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DEVELOPMENT OF COMMERCIAL DRAG-AUGMENTATION SYSTEMS FOR SMALL SATELLITES

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ABSTRACT

In the framework of the ESA CleanSat programme Cranfield University is developing a family of drag augmentation system (DAS) modules to enable small satellites in Low Earth Orbit (LEO) to comply with space debris mitigation requirements.
There are currently two mature Cranfield DAS designs based on deployable Kapton sails using stored energy for deployment. One concept is Icarus and it is currently on-board the UK's TechDemoSat-1 (launched 8 July 2014) and Carbonite-1 spacecraft (launched 10 July 2015). The second concept is the de-orbit mechanism (DOM) module, which is due to fly as technological demonstrator on the upcoming ESA ESEO mission.
The key drivers used during the design process were: low cost, low mass, easy testability, safety, reliability, and avoidance of additional debris production. These drivers matched with top-level requirements, from a potential customers perspective (e.g.: satellite integrators), which were defined during the CleanSat study. Other relevant requirements for the DAS included demisability, performance (in terms of orbital decay), area-to-mass ratio, functionality, lifetime, and environment compatibility.
This paper discusses the compliance of the Cranfield DAS designs with the identified requirements, and illustrates the scalability via application to several case study missions (500 kg and 200 kg LEO satellites).
The two most challenging aspects to assess were compliance with the lifetime required for storage on ground and pre-deployment on orbit, and the effect of the orbital environment (radiation, ATOX, debris) on the sail.
The study has provided useful input to explore new concepts based on the heritage designs; these concepts are evolutions of the DOM unit and hybrid designs. The hybrid design combines aspects of the Icarus and the DOM concepts to reduce the limitations of the respective individual devices and improve scalability, adaptability and manufacturability.
In addition, this work is helping to achieve commercial readiness for the technology. This will enable development of a commercial DAS offering that will be an attractive solution for small satellite integrators, allowing them to meet debris mitigation requirements.

Keywords: de-orbit; DAS; drag augmentation; debris mitigation; CleanSat.

1. INTRODUCTION

CleanSat represents the programmatic response of ESA to support the European industry in designing technology for debris mitigation to reduce the production of space debris. ESA Clean Space is coordinating the CleanSat programme, which is addressing three key areas for spacecraft to be launched:

- Design for Demise: to ensure spacecraft comply with the casualty risk on ground (shall not exceed 10^{-4} [1]) and that they demise upon re-entry.
- De-orbiting systems: to remove satellites in LEO within 25 years after the end of mission, ideally without reduce mission efficiency.
- Passivation: to permanently deplete or make safe all source of stored energy on-board (mainly propulsion and power subsystems).

CleanSat is supporting LEO platform evolution through technology assessment and concurrent engineering studies [2] to mature and develop suitable ”building block” (BB) that will help satellite designers and manufacturers make their future satellites compliant with current best practice guidelines and regulations.

In the framework of the CleanSat programme Cranfield University is developing a family of drag augmentation system (DAS) modules to enable small satellites in Low Earth Orbit (LEO) to comply with Space Debris Mitigation (SDM) requirements [3]. The disposal manoeuvre approach selected, allowing a clearance time within the 25 years after the End of Mission (EoM), is augmenting the spacecraft orbital decay by deploying a device so that the remaining orbital lifetime is compliant (ISO 24113 [4]).
Cranfield has already supplied two research de-orbit modules which are now in orbit and is building a third flight model, and thus has unrivalled flight heritage for

the technology. The CleanSat study helped evaluating the design options, manufacturing, AIT and operational factors to define DAS modules which designers can use to ensure future satellites compliance with SDM requirements.

This paper is aimed at potential users of the DAS technology, as well as at developers of similar technologies who might face analogous issues, and find it useful to compare modelling techniques and approaches adopted. Finally, it is of relevance to policymakers, who can have an overview of what can be achieved using the DAS modules.

2. DAS BASELINE DESIGNS

The aim of this part of the study was to develop a baseline design, which is suitable for development into a technology BB. The study considered four design concepts for a DAS module. The first two concepts (here referred as heritage designs) are already at TRL 6; while the other two DAS, which are derived from the previous concepts, are at an earlier design stage.

Despite differences in the layout, the DAS modules developed by Cranfield share the following characteristics:

- Satellite drag is increased by deploying a lightweight membrane supported by rigid booms;
- Deployment is achieved using stored spring energy;
- DAS should pose negligible risk to the host satellite's operation;
- Deployment is controlled by the host spacecraft;
- DAS design is compatible with low-cost manufacturing and testing facilities.

In addition, the Cranfield DAS modules are typically suitable for small sized satellites (m<1000 kg) in LEO with altitudes approximately below 800 km.

2.1. Heritage designs

There are currently two mature Cranfield DAS designs based on deployable Kapton sails using stored energy for deployment: the Icarus design (with flight heritage), and the De-Orbit Mechanism (DOM) design (flight qualified).

2.1.1. Icarus design

The DAS concept with flight heritage is currently onboard the UK's TechDemoSat-1 (launched 8 July 2014) as Icarus-1 payload [5] and on the Carbonite-1 [6] spacecraft (launched 10 July 2015) as Icarus-3 payload.

