

Parametric Study of Deorbit Sail Sizes for Small Satellites

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Abstract—The need for reliable and scalable post-mission disposal solutions is inevitable for small satellite missions in Low Earth Orbit (LEO). For spacecrafts without onboard propulsion system passive, fail-safe deorbit technologies can be essential to ensure compliance with orbital lifetime mitigation requirements. Deorbit modules using dragsails provide a robust approach and limited operational complexity without propellants, active control, or continuous ground intervention. This paper shows a parametric analysis of drag sails for a wide range of satellites, focusing on deorbit duration as a function of sail size, spacecraft mass, initial orbital altitude and solar activity. The primary objective is to present a general performance overview of the drag sail concept that enables rapid responses to potential customers regarding ADEO module selection and mission planning. Besides, the analysis identifies the physical limits of the applicability of this method, supporting strategic decisions on future product development.

As the ADEO family of deployable dragsail modules offers scalable solutions tailored to different satellite configurations and mission needs, this study is structured around two topics. Firstly, the selection of the most suitable ADEO module for a given small satellite mission. Secondly, the key results from the analysis of maximum initial altitudes up to which each sail configuration can meet the five-year deorbit requirement, evaluated based on a representative range of spacecraft masses.

In addition to deorbit performance, the analysis highlights drag sails as an effective risk mitigation solution for small satellites by shortening orbital lifetime, reducing long-term collision probability, and providing predictable end-of-life behaviour independent of spacecraft operability.

I. INTRODUCTION

Since the dawn of spaceflight, the rate of launches and the amount of space craft in orbit has been growing exponentially. Today, about 50400 objects are in LEO, many of which are already out of service [1]. This brings an ever-increasing risk for collisions, each collision creating more and smaller projectiles. After surpassing a critical threshold, the debris-multiplication would accelerate itself, potentially denying access to space for mankind for decades. This is known as the Kessler syndrome [2].

Agencies and companies are proactively working together to avoid this catastrophe. The European Space Agency (ESA), the Federal Communications Commission (FCC) and SpaceX independently implemented requirements for space-debris mitigation. One common denominator is the 5-year rule. According to ESA's requirements for example, the spacecraft needs to clear its orbit within 5 years after end of mission. If it

lacks manoeuvre capabilities, the time even starts running directly after launch [3].

In parallel, various technical solutions have been proposed and developed to deorbit spacecraft quickly and reliably. One in-orbit verified solution is ADEO [Fig. 1], a family of drag sail modules developed by HPS. The family consists of several ADEO modules of different sizes, tailored to various regimes of spacecraft size and altitude. Some ADEO modules have been developed in collaboration with ESA, others are tailored for the emerging new space market.

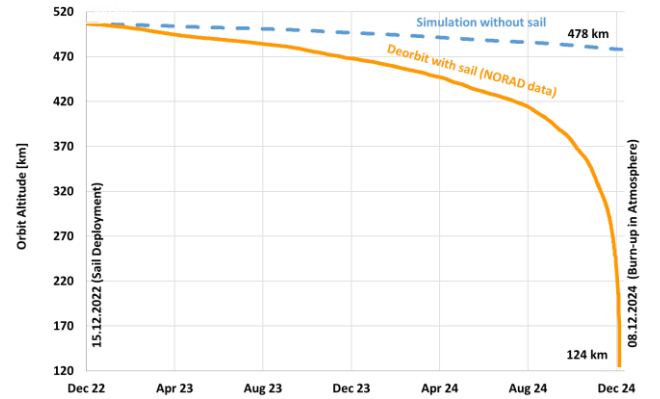


Fig. 1. Actual orbital decay of Dauntless David mission with ADEO-N

The product family includes various configurations of the ADEO sail: the ADEO-N series is designed for small satellite missions of 20-250 kg, while the ADEO-M and ADEO-L target larger satellites, 100-700 kg and 500-1500 kg. The ADEO-N series corresponds to sail sizes of $5 \pm 2 \text{ m}^2$, while ADEO-M covers areas up to 15 m^2 . Smaller versions are available as well for picosatellites (ADEO-P) with a sail size of 1.7 m^2 , and CubeSats (ADEO-C) in particular with 3.4 m^2 sail, and with the option to configure the sail size according to customer needs. The ADEO modules are designed to survive the harsh environmental conditions in space, including the UV, ATOX, and temperature environment.

