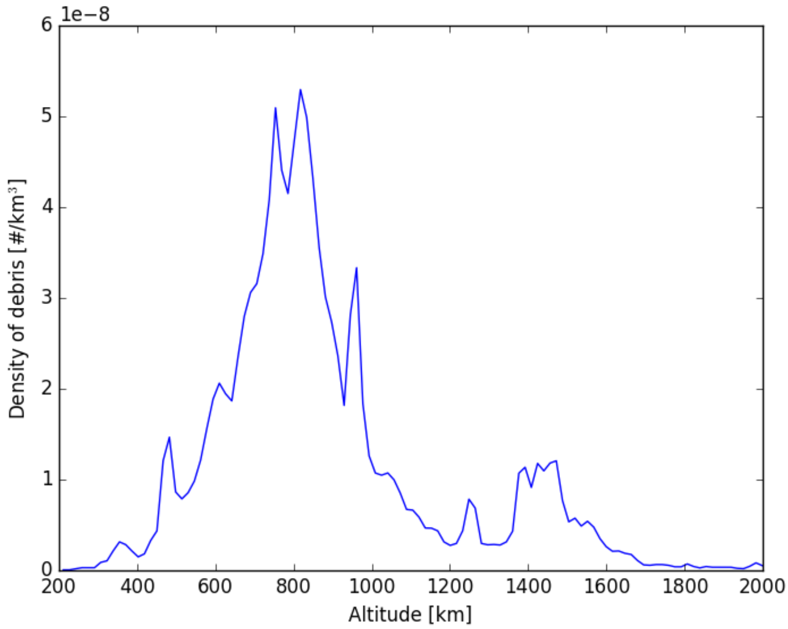

Figure 2: Density of catalogued space objects as a function of the altitude (at apogee) in LEO.

desired debris and to bring it properly to the center of an orbital domain assigned to this particular SC. Typically, one satellite could be in charge of cleaning all the debris orbiting around its own initial orbit, with a discrepency of ± 1° in inclination and RAAN, and ± 30 km in altitude. After a portion of orbit is cleaned, the satellite goes down to a lower orbit for further operation until the propellant tank is empty. Once there is no propellant left, the SC ultimately docks on the storage vessel. The sequence would start with a pulse to slightly decrease the satellite altitude, hence increasing it angular velocity. Once the chaser is at the same angular position as the debris, the propulsion system provides an other pulse to reach the debris vicinity (approximately 10 m). The first step of the capture phase could consist in de-tumbling the debris. A contact system will be used to catch the debris and to carry it away to a trash orbit where it is released on a temporary storage tank.

Technological choices

The design baselines for our Cubesat are in the process of being established. The main assumptions made in our models are as follows:

- Several standard parts for Cubesats are manufactured by several companies throughout the world (ISIS, Clyde Space). The change in attitude of the Cubesat is a crucial capability at each phase of the operations, for catching, transporting the debris, recovering from a spin or just cruising from one point to another
- Plasma Thrusters operate exclusively with electric power and feature low propellant mass consumption. For example, ThrustMe Start-up designed a plasma thruster operated with iodine which is about 20 times more affordable than xenon.[18] A finer estimate for operation time and required propellant resource will be provided in the next section.
- Several tests were made to design the crucial device that will catch the debris. At present, the most relevant idea is an expandable loop that can be orientated and tied around the debris as it is shown in Figure 3. This solution is compatible with debris having a large aspect ratio, but may be inadequate to catch e.g. spherical objects. The solution could be improved by adding another loop in a plane perpendicular to one of the first loop.
- It would be mandatory to have a visualization capacity of the debris capture, in order to change the Cubesat attitude, according to the relative position of the space debris.

The debris can have any position, orbit, and attitude angles. They can also have complex rotative motions around multiple axes. 3U Cubesats are standard on launch platforms and are becoming affordable. New services propose

