ESTCUBE-2 PLASMA BRAKE PAYLOAD FOR EFFECTIVE DEORBITING

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ABSTRACT

In this paper, a novel, efficient, lightweight and cost-effective method for satellite deorbiting is presented. The plasma brake deorbiting technology is based on the electrostatic Coulomb drag effect utilising the momentum exchange between the atmospheric ion flow and a negatively charged body. ESTCube-2 is a three-unit CubeSat that will deploy an electrostatically charged tether to test deorbiting with the help of a plasma brake. The satellite will be equipped with a 300 m long tether, a tether deployment system and a high voltage supply to charge it up to 1 kV. Eventually, the system will occupy up to half a CubeSat unit with a tether mass around 22 g or less. It is estimated that such a system can reduce the orbital altitude of an average-size (4 kg) nanosatellite from 700 km to 500 km in half a year. While simpler deployment strategies can be utilised for operational missions, centrifugal deployment will be used on ESTCube-2 to provide precise measurements of the Coulomb drag effect with the tether spinning up and down the plasma stream. All of the satellite bus subsystems will be integrated into one miniaturised system occupying 0.6 units. During the development of the integrated satellite bus, the main requirements were to provide 45 Nms of total angular momentum in order to deploy the whole tether, to provide a means of deployment verification, and to provide up to 3 W of electrical power for the payload. The satellite is planned to be ready by the end of 2018 and launched in the first half 2019. Moreover, the plasma brake is scalable for larger satellites by increasing the length of the tether. The thin wire (<50 μm), that the tether is made of, creates minimal risk to other satellites in the event of accidental collision and is comparable with the impact from micrometeoroids and already existing debris impacting satellites continuously. Deorbiting time of theoretically damaged tether is analysed in the paper. This paper also presents alternative implementation of the payload – propellantless propulsion (E-sail). The ESTCube-2 satellite will evaluate an E-sail effect and will perform actual end-of-life deorbiting.

Key words: ESTCube, plasma brake, E-sail, space debris.

1 INTRODUCTION

The world of artificial satellites began in the middle of last century with launching the first object into orbit known as Sputnik-1 [1] by Russia (former Soviet Union). The first decade of the Space Era [2] brought fascinating opportunities to develop and demonstrate technological, social, economical, and safety (military/defence) aspects into people's lives. However, along with the opportunities it brought an unprecedented growth of uncontrolled man-made space objects (MSOs), the quantity of which is continuing to multiply presently. Consequently, it resulted in the unavoidable negative effect of ecological change in Low Earth Orbit (LEO) space exploration by increasing the risk of collision with the functional satellites and those to be launched in the future. Moreover, it also creates the risk of uncontrolled re-entry of upper-stage rockets, final stage vehicles, satellites, and their parts over populated or industrials areas. In the case of heavy and large enough objects, or those made out of materials with high melting temperatures, the probability of surviving the travel through Earth's atmosphere is above zero.

A study by the Inter-Agency Space Debris Coordination Committee (IADC) has shown that the population of space debris will increase due to collisions, even if nothing new is launched [3]. Each collision will create thousands of new debris that consequently will result in more collisions (Kessler syndrome). Collisions equivalent to that of the tragic Cosmos 2251 and Iridium 33, are predicted to take place every five to nine years. The measurable evidence is the recent collision of a millimeter-size particle with a solar panel on the

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Sentinel-1A satellite.

The number of MSOs that are still orbiting Earth and have been listed by the automatic warning system for dangerous situations in LEO was 12 376, as of 31 August 2015 [4]. One should mention that the current situation of space debris population is alarming and slowly switches its phase from inactive to absolutely aggressive.

In addition to the aforementioned statements, new missions will be designed in the upcoming years with continuous launching of spacecraft fleets and constellations, especially in LEO for Earth Observations (EOs), as well as a large number of educational nanosatellites. Launching CubeSats is typically more expensive than building one, which has escalated an enormous interest in low-cost small launch vehicles (LV). Twenty-two emerging small booster systems with payload capacity ranging from 4 kg to 2480 kg are under development, and some are currently operational [5, 6]. The trend indicates the global market's need in injection of a single or small number of flysharing smallsats into LEO.