The parameter study presented in this paper defines and validates an approach for orbit propagation simulations and illuminates the performance of the dragsail concept in LEO. The software used are the modules CROC (cross-section of complex bodies) and OSCAR (orbital spacecraft active removal) of the DRAMA (debris risk assessment and mitigation analysis) suite, version 3.1.0. [4]. The analysis is split in two parts. The first is the parameter validation analysis,

in which a set of simulation parameters is selected based on correlation between simulations and historic mission data. The second part is the generic deorbit performance prediction, which shows which advantage the dragsail concept can bring to a broad field of spacecraft configurations and orbits.

II. PARAMETER VALIDATION ANALYSIS

A. Objective

The objective of the parameter validation analysis is to set the parameters for the generic deorbit performance predictions based on specifications as well as correlation of simulation results with historic mission data.

B. Method

The deorbit using drag augmentation is mainly determined by the drag force created in the remaining atmosphere in Low Earth Orbit that acts on the spacecraft. Solar radiation pressure also acts as an influence but has minor effects in LEO compared to the drag force.

The drag acceleration can be calculated with (Wertz et al. [5] p. 214-):

$$a_D = -\frac{1}{2} \cdot \rho \cdot \frac{C_D \cdot A}{m} \cdot V^2 \quad (1)$$

Where ρ is the atmospheric density, C_D is the drag coefficient of the satellite, A is the cross-sectional area, m is the satellite mass and V is the velocity [5].

The cross-sectional area and the mass depend on the spacecraft specifications, the velocity depends on the orbital parameters. These parameters are mission specific. The drag coefficient C_D is a non-dimensional coefficient to describe the aerodynamic drag acceleration or force. The density ρ depends on the orbital parameters, mainly the altitude but also the date of the deorbit due to changing solar activity, which influences the atmosphere. DRAMA OSCAR uses the atmosphere model NRLMSISE-00 [4] to model the density. OSCAR offers five different methods to predict future solar activity based on ISO and ECSS standards, namely Latest prediction, Best case and worst case, Constant solar and geomagnetic activity, ECSS repeated solar cycle and Monte Carlo sampling [4].

Solar radiation pressure also needs to be considered in the simulation. DRAMA models the SRP using the reflectivity coefficient C_r [4].

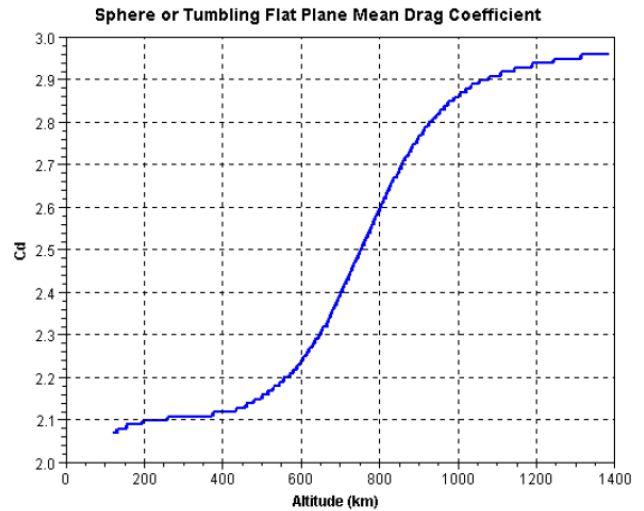
To ensure a reasonable trade-off between simplicity and accuracy when conducting deorbit simulations with DRAMA, both coefficients need to be defined and a method to predict solar activity needs to be chosen.

The drag coefficient is subject to considerable uncertainty, especially for complex spacecraft. Its determination is additionally complicated by its dependence on the orbital altitude, which is displayed in Figure 2.

