Scalable PnP Drag Sail Module Deorbit System for LEO Satellites

Preprint · August 2022

DOI: 10.13140/RG.2.2.17377.38244/1

Scalable PnP Drag Sail Module Deorbit System for LEO Satellites

[1]Anshuman Shukla, [1]Pranav Milind Sawant and [2]K.C. Mohite

[1]Student, C.P.V.N Kaimganj, U.P., India;

[1]Student, Army Public School, Pune, India;

[2]Principal, CT Bora College, Savitribai Phule Pune University, Pune, India;

Email: {[1]anshuman.contact07@gmail.com, [1]pranavsawant132@gmail.com, [2]kcmohite@gmail.com}

Abstract

The increase in use of Low Earth Orbits (orbits typically considered ranging from 250km to 2000km) for research, student, and global constellation satellites has increased the satellite density by large, posing a threat of collisions with other satellites and debris satellites. To avoid this, satellites are recommended to be deorbited as soon as possible after their mission lifetime. Various deorbiting techniques are being worked upon, especially passive ones as their size, mass and power budgets are usually lower than that of active ones. Drag sails are one such passive method popular in the field. In this paper, we propose a plug and play design of a drag sail module, different from the mechanical boom and electrical methods currently used, using Commercial-Off-The-Shelf (COTS) components for cubesats and satellites up to 500kg in mass for LEO orbits with altitude up to 1100km. A conceptual design is proposed, keeping the mission requirements from standard sample missions and trajectory analysis results in focus. This is followed by a conceptual control system analysis and cost analysis. The research provides a ready to use, scalable solution for the mentioned problem with COTS components and standard processes for trustworthy acquisition.

Keywords

Drag Sail, Satellite Deorbiting, Passive Deorbiting Systems, Sail Deployment, Plug and Play Design

Abbreviations

LEO - Low Earth Orbits

PnP - Plug and Play

COTS - Commercial-Off-The-Shelf

ADM - AirDragMod

IADC - Inter-Agency Space Debris Coordination Committee

UTIAS SFL - University of Toronto Institute for Aerospace Studies Space Flight Laboratory

ATOX - Atomic Oxygen

UV – Ultraviolet

WSCEA - Whole Solar Cycle Effective Area

ESA- European Space Agency

1. Introduction

In recent years, Low Earth Orbit (LEO) satellites have been constantly facing the problem of space debris requiring them to perform collision avoidance maneuvers [1]. With the growing consciousness for space debris, it is only a matter of time before satellite makers are compelled to incorporate a deorbiting system.

Heavier satellites could carry some extra fuel to perform deorbiting burns but are constrained by mass. Cubesats, due to their small size, don't face this often but with their growing popularity, it is important to ensure their timely deorbit. A collision avoidance maneuver is almost impossible for cubesats lacking any active control system owing to its limited mass, size and cost constraints, which reinforces the need for such a passive deorbiting system in order to avoid such scenarios in future.

Since deorbiting isn't an integral phase of a mission and doesn't affect its operations directly, a deorbiting system would have to be as "invisible" as possible viz. low mass, low volume, low operational power usage. Although the launch costs have significantly fallen [2], mass still affects satellites power usage during necessary attitude control thus affecting mission life.

The two most discussed and worked upon passive techniques for deorbiting are drag sail and plasma brakes. While plasma brakes are good for cubesats in LEO orbits, they aren't found to be equally effective in upper orbits for satellites of higher mass [3]. So, in this research paper, we propose a drag sail design which is easily scalable, easy-to-implement and uses low power. A drag sail deorbiting system is a system which uses a thin sail to maximize drag area of a satellite thus increasing perturbations of aerodynamic drag force.

$$F_d = \frac{1}{2}\rho A C_d v^2 \qquad (1)$$

Equation (1) is a basic equation governing aerodynamic drag on a surface. Here, F_d is the drag force, ρ is the density of the medium, A is the cross-sectional area i.e., area perpendicular to atmospheric flow vector, C_d is the coefficient of drag generally taken 2.2 for LEO orbits, and v is the orbital velocity of the craft. While the atmosphere of Earth grows thinner as higher orbits are attained, it still causes significant degradations in orbit of satellites in LEO over time which need to be accounted for using boost burns. A drag sail increases the rate of orbital decay, in compliance with current Space Debris Mitigation best practices and guidelines [4].