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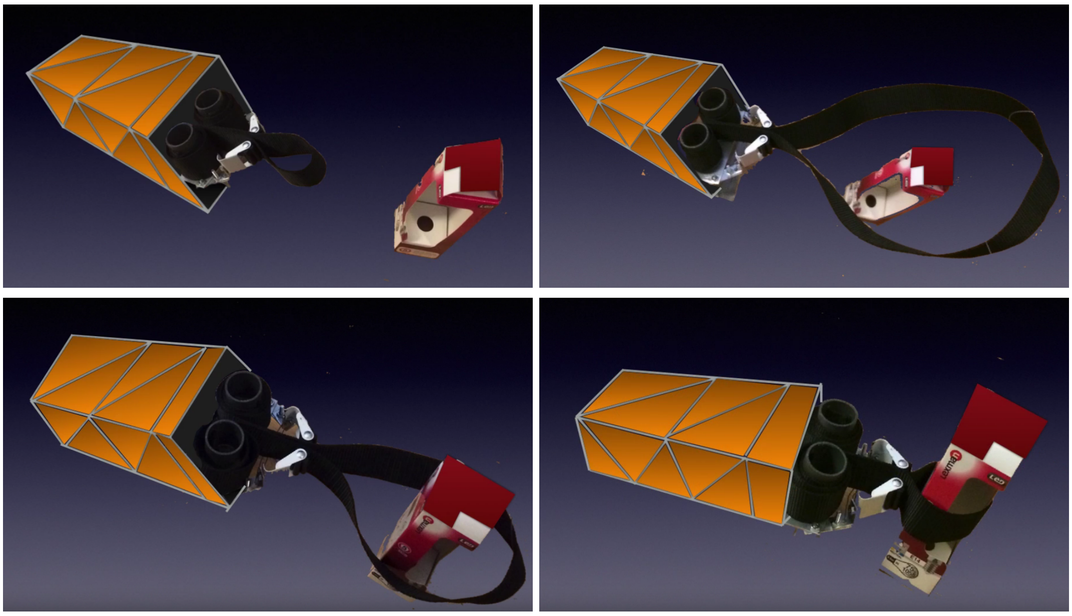

Figure 3: Demonstrator of the debris catching device.

3U launches for 180 000$ (Rocket Lab). Again, our main concern is to have the more economical and progressive approach. However, first design sketches may suggest that a 6U Cubesat would be necessary. This is in agreement with former studies that were conducted to demonstrate ADR technologies with 8U and 4U Cubesats.[17]

The plasma propulsion system (PPS) requires about 120 W of electrical power during operation. So the power on board should be at least 150W to sustain the electronics and minimal communications systems. With a yield of 26% for the solar panels, and 1 kW/m^2 of sun light (in normal incidence), this leads to 0.42 m^2 of solar panels. The 6 cubesat units will be aligned in one row and each of the lateral face will be covered by two layers of solar panels at launch that will be deployed once in orbit. This provides a surface of 0.48 m^2.

Cubesat maneuvers

Phase angle

In order for a SC to catch space debris orbiting at an altitude h, the easiest way is to reach the approximate same orbit and to decelerate by taking the benefit of Kepler's 3$^{\text{rd}}$ law for circular orbit:

$$\omega^2(R_T + h)^3 = GM_T \tag{1}$$

where ω is the angular velocity, G is the universal gravitational constant, and M_T is the mass of the Earth. It yields from 1 that it is possible to increase the velocity of the SC by decreasing its orbit altitude. On the contrary, increasing the altitude would lead to a decreased angular velocity. Mathematically speaking:

$$\omega(h) = \sqrt{\frac{GM_T}{(R_T + h)^3}} \tag{2}$$

A variation of h induces an absolute variation of ω such that, to the first order:

$$\Delta\omega = \frac{3}{2}\sqrt{\frac{GM_T}{(R_T + h)^5}}\Delta h \tag{3}$$

Or equivalently,

$$\Delta h = \frac{2}{3\sqrt{GM_T}}(R_T + h)^{5/2}\Delta\omega \tag{4}$$

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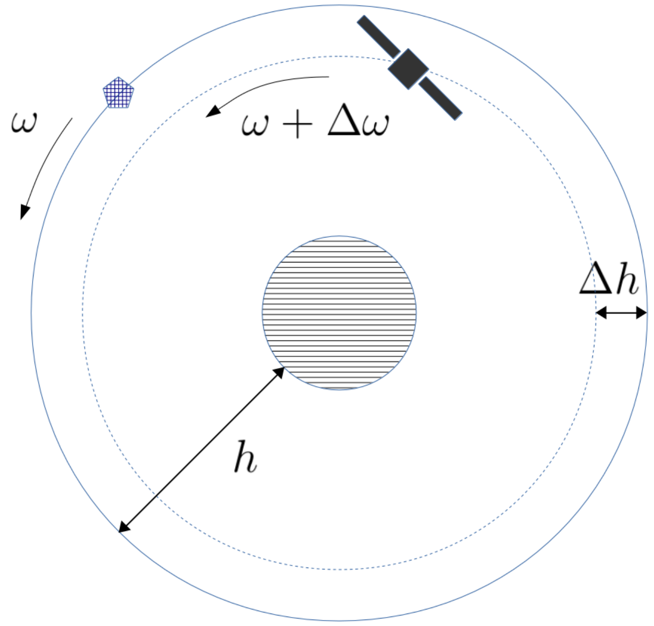

Figure 4: Diagram of a simple maneuver.