Fig. 2. Drag coefficient diagram from STELA user manual **Error! Reference source not found.**

DRAMA only offers to input a constant value for the entire deorbit. For these types of analyses, mostly $C_D = 2.2$ has been used as an approximation. For example, ISO 27852 [7], p. 29) recommends its use: "A reasonable value of the dimensionless

drag coefficient, C_D , is 2.2 for a typical spacecraft". It is also mentioned by [5], p. 214 as well as the NASA report on State-of-the-Art of Small Spacecraft Technology published in 2025 [8] and Cook, 1965 [9]. The use of a fixed value is a



simplification that does not completely describe the physical circumstances and introduces an error to the simulation, but significantly decreases its complexity. A value of $C_D = 2.2$ has been used widely in literature and serves as an acceptable approximation.

The Reflectivity Coefficient C_r describes the reflectivity of a surface. The value ranges between 0 and 2, with $C_r = 0$ corresponding with a completely translucent body, $C_r = 1$ with an absorbing "black body", and $C_r = 2$ with a perfectly reflecting body [10]. Since a typical satellite will likely both absorb and reflect parts of the light, the value should lie between 1 and 2. ISO 27852 [7] uses $C_r = 1.5$ as the starting hypothesis condition for a simulation. Braun et al., 2016 [11] state that ISO 26872:2010 recommends the use of at least $C_r = 1.5$, but the source cannot be verified since the standard has been withdrawn. The literature research results on the reflectivity coefficient are not entirely conclusive but suggest that the value should lie between 1.5 and 1.9. A value of $C_r = 1.5$ is chosen for the analyses based on the ISO standards mentioned.

The five solar activity models offered by DRAMA follow different approaches. The first, latest prediction, is recommended by ISO 27852 [7] and uses a historical mean cycle based on the solar cycles 10 to 23 and compares it with available ESA up-to-date data to predict the future behaviour of the current cycle. The second option, Best Case and Worst Case, allows the user to enter a confidence interval in percent, which defines the bottom and top value of solar activity based on the historic solar cycles, which are used for the simulation. The third option, constant solar and geomagnetic activity, uses constant values for the entire deorbit time. The fourth option, repeated solar cycle, is based on the space environment standard of the ECSS. It uses the 23rd solar cycle for all future predictions. The last option, recommended by ISO 27852 [7], uses the Monte Carlo method to sample the solar activity by randomly drawing data from the preceding five solar cycles. The four options (excluding Best Case and Worst Case, since it returns the same value as Latest Prediction as well as two values based on the best- and worst-case scenario) were evaluated by comparing the simulations to factual deorbit data from historical reference missions to evaluate their accuracy.

The reference missions were selected with regard to the following criteria: high diversity of dimensions, mass and orbital altitude; certainty that no other deorbit devices/propulsion were used, except drag augmentation and completeness of available data. Based on these criteria 30 reference missions were selected, including drag sail tests like NanoSail-D2 and CanX-7, large satellites like BeppoSAX and ASCA, rocket bodies and CubeSats deployed from the ISS.

The cross-sectional area of the satellites is determined with the DRAMA tool CROC based on the researched dimensions and assuming a randomly tumbling state of the un-controlled satellite, in accordance with ISO 27852:2024 [RD4].

Thirty spacecraft being analysed with four different options respectively means that 120 simulations are conducted.

C. Results

The simulation results show that Latest Prediction is the best option, since it has the average absolute error with a value between 17% and 22.4% and a similar standard deviation to Monte Carlo and ECSS sample cycle. Constant Solar Flux has the highest error rate and is clearly not usable for deorbit predictions. Monte Carlo and ECSS return similar results, as they are both significantly more accurate than Constant Solar Flux but less accurate than Latest Prediction, but show a slightly better standard deviation. Based on this analysis, the use of Latest Prediction with up-to-date ESA data is recommended. The errors of the simulation cannot be exclusively attributed to the solar and geomagnetic activity since many simplifications were used in these simulations.

CD = 2.2 and Cr = 1.5 are set for future simulations according to the conducted literature research.