This issue was regulated with a limit in the orbital post-mission lifetime of 25 years or 30 years after launch for all satellites in LEO [7]. The removal of a satellite can be achieved by lowering orbital altitude with further burning in upper stages of the atmosphere, or by controlled re-entry; rarely the removal can be achieved by the maneuvering of the satellite to a disposal orbit above LEO or geosynchronous orbit (GEO) that is hardly achievable for small spacecrafts.

Space junk combating techniques have been proposed over the globe in recent years. The methods vary from capturing satellites into the bag [8], contactless deorbiting by employing an ion beam [9], using passive electrodynamic drags [10], and a deorbiting sail [11]. Commercial deorbiting services can potentially be executed by D-Orbit and Astroscale. Deorbiting can also be achieved by implementing electrical (which are heavy) or chemical propulsions (which have a high power consumption), which would require a fully functional satellite at the end of the mission and might thus violate safety issues for nanosatellites. Relatively thick electrodynamic tethers create a high risk of collisions. The aerodynamic sail has a complicated deployment system and can be easily damaged by micrometeorites and extant debris.

The rationale behind using the aerodynamic brake has been seriously called into question because the method does not decrease the area-time product and hence it does not really lower the probability of collision [12].

2 PLASMA BRAKE EXPERIMENT

The plasma brake deorbiting technique taps momentum from the ionospheric plasma by employing a long, thin and conductive tether(s) that is attached to the disposal object. The disposal object is typically an artificial satellite at the end of its mission. The satellite moves through relatively immobile (in comparison to the 7-8 km/s orbital speed) ionospheric plasma, and if the tether is charged negatively by technical means, there results a friction force between the tether and the plasma ram flow, more known as Coulomb drag plasma brake [13, 14]. The Coulomb friction slowly brakes the satellite orbital speed, consequently lowering the orbital altitude. The system has low mass (<0.5 kg) and power consumption (~3 W), moreover it is safe to other space assets, even if unavoidable collision with tether occurs. The schematics of the gravity-stabilised experiment for a nanosatellite is shown in Figure 1. While simpler deployment strategies (e.g. gravity-stabilised tether) can be utilised for operational missions [25], centrifugal deployment will be used on ESTCube-2 to provide precise measurements of the Coulomb drag effect with the tether spinning up and down the plasma stream.

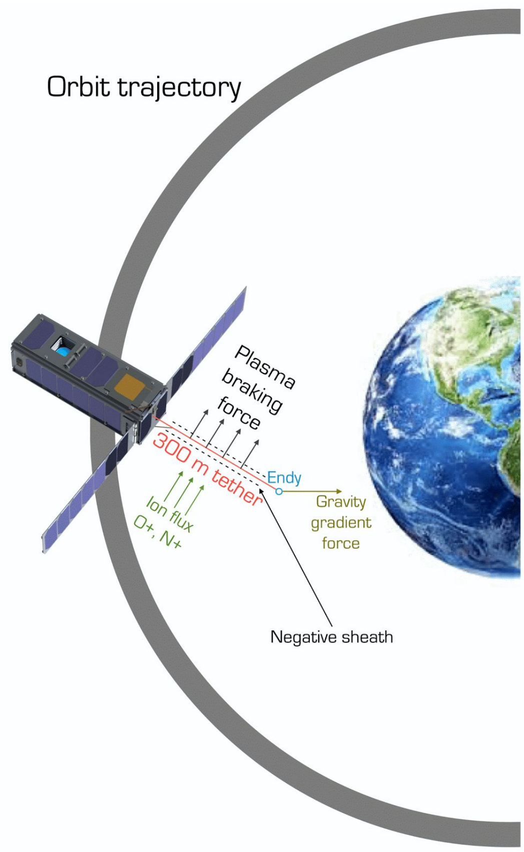

Figure 1. Schematics of the plasma brake experiment with a gravity-stabilised tether

The thrust develops by momentum exchange between the negatively charged body and the ionospheric plasma – it is **not** a traditional electrodynamic tether, where the thrust occurs by the current flowing in the tether [15]. In order to minimise the power budget for the deorbiting phase, the tether has to gather as low of a current as possible, and hence must be as thin as possible. The payload requires an electron collecting surface which is the satellite body itself. The braking force is controllable by tuning the voltage. The system can only lower, not raise the orbital altitude.