The Icarus DAS consists of a thin aluminium frame, fitted around one of the external panels of the spacecraft, in which four trapezoidal Kapton sails and booms are stowed and restrained by a tough band. Deployment is achieved by cutting the band, activating cord cutter actuators, which allows the stored energy in spring hinges to unfold the booms and sails. The design layout and the release mechanism for deployment are discussed in [5].

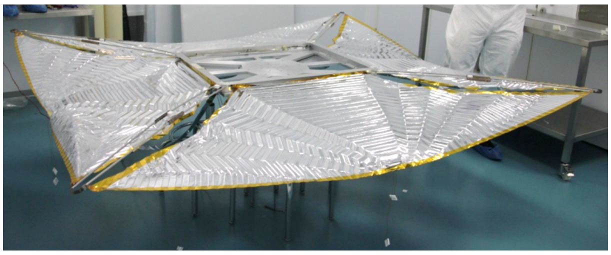

Figure 1. Icarus-1 in deployed configuration in Cranfield clean room.

2.1.2. DOM design

The second concept is a compact DAS module supplied in a small cuboid outline. The DOM module can be attached wherever appropriate on the host spacecraft and a technological demonstrator is due to fly on the upcoming ESA ESEO mission [7].

The DOM does not have flight or qualification heritage at full system level; however it draws on qualification heritage from the Icarus payloads for the cord cutters, the aluminized-Kapton sails, and using stored spring energy for deployment.
The mechanical design is different from Icarus, in fact the four triangular sails and the four booms are rolled up around a central spool in the middle of the device and held in the stowed configuration by Kevlar cords. In particular the booms act as tape springs themselves, while in Icarus there are only tape spring hinges.

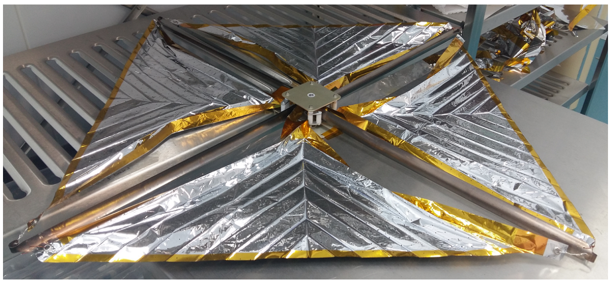

Figure 2. DOM Flight Model in deployed configuration in Cranfield clean room.

Once the ARM and FIRE command are sent, by closing a series of relay switches and thereby activating two cord cutters, the Kevlar cords are cut and the strain energy stored in the booms is transferred into kinetic energy about the central spool, resulting in deployment. A more detailed design description is presented in [8].

2.2. New design concepts

The new concepts are being developed considering the gap in the family of heritage designs.
For example there is a gap in the family of DAS for a device that is compatible with a satellite that may not have rectangular panels, or four edges available and also does not have a panel that is free of protrusions [9].

2.2.1. DOM evolution concept

The first new concept is a very simple adaptation, which focuses around the current DOM. One of the main constraints on the DOM is that it cannot be deployed when there are protrusions on a panel as these prevent the sail from fully deploying. By adapting the configuration to allow for these protrusions it can greatly increase the number of satellites that are compatible.

The DOM can be adapted by removing one of the four sail segments within the concept (Fig. 3). Through careful placement of the DOM on the corner of a satellite panel, the booms can be orientated to allow for the three remaining sail areas to deploy without obstructing the S/C panel and allowing to accommodate protrusions such as antenna, etc.

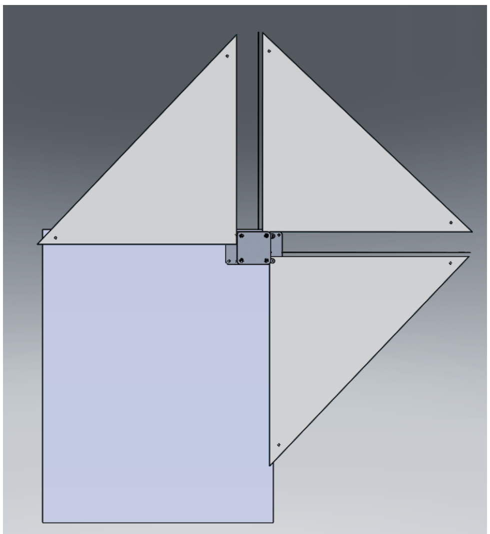

Figure 3. DOM evolution concept for corners.

Moreover, as the device is small and light, multiple DOMs could be placed around the panel to maximise the additional drag area, in addition, this design concept also decreases the wasted sail area of the design.

In addition to removing one of the sail segments, the design can be altered to deploy only three booms and two sail segments. The DOM evolution can be positioned on the edge of the panel instead of the corner, to accommodate protrusions.
With the reduction of wasted sail area and a boom, it reduces the overall mass of the device, which can potentially make it more compact.

This design only requires one corner of a satellite panel free in order to interface with the satellite; furthermore, with multiple corners available, a larger sail area can be produced as the unit is small and compact several can be used on one satellite.
This design is able to scale for larger and smaller satellites. As the device is a small compact unit, for larger satellites more devices can be implemented to increase the sail area.
In terms of increasing the sail area a single unit can produce, this is constrained by the boom length.