III. GENERIC DEORBIT PERFORMANCE PREDICTION

A. Objective

The objective of the generic deorbit performance prediction is to sketch out the realm of feasibility of the drag sail concept measured by the compliance to the 5-year rule.

B. Method

The described objective requires relevant S/C geometries and masses to be simplified, to achieve general applicability, minimal simulation time and easy understanding. Similarly, the space of all possible and relevant initial orbits has to be simplified, again seeking a trade-off between applicability and simulation time and complexity.

The values for drag and reflectivity coefficient, as well as the most suitable method to model solar activity were determined in the previous analysis. The starting date for the deorbit simulation needs to be varied over one solar cycle, averaging the results, in accordance with the ESA Space Debris Mitigation Requirements [12]. Eleven starting dates with a one-year spacing are chosen in solar cycle 25.

For the deorbit simulations, consistent orbital parameters have to be chosen, based on statistical data, since the results should represent the most common cases for satellites. The inclination is chosen to 97.5 degree based on satellite distribution data from the Satellite Database of the Union of Concerned Scientists [13]. It is assumed that most LEO orbits are nearly circular, therefore have an eccentricity of close to 0. [13] provides a list of current satellites with orbital parameters and operational details, where the average

eccentricity of all listed LEO satellites (over 6000) equals 0.0009058. This value is close enough to zero that it is deemed a fair approximation to conduct every simulation with an eccentricity of 0. The RAAN is decided to have a value of 0°, since differing RAAN values have a low impact on deorbit time. The argument of perigee does not apply, since a circular orbit does not have a perigee. The mean anomaly is set to 0°.

To model satellites in general with a reasonable compromise between simplicity and accuracy an extensive list of existing satellites (93 satellites) with different dimensions, mass, and solar array size is created. The criteria for the selection were: high variety in dimensions, mass and solar array size; close to equal number of CubeSats and non-CubeSats and reliable available data on dimensions, mass and solar array size. CubeSats and non-CubeSats are evaluated separately, since they are estimated to vary greatly in density and solar array size. From the researched data functions are developed to describe the correlation between the dimensional variables mass, volume and solar array size, seeking the closest approximation. Ten standard satellites (5 CubeSats and 5 non-CubeSats) are defined. The standard CubeSats are: 1U, 3U, 6U, 12U and 16U and the non-CubeSats are: 100 kg, 200 kg, 300 kg, 600 kg and 1000 kg. Their dimensions are averaged, using the approximation function determined before, based on an input variable. For CubeSats, the input variable is the volume, since these satellites have predefined form factors and therefore volumes according to the CubeSat Standards [14]. From the standardized volume, mass and solar array size are approximated. For non-CubeSats the mass is the input variable, since it is the defining parameter. Based on the mass, volume and solar array area are approximated. With these dimensions, CROC models can be designed to simulate the cross-sectional areas.

Different HPS sail models will be used for the deorbit simulations. The available models range from a sail size of 1.4 m² to 25 m² [Fig. 3] (100 m² hypothetically) and are therefore applicable to a large variety of satellites. These sails are integrated in the CROC models for the deorbit simulations with sails.

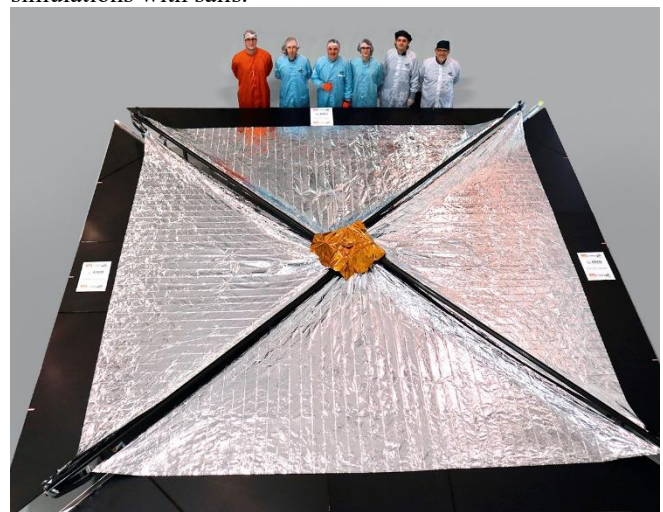


Fig. 3. ADEO2 PFM ambient deployment at DLR with 25 m² sail

For each case the highest possible starting altitude, that still fulfils the 5-year deorbit requirement, is determined in iterative simulations. Every one of the ten satellites is

simulated without sail and with all possible sail choices. Possibility means here that the sail mass does not exceed the total mass of the satellite.

