LEO SPACECRAFT ACCELERATED ORBITAL DECAY BY ENHANCED AERO-BRAKING: THE BASELINE DESIGN OF THE END (END-OF-LIFE NATURAL DE-ORBITING) SYSTEM

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1 Abstract

The scope of this paper is to present an assessment about the possibility of applying an uncontrolled de-orbiting method to LEO (Low Earth Orbit) spacecraft (S/C) at the end of their service life, through the implementation of a specific activable device aimed to increase the vehicle aerodynamic drag, accelerating so its orbital decay. The reason that has motivated this proposal is the attempt to provide a simple and reliable solution as a partial (i.e. limited to LEO operations) answer to the growing concern about the problem of the orbital debris, that is already pushing most of the Space Agencies around the world to impose rules and guidelines relevant to their mitigation.

2 The Problem of Orbital Debris Accumulation - Issue and Enforcement of Relevant Regulations

According to some observer [1], pag. 6-1, about 2000 tons of debris have been so far accumulated in Earth orbits as a consequence of the uncontrolled growth of the number of both launch vehicles upper stages abandoned in LEO and satellites that have terminated their operational life.

Recently, however, various National Space Agencies around the World have tried to focus the attention of the Space Community on the medium and long term negative consequences of this practice, with particular regard to the risk of collision of some debris with manned or unmanned operational spacecraft.

In Europe the document [2] has been issued by ESA (European Space Agency), ASI (Agenzia Spaziale Italiana) and other European National Space Agencies on September 2007. At point A) of the sub-paragraph 6.6.2 (Disposal) of paragraph 6.6 (End-of-Life Measures) of such a document it is reported the guideline SD-OP-03, whose text quotes:

"The operator of a Space System should perform disposal manoeuvres at the end of the operational phase to limit the permanent or periodic presence of its space system in the protected regions to a maximum of 25 years.
This can be achieved, in decreasing order of preference:

- *Either by performing a direct re-entry of the space system*
- *Or by limiting the orbital lifetime of the space system to less than 25 years after its operational phase*
- *Or by transferring the space system to a disposal orbit"*

The protected regions mentioned in the quote are the LEO protected region (defined as all the space volume within 2000 km of altitude from the Earth surface) and the GEO (Geostationary Earth Orbit) protected region.

It is clear, indeed, that the subject of the orbital debris regulation will become important ever more, and that even in the short term it will pose non-trivial constraints in the design of both the LEO Spacecraft configurations and their missions.

On the other hand, the authorities around the world interested in the application of the relevant regulations on this matter will likely try to enforce them over and over, aiming to develop new procedures and methodologies (like, for instance, those ones described in [3]) in order to be sure that such regulations be correctly obeyed by the various S/C owners.

It is therefore wise, from now on, to start thinking about some sort of "activation-piloted" LEO spacecraft de-orbiting method (not necessarily controlled) that could be as much as possible simple, reliable and economical.

3 Outline of the END Concept

As said above, the study reported on this paper consists of an analysis and a technical and economical feasibility assessment about the possibility of applying an uncontrolled de-orbiting method for LEO spacecraft (orbit apogee altitude < 2000 km) that are supposed to be compliant with the rules concerning the limit casualty due to their fragmentation process during the atmospheric re-entry.

The main idea, already outlined in [3], is indeed based on the implementation of a specific device (to be activated at the end of the spacecraft operational life) aimed to increase the vehicle aerodynamic drag, and therefore to exploit the consequent enhanced aero-braking in order to accelerate a spacecraft orbital decay (rather than executing a direct retro-firing maneuver using propulsion systems aboard), so to limit the orbital lifetime well within the 25 year time constrain mentioned in the previous paragraph as one of the three options recommended for spacecraft “end-of-service” disposal.

This solution, that has been called END (End-of-Life Natural De-orbiting), appears simple and reliable, since, after the quick activation of the relevant device, no other operation is needed, and the vehicle is supposed to go through a progressive uncontrolled orbital decay.

In comparison to it, the classical retro-firing maneuver requires not only additional propellant for the main engine (w.r.t. the relevant amount accounted for its operational life) and additional weight and volume for its tank(s), but also the obvious full availability of a complete Attitude Control System in order to ensure the proper alignment of the thrust vector during the retro-firing phase.

On the other hand, as already mentioned, the application of this augmented aero-braking de-orbiting strategy has to be limited to spacecraft for which it is verified the compliance with the rules concerning the limit casualty due to their fragmentation process during the atmospheric re-entry.

If this was not the case, the relevant orbital decay should be controlled and steep enough (perigee altitude < 60 km, as mentioned in [4], pag. 3, and [5], pag. 10) to ensure a well-defined impact area and location, but these requirements, and the complexity that they bring along, are obviously out of the applicability range of the proposed solution.

4 Parameters Affecting a LEO Spacecraft Orbital Lifetime

The primary factors affecting a spacecraft orbital lifetime are:

- Initial (end-of-service) orbital altitude
- Spacecraft ballistic coefficient
- Solar flux intensity

The initial orbital altitude is obviously an input datum of the problem that simply depends on the type of S/C operational use.

