A GLOBAL STRATEGY TO CLEAR THE "SPACE PROTECTED ZONES" FROM THE SPACECRAFTS, AT THEIR END OF OPERATIVE LIFE.

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ABSTRACT

The increasing number of orbiting spacecrafts is "pushing-up" the problem of controlling the risk of collision with orbital debris. Several European Space Agencies (ASI, BNSC, CNES, DLR and ESA) recently decided to adopt a common "European Code of Conduct for Space Debris Mitigation". This paper focuses on the Code Requirement relevant to clear the space volume around Geosyncronous Earth Orbit (GEO) at the satellite end of life, as well as the one that imposes a maximum residual orbital life of 25 years to defunct Spacecrafts (S/C) orbiting up to 2000 Km of altitude.

The optimal technical solution to meet such requirements is significantly different basing on mission and S/C characteristics, namely the orbit altitude, the S/C dimensions and ballistic coefficient, the dry mass, the on board propulsion type.

This paper intends to help the S/C designers to identify the most promising technical solutions that guarantee the compliance with the Space Debris Mitigation Code requirements, and minimise the impact on their S/C.

After an introduction, where the Code Requirements are recalled and commented, the paper defines the orbit altitude ranges for which an homogeneous debris control strategy can be efficiently developed. Then, the core part of the paper is constituted by the analysis of each orbital ranges above, and to the consequent definition of the most promising approach for clearing the orbit after the End Of Service (EOS). Such approaches include: a)"do nothing" where Code Requirements will be met by any S/C original design, b) utilise aero-brakes to accelerate the natural de-orbiting, c) utilise a propulsion module to de-orbit from Low Earth Orbit (LEO) or, d) to increase orbital altitude above the LEO protected one, e) utilise the S/C own propulsion to perform the graveyard firing of GEO satellites, supported by the use of an accurate, and reliable, residual propellant gauging system that assures the availability of the propellant for the manoeuvre. In addition to a) to e) technical solutions for the future space mitigation systems should be conceived to actively remove the large and intact satellite or upper stage debris. Its design should consist of a spacecraft having on board a proper randez vous and docking system, able to approach the debris, assume an identical tumbling motion, rigidly grab it and manoeuvre to a disposal orbit.

1. BACKGROUND

The concept that (natural) debris will constitute a significant hazard to the Space Travel is known since the 1940 (see [1] para 21.2). From that time on, the space missions have contributed to increase such risk by creating a large quantity of "non natural" (man made) debris whose number and total mass is continuously growing in space.

In spite of this, no binding rules have yet been adopted by the United Nations to counteract the debris risk (see [2]), even if 67 UN Countries have already declared this as a critical issue. Present status (see [8]) is that a set of seven common guidelines aimed just to limit the negative impact of the progressively growing orbital debris, have been defined and endorsed by a group of Nations and Organisations, on a voluntary basis.

As a matter of fact, sooner or later, the number of accidental collisions generated by flying debris (and the consequential economical, social and political impacts) will force Countries and UN to impose specific requirements in order to control the debris risk.

2. CODE REQUIREMENTS

The requirements and guidelines developed and adopted by ASI, BNSC, CNES, DLR and ESA are defined on [5] & [6]. The requirements that involves Satellite de-orbiting – re-orbiting aimed to clear the so called "Space Protected zones" are shown on figure 1 and are summarised here below for LEO and GEO. This paper focus is limited to such requirements.

<u>Low Earth Orbit (LEO) region</u>:

The Operator of a space system should perform disposal manoeuvre 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**…. (from requirement SD-OP-03 of RD7).

<u>Geosyncronous Earth Orbit (GEO) region</u>:

Spacecraft that have terminated their mission should be manoeuvred far enough (note 1)) away from GEO so as not to cause interference with space systems still in Geostationary orbit. The manoeuvre should place the S/C in an orbit that remains above the GEO protected region …. (from requirement SD-OP-04 of [7]).

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Note 1): "far enough" is quantified on [3]; it is a function of aspect area to dry mass ratio, and amounts roughly to + 235 Km above GEO.