2.2.2. Hybrid concept

This concept is derived from component technologies used in Icarus and DOM and conceptually builds on the strengths of each to improve scalability, adaptability and manufacturability. The hybrid concept is at an earlier design stage and requires development to achieve TRL 6 alongside the other DAS.

The philosophy of the hybrid concept is to achieve a design composed of discrete self-contained modules, which can be integrated in a variety of configurations to adapt to different host satellite architectures. The modules derived from the heritage designs are:

- Boom module with release mechanism: allows different lengths of boom and can contain a sail quadrant if required.
- External sail cartridge: allows different widths and lengths of sail.

The boom module is based on the DOM design, but will contain either a single boom and no sail, or two booms and only one sail segment. Longer booms can be stored within the same size of the unit.

The external sail cartridge is derived from Icarus. One of the main issues noted from the critical analysis performed on the current DAS was the sail folding. The folding of the sail is a very time consuming process that can also lead to tearing and damaging the sail. This issue is exacerbated through the need for repeating the process for deployment testing and so a way to mitigate this issue is required.
The cartridge would contain the stowed drag sail, and would connect to two boom module units. When the two booms are deployed, the drag sail is pulled out of

the cartridge. The sail can be rolled on a spool within the cartridge, reducing the complexity of the folding patterns required and simplifying the stowing process.
Different configurations, using the two modules, can be implemented depending on the satellite host.

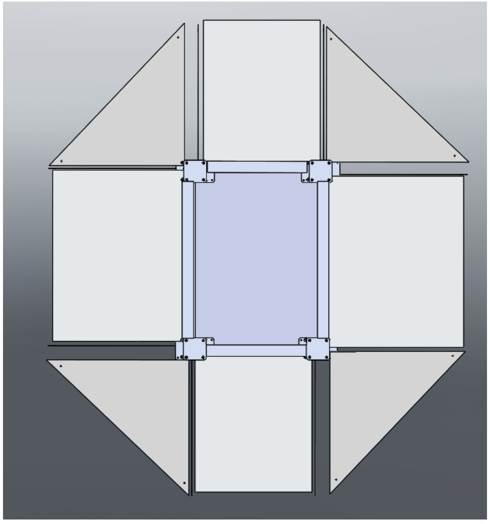

Figure 4. Hybrid concept with eight booms.

The hybrid concept is more scalable than the original Icarus as the drag sail area produced is no longer constrained by the panels side length, it is now constrained only by the boom length (as in the DOM).The concept can also be used as a single edge device.

3. REQUIREMENTS

In this section we present the main design drivers (i.e. the internal requirements) for the DAS and the customer's requirements matured during the Concurrent Design Facility (CDF) for the CleanSat BB study. The match between the internal requirements and the top-level requirements from the potential customer's perspective is shown and discussed.

3.1. Design drivers

The key drivers used internally during the design process were:

- Reliability: ensuring the proper function of the DAS device.
- Low mass: the DAS mass must be less than the propellant mass required for de-orbit.
- Low cost: use of COTS, avoidance of complex systems to ensure commercial viability.
- Simple design: to be easily assembled and interfaced with the satellite.
- Simple interfaces: to ensure minimum impact on the host satellite and minimal power required for the deployment.
- Testability: to be easily testable in 1g and to ensure repeatability.
- Safety: avoiding risk of premature deployment and any risk of damage to the host satellite.
- Scalability: to ensure it can be compatible to a wide range of commercial satellite platforms.
- No additional debris production: the DAS shall not create debris when being deployed and should not fragment when struck by space debris.

3.2. Customer's requirements

Table 1 lists the requirements established during the first CDF meeting and their verification method. These requirements are based on inputs from key players in the European space industry, and are intended to capture the customer's perspective (typically for a Large Satellite Integrator, LSI). In some cases, the "requirements" are actually targets rather than strict requirements, however expressing them as requirements gives a clear statement of potential user needs and expectations.

As can be seen from Table 1 the Cranfield design drivers matched with different top-level requirements. These are safety and reliability, system level requirements in terms of command, power, state modes, interfaces to ensure minimum impact, and physical properties (mass, area-to-mass ratio implying scalability).
In particular, in terms of safety the host spacecraft must be assured that there is negligible risk (<0.001) that the DAS will jeopardise the primary mission. The reliability of the DAS maintains a strict 95% as it is assumed a required overall spacecraft system reliability of 90% according to [4] and specified in the ESA SDM Verification Handbook [1].

At the current TRL the DAS is able to meet most of the requirements, however the two most challenging aspects to assess were the compliance with the lifetime required for storage on ground and pre-deployment storage on orbit, and the effect of the orbital environment on the sail.
The storage lifetime of 10 years was a challenging aspect to verify. A 10 year storage period on ground may not be achievable, moreover verification of this requirement may be incompatible with other requirements. A servicing may be needed to check and re-tensioning of a safety protection band on the DAS. The compliance for the storage time on orbit requires either assumptions about a nominal host mission or definition by the customer of their specific mission plan.
Both the storage lifetimes require significant further analysis and testing to validate practical component lifetimes. The environmental effects on the lifetime of the DAS are discussed in details in the analysis section.