The spacecraft ballistic coefficient (BC) is defined as

$$BC = m / (A_{REF}\, C_D)$$

where m is the spacecraft mass, A_{REF} its aerodynamic reference area and C_D its aerodynamic drag coefficient.

For a given altitude and for a given level of solar flux, the lower is the ballistic coefficient, the higher is the value of the satellite apogee radius decay.

The solar flux level fluctuates with a 11 year cycle, inducing, at a given altitude, a maximum value of atmospheric density during the period of maximum solar flux, and, conversely, a minimum value of atmospheric density during the period of minimum solar flux.

Apart from the solar flux effect, other major LEO perturbations (that indeed have not been considered in the present analysis) are those ones due to solar radiation pressure, Earth oblateness (J2 zonal coefficient), Moon and Sun gravitational forces and aerodynamic lift.

The solar radiation pressure (SRP) tends to lower the orbital altitude of a S/C if part of its orbit passes through the Earth’s shadow, since in the shadow phase there is no compensation for the SRP force directed toward the Earth during the opposite phase, i.e. when the S/C orbit position is between the Sun and the Earth.
This phenomenon, however, becomes appreciable at altitudes above 800 km from the Earth Surface [6], pag. 146. Therefore, since, as it will be shown in the following paragraphs, the range of application of the proposed de-orbiting strategy features, in terms of initial circular orbit altitude, an upper limit value of about 850 km, it has appeared reasonable not to take this perturbation into account.

The Earth oblateness (J2 influence) has no direct effect on the orbital radius decay, since its perturbations involve the orbit RAAN (Right Ascension of Ascending Node) and the perigee argument only.

The Moon and the Sun gravitational forces have not been included, since their influence become important when the apogee radius of the relevant Earth orbit is above 30000 km [5], pag. 10, value by far above the 2000 km upper limit of a LEO.

aerodynamic lift has been assumed to be zero since, typically, space vehicles (with the only exception of re-entry vehicles, that don't fall in the class under our analysis) are designed as bodies not aerodynamically shaped. Therefore, their drag coefficient value is usually quite higher than the lift coefficient, and, in any case, the lift influence (positive or negative) on the orbital lifetime will depend upon the history of the spacecraft attitude during the uncontrolled orbital decay (a rather random variable), so that it seems reasonable to assume its time-average value equal to zero.

5 *Initial Circular Orbital Altitude vs Ballistic Coefficient for a 25 Year Circular Orbit Decay Time*

Although the END concept can be applied, in principle, to LEO S/C placed on generic elliptic orbits, for the sake of simplicity the present analysis has been referred to cases for which the S/C operational (end-of-service) orbit is circular.

Moreover, as described in the following paragraph, for the retro-firing option it turns out that, when the S/C end-of service circular orbit is below 1000 km, the most efficient strategy is the one of the Hohmann transfer to lower circular orbits featuring a S/C orbital lifetime of 25 years.

For the reasons mentioned above it has been necessary to identify the relationship between a S/C initial orbital altitude and its ballistic coefficient for a constant circular orbit decay duration of 25 Years.

At this purpose, a set of ballistic coefficient values has been selected in the range from 1.0 kg/m2 to 200 kg/m2 (this range should include the vast majority of the commercial satellite market).

For each one of these BC values a series of orbital decay calculations has been performed, taking into account the periodic effect of the 11-year solar flux cycle, for a range of 7 initial orbital altitude values going from 500 km to 1100 km, at intervals of 100 km apart.

The influence of the level of the 11-year cycle solar flux has been considered at time interval of 6 months, so that, for each value of ballistic coefficient and for each value of initial circular orbit altitude, 21 cases (each one with a different level of solar flux) have been run.

From each set of 21 cases the one with the shortest orbital decay lifetime has been selected for our analysis, so to have, from our standpoint, the most conservative assessment with regard to the 25-year orbital decay constraint.

From the above mentioned selected results of such calculations it has been derived, for a constant orbital decay lifetime of 25 years, the functional dependence of a S/C initial orbital altitude upon its ballistic coefficient.

Belows, the graphical representation (with a logarithmic scale on the abscissa axis) of this dependence is reported.