The ESTCube-2 experiment will be executed using a 300 m multi-wire tether with a single wire diameter of 35 μm and a total of width of approximately 2 cm. The single wire has a high risk of being cut by extant debris and micrometeoroids. The tether weighs 22 g (7.5e-5 kg/m) if made of gold. Other requirements include being conductive, providing sufficient mechanical strength, tolerating temperature variations, and resistance of ion sputtering and atomic oxygen (ATOX). If the tether or its part were to collide with other spacecraft at orbital hypervelocity, no significant harm or damage to the collided object would occur. Linear scratches resulting from such an accident would be equivalent to ones that spacecraft experience constantly under the nominal operation. A hypothetical loose tether piece deorbits passively in one week from an orbital altitude of 600 km. Typical atmospheric density for this altitude is 2.4e-13 kg/m^3. Considering the tether's cross sectional area and mass per length properties, the resulting acceleration is 1.3e-4 m/s^2. Decreasing the altitude by 200 km requires 100 m/s delta-v, which will be obtained in nine days by the conservative estimation. However, if passive electrodynamic and electrostatic effects as well as the atmospheric density function in relation to altitude are taken into account, deorbiting time becomes even shorter.

P. Janhunen's particle-in-cell (PIC) simulations study [16] using a supercomputer fitted the function in order to obtain the equation for plasma brake thrust per unit length calculation:

$$\frac{dF}{dz} = 3.864 \times P_{dyn}\sqrt{\frac{\varepsilon_0 V}{en_0}}exp(-V_i/V), \qquad (1)$$

where $P_{dyn} = m_i n_0 v_0^2$ is dynamic pressure, m_i is the ion mass, v_0 is the plasma flow speed relative to the satellite (assumed to be perpendicular to the tether). $V_i = \frac{1}{2}m_i v_0^2/e$ is the bulk ion flow energy in voltage units.

$$V = \frac{V_0}{ln(\lambda_D^{eff}/r_w^*)}, \qquad (2)$$

where $r_w^* = \sqrt{br_w}$ is the tether's effective electric radius, r_w is the tether wire radius, b is the tether width (~2 cm), and $\lambda_D^{eff} = \sqrt{\varepsilon_0 V_o/en_0}$ is the effective Debye length.

From Equation (1), one can infer that the thrust is proportional to the plasma density that consists mainly of oxygen ions, protons, and also some helium ions. The mass of the oxygen atom is 16 times larger than the proton mass. Therefore, oxygen-rich ionospheric plasma plays a big role in deorbiting efficiency.

Typical O^+ plasma density is 3e+10 m^{-3}. The ESTCube-2 voltage for the experiment is 1 kV. Taking into account Equation (1), the thrust per tether length produced by the payload is 85 nN/m or 2.5e-5 N for the entire length. A four-kg satellite will experience a deceleration of 6e-6 m/s^2 or 200 m/s delta-v per year. Hence, it will lower the ESTCube-2 orbital altitude from 700 km to 500 km in six months, which requires 100 m/s delta-v.

The mission failure risk is concerned around the tether being damaged or cut. In the case of the ESTCube-2 tether, the 10 μm and 3 cm fluxes are relevant, as the first is a hazard to the single tether's wire and the second impactor is capable of cutting the entire tether at once. A typical two-cm wide and 300-m long tether has an area of 6 m^2. According to the MASTER-2009 model version 7.02 for the year 2025, meteoroid and debris fluxes at 800 km altitude for 3 cm and 10 μm are equivalent to 1.6e-5 m^{-2}/year and 328 m^{-2}/year respectively. It results in 0.0096% single-blow breaking probability per year or ~0.02% per two years, the duration of the mission. A single wire's surface area is 1.08e-5 m^2 with a corresponding flux of 10 μm. The breaking probability is 0.35% per year or 0.7% for the mission duration. Taking into account the tether's four segments, the breaking probability per tether unit cell is equivalent to 0.007^4=2.4e-9 in two years. If one will assume that cells are 10 cm long, a 300-m long tether will have 3000 cells. It results in a total breaking probability of $3000 \times$ 2.4e-9=7.2e-4%. Hence, the total breaking probability of such tether is 0.02+7.2e-4 $\approx$ 0.021% over two years.