Table 1. Consolidated requirements as agreed at CDF for CleanSat study. For Verification methods: D=design, A=analysis, T=test

ID	Requirement topic	Requirement description	Verification method
BB13-CU-01	Command	The device shall deploy as commanded by the host spacecraft.	D
BB13-CU-02	Demisability	The device shall be fully demisable, with no debris over 15 Joules (kinetic energy) reaching the surface.	D, A
BB13-CU-03	Safety	The risk of premature deployment of the device shall be 0.001 (goal=10^{-4}) or less.	D, A
BB13-CU-04	Reliability	The reliability of the device (predicted successful deployment at end of host's nominal mission, evaluated at time of integration) shall be superior to 95%.	D, A, T
BB13-CU-05	Performance	Once deployed the device shall ensure de-orbit of the host spacecraft within 25 years. Two test cases shall be used to demonstrate compliance: (1) 500 kg mass, (2) 200 kg mass, both at 600-800 km altitude.	D, A
BB13-CU-06	Deployed area	Random tumbling of the spacecraft shall be assumed to estimate the effective area of the deployed device.	A
BB13-CU-07	Area-to-mass ratio	The ratio between deployed surface area and subsystem mass shall be better than 1 m^2/kg (threshold), 2.5 m^2/kg (goal)	D, A
BB13-CU-08	Functional	The device shall be capable of deploying successfully on a host spacecraft rotating at up to 0.2 deg/s about any axis.	D, A
BB13-CU-09	Functional	The device shall not require any electrical power from the host spacecraft once deployed.	D
BB13-CU-10	Lifetime	The device design shall be compatible with 10 years ground storage, without need for complementary re-acceptance testing at the end of the storage period.	D, A, T
BB13-CU-11	Lifetime	The device shall be able to operate successfully after an operational host satellite period of 10 years in LEO.	D, A, T
BB13-CU-12	Environment	The device shall ensure the expected performance under the radiation conditions observed during the operational lifetime and the disposal phase.	D, A
BB13-CU-13	Environment	The device shall ensure the expected performance under the ATOX environment of a worst-case of de-orbit from 600 km, 25 year re-entry time.	D, A
BB13-CU-14	Environment	The device shall ensure the expected performance under the debris/meteoroid environment of a worst-case of de-orbit from 800 km, 25 year re-entry time.	D, A
BB13-CU-15	States	The device shall have three primary discrete states: stowed, deploying and deployed.	D
BB13-CU-16	Subsystem	The device shall be a separate sub-system with clearly defined interfaces to the host.	D
BB13-CU-17	Cost	Target figures for threshold and goal were defined by the LSIs.	A
BB13-CU-18	Mass	The device mass shall be inferior to 5 kg as a goal, 10 kg as a maximum.	D
BB13-CU-19	Volume	The volume of the undeployed device shall not exceed 10 litres.	D

The technical work performed in the later part of the study consisted of evaluating Cranfield's design concepts against these requirements, and using the requirements to establish the most promising DAS concept design.

4. DESIGN ANALYSIS

Detailed analyses have been carried out to assess the performance and compliance of the Cranfield DAS against the requirements identified by the LSIs.
The analyses performed include several case study missions (500 kg and 200 kg LEO satellites) to evaluate the potential reduction of orbit lifetime after EoM, the effects of atomic oxygen erosion and the debris risk during the de-orbit phase.

4.1. Mission case studies

The compliance to the performance requirement BB13-CU-05 was assessed by means of two main test cases (see Table 2), which were proposed during the CDF session:
(1) 500 kg S/C with solar panel deployable wings, and with AstroBus platform features and geometry;
(2) 200 kg S/C with body-mounted solar panels, similar to the Myriade platform geometry.

Table 2. Spacecraft characteristics

	S/C characteristics	
Mass	500 kg	200 kg
S/C body	1.0 m x 1.0 m x 1.8 m	0.6 m x 0.6 m x 1.1 m
Solar array	2 deployable wings (0.9 x 1.4) m^2	Body mounted
Drag area random tumbling	3.56 m^2	0.84 m^2
Mass-to-area ratio	140.45 kg/m^2	238.10 kg/m^2

The 25 year requirement constraint is the driver for the mission case studies. For each of the two S/C configurations the following analyses were performed:

- Maximum altitude: S/C at 800 km of altitude with no DAS, to estimate the area of the sail needed to re-enter in 25 years.
- Existing design limit: S/C with largest DAS area achievable with heritage design, to compute the maximum orbit altitude to ensure 25 year re-entry.
- New design and alternative configuration limit: same as before but with the new design concepts applied.

The preliminary assessment was performed with a simplified decay model for circular orbits, and then verified with the CNES tool for end of life analysis STELA [10]. In the decay model the atmospheric scale height and average density values, function of the constant solar flux (solar mean), were extracted from the tables in [11]. The density and scale height were calculated with linear interpolation for the different altitudes from the reference values. The satellite's lifetime was imposed at 25 years after the EoM, then, with the assumed CD (2.2), the required mass-to-area ratio for the specific orbit altitude was obtained.
In STELA we assumed a mean constant solar flux to avoid dependence of the results with the simulation date. All the simulations were set with the starting date on the 2016-12-01 and time T00:00, however considering average solar activity, this does not affect the results. The orbits considered are sun-synchronous (SSO) and circular (e=0).