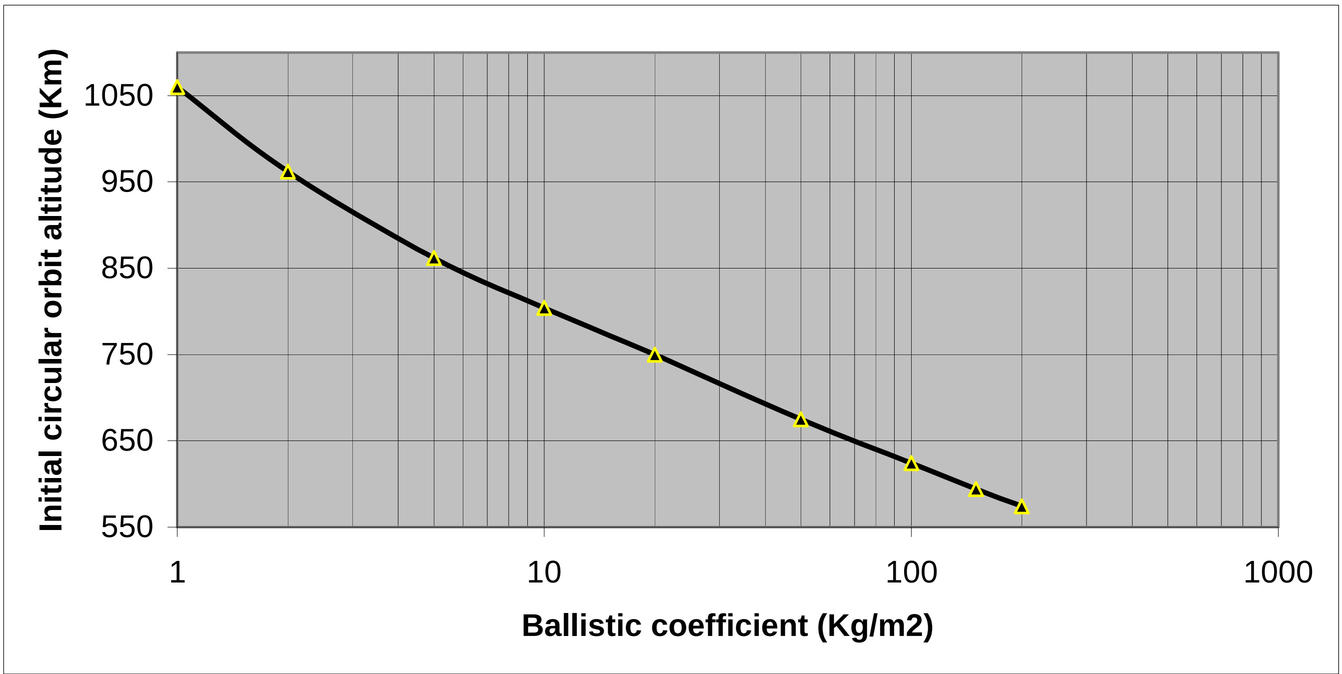

Fig. 1 - Initial S/C Circular Orbit Altitude versus Ballistic Coefficient for a constant Orbital Decay time of 25 years

This 25 years iso-decay curve, in which the initial orbital decay altitude varies from 1060 km (for a BC equal to 1.0 kg/m2) to 575 km (for a BC equal to 200 kg/m2), can be then considered as the borderline between the region where no piloted action is needed for ensuring a complete de-orbiting within the 25 years period (region below the curve) and the region where this action is indeed required (region above the curve).

6 Active De-Orbiting Through Impulsive Retro-Firing

The most direct and fastest way of de-orbiting a spacecraft is the one of executing an impulsive retro-firing maneuver (like the one usually adopted by manned spacecraft for initiating the re-entry phase of the flight) by means of a chemical rocket thrust acting in the direction opposite to the direction of motion, in order to provide the vehicle with a negative ΔV.

This option, therefore, requires an additional amount of propellant and increased tank size aboard the S/C (w.r.t. to its service life budget), so to allow for the above mentioned end-of-service de-orbiting maneuver.

In order to carry out a critical comparison in terms of mass and costs between the aero-braking and the retro-firing de-orbiting modes, it is therefore necessary to perform an assessment of the propellant mass budget relevant to the latter case (i.e. required to execute the de-orbiting maneuver of a satellite from its operational circular orbit).

At this purpose specific calculations have been carried out, whose main features can be summarized as follows:

- Three values selected for the S/C initial mass (start of the de-orbiting phase): 500 kg, 1000 kg and 1500 kg
- Monopropellant Hydrazine (N_2H_4), with Isp = 220 sec, selected as the S/C rocket propellant, since it is the choice adopted by the vast majority of the LEO S/C having their own propulsion system
- Performed budget evaluations of ΔV and of additional propellant mass relevant to direct single retro-firing to an elliptic orbit with apogee altitude equal to initial de-orbiting altitude (comprised between 600 km to 900 km) and perigee altitude equal to 90 km
- Performed budget evaluations of ΔV and of additional propellant mass relevant to retro-firing Hohmann transfers to final circular orbits at altitudes of 575, 595, 625 and 675 km, starting from circular orbit altitudes comprised between 600 km and 900 km

With regard to the cases of direct single retro-firing de-orbiting to elliptic orbits with a low perigee altitude, the reason why no simulation has been carried out for perigee altitude value > 90 km is that, as reported in [6], pag 158, "*Reducing perigee to 100 to 150 km could result in several orbits over which apogee decreases before the spacecraft re-enters and might not allow adequate control of the deorbit conditions.*"

Moreover, the non-immediate decay (i.e. multiple orbit passes) of a S/C elliptic orbit with a low perigee altitude increases considerably the risk of collisions with other S/C's when crossing the altitude gap between its apogee and its perigee.
As a matter of fact, the same above mentioned source, [6], pag. 157, implicitly suggests, for the direct single retro-firing de-orbiting maneuver, a perigee altitude ≤ 50 km.

Indeed other sources, like for instance [4], pag. 3 and [5], pag. 10, suggest to assume, for a direct retro-firing de-orbiting maneuver, a perigee altitude ≤ 60 km.