Therefore, the phase angle of the target can be reached by a simple change of altitude.

Let us assume in the followings that the SC has a dry mass called m_0 and carries a mass of propellant m_p. A good approximation for accelerating (or decelerating) the SC is to provide a thrust T in the direction that is tangential to the circular trajectory. Under this assumption, the mechanical power applied on the SC during maneuver is just (in absolute value):

$$P = Tv = T(R_T + h)\omega(h) \tag{5}$$

The sign is positive when the thrust is in the same direction as the velocity and negative when the two directions are opposite one to another. SC altitude would rise in the former case, and drop in the latter.
At a given altitude h, the SC has a mechanical energy E_m that reads:

$$E_m = -\frac{GM_T(M + m_p)}{2(R_T + h)} \tag{6}$$

So a change in altitude is related to a change in E_m and in the propellant mass such that:

$$\Delta E_m = \frac{GM_T(M + m_p)}{2(R_T + h)^2}\Delta h - \frac{GM_T}{2(R_T + h)}\Delta m_p \tag{7}$$

Given that the thrut is $T = I_{sp} g \dot{m}$ – where I_{sp}, g, and $\dot{m}$ are respectively the specific impulse, the gravitational acceleration at the surface of the Earth and the thruster mass flow rate – the propellant mass required to shift the orbit is the following:

$$\Delta m_p = \frac{\Delta p}{I_{sp} g} \tag{8}$$

Consequently, the conservation of mechanical energy yields to:

$$P\Delta t = \Delta E_m \tag{9}$$

where Δt is the duration over which the thrust is applied.
Therefore, by using equations 5, 7, and 9, the required pulse Δp for changing the orbit by Δh is estimated as:

$$\Delta p = T\Delta t = \frac{1}{\frac{2}{v} + \frac{1}{v_{ex}}} \frac{\Delta h}{R_T + h}(m_0 + m_p) \tag{10}$$

Finally, replacing Δh with equation 4 simplifies the formula by:

$$\Delta p = \frac{1}{3\left(1 + \frac{v}{2v_{ex}}\right)}(R_T + h)(m_0 + m_p)\Delta\omega \tag{11}$$

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Description	Variable	Value	Unit
SC mass	m_0	8.0	kg
Propellant mass	m_p	1.0	kg
Electric power for propulsion	P_{elec}	120	W
Thrust to power ratio	t_w	3.0×10^{-5}	N/W
Specific impulse	I_{sp}	3 000	s
Thrust	T	3.6	mN
Thruster mass flow rate	$\dot{m}$	1.2×10^{-4}	g/s
Angular velocity variation	$\Delta\omega$	1.2×10^{-6}	rad/s

Table 1: Test SC characteristics

Orbit altitude	h	[km]	300	1 400	36 000
Altitude shift	Δh	[km]	4.7	6.9	474
Required pulse	Δp	[Ns]	25	29	154
Time to shift orbit	Δt	[hours]	0.75	0.88	4.8
Used propellant mass	Δm_p	[kg]	0.0009	0.001	0.005

Table 2: Phase angle maneuvers at various altitudes

Let us consider the case of a SC that has the characteristics summarized in the following table. The reference thruster technology is a gridded thruster operated with iodine.

The angular velocity variation was calculated assuming that it would take 1 month to go to the opposite side of the orbit: $\Delta\omega = \frac{\pi}{1\ \text{month}}$. The main quantities discussed here were computed for altitudes of 300 km, 1 400 km (LEO), and 36 000 km (GEO) (Table 2).

The first thing to verify is that first order approximations used in section 2 were correct. We verify *a posteriori* that:

$$\begin{aligned} \Delta h &\ll h \\ \Delta m_p &\ll m_0 + m_p \\ \Delta t &\ll 1\ \text{month} \end{aligned}$$

So the analysis that was carried out here is relevant and the assumptions made are legitimate. If instead of $\Delta\omega = \frac{\pi}{1\ \text{month}}$, we had chosen $\Delta\omega = \frac{\pi}{1\ \text{hour}}$, the approximations made here would not have been correct and the conclusions would have been the opposite. In conclusion to this subsection, reaching the phase anfle of a debris can be performed at very low expenses in terms of propelant consumption, as long as the time given to achieve the maneuver is not too short.