For every of the 10 standardized satellites 8 (different sails including no sail) times 11 (different simulation starting dates), yielding 88 simulations are conducted. From the resulting altitudes the arithmetic mean is calculated for every satellite-sail combination, as well as the minimum and maximum altitudes determined.

C. Result

The densities also vary largely between CubeSats and non-CubeSats, as has been predicted. The non-CubeSats have an average density, calculated from the satellite mass m and volume V of:

$$\rho = \frac{m}{V} = 233.3 \frac{\text{kg}}{\text{m}^3} \quad (2)$$

And the CubeSats:

$$\rho = \frac{m}{V} = 1116 \frac{\text{kg}}{\text{m}^3} \quad (3)$$

The results of the simulations are displayed in the following diagram, where the lines always represent the averaged results for every satellite and the markings over and under the lines show the minimum and maximum results for this satellite.

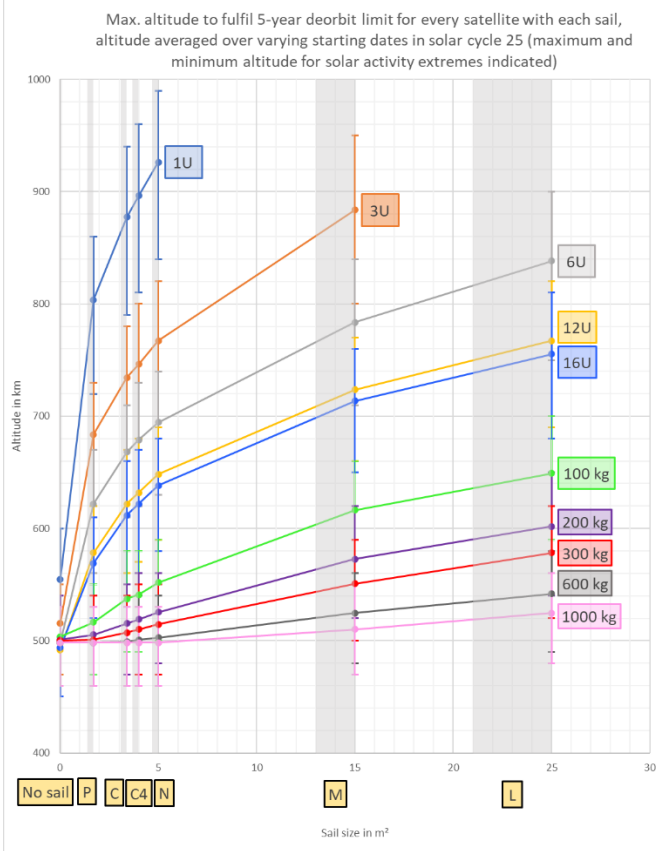


Fig. 4. Recommended sail size depending on the altitude and the spacecraft mass

IV. CONCLUSION

This study confirms that drag sail modules such as the ADEO family provide a reliable and scalable solution for passive deorbiting of small satellites in LEO. Based on the simulations with multiple sail and spacecraft configurations, and varying solar conditions, the maximum initial altitude meeting the five-year requirement was determined and statistically evaluated.

The results show that drag sails are highly effective for satellites up to ~300 kg, enabling compliance with the deorbit rules. For larger spacecrafts, they remain viable, though the improvement in maximum starting altitude decreases, reaching a practical limit of around 600 km. Using averaged ESA-recommended solar conditions ensures robust and predictable performance.

Overall, this analysis provides a practical framework for satellite operators to assess drag sail suitability and select the appropriate ADEO module, while supporting safer and more sustainable space operations.

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