3. LEO RANGE ANALYSIS

The LEO protected altitude range is presented in Fig 2. It can be subdivided in the following 3 sub-regions, for which a specific de-orbiting approach can result as more convenient:

- Lower altitude LEO range (0 to 500 Km altitude)
- Intermediate altitude LEO range (500 to 1500 Km altitude)
- Higher altitude LEO range (1500 to 2000 Km altitude)

a) Lower altitude LEO range

At low altitude the presence of a dense Earth Atmosphere causes a very significant drag over the S/C surface in the direction of the orbital velocity. Consequence of this consistent drag force, that is progressively growing with the lowering of the S/C altitude, is a "quick" reduction of the S/C altitude.

The altitude reduction rate is depending, in addition to the already mentioned S/C altitude, by the S/C ballistic coefficient Bc defined as the ratio between the S/C mass (m) and the product of the S/C aerodynamic coefficient Cd and the S/C drag equivalent surface (A):

$$Bc\ (Kg/m^2) = m\ (Kg)/(Cd\ (-) * A(m^2))$$

And with A that, as first approximation can be taken as the average S/C cross section exposed to the atmosphere drag.

Since for the most part of S/C's (see [1]) the following figures can be assumed:

$Cd=2.5$

$Bc= 50$ to $200\ Kg/m^2$

And taking into account the significant influence of the solar radiation, it is possible to identify the range of S/C mass & ballistic coefficient that generates an altitude reduction rate that assures the S/C natural de-orbiting to ground within the specified limit of 25 years.

The analysis led to the diagram on Fig 3 that has been developed without any contingency margin and basing on two different hypothesis as far as the solar radiation is concerned: a constant and mean flux, and a variable flux over its natural period of 11 years.

Calculation performed, as an example, for a S/C with a ballistic coefficient of 100 Kg/m^2, results in an expected orbit altitude threshold (for natural de-orbiting within 25 years) of about 600 Km. To this value a contingency margin should be added, leading to conclude that, if the nominal operative orbit altitude is </= to 500 Km, this does not require any specific design adaptation to meet the Code Requirement subject of this paper.

Note: *The code also defines an upper limit of the probability to cause a serious injury to, or death, of a single person due to the re-entry (named casualty risk): this figure is depending from the Launch Site and has a standard value of 1 E-4 but for launches from France (i.e. including the European Launch Site in Kourou) the more stringent figure specified by the CNES [9] shall be applied..*

b) Intermediate altitude LEO range

Within this altitude range the residual density of the atmosphere still consents the utilisation of drag forces to drive the de-orbiting of the S/C, but requires an augmentation of the drag effect by implementing additional drag surfaces.

The equivalent density of the Earth atmosphere is shown on fig 4, where the word "equivalent" makes reference to the fact that figures include the effect of the solar radiation pressure. Three curves are shown for minimum, mean and maximum expected values, taken from [1]. It shall be pointed out that minimum curve is the conservative side for calculating the de-orbiting time (i.e. in case of higher densities the S/C falls down earlier so meeting, with more margin, the requirement of 25 years minimum).

The fig 6 shows the behaviour of environmental torque / forces as function of orbit altitudes. It is evident the dominance of atmosphere drag forces below an altitude of 600-700 km while, above such altitude, solar radiation is prevailing; note that the magnetic torque/force is not applicable for an inert S/S.

What above suggests the utilisation of deployable drag shield as an attractive approach, mainly for S/C without an on board propulsion, to reduce the ballistic coefficient at the end of the S/C Operative life, and in order to force its de-orbiting within the requested 25 years.

It is evident that, for progressively increasing altitudes, the atmosphere density drops, the effectiveness of the drag is reduced, and the shield surface to be deployed in order to assure the 25 years limit compliance, drastically increases (with relevant mass and volume impacts on the S/C).

At the top of the intermediate altitude range the mass and volume impacts of the required drag shield are equivalent to the one caused by the addition of a simple monopropellant orbit transfer module.

It is evident that such threshold altitude strongly depends on a) the presence or not of a propulsion system on board the S/S, b) its compatibility with firing with the S/C configuration at EOS, and c) on its reliable operability at the end of the S/C operative mission (see note below): in such a case the implementation of the deployable shield will be less convenient than to add few kilograms of propellant to carry out the de-orbiting burn

Note: the Code defines a minimum level of reliability of the de-orbiting operation of 0.9 (this shall take into *account that such manoeuvre takes