2.1 Electric solar wind sail (E-sail)

The ESTCube-2 deorbiting payload also serves as a testbed for estimating the electric solar wind sail's (E-sail's) [17] capability to produce thrust in LEO, and full performance system in lunar vicinity on-board ESTCube-3. If an equivalent tether is charged positively with a high voltage, consequently removing electrons from the system by employing electron emitters, it will turn to propellantless propulsion. The thrust [18] is generated by extracting momentum from the solar wind protons flow and is transferred to the spacecraft by thin charged tethers which deflect the proton flow. Such a system will work in the entire Solar System, where solar wind blows. An exception to the scope is inside the magnetospheres of planets.

In the case of the ESTCube-2 satellite, the E-sail effect will be detected by changes in the spin rate with a partly deployed tether or lower voltage, as there are limitations in LEO. The ESTCube-3 spacecraft will test a fully deployed tether with a voltage of 5-10 kV. The expected force per tether length is 250 nN/m at 10 kV, which provides 75 µN of thrust and an acceleration of 1.88e-5 m/s^2 for a 300 m long ESTCube-3 tether. The thrust will be detected by an on-board accelerometer(s).

3 ESTCUBE-2 AND ESTCUBE-3

The ESTCube-2 satellite is a three-unit CubeSat that will test a novel, scalable, lightweight and effective deorbiting technique based on the Coulomb drag effect. The same system will be involved in testing the E-sail by changing polarity and employing an electron emitters. The second and third ESTCube generations will have a miniature integrated bus (~0.6U) as an individual system, which will include all basic subsystems, a star tracker (ST), three reaction wheels (RW), and batteries in order to maximise the volume for payloads [19]. Among them is a dual-optical multispectral imaging system, C-band communication, and a thin protective coating experiment. Moreover, ESTCube-2 serves as a testbed for the ESTCube-3 mission beyond Earth orbit, where it will test the E-sail in its native environment – that of the solar wind. The main requirements for centrifugal tether deployment are addressed to the Attitude and Orbit Control System (AOCS) referencing the satellite's need in the relatively high spin rate around its X-axis. An exploded view of the satellite is shown in Figure 2.

The spacecraft is developed by the Estonian Student Satellite Foundation in cooperation with Tartu Observatory (Estonia), the University of Tartu (Estonia), ESTSat (Estonia), the Finnish Meteorological Institute (Finland), Ventspils University College (Latvia), GOMSpace (former NanoSpace, Sweden), the Laboratory of Thin Film Technology (Estonia), and Captain Corrosion (Estonia).

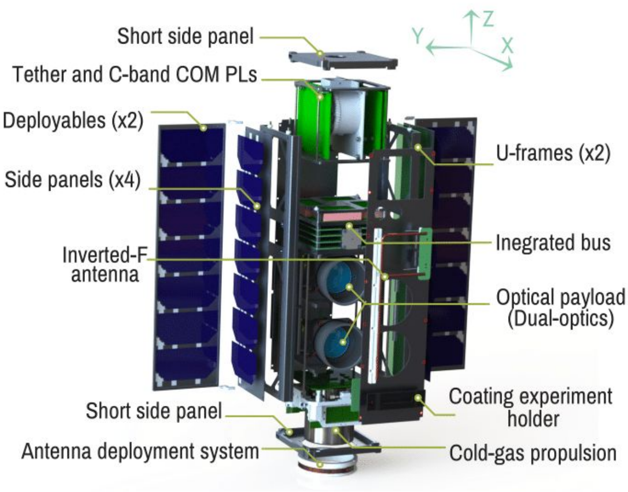

Figure 2. ESTCube-2 exploded view

3.1 Deorbiting payload

The initial design of the deorbiting system has been introduced in the ESTCube-1 payload [20]. The satellite reached a recorded spin rate of 840 deg/s, however the deployment of the 10 m tether failed due to the malfunction of the deployment system [21]. More robust design is implemented in the Aalto-1 satellite [22] which is scheduled to launch in April 2017 from India (PSLV). The ESTCube-2 payload will have a similar design to that of Aalto-1 and will be located in the long end (Z-plus) of the satellite. The name "Z-plus" comes from the axial direction. The Coulomb drag payload will occupy around half of the CubeSat unit and will be positioned perpendicularly with respect to the main satellite bus and Z-minus payload block. It will consist of two Printed Circuit Boards (PCBs): motor and high voltage (HV) supply boards. Figure 3 represents a computer-aided design (CAD) of satellite with the position of the tether payload integrated into the satellite. Endy is a 2.5 g mass at the end of tether, its main function being to keep the tether stretched by means of the centrifugal force while deploying and operating the payload.