Orbit raising

The launch altitude cannot always be customized for cubesats so the orbit raising has to be taken into account in mission design. Let Δm be the propellant mass needed to generate a circular orbit raising from an altitude h_1 to a higher altitude h_2. The velocity budget is provided by the vis-viva equation:[4]

$$\Delta v = \sqrt{\frac{GM_T}{r_1 r_2}} \left(\sqrt{\frac{2}{r_1 + r_2}} (r_2 - r_1) - \sqrt{r_2} + \sqrt{r_1} \right) \tag{12}$$

This equation is valid when the pulses are generated at optimal points only (perigee and apogee) and that when the pulse duration is much smaller than the orbit period. To reasonably satisfy these conditions, and to keep a good power efficiency for orbit raising (otherwise the energy balance that was performed above would become wrong), it is assumed that the thrust is generated only during 1/24$^{\text{th}}$ of the orbit period T_{orbit}, once at each revolution. Hence, the time for orbit raising is evaluated as:

$$\begin{aligned} T_{OR} &= \sum T_{orbit} && (13) \\ &= \sum 24 \frac{\delta m}{\dot{m}} && (14) \\ T_{OR} &= 24 \frac{\Delta m}{\dot{m}} && (15) \end{aligned}$$

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where $\dot{m}$ is the mass flow rate, δm is the mass used for one pulse, Δm is the total mass budget for performing the orbit transfer. The sum performed in equation 15 is made over the orbits necessary to achieve the orbit raising.

The propellant mass is evaluated using Tsiolkovski equation:[8]

$$\frac{\Delta m}{m} = \exp\left(\frac{\Delta v}{v_{ex}}\right) - 1 \tag{16}$$

Numerically, assuming that the initial launch is at 500 km of altitude (Rocket Lab launcher) and assuming the following SC characteristics:

- $m = 10$ kg
- $T = 3.6$ mN
- $P_{elec} = 120$ W
- $\dot{m} = 1.2 \times 10^{-4}$ g/s
- $I_{sp} = 3\,000$ s

the cost for orbit raising is summarized in the table bellow:

h	[km]	800	1 200	2 000
Δm	[g]	21	47	97
T_{OR}	[days]	47	107	220

Changing the orbit inclination

Once the SC is approximately on the same orbit as the debris, we have seen that it was quite inexpensive in terms of propellant mass consumption to reach a space debris phase angle. Another maneuver that the SC should be able to accomplish is a change of orbit inclination. As we will see, this costs a relatively larger amount of propellant.

We assume in the first place that the debris orbits are circular. The orbit is entirely defined by its inclination, RAAN, and altitude. It is assumed here that the target debris has the same altitude as the chaser, but with different angular parameters. Originally, there is an shift angle α between the two orbit planes. α is defined by the inclination angle θ and the RAAN ϕ as:

$$\alpha = \int d\alpha = \int \sqrt{d\theta^2 + d\phi^2} \tag{17}$$

So the angular momentum of the SC has to be modified using electric propulsion. The velocity budget for a change of orbit plane by an angle α is given by the following formula:[4]

$$\Delta v = 2v \sin\left(\frac{\alpha}{2}\right) \tag{18}$$

m_0 is here the dry mass of the SC and m_p is the propellant mass remaining once the satellite is in orbit. Tsiolkovski equation (16) yields the propellant mass required for this maneuver. Generally speaking, a fraction of the propellant mass has to be kept for orbit raising and altitude shift but most of it is used for the change in orbit inclination. Maneuvers that drive a change of the orbit plane require huge amounts of propellant, so we will assume in the followings that these will be performed for small angles only and we will write the previous formula to the first order, with small values of α.

The total angular path that the chaser can achieve hence writes as:

$$\alpha = \frac{v_{ex}}{v} \ln\left(1 + \frac{m_p}{m_0}\right) \tag{19}$$

A core feature of the chaser designed by Share My Space is its high Δv capacity. The high density of iodine conveniently stored in solid state leads to a very high storage capacity. To roughly estimate the total angle α, one can assume that $m_p \approx m_0$ and $\frac{v_{ex}}{v} \approx 4$. This yields to $\alpha \approx 0.9\pi$, which is very large compared with usual satellite performances.