As said earlier, the above mentioned analysis aimed to evaluate the additional propellant mass that a spacecraft propulsion system would feature w.r.t. the case for which no retro-firing de-orbiting was envisaged in its mission profile.

From the analysis of the above mentioned data it has come out the very important consideration that, *in the range of the S/C ballistic coefficient values (50 kg/m^2 to 200 kg/m^2) and of the initial circular orbit altitude values (600 km – 900 km) of relevant pratical interest, the strategy of de-orbiting a S/C by means of a retro-firing Hohmann transfer to a 25-year decay circular orbit (whose altitude value is function of the S/C ballistic coefficient) is always convenient, in terms of ΔV (and then in terms of propellant mass budget for a given propellant Isp), w.r.t. to the single retro-firing direct de-orbiting maneuver to 90 km perigee altitude.*

For completeness of information it has to be mentioned (although not shown in the present document) that the single retro-firing direct de-orbiting maneuver to the 90 km perigee altitude becomes convenient, from the propellant mass budget standpoint, w.r.t. the Hohmann transfer retro-firing de-orbiting to a 25-year decay circular orbit, when the value of the initial circular orbit altitude is above 1000 km.

In any case, for the reason described above, ***the strategy of the Hohmann transfer to the circular orbit of 25-year orbital decay lifetime has been taken as the reference basis for the retro-firing de-orbiting option to be compared with the END (aero-braking) de-orbiting option.***

The additional S/C structural mass due to the boarding of the extra propellant required for the retro-firing de-orbiting maneuver has been assumed to be equal to that one of a cylindrical shell having the same diameter of the payload adapter upper base and the length necessary for housing an additional spherical tank whose internal volume is sized for containing the additional propellant mass, neglecting instead the mass of the tank itself.

7 *Enhanced Aero-Braking De-orbiting through Augmentation of the Spacecraft Aerodynamic Drag Area*

As we have seen, the orbital lifetime of a LEO S/C is strongly affected, for a given initial orbital altitude, by its ballistic coefficient BC. The lower is the S/C BC value, the shorter turns out to be its orbital lifetime.

A S/C ballistic coefficient can be reduced by increasing its aerodynamic reference area and/or its aerodynamic drag coefficient C_D. Generally speaking, an increase of the S/C aerodynamic reference area (and, to some extent, of the aerodynamic drag coefficient C_D) can be achieved through either deployable spoiler panels or inflatable vessels shields.

If the relevant aerodynamic reference area considered is the projected area normal to the direction of motion, then the value of the aerodynamic drag coefficient C_D of a blunt body (like a non-aerodynamically shaped S/C) under the conditions of free-molecule flow regime (like in the atmosphere layers above 100 km of altitude, where the air cannot be considered a continuum medium anymore) generally falls in the range between 1.5 – 2.5, with an average value around 2.2 (see [8], pag. 4).

It is however clear that the capability of lowering the ballistic coefficient value through an increase of C_D is in any case relatively limited, and that, for this purpose, the major effort has to be devoted to the increase of the aerodynamic reference area.

At this regard the solution of the inflatable vessels appears to be the more appropriate since, with the due limitations, it can be considered as the less intrusive from the spacecraft-owner standpoint, while, on the other hand, the relevant technology appears to have reached a sufficient level of maturity [9].

In order to perform a preliminary comparison between the retro-firing de-orbiting option and the one making use of an inflatable aero-braking shield, AVIO has requested AERO-SEKUR to carry on a preliminary analysis in order to identify the most promising configuration, that is turned out to be the one featuring a membrane shield stiffened by radial inflatable beams.

From the time of the END module assembly to the event of the shield deployment both the membrane shield and its inflatable beams are supposed to be packed inside a metallic (aluminum alloy) cylindrical shell interposed between the payload and the LV payload (P/L) adapter. This cylindrical shell will have to reply exactly, at both of its ends (i.e. toward both the adapter and the payload), the original mechanical and electrical payload/adapter interfaces, so that this device turns out to be indeed minimally invasive in the frame of the LV/PL assembly.

AERO-SEKUR has identified the value of 40 m^2 as the maximum value of aero-braking shield area that can still ensure the pratical feasibility of this inflatable system.

The required length of the relevant cylindrical shell capable of housing an END aero-braking system with a 40 m^2 of aerodynamic shield area has turned out to be about 350 mm. Assuming a wall thickness of this aluminium-made shell of about 3mm (its diameter is equal, by definition, to the 937 mm of the adapter) and a radial width of the flange of 30 mm c.a., the mass of this shell should be around 16 kg.

In case the relevant S/C is equipped with its own propulsion system for the operational phase of the mission, then two distinct thermal protection components will be required, one as a logical elongation of the nozzle divergent section through the inner longitudinal part of the END module, the other one as a disk to be applied to the bottom side of the module (as a barrier against the radiative heat coming from the rocket plume). The cumulative mass of this thermal protection components will be in any case not higher than 2 kgs.

Lastly, the electrical components of the module (payload/adapter connection cable + connectors) are supposed to have a total mass not higher than 1 kg.