The deployment of a 300 m long tether will be accomplished by employing the centrifugal force that will be executed in the Z-plus and Z-minus directions by spinning the satellite around the X-axis that approximately crosses the geometric center. In order to achieve a high spin rate, the spacecraft has to generate 45 Nms of torque. The deployment will proceed in two steps: firstly deploying 10 m of tether which requires 0.14 Nms angular momentum and 170 deg/s angular velocity; secondly an additional angular momentum will be created with a partly deployed tether (by actuators or Coulomb drag force itself) which will result in full deployment. The tether tension should be limited to

1 cN and no contact with sidepanels should occur, meaning that angular acceleration should not exceed 95e-3 deg/s^2.

Since ESTCube-2 and ESTCube-3 have to operate in different environments, the spinning will be achieved either by magnetic coils [23] (in LEO) or cold gas propulsion with reaction wheels (in LEO or the solar wind environment). Both satellites will be equipped with approximately 100 g of butane propulsion and three RWs. The propulsion has four nozzles pointing in different directions, in the Z-minus direction for ESTCube-2 and for ESTCube-3 in the Y-plus and Y-minus directions, meaning two in each. This will create 1 mN of thrust at a temperature of 15 ºC and will require an average of 1 W of power per quarter-hour of tank heating. Each RW is able to generate 0.1 mNm of torque [24]. The spin axis pointing accuracy has to stay within two degrees of error. By employing RWs and the ST, it is possible to achieve fine pointing with absolute accuracy under one degree. This is feasible, as other payloads' pointing requirements are more strict, such as that of the EO payload, with an absolute accuracy of 0.25º in order to prevent pixels blurring during exposure.

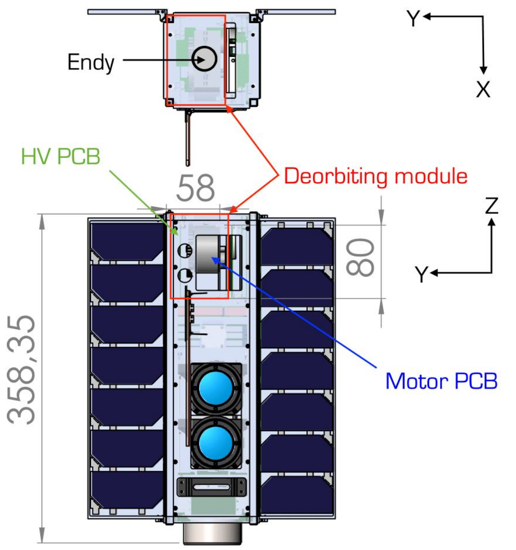

Figure 3. Deorbiting payload location (dimensions are in mm).

3.2 Motor board

The motor PCB has dimensions of 94x90mm and consists of a reel, reel enclosure, reel lock, stepper motor with controlled electronics, tether with an endy attached to the tip, endy lock, slip ring in order to charge the tether (connected to HV PCB), electron emitters for the E-sail experiment, and miniature camera. Electrons will be emitted through two round holes next to the deorbiting module shown in Figure 3, and will only be used for the E-sail experiment. It will have a more robust design in comparison to the ancestors, a space qualified motor, and an extensive testing campaign. Considering issues encountered during the ESTCube-1 mission [21], the following will be added to the design: a sensor for detecting a reel rotation, motor feedback, and a miniature camera (or equivalent sensor) for detecting endy.

The motor board will be connected to the payload bus, which is an RS485 bus between an on-board computer system (OBCS) and the satellite payloads. Communications will use CubeSat space protocol (CSP) which takes care of collisions, checksums, ordering, and addressing. On top of that, a simple command structure will be built, consisting of commands and arguments. In addition, the deorbiting module will receive unregulated voltage (6.6-8.4 V) from the electric power system (EPS).