For the second group of analyses, the achievable sail area was computed with the current Icarus design for the two case studies, and with the DOM evolution designs for the 500 kg S/C case.
The mass-to-area ratio of the S/C was then re-calculated for a random tumbling configuration (factor of 2 assumed as in [12]), including also the expected additional mass of the DAS. Using the previous model, the altitude limit for the 25 years maximum re-entry time is then obtained, a verification with STELA followed, as before.

4.1.1. Maximum altitude

The mass-to-area ratio is initially obtained from the simple model, and then refined with STELA simulations. As expected, the area needed for the 500 kg case to be de-orbited within 25 years from 800 km of altitude is not feasible (Table 3); moreover, it is not achievable with the current existing design.

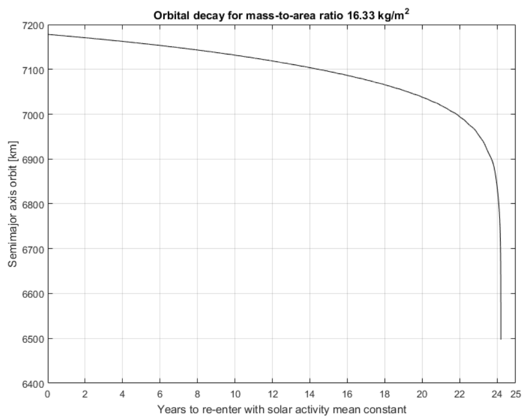

Figure 5. Decrease of semi-major axis for S/C with m/A=16.33 kg/m^2 and required sail area above 50 m^2 to comply with the 25 years requirement.

For the smaller spacecraft the drag sail area required is 23 m^2 for a S/C with random tumbling orientation. Despite the smaller area required when compared with the 500 kg S/C, this is still not achievable with the current existing design. If aerostability is achieved, the drag area needed to be provided by the sail is halved; only in this case a refined DAS design could be feasible and would allow a compliant re-entry.

Table 3. Summary of results for 500 kg S/C and 200 kg S/C for extreme case at 800 km altitude. Note: Total drag area is considered as random tumbling drag area.

Maximum altitude 800 km results		
Spacecraft case	**500 kg**	**200 kg**
Mass-to-area ratio needed	16.33 kg/m^2	16.33 kg/m^2
Total drag area needed	30.61 m^2	12.25 m^2
Drag sail area	54.12 m^2	22.81 m^2
STELA decay time	24.19 y	24.18 y

4.1.2. Existing design limit

As mentioned before, the achievable sail area was computed with the current Icarus design for the two spacecraft case studies. In Table 4 the performance results achievable with the Icarus design are presented.
Two configurations (Fig. 6, 7) were considered for the 500 kg S/C: A) - Sail deployed in a plane parallel to the solar panels; B) - Sail deployed in a plane perpendicular to the solar panels.

As expected, the random tumbling area of configuration B is better than configuration A; however, the sail deployment is more constrained, in particular should be taken into account where the frame is going to be attached (on the side, and not the edges) to avoid interference with the deployed wings. In this case having a dihedral angle for the sail and booms deployment would be probably beneficial.
The configuration selected for the 200 kg is instead the one that maximizes the achievable area with the S/C geometry. The configuration for this case resembles the S/C with the Icarus-1 and 3 currently on-orbit.

Table 4. Summary of results for 500 kg S/C and 200 kg S/C applying existing DAS Icarus design. Note: Total drag area calculated with STELA random tumbling model.

Existing design limit - Icarus			
Spacecraft case	**500 kg - A**	**500 kg - B**	**200 kg**
Drag sail area	11.69 m^2	13.77 m^2	4.50 m^2
Total drag area	7.97 m^2	8.7 m^2	2.79 m^2
DAS mass	6.18 kg	7.01 kg	3.30 kg
Total m/A	63.51 kg/m^2	58.28 kg/m^2	72.87 kg/m^2
Limit altitude 25 y	685 km	692 km	674 km
STELA decay time	24.77 y	24.81 y	24.75 y

4.1.3. New design limit

The achievable sail area was computed with the proposed DOM evolution design for the 500 kg S/C case (see Table 2 for S/C features) only. This case is the most demanding in term of DAS performance.
Different configurations were analysed to evaluate the performances in term of limit altitude to comply with the 25 year requirement, and area-to-mass ratio. The following configurations were chosen:

- DOM evolution design with standard sail shape corner configuration,
- DOM evolution design with ”fan” sail shape (scalloped sails) corner configuration,
- DOM evolution design with ”fan” sails with two different configurations.

It must be clear that the achievable sail area is not necessary additional area to the S/C drag surfaces, there can be overlapping with solar panels in parallel planes and/or other panel surfaces.
In the DOM evolution designs considered the limit is given mainly by the length of the boom to be self-supported in 1 g. This is a limit to be easily testable in 1 g, however if this constraint can be removed the upper limit of the sail area could be increased. Care will be needed however to confirm that the testing performed is still valid and robust. For this reason the maximum boom length considered in the DOM evolution designs is 1.5 m, however 1 m boom length, which is also shown as comparison, gives more confidence at the current stage of the DOM design. The mass of the DOM evolution was calculated from the design parameters analysis performed as shown in [13].