Although not a new concept, significant work has only been done in the past decade with TechEdSat4exo-brake launched in 2014 and successfully deorbited with a $0.35m^2$ of drag sail area. TechDemoSat-1 launched in 2014 with $6.2m^2$ of drag sail area, dragNET™ - a mission which successfully deorbited a Minotaur upper stage in 2016 utilizing a $14m^2$ drag sail area, CANX-7 (Canadian Advanced Nanosatellite eXperiment-7) [5] in 2017 with a $5m^2$ of drag sail area, removeDebris in 2018 with a drag sail area of $0.35m^2$. There are more missions currently in orbit or to be launched including the ADEO-N2 subsystem of 1U (10cm x 10cm x 10cm) size capable of stowage of $5m^2$ of drag sail by ESA's Clean Green Space Initiative. [6]

In this paper, we propose a Plug and Play (PnP) module design standardized using Commercial-Off-The-Shelf (COTS) components as a ready to use solution for the purpose of deorbiting. PnP design philosophy enables quick and reliable installation. This deorbiting module has been designed in two configurations: one for cubesats and other for heavier satellites of up to ~400kg in an orbit of up to ~900km. The cubesat configuration has two further variations. The design is inspired by JAXA's IKAROS [7] mission which was the first interplanetary solar sail propulsion mission launched in 2012 and performed a flyby of Venus. It weighed 300kg and consisted of a 14m x 14m spinning sail. It used angular momentum leftover after separation from its rocket stage to deploy 4 tiny masses, which due to centrifugal force extended deploying the sail attached to it. This technique is much more efficient than electrically or stored mechanical energy deployed systems as we would further discuss in this paper.

Being aware of all constraints, this work proposes a drag sail module- AirDragMod (ADM) design to deorbit cubesats and satellites of mass up to ~400km up to an orbital altitude of ~900km. Thus, in order to justify the capability of the design, a trajectory analysis of simulations ran in FreeFlyer® has been performed, keeping in mind the results, the design has been proposed and is divided into three phases: i. Conceptual design of both configurations, ii. Sizing of drag sail and module dimensions are calculated, iii. Control System conceptual analysis Then a comparison and cost analysis has been shown, and lastly concluded with the scope of further research on this.

2. Requirement Analysis

In order to determine the design requirements of AirDragMod, two different reference missions have been chosen, one for each configuration. For the cubesat configuration, CANX-7 [5] is the chosen mission, as it was a successful mission, the data from which is publicly available. CANX-7's main objective was to demonstrate the use of an external drag sail module. It was launched in a Sun Synchronous Orbit (SSO) with an altitude of 688km and an inclination of 98°. It had an ADS-B receiver as its payload. The 3U (10 cm x 10 cm x 34 cm) Cubesat platform weighed 3.75 kg and relied on a purely magnetic attitude determination and control system. 2U was allocated to space- craft housekeeping, payload systems and 1U was allocated to its 4 dragsail modules stacked in pairs of two one above another. The objective of the dragsail module is to assist with the quickest deorbit possible in coherence with the upper limit as set by Inter-Agency Space Debris Coordination Committee (IADC) guidelines which is 25 years [4]. In actuality, the deorbit time would be much less. The dragsail was deployed in 2017. It is expected to deorbit in roughly 5 years. This would later be used to derive the design dimensions.

Considering these characteristics of the CANX-7 mission, the primary objective of ADM cubesat configuration would be to deorbit a cubesat up to 10U from LEO (up to 1100 km) orbit to below 100 km. In addition, it would contribute to development of COTS Plug and Play (PnP) passive deorbiting system for cubesats and the advancement of dragsail induced deorbiting technology. For this, a size constraint of 1U (10 cm x 10 cm x 10 cm) form factor is set which implies a mass constraint of 1 kg. This configuration design must also be compliant with the CubeSat design specifications and secondary payload launching systems. For easy development and to reduce development cost and time, space-tested components and materials must be used and is a requirement. Because of creating a PnP module, other functional requirements arise which include: i. The ADM module should be easily attachable and its dimensions adaptable to host satellites. ii. Shall be independently deployable and act as a completely independent system using minimum host satellite power budget and components.