So, the total mass of the END module will be made up of the metallic cylindrical shell mass (16 kg for a shell with a length of 350 mm) + the thermal protection layers mass (if required, 2 kg) + the electric cable and connectors mass (1 kg) + the mass of the aero-braking system itself.

The total aero-braking system mass and volume data, as a function of the aero-braking shield area in the range 1.0 m^2 - 40 m^2, relevant to the selected configuration, are reported in the table below, along with the mass and cost data of the corresponding complete END module. This data base has been used for extrapolating the mass and cost data referring to 25-year circular orbit decay missions making use of the END module aero-braking system.

Aero-Braking Shield Area [m^2]	Aero-Braking System Mass [kg]	Aero-Braking System Volume [liters]	END Module Total Mass [kg]	END Module Total Cost [k€]
1.0	1.5	30	20.5	31.0
2.0	1.6	30	20.6	31.8
3.0	1.7	30	20.7	32.6
4.0	1.8	30	20.8	33.4
5.0	1.9	30	20.9	34.2
10.0	2.9	130	21.9	58.3
20.0	4.7	130	23.7	66.6
30.0	6.4	210	25.4	74.5
40.0	8.1	210	27.1	82.8

Table 1 – Aero-Braking System and END Module Characteristics vs Shield Area

Having taken the ARIANESPACE 937 as the reference adapter (foreseen for the VEGA Launch Vehicle too), a rough drawing study has been performed by AVIO, in order to evaluate the housing of the END system deployable components inside a cylindrical case with the same diameter as the one of the 937 adapter upper base, and also its interface arrangement with both the payload adapter and the payload itself.

Some views of the relevant CATIA DWG's are shown below.

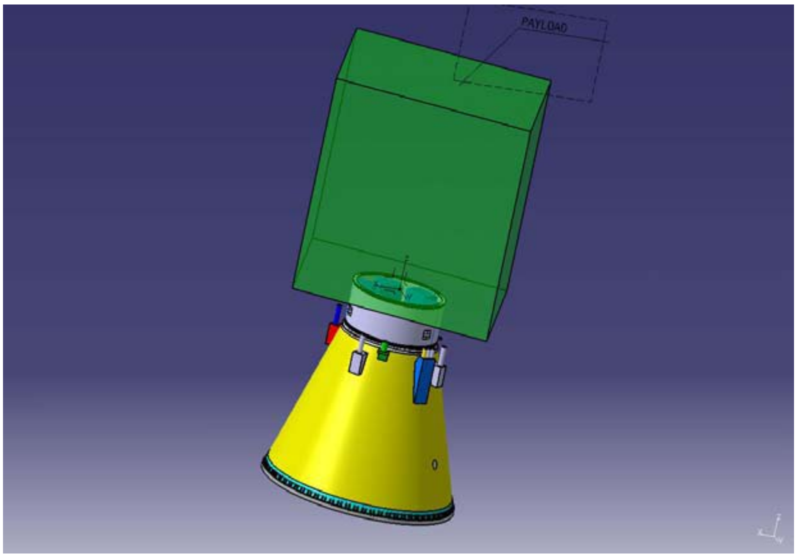

Fig. 2 - Adapter/END/Payload Assy before END/payload separation from the adapter

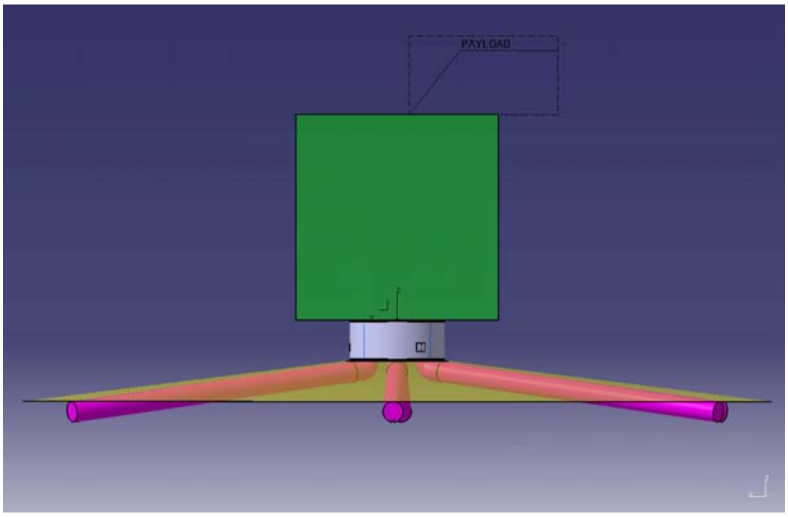

Fig. 3 - END/Payload Assy after end-of-service aerodynamic shield deployment – Lateral view