More accurate results could be derived using a code for low thrust orbit transfer instead of vis-viva equation that is only applicable for Hohmann orbit transfer. Particularly, orbit raising and change of inclination orbital maneuvers can be performed together so as to reduce propellant consumption and time of transer. This was done by the authors using equation found in a technical report published by the University of Colorado that is available online.[12] The method and findings are described briefly in the last section.

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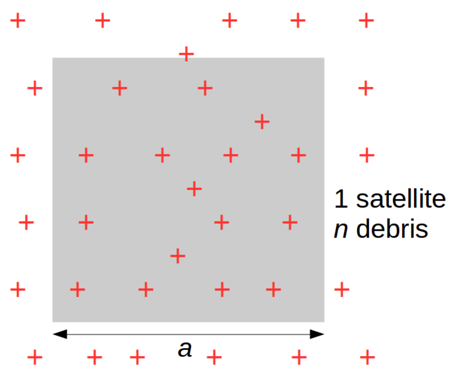

Figure 5: Diagram of the orbital domain that one chaser has to clean up. The red crosses represent the target debris.

Constellation management

Let us assume that one SC is responsible for cleaning a solid angle of the spherical shell parametrized by θ (inclination) and ϕ (RAAN). The positions of the debris near a given altitude in the orbit space are represented in a plane (θ, ϕ), where θ is the orbit inclination and ϕ is the RAAN. Locally, this area is assumed to be a square of a side a, where:

$$a^2 = \frac{\Omega r^2}{p} \tag{20}$$

p being the number of satellites in the constellation, Ω the target solid angle of orbit space that needs to be cleared of debris, and $r = R_T + h$ the SC distance to the center of Earth.

The chaser operation is divided in several steps that are repeated for each debris:

1. It starts from the center of the square
2. Reaches the next debris orbit (the average distance is $\lambda \approx 0.38a$ for a square)
3. Changes its altitude to reach the debris phase angle
4. Captures the debris - this step is crucial but is not fully in the scope of this paper.
5. Returns to the inclination angle corresponding to the center of the square (again $\approx 0.38a$)
6. Moves to the altitude of the garbage orbit for temporary storage.

The storage tank features minimum Earth control systems and the docking operation of the chaser to release the space debris could get inspiration from the docking of the Automated Transfer Vehicle (ATV) on the ISS for example. An inflatable structure design that would fit into a very reduced volume at launch is contemplated. This technology could capitalize on the know-how developed for NASA's Bigelow Expandable Activity Module.[14] An idea would be to use this structure to refuel the chaser with propellant. At the end of its lifetime, the chaser docks on the storage tank and remains there until the storage vessel is transported to recycle all the debris together.

We assume that the distribution of space debris in the solid angle Ω is uniform. If one takes only into account orbit plane change maneuvers, the SC performs in average a maneuver corresponding to an angle of $\frac{0.38a}{r}$ to reach one debris, and again the opposite maneuver to reach the storage orbit. Therefore, the SC performs over its lifetime a maneuver corresponding to a total inclination angle of:

$$\alpha_{tot} = 2 \times 0.38 \frac{an}{r} \tag{21}$$

where n is the number of space debris collected by one SC. Using equation 20, this is also:

$$\alpha_{tot} \approx 0.76\, n \sqrt{\frac{\Omega}{p}} \tag{22}$$

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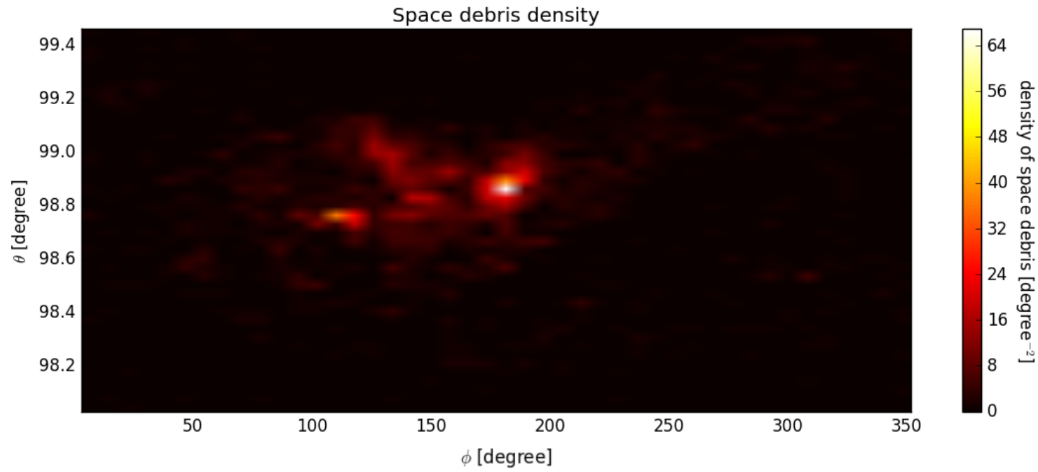