Table 5. Summary of results for 500 kg S/C applying DAS with DOM corner configuration 1.5 m booms. Note: Total drag area calculated with STELA random tumbling model.

DOM evolution design limit - corners configuration		
Configuration	**Standard corners**	**Fan corners**
Drag sail area	6.75 m^2	10.61 m^2
Total drag area	5.36 m^2	7.29 m^2
DAS mass	1.80 kg	2 kg
Total m/A	93.62 kg/m^2	68.86 kg/m^2
Limit altitude 25 y	655 km	679 km
STELA decay time	24.88 y	24.91 y

In this configuration two DOM evolution devices are attached on opposite corners of the S/C panel parallel to the solar panels, resulting in a configuration similar to Icarus in terms of deployed drag area. In this case the two DOM deployed sails are slightly overlapping each other, however this is not a problem if the two DOM are integrated with plate supports of different thickness (or one sunk in

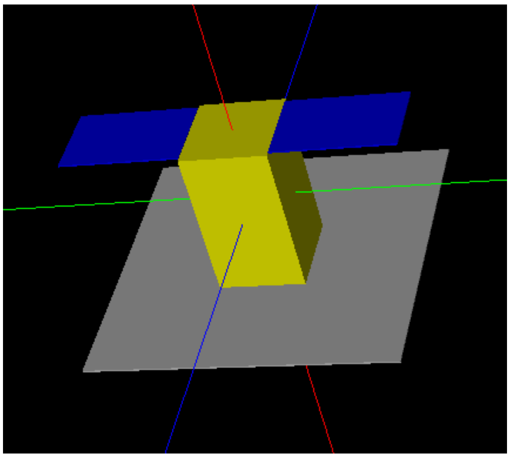

Figure 6. Configuration A: Sail deployed in a plane parallel to the solar panel deployable wings.

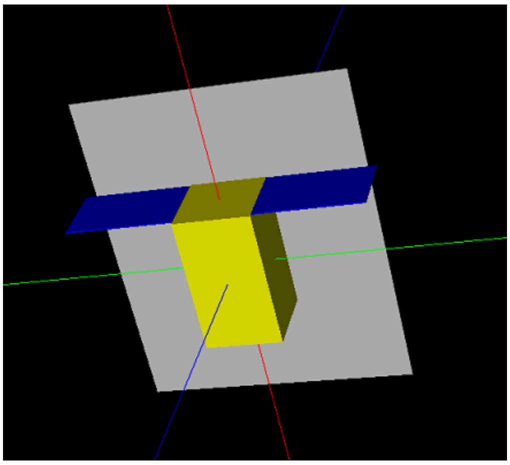

Figure 7. Configuration B: Sail deployed in a plane perpendicular to the deployable wings.

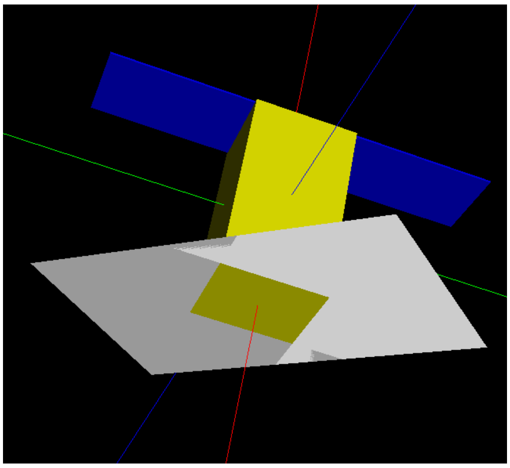

Figure 8. Configuration with two DOM 1.5 m booms - Sails deployed in a plane parallel to the solar panel deployable wings.

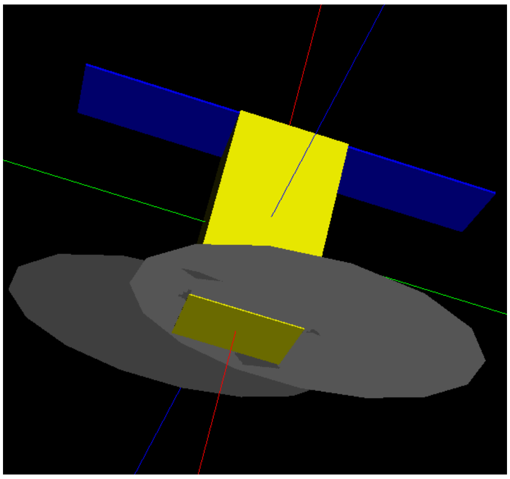

Figure 9. Configuration with two DOM 1.5 m booms with scalloped sail - Sails deployed in a plane parallel to the solar panel deployable wings.

the panel and the other not). In this way the sails deploy in parallel but different planes.

To gain additional drag area, without modify the design adopted in the previous configuration, the triangular sail shape can be modify in a "fan" shape, i.e. the external edge will be circular instead of straight.
As can be seen (Table 5)the main advantage of this design is a better area-to-mass ratio of the device, having the same boom length and only minor additional mass given by the sail. The additional sail area respect to the standard triangular shape is given by a factor of $pi/2$. As a consequence, the altitude limit to meet the 25 years requirement is higher.