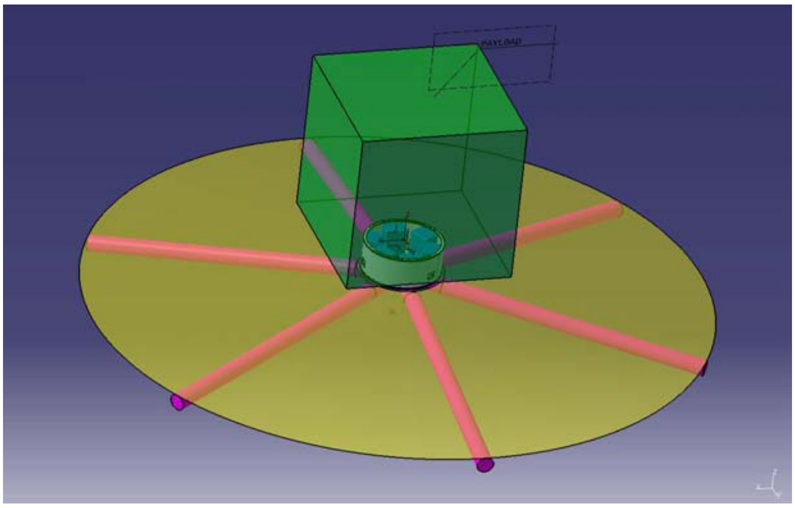

Fig. 4 - END/Payload Assy after end-of-service aerodynamic shield deployment – Overall 3D view

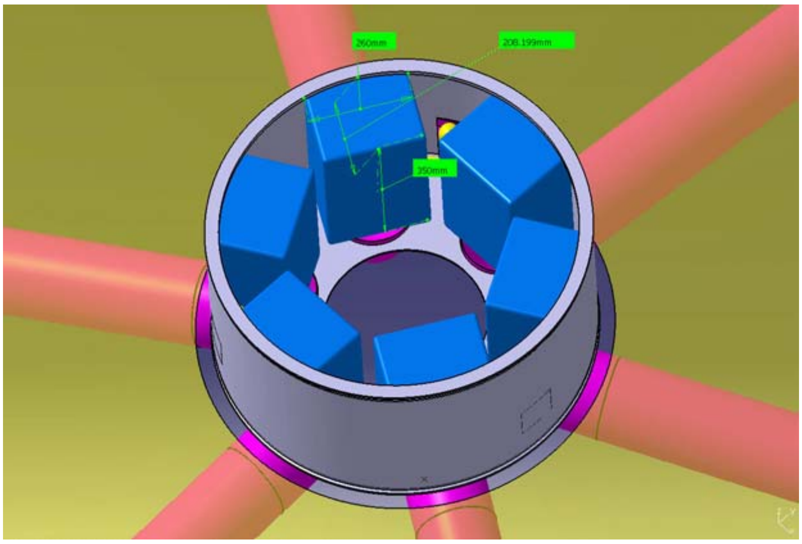

Fig. 5 – Internal arrangement of the END module cylindrical shell (Payload not shown for view convenience)

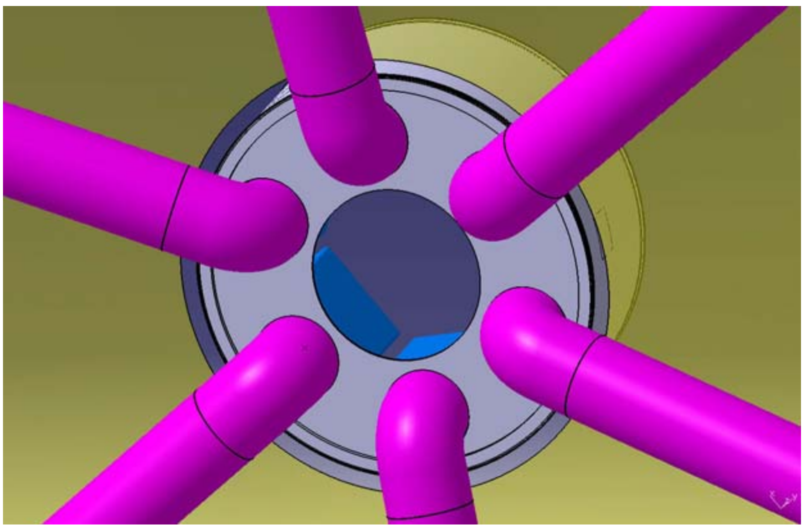

Fig. 6 – Bottom view of the END module after aerodynamic shield deployment
(Payload not shown for view convenience)

Another point to be remarked is that a side benefit coming from the adoption of the END module is the damping, toward the payload, of the axial loads induced by the launch vehicle.

Critical Aspects of the END De-orbiting Strategy

The critical aspects involving the application of the END aero-braking concept to S/C end-of-service de-orbiting can be summarized as follows:

- Reliability of the system deployment and inflation operations after a long period (years) of “packaging”
- Selection of a specific technology that can ensure a sufficiently satisfactory rigidisation of the vessel material after completion of the inflating process
- Accurate evaluation of the aero-thermal loads acting on the inflated structure, in order to make a robust assessment about the orbital altitude value at which such a structure will likely start the progressive mechanical degradation with the consequent problems of both aero-braking capability loss and fragments generation

8 Cost assessments for the Retro-Firing and the END De-orbiting Options

What remains to be assessed in the trade-off analysis between the retro-firing and the aero-braking de-orbiting options are the economical aspects.