Figure 6: Distribution of tracked space debris in terms of inclination angle θ and right ascension of the ascending node (RAAN) ϕ for altitudes between 750 km and 850 km, eccentricities lower than 0.1, and inclination angles θ between 98° and 99.5°. Unclassified data form space-track.org.

Using the condition imposed by equation 19, the minimum number of SC p is related to the target number of debris per SC n by:

$$\frac{n}{\sqrt{p}} \approx \frac{1.32 v_{ex}}{v\sqrt{\Omega}} \ln\left(1 + \frac{m_p}{m_0}\right) \tag{23}$$

There is a total of N debris spread in the solid angle (in the orbit space) Ω that need to be removed. Given that $N = pn$, the number of satellites required to clean up this region is approximately:

$$p = 0.83 \times \left(\frac{Nv}{v_{ex}\ln\left(1 + \frac{m_p}{m_0}\right)}\right)^{2/3} \Omega^{1/3} \tag{24}$$

This formula is of high interest for the design of the clean-up missions because it relates the target debris characteristics – the number of debris N, their approximate altitude (given through the orbit velocity v) and the solid angle in which they are spread Ω – to the propulsion system performances – the exhaust velocity v_{ex} and the propellant mass relative to the dry mass m_p/m_0. This formula was derived under the assumptions that most of the propellant consumption came from the change of angle maneuvers, that the distribution of debris is uniform over the domain covered by one SC, and that the domain covered by each satellite is a square. These are strong assumptions so one should not expect fine results from this equation. However, it can yield relevant orders of magnitude and scaling laws, as it will be shown in the next section.

Results and discussion

Let us now focus on the near SSO orbit region where the debris density is the highest. There are 1937 tracked debris between 750 km and 850 km of altitude (as of April 22nd 2017) with inclination orbits between 98° and 99.5°. The distribution is shown in Figure 6. This corresponds to a solid angle of $\Omega = 0.16$ steradian. The domains covered by each SC in orbit space are not exactly squares and the density is not very much uniform in each square. Each chaser has 3 000 s of specific impulse, 8.0 kg of dry mass and carries 8.0 kg of iodine at the beginning of the mission. Simple application of equation 24 yields to a total of 38 satellites required to clean up those 1 937 debris.

In order to assess the validity of this result, low thrust orbit transfer formulas provided by S. Miller (University of Colorado)[12] were implemented. It was found that a constellation of 38 satellites each responsible of an area corresponding to 98°< θ < 99.5° and spread along the ϕ direction could collect up to 1 075 of the 1 937 debris of the domain of investigation. As shown in Figure 6, the distribution of space debris is not uniform and there are more space debris at around $\phi \approx 180°$. The concentration of chaser was also set higher in this area. For each chaser, the storage orbit was defined as the barycenter of the angular coordinates of all the debris orbits. The debris that were located closer to the storage orbit were caught first.

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Conclusion

A preliminary Cubesat design was proposed for multi-debris removal missions that require high Δv and high flexibility. In the prospect of removing current space junk, an emphasis should be set on polar to SSO orbits at about 800 km of altitude. Mission design could also anticipate the need expressed by operators of mega-constellations such as One Web to remove dysfunctioning microsatellites that would become a major threat for the other satellites of the constellation. Simplified formulas were given to estimate the relative costs for catching up a debris that is on the same orbit, raising the orbit, and changing the orbit angular parameters. These results were partially validated with exact equations for low-thrust orbit transfers. A statistical approach yields an estimate of the global cost of removing several debris with one SC. The design of the capture system still needs to be validated and a study is currently being conducted in order to assess the flexibility of the various kinds of thrusters available on the market. Electric propulsion with iodine is very suitable for small space debris removal but at the expense of an increased mission time. The temporary storage aspect was not much emphasized in this paper. Once the debris are concentrated in just a few on-orbit storage sites, it will be much easier to deal with them on the long term, for instance by reprocessing them on a space station in LEO (but this would require again a high consumption of propellant), or send them to the Moon to sustain future lunar economy.

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