The DOM evolution design is pretty flexible and different configurations can be used depending on the S/C and presence of appendages, protruding parts, etc. Here we present one example of possible configuration among the ones considered. This configuration has three DOM evolution devices: two of them with 1.5 m boom length and 3 sails each (corner configuration type design), the other with 1 m boom and edge configuration. All the three devices deploy their sails in planes perpendicular to the solar panels.

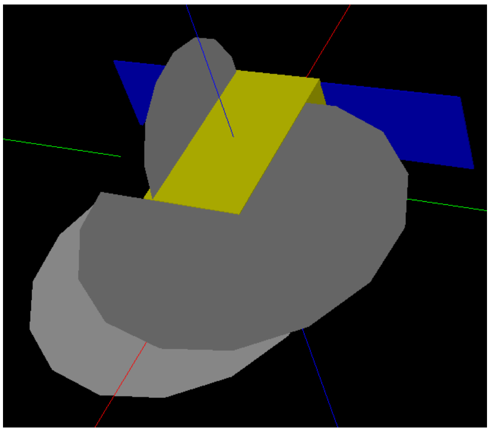

Figure 10. Configuration with 3 DOM evolution devices - Sail deployed in planes perpendicular to the solar panel deployable wings.

4.2. ATOX analysis

The compliance to the ATOX environment requirement BB13-CU-13 was assessed to evaluate the degradation of the vapor-deposited aluminum (VDA) Kapton sail with respect to the orbital decay.
The case suggested to be analysed is a re-entry starting from an orbit altitude of 600 km and decay time of 25 years (see Table 6). This represents the worst case for the VDA Kapton sail (to be noted the VDA is on both sides of the sail), as the spacecraft will remain longer time in low and very low orbit, where the ATOX levels are higher.

The ATOX flux erodes spacecraft surfaces and also causes drag on the spacecraft. Since the same flux is

Table 6. Summary of results for ATOX decay simulation.

ATOX case 600 km 25 y	
Spacecraft mass	500 kg
Mass-to-area ratio	192 kg/m^2
Random tumbling drag area	2.6 m^2
STELA decay time	23.84 y

responsible for both effects a relationship can be established between ATOX erosion rate and orbital energy loss for orbit heights where ATOX is the dominant atmospheric gas.

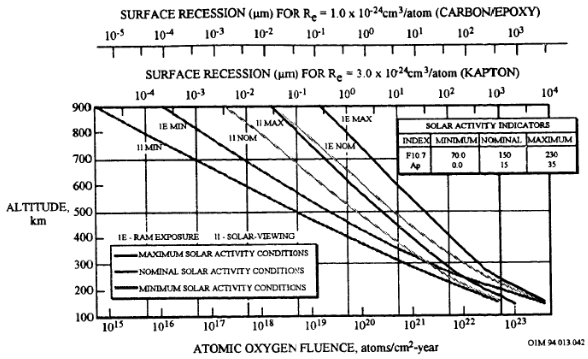

Figure 11. Atomic oxygen fluence for the ram (1E) and sun-facing (1I) directions, Figure 2-1, 4-1 from [14].

Figure 11 gives atomic ATOX fluence as a function of orbit altitude and solar activity for the ram (1E) and sun-facing (1I) directions.
Considering a ram facing surface with average solar activity conditions the ATOX fluence (1E NOM) at different altitudes can be extrapolated from Fig. 11. From the decay profile obtained with STELA simulator the time spent at the different altitude ranges can be derived. In this way, having the ATOX reaction efficiency of the Aluminized Kapton (1 x 10^{-25} cm^3/atom) from [14], a preliminary estimation of the surface degradation can be performed. For each 50 km altitude range considered (see Table 7) the value of fluence corresponds to the lowest altitude of the interval (worst condition than expected).
Since the spacecraft is assumed to be randomly tumbling, each side of the drag sail (both covered by VDA) is effectively exposed directly to the ATOX flux for only a quarter of the time (so for a 2-sided sail, the effective drag exposure is 2 x 1/4 = 1/2 which is the effective drag area factor assumed for a randomly tumbling flat surface).
Although from this first rough estimate it can be seen that the total surface degradation exceeds the thickness of the VDA Kapton sail (which is 25 μm); the initial erosion on any one surface is barely 1 μm for the first half of the de-orbit period (note that the estimate uses worst case assumptions of erosion rate). However, this is still close to the thickness of the Al coating of the sail and so it is possible that the sail will begin to erode more quickly around this time.
Design precautions, which could be included to mitigate this effect, include using thicker Al coatings and/or mul-

Table 7. Surface degradation for aluminised Kapton with decay profile of ATOX simulation 600 km 25 y decay.