3.3 High voltage board

In order to charge the tether, an HV direct current (DC) to DC converter, whose HV output is attached to an anode, will be used. A cathode could be either cold or hot, the latter meaning electrically heated. Currently, hot oxide-coated and thoriated cathodes are being considered, since the emission current of a plain tungsten cathode is not enough to conduct an E-sail experiment in LEO based on our estimations. However, solely for the deorbiting mission, either can be implemented. The biggest constraints in the cathode selection are the size and power consumption. For ESTCube-2, in the case of hot cathode, the payload will require 6 W of power, and for the cold, 1 W of power and more space.

The anode and cathode are about 1 mm apart, so that less voltage is needed to overcome Child's Law as shown in Equation (3). This states that the space-charge limited current in a planar diode is proportional to the three-halves power of the anode voltage and the inverse value of the square of the distance between the cathode and the anode.

$$J = \frac{Ia}{S} = \frac{4\varepsilon_o}{9}\sqrt{2e/m_e}\frac{V_a^{3/2}}{d^2}\,, \qquad (3)$$

where J is the current density, I_a is the anode current, S is the anode surface inner area, e is the magnitude of the charge of the electron, and m_e is its mass.

4 SCALABLE SYSTEM

The payload has been designed for experimental purposes. If the in-orbit results correspond with expectations, the system will be customised for bigger satellites (above 500 kg) and higher orbital altitudes (up to 1200 km). Taking into account Equation (1), it can be achieved by either increasing the number or length of tethers, or both. The limitations in numbers are determined by the deployment method, in length by manufacturing and collision probability. The main requirement for such a payload is to be independent, meaning to be able to decrease the orbital altitude of the spacecraft in the case of failure or at the end of a mission without requiring a satellite subsystems.

Centrifugal deployment is not preferred in the long term and would require a fully functional spacecraft and AOCS. The tether can be deployed by a miniature cold gas (CG) thruster on its tip, which makes deployment more complex, consequently increasing the price and decreasing the reliability of the system. A gravity-stabilised tether attached to a short tape tether, that is initially deployed by the spring, has been studied by FMI [25]. The study showed the feasibility of such an independent system that is limited to two tethers – one deployed downward (towards Earth) and other upward (to higher altitudes in relation with the satellite).

Taking into account the thrust parameters from Section 2 (85 nN/m), a 5 km plasma brake tether can reduce the orbital altitude of a 260 kg object by 100 km per year. Employment of a second tether accelerates the process proportionally.

5 CONCLUSIONS

This paper presents a novel deorbiting method for satellites in LEO. The experiment and thrust evaluation will be conducted on a 3U nanosatellite at the end of mission. The thrust is created by the momentum exchange between a negatively charged object and ionospheric plasma based on the Coulomb drag force. The negatively charged body is a long, thin, conductive multi-wire tether with estimated thrust of 85 nN/m per length for ESTCube-2 payload parameters. The tether does not create a risk either for operational satellites or for debris augmentation. It has been analysed for a collision probability, resulting harm, and disposal time in this paper.

The system can be implemented purely for lowering the orbital height, not for increasing the altitude as in the case of traditional electrodynamic tether. ESTCube-2 is predicted to be launched in the first half 2019, and will brake the orbital altitude from 700 km to 500 km in six months.

6 ACKNOWLEDGMENTS

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7 ABBREVIATIONS AND ACRONYMS

MSO	Man-made space object
LEO	Low Earth Orbit
IADC	Inter-Agency Space Debris Coordination Committee
EO	Earth Observation
LV	Launch Vehicle
GEO	Geosynchronous Orbit
ATOX	Atomic Oxygen
PIC	Particle-in-cell
E-sail	Electric solar wind sail
RW	Reaction Wheel
ST	Star Tracker
AOCS	Attitude and Orbit Control System
OBCS	On-board computer system
EPS	Electric Power System
CSP	CubeSat Space Protocol
PSLV	Polar Space Launch Vehicle
PCB	Printed Circuit Board
HV	High voltage
CAD	Computer-aided design
DC	Direct Current
CG	Cold Gas
FMI	Finnish Meteorological Institute
TO	Tartu Observatory
UT	University of Tartu

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