At this regard, we state or recall hereafter the assumptions that have been made:

- Each cost comparison between the two options has been performed for a same 25-year circular orbit decay de-orbiting mission
- For the retro-firing option, the additional costs of the required bigger propellant tanks w.r.t. to the baseline ones (sized w/o taking into account the retro-firing de-orbiting phase of the mission) has been neglected
- For the retro-firing option, the additional cost of the S/C structure modification due to the boarding of the extra propellant required for the retro-firing de-orbiting maneuver has been assumed to be equal to that one of a cylindrical shell having the same diameter of the payload adapter and the length necessary for housing an additional spherical tank whose internal volume is sized for containing the additional propellant mass.
- For the retro-firing option, the cost of the additional propellant mass accounted for has been twice as much the amount necessary for the retro-firing de-orbiting phase, since canonically, it has to be considered, during the pre-launch operations, the possibility of being forced (for whatever reason) to dump out the amount of liquid propellant loaded the first time.
- The cost of the aero-braking system itself and of the complete END module, as already shown on Table 1, has been assessed as a function of the aerodynamic shield area.

Taking into account an over-head cost of about 25% due to transportation, handling, transfer tank maintenance, etc., the overall costs/kg (updated as June 2008) of the Monopropellant Hydrazine (N_2H_4) taken into considerations for our analysis have turned out to be 340 € / kg.

Moreover, an important consideration to be done is that the mass saving provided by the END system w.r.t. the retro-firing one gives the payload owner the benefit of a launch cost saving ranging around 20.0 k€/kg.

Alternatively, the mass of propellant saved for the elimination of the retro-firing de-orbiting maneuver could be used for extending the operational life of the S/C, increasing so the gain margin of the S/C owner.

The total cost saving (***Δ Cost Total***) of the END system w.r.t. the retro-firing option, for the same de-orbiting mission, has been defined as follows:

Δ Cost Total = Total cost of the complete END module
- Cost of the additional propellant and structure relevant to the retro-firng de-orbiting option
- ***Δ Cost mass***

where

Δ Cost mass **=** 20.0 k€ x (END module kgs – additional kgs due to the adoption of the retro-firing strategy)

9 Overall Mass and Cost Comparison between the Retro-Firing and the END Circular Orbit Decay Options

The overall results of the trade-off analyses (all referring to 25-year circular orbit decay missions) described in the previous paragraphs are collected in the following table, below which the relevant legenda is reported.

h_0 [km]	*S/C M_0 [kg]*	*S/C BC_0 [kg/m²]*	*Δ Mass [kg]*	*Δ Cost System [k€]*	*Δ Cost Mass [k€]*	*Δ Cost Total [k€]*
650	500	200	**- 7.6**	+ 28.8	- 152.0	**- 123.2**
650	500	100	**0.0**	+ 30.0	0.0	**+ 30.0**
650	500	50	**N/A**	N/A	N/A	**N/A**
650	1000	200	**- 16.8**	+ 27.4	- 336.0	**- 308.6**
650	1000	100	**- 4.5**	+ 29.8	- 90.0	**- 60.2**
650	1000	50	**N/A**	N/A	N/A	**N/A**
650	1500	200	**- 25.7**	+ 32.5	- 514.0	**- 481.5**
650	1500	100	**- 7.6**	+ 29.6	- 152.0	**- 122.4**
650	1500	50	**N/A**	N/A	N/A	**N/A**
750	500	200	**- 18.3**	+ 51.4	- 366.0	**- 314.6**
750	500	100	**- 12.5**	+ 48.8	- 250.0	**- 201.2**
750	500	50	**- 6.9**	+ 39.8	- 138.0	**- 98.2**
750	1000	200	**- 37.7**	+ 52.6	- 754.0	**- 701.4**
750	1000	100	**- 26.0**	+ 54.8	- 520.0	**- 465.2**
750	1000	50	**- 14.7**	+ 55.2	- 294.0	**- 238.8**
750	1500	200	**- 57.2**	+ 53.4	- 1144.0	**- 1090.6**
750	1500	100	**- 24.4**	+ 56.8	- 488.0	**- 431.2**
750	1500	50	**- 22.6**	+ 57.7	- 452.0	**- 394.3**
850	500	200	**- 24.5**	+ 72.0	- 490.0	**- 418.0**
850	500	100	**- 18.8**	+ 72.9	- 376.0	**- 303.1**
850	500	50	**- 13.3**	+ 73.2	- 266.0	**- 192.8**

Table 2 – Overall Mass and Cost Comparison between the retro-firing and the END de-orbiting options (Isp = 220 for the retro-firing option)

h_0 = S/C Initial Circular Orbit Altitude (at the end of its operational service)

S/C M_0 = S/C Initial Mass (at the end of its operational service)

S/C BC_0 = S/C Ballistic Coefficient (at the end of its oper. service, before aero-braking shield deployment)

Δ Mass = Mass saving of the END module w.r.t. the retro-firing option (for the same 25-year decay de-orbiting mission)

Δ Cost System = Cost difference between the END module and the retro-firing option additional costs

Δ Cost Mass = Cost saving due to the mass saving at launch

Δ Cost Total = Overall cost saving of END system w.r.t. retro-firing option (inclusive of mass saving at launch)

It can be seen that, in the range of initial circular orbits between 650 km and 850 km, END shows, in general, appreciable mass and cost savings over the retro-firing option. For h_0 = 650 km and S/C BC = 50, END is not applicable in the S/C M_0 range analysed (500 - 1500 kg), since in such a case the orbital decay time is already lower than 25 years with the S/C baseline configuration. Conversely, for h_0 = 850 km and in the S/C BC range analysed (50 – 200 kg/m²), S/C's with M_0 > 500 kg would require an aerodynamic shield area greater than the maximum feasible one (40 m²).