Altitude range (km)	ATOX fluence (atoms/cm^2 year)	Time at h (years)	Time % years	Surface deg (cm)	Surface deg (μm)
600-550	4.00E+20	12.5	52.15	0.0005	5
550-500	8.00E+20	5	20.86	0.0004	4
500-450	1.00E+21	3	12.52	0.0003	3
450-400	2.50E+21	2	8.34	0.0005	5
Below 400	1.00E+22	1.5	6.26	0.0015	15
Total Degradation				0.0032	32

tiple layers or thicker aluminised Kapton. Nevertheless, undercutting of aluminized Kapton is not predictable. In addition, the material is perforated (and will suffer micrometeoroid damage), which is advantageous during depressurization but at the same time it can lead to faster degradation. On this purpose, the topic requires further study, moreover the data available on the degradation on the VDA Kapton are sparse and the effect are not known well.
However, it must be noted that the decay profile considered is the worst case not only for the degradation of the VDA Kapton sail, but also in term of mass-to-area ratio of the S/C. Indeed, the resulting random tumbling area is 2.6 m^2 for a S/C of 500 kg mass, this drag area is even smaller than the one simulated in the mission case studies without any DAS on-board. This means that:
1) the 500 kg S/C (see Table 2), even without any DAS, would decay much faster from 600 km of altitude and so less affected by degradation;
2) The resulting tumbling area would include a very small sail area other that the S/C surfaces.

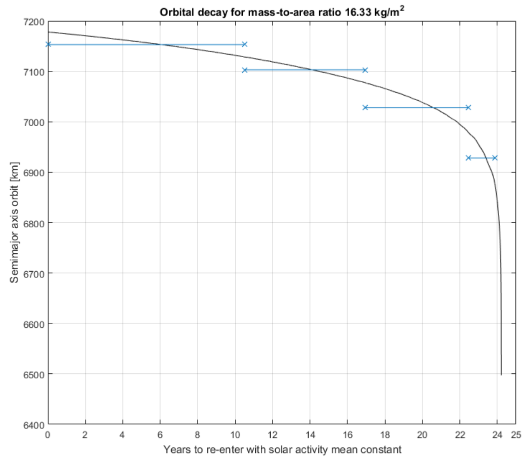

Figure 12. Decrease of semi-major axis from 800 km with constant time intervals considered.

4.3. Debris risk analysis

The case analysed to assess the risk of debris collision (requirement BB13-CU-14) with the DAS is a re-entry starting from an orbit altitude of 800 km and decay time of 25 years. This is the worst case in terms of debris environment. The decay profile is the same as the 500 kg S/C case presented in Fig. 5 (see Table 3). The case represents the debris risk for the (maximum 25 years) deployed lifetime only.
MIDAS tool within DRAMA was used to assess the collision risk and impact flux. The orbit altitude was assumed be constant for distinct time intervals, so multiple target orbits were defined (see Fig. 12).
The worst case for the impact vs sail area has been considered, i.e. the 500 kg S/C, which needs a bigger sail area than the 200 kg ones.

The critical size (diameter) of the particle that will lead to (partial and complete) failure of the sail was selected with respect to the boom width of the DOM design.
The reference width is 28 mm and the critical size is debris with diameter $d > 1/4w\_boom$, i.e. 7 mm. The debris range considered in the simulation is 7 mm to 1 m, it is assumed that size above 1 m will damage also the S/C so they were already assessed by the S/C integrator.

Table 8. Summary of boom surface area for the simulation performed.

Parameter	Value
Area sail (m^2)	54
Boom length (m)	5.196
Boom width (m)	0.028
Single boom area (m^2)	0.145488
Total 4 booms (m^2)	0.581952

The surface defined for the impact flux analysis is a random tumbling plate with cross sectional area equivalent to the sum of the exposed area of four booms. The sail area is not considered as source of debris, in addition the potential impact will only leave an hole to the VDA Kapton surface.
The scenario considered in MIDAS is Business as usual. In Table 9 the different time intervals with constant altitude and correspondent result for cumulative probability of collision are shown.

Table 9. Time intervals with constant altitude as shown in Fig. 12 and collision risk results.

Interval	Altitude range (km)	Average a (km)	Time from launch (years)	Time at average h (years)	Cumulative probability of collision
1	800-750	7153	10.5	10.5	P<0.0014
2	750-700	7103	16.93	6.43	P<0.0007
3	700-600	7028	22.47	5.54	P<0.0004
4	600-500	6928	23.86	1.39	P<0.00004

As expected, the first interval at 775 km, it is the one with highest risk; however, the probability of collision remains below 0.0014.

5. CONCLUSIONS

The requirements definition for typical LEO satellites of interest to European LSIs and their detailed analysis relative to DAS show there appears to be a valid role for drag augmentation systems for EoL de-orbit of some satellites in significant LEO regions.

More rigorous analysis is needed for what concern the attitude dynamics, atmospheric demise and to quantify the benefits to the space environment of drag augmentation systems, to lay a firm foundation for wider use of drag augmentation for debris mitigation.
Risks such as atomic oxygen erosion, damage by other debris and micrometeoroids have been quantified and can be mitigated.
Storage lifetime on ground and in-orbit before deployment requires significant further analysis and testing to validate practical component lifetimes.

The study has provided useful input to explore new concepts based on the heritage designs. The DOM Evolution concept, especially with several units installed on one satellite, is probably the most effective method in the short-term for increasing a satellite's drag area. The most promising concept for development is a hybrid concept using components of the two Cranfield DAS heritage designs.

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