It should, as mentioned, be capable of deorbiting well within IADC guidelines. It should also be pointed out that while CANX-7 has been used as a reference mission for the cubesat configuration of ADM, it is not limited to the CANX-7 type of mission. This configuration in itself is intended to demonstrate adaptability to other UTIAS SFL spacecraft such as the Generic Nanosatellite Bus (GNB) and Nanosatellite for Earth Monitoring and Observation (NEMO) bus designs. An important requirement is the testability of the design in a 1g environment or another controlled environment easily creatable. [8]

The second configuration applies for satellites greater than 10U (1m x 1m x 1m) in volume and greater than 10kg in mass. Since there are no flown missions of satellites greater than the above-mentioned specifications which successfully deorbited using a drag sail in authors' knowledge, a result from 'ADEO De-Risk Dynamic Analysis' or ADDA [9] which was used by ESA for ADEO-N mission for the purpose of de-risk activity analysis ADDA following directly the ADEO (Architectural Design and Testing of a De-orbiting Subsystem) activity, is used for reference. Its purpose was the verification of the feasibility of de-orbiting a spacecraft using a drag sail. One of its finding was that the drag sail shortens the post mission lifetime significantly, for example, a $25m^2$ sail on a 300kg satellite from a 600km orbit de-orbits the satellite 97% faster in 5 years instead of 140 years. This would later be used to compare simulations.

To sum up, the requirements for the second configuration of ADM are mostly similar to that of the cubesat configuration. The objective of this configuration would be to deorbit a satellite of mass up to 500kg in up-to an orbit of altitude 1000km. For this, a size constraint of 2U (20 cm x 10 cm x 10 cm) is set which also implies a mass constraint of 2kg. It is ensured to be made of COTS components which are space tested for easy development and to reduce development cost and time. Similar to the cubesat configuration, it has other functional requirements due to being a PnP configuration viz. i. The ADM module should be easily attachable and its dimensions adaptable to host satellites. ii. Shall be independently deployable and act as a completely independent system using minimum host satellite power budget and components.

Both the configurations' system and their materials must also withstand the harsh environmental conditions in space such as the ATOX, UV and temperature environment.

3. Optional Increase In Deorbit Rate

To increase the rate of deorbit by ADM simulations were performed. The results are combinations of fixed-attitude and variable-attitude simulations. The varying attitude simulation results are needed as the fixed-attitude simulations fail to account for the time varying nature of the spacecraft's projected drag area. This could be accounted for using an active control system which is not viable for the lifetime of de-orbit.

Aerodynamic perturbations are not accounted for below 450km, nor is geomagnetic disturbance due to residual magnetic dipole moments above 650 km. Assuming a constant drag area could yield very misleading results and thus fixed-attitude simulation results are needed.

To assess this, short term attitude simulations were used to evaluate the de-orbit performance in slices of an entire ~11 year solar cycle and combined appropriately to arrive at a result which approximates the performances. The output (WSCEA [5]) can be compared against the constant area simulations performed on FreeFlyer®. The results in fig. 1 were obtained on two configurations of 0-degree angle between drag sailplane and residual dipole for 5 year deorbit profile and 90 degree for 10 year deorbit profile. The orbit considered was 700km altitude, 98.1-degree inclination orbit with LTAN ranging 500 HHMM to 2800 HHMM.

These results indicate that providing occasional stabilization in the form of nadir pointing/PID control to keep drag area constant would increase the rate of deorbit rate and thus decrease deorbiting time. The most effective stabilization being at ~1000 HHMM LTAN and ~2000 HHMM LTAN, as per the results, thus maximizing the duration of higher drag area. Similarly, stabilizations could be performed following simulations of respective missions using the ADM in future.

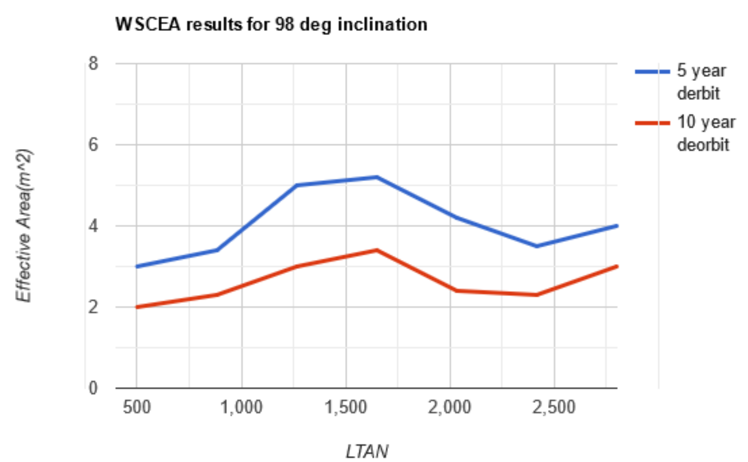

Figure 1: WSCEA results for mentioned orbit and inclinations

4. Adm Design

This section presents the design of the AirDragMod (ADM) which is the core of the paper which was carefully thought through several iterations. The main stages can be distinguished: the conceptual design, the deployment mechanism and design overview. These phases are briefly introduced in the following subsections.