10 Conclusions

An assessment has been carried out about the possibility of applying an uncontrolled de-orbiting method to LEO spacecraft at the end of their service life, through the implementation of a specific activable device aimed to artificially increase the vehicle aerodynamic drag, accelerating so its orbital decay. The reason that has motivated this proposal (whose concept has been called END) is the attempt to provide a simple solution as a partial (i.e. limited to LEO operations) answer to the growing concern on the problem of the orbital debris, that is already pushing most of the Space Agencies around the world to impose rules and guidelines relevant to their mitigation.

This assessment has consisted of a comparison, from both the technical and economical standpoint, between the proposed END option (aero-braking de-orbiting) - whose implementation has been shown to be minimally invasive toward both the payload and the launch vehicle - and the classical de-orbiting strategy based on impulsive retro-firing maneuver, for a series of LEO 25-year circular orbit decay missions.

It has been shown that the END solution appears to be convenient in terms of performance, costs and simplicity, for most LEO S/C in the range of operational circular orbit altitudes comprised between 650 km and 850 km.

Indeed, below 650 km only S/C with high values of the ballistic coefficient (> 150 kg/m^2) feature, in the worst case, an orbital decay lifetime (slightly) higher than 25 years, so that no real de-orbiting strategy is actually strictly required.

On the other hand, above 850 km the present solution is not applicable, since even implementing an aero-braking shield with the largest feasible aerodynamic area (40 m^2) to spacecraft with relatively low value of ballistic coefficients (50 kg/m^2), it does not allow a S/C de-orbiting time shorter than 25 years. However, a possible extension of the application range of END might be its sequential combination with a retro-firing maneuver, when the latter is not sufficient by itself.

Actually, in the ranges of the parameters taken under analysis (end-of-service S/C mass from 500 to 1500 kg, S/C ballistic coefficient from 50 to 200 kg/m^2, C_D equal to 2.2, Isp values of 220 sec for the retro-firing mode, initial circular orbit altitude comprised between 650 and 850 km) the gain of mass and cost offered by the END option w.r.t. the retro-firing one, for the same de-orbiting mission, turns out to be quite interesting, having also considered that the mass saving provided by the END de-orbiting system w.r.t. the retro-firing strategy gives the payload owner the benefit of a launch cost saving of about 20 k€/kg.

Alternatively, the mass of propellant saved for the elimination of the retro-firing de-orbiting maneuver could be used for extending the operational life of the S/C, increasing so the gain margin of the S/C owner.

Besides, a collateral benefit coming from the adoption of the END module is the damping, toward the payload, of the axial loads induced by the launch vehicle.

In any case, for those specific S/C's having no propulsion systems (like many scientific satellites) the retro-firing de-orbiting option is obviously not applicable, and the END de-orbiting strategy turns out to be the only reasonable solution.

A final consideration to be done is that a deeper investigation is required in the areas of the reliability of the system deployment and inflation operations after a long period (years) of "packaging", of the selection of specific technologies capable of ensuring a sufficiently satisfactory rigidisation of the vessel material after completion of the inflating process and of the accurate evaluation of the aero-thermal loads acting on the inflated structure, in order to make a robust assessment about the orbital altitude value at which such a structure will likely start the progressive mechanical degradation with the consequent problems of both aero-braking capability loss and fragments generation.

11 Reference Documents

[1] NASA NSS 1740.14 – "*Guidelines and Assessment" Procedures for Limiting Orbital Debris"* – 1995

[2] ESA et al. - "*European Code of Conduct for Space Debris Mitigation*" – 2007

[3] Balduccini et al. – "*A global strategy to clear the 'Space Protected Zones' from spacecraft at their end of operative life"* – 5th European Conference on Space Debris - Darmstaad - 2009

[4] Janovsky et al. – "*End-of-Life De-orbiting Strategy for Satellites"* - DGLR-JT2002-028 - Deutsher Lift und Raumfahrtkongress - 2002

[5] OHB-System AG - "*End-of-Life De-Orbit Strategies"* - EOL-OHB-ES-001 – 2002

[6] Wertz, J.R. – "*Space Mission Analysis and Design"* – 3rd Ed. – 1999 – Kluwer Academic Publishers

[7] Pisacane, V. - "*Fundamentals of Space Systems*" – 2nd Ed. – 2005 - Oxford University Press

[8] Toit et al. - "*Using Atmospheric Drag for Constellation Control of Low Earth Orbit Micro-satellites*"

[9] Langlois et al. – "*Inflatable Technology Activities"* – ESA Technical Dossier TOS-MCS/2003/827/LN/RR - 2003