4.1 Conceptual Design

During the conceptualisation of ADM's design, the selection of main configuration elements is performed. For that figure of merits defined from mission analysis defined from the mission analysis section is used. For ADM, three main elements studied are: the drag sail, the rotator, and the ADM structure layout. The design criteria followed during the decision-making process has been influenced by six key aspects: 1. fulfillment of deorbit requirements, 2. better approach to deployment than existing methods, 3. maximum drag sail area while minimum ADM volume occupied, 4. generation of a PnP module, 5. Preference for COTS components and space tested technology, 6. manufacturable design at minimum cost.

The ADM has been designed in accordance with the results. One of the objectives of this paper is to propose a scalable PnP module solving problems existing with the current techniques of deployment, specifically the stored mechanical energy boom deployment technique used in CANX-7. The ADM is based upon the Solar sail model used in the IKAROS mission by JAXA. The idea behind the deployment was that angular momentum of the module would cause four mass blocks connected with sail ends to extend due to centrifugal force.

Both configurations of the ADM use a similar technique to deploy their stowed sails. Based on this and the previously obtained result, four critical components were chosen. These are 1. Sail, 2. Rotator 3. Connector.

The first element chosen encompasses the sail configuration and the characteristics of its membrane. The influence of this element of the design requires selection of the configuration of sail. A square shaped configuration was chosen due to providing the best drag to mass ratio as well as having a broad-space heritage. The chosen sail configuration is a single unit with intrinsic divisions into 4 triangle quadrants. This configuration is perfect for deployment using angular momentum as the individual mass tips could uniformly spread out the sail and stay at maximum distance on its diagonal. The average size of sail to be used would stay $7m^2$. The proposed material of the sail is a metalized polymer similar to the one used for the sail of the CANX-7 mission. The polymer proposed to be used is a 12.7 micrometer Kapton film with a 300-Angstrom Aluminum coating capable of handling greater than 200° temperatures. This makes the mass/m^2 equal to ~7 for cubesat configuration and ~4 for sat configuration. The aluminum coating allows achieving high equipotential space-craft structures and avoids excessive charging of membrane. The material selection was in accordance with COTS philosophy and objective to use tested components. Secondly, since no boom deployment mechanism will be used, the absence of boom configuration would otherwise cause the deployed sail to fold in on itself as the masses loose tension. The spacecraft will not be kept in a constant revolving state as the increase in mass would mean increased power usage to keep a controlled spin rate; it increases communication hindrances. To solve this, a photopolymer in the form of epoxies or nitrile rubber would be used in the sail along its diagonals (fig. 2). Photopolymers harden under sunlight undergoing a process called curing [10]. The material selection again follows COTS philosophy and a preference to those already tested in space would be given. To be noted, this is only for cubesat configuration.

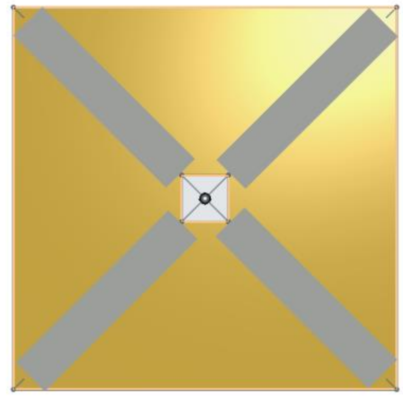

Figure 2: Photopolymer application at diagonals of sail

The second element of importance is the Rotator. It derives aspects of stowage from IKAROS probe design. The primary functions of the Rotator include stowage and deployment. The Rotator element is a cylindrical probe of diameter and length not exceeding 20 cm in both configurations. The sail is folded in a rolled 4 petal fashion similar to IKAROS. The IKAROS deployment mechanism utilizes the 4 stoppers which aid in deployment of the rolled petals using relativistic rotation mechanism (motor drive) and eventually are released to allow the expansion of then fully deployed petals into a square sail. The tip masses attached to sail ends are also stowed in Rotator and are the first ones to deploy on rotation aiding in the deployment process by the tension created due to experiencing centrifugal force. Each tip mass is 50g in mass. Angular momentum to deploy the sail is generated by a reaction wheel. Adhering to COTS and space tested component requirements, the reaction wheel suggested is CubeSpace™'s CubeWheel® Small+. It is the only rotating module and is further connected to the host space-craft by a Connector module.

The Connecter module is a passive module, not present in cubesat configuration, connecting the Rotator to the host satellite. It is fixed to the satellite on one end and extends a cylindrical rod until the end of the Rotator. The Rotator rotates around this rod of connector as the axis. The Connecter furthermore houses a housekeeping computer controlling the entire module primarily providing needed power control. To dump excess momentum and avoid tumbling in the satellite caused by this rotation, the connector consists of magnetorquer rods. Components chosen based on

COTS philosophy and space tested systems are NCTR-M002 Magnetorquer Rod. This isn't present in cubesat configuration as the Rotator is directly attached to the satellite if it is capable of holding together during the deployment phase. This is done to keep the mass and power usage low. Figure 3 shows the design of Connector and Rotator.

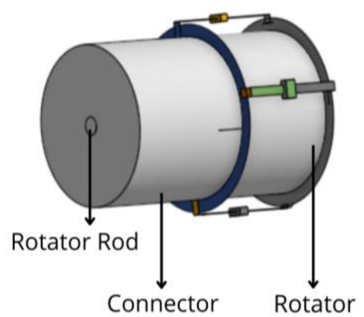

Figure 3: Connector, Rotator, Rotator Rod

4.2 Deployment Mechanism

Deployment process can be initiated from ground or host spacecraft. Once initiated, the power supply to the reaction wheel is regulated and increased to have the Rotator rotate at 5 rpm over a span of 6 seconds. As the rotation speed increases gradually, at 2 rpm the tip masses are released which are clutched mechanically. This creates a centrifugal force experienced by the tip masses which extends the sail-petals gradually connected to the Rotator by tethers. The spin rate is now increased to 25 rpm as the petals extend out while stoppers hold the membrane through the relative rotation mechanism. The spin rate gradually decreases as the petals extend half their size to ~15 rpm as no new torque is being produced. The spin rate on full extension of petals is ~3 to 4 rpm and on full extensions, the stoppers are released. As the stoppers “fall down”, the final stage of deployment begins and the sail starts acquiring its square shape from petals. By this point, the rpm is low at ~2. In cubesat configuration, this rate is kept for an hour under sunlight post deployment for the sail to harden at its diagonals after which the dumping devices are used to dump the rotation. In the other configuration, the Rotator is kept spinning at the rate as the Connector’s dumping magnetorquer dumps excess rotation and draws power from the satellite. Figure 4 represents the deployment sequence.

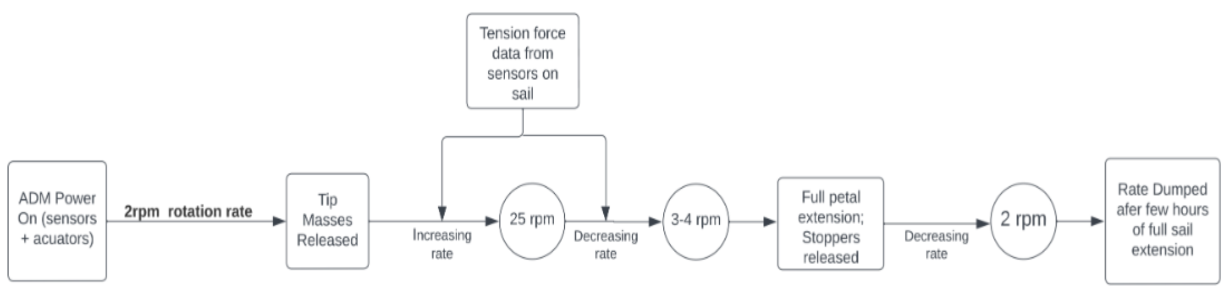

Figure 4: Deployment Sequence

4.3 Design Overview

To fulfill its initial objectives, the design should provide the said results within the cost and mass constraints minimizing both of them. Table 1 gives an estimation of cost and mass for both configurations, also justifying the division of design into two configurations visible by the considerable mass and cost difference mentioned in the table apart from the size difference described in Conceptual Design. The components are COTS and space tested on multiple missions previously thus adhering to initially said philosophies.

Table 1

Components and Cost

	Component	Mass	Estimated Cost (USD)
Cubesat Config	Chassis	700g	2500
	Reaction wheel	90g	5950
	Sail	~50g	500
	Dumping rod	30g	1200
	Electronics	1500g	700
		NET: ~ 1100g	NET: ~10k USD
Sat Config	Chassis	1000g	~3100
	Reaction wheel	90g	5950
	Sail	~28g	250
	Dumping rod	30g	1200
	Electronics	200g	800
		NET: ~ 1400g	NET: ~11k USD

5. Conclusions

This research offers an alternative deployment mechanism conceptualized in the form of above-mentioned design which is clearly plausible in theory. More testing of the mechanism in controlled environments along with prototyping and testing the design should be the next steps in developing a full-fledged design. The mechanism and sequence given could be schematized and tested on the prototype. More research is needed towards various deorbiting mechanisms to avoid increasing the already growing space debris problems.

6. Nomenclature

F_d = drag force,

ρ = density of the medium,

A = cross-sectional area i.e., area perpendicular to atmospheric flow vector,

C_d = the coefficient of drag

v = orbital velocity of craft

References

[1] Darren McKnight, Matthew Stevenson, Chris Kunstadter, Rohit Arora, "UPDATING THE MASSIVE COLLISION MONITORING ACTIVITY - CREATING A LEO COLLISION RISK CONTINUUM", Proc. 8th European Conference on Space Debris (virtual), Darmstadt, Germany, 20–23 April 2021, Ed. T. Flohrer, S. Lemmens & F. Schmitz, published by the ESA Space Debris Office, pg 1-9

[2] Harry W. Jones, "The Recent Large Reduction in Space Launch Cost", 48th International Conference on Environmental Systems, 8-12 July 2018, Albuquerque, New Mexico, pg 1-10

[3] G. Sánchez-Arriaga, J. R. Sanmartín, E. C. Lorenzini, "Comparison of Technologies for Deorbiting Spacecraft From Low-Earth-Orbit at End of Mission" (Accepted Manuscript)

[4] NASA, IADC -02-01, "IADC Space Debris Mitigation Guidelines", Revision 2, March 2020

[5] Shmuel, Barbara & Hiemstra, Jesse & Tarantini, Vincent & Singrayar, Fiona & Bonin, Grant & Zee, Robert. (2012). "The Canadian Advanced Nanospace eXperiment 7 (CanX-7) Demonstration Mission: De-Orbiting Nano- and Microspacecraft.", Conference: Small Satellite Conference, Logan, Utah, USA

[6] Jennifer L.Rhatigan, Wenschel Lan, "Drag-enhancing deorbit devices for spacecraft self-disposal: A review of progress and opportunities", Journal of Space Safety Engineering, Volume 7, Issue 3, September 2020, Pages 340-344, DOI: 10.1016/j.jsse.2020.07.026

[7] Osamu Mori, Yoji Shirasawa, Yuya Mimasu, Yuichi Tsuda, Hirotaka Sawada, Takanao Saiki, Takayuki Yamamoto, Katsuhide Yonekura, Hirokazu Hoshino, Junichiro Kawaguchi & Ryu Funase, "Overview of the IKAROS mission", pp 25-43 Advances in Solar Sailing, February 2014, DOI: 10.1007/978-3-642-34907-2_3

[8] F. M. Pranajaya, R. E. Zee, "THE NEMO BUS: A THIRD GENERATION HIGH-PERFORMANCE NANOSATELLITE FOR EARTH MONITORING AND OBSERVATION", 24th Annual AIAA/USU Conference on Small Satellites, SSC10-VI-8

[9] Sinn, Thomas & Garcia Hemme, Hugo & Schmid, M & Vogt, C & Ernst, Ruwan & Riemer, A & Hahn, R & Spröwitz, Tom & Seefeldt, Patric & Zander, Martin & Meyer, S & Bunte, K & Weikert, Sven & Brandt, S & Koch, M & Falken, A & Cardone, Tiziana. (October 2018). "THE ADEO PASSIVE DE-ORBIT SUBSYSTEM: REFERENCE MISSION SELECTION AND PRELIMINARY DESIGN OF PROTO FLIGHT MODEL.", 69TH INTERNATIONAL ASTRONAUTICAL CONGRESS, Bremen, Germany

[10] Saeed Bastani, Majid Mohseni, Chapter 7 - UV-Curable Nanocomposite Coatings and Materials, "Handbook of Nanoceramic and Nanocomposite Coatings and Materials", 2015, Pages 155-182, DOI: 10.1016/B978-0-12-799947-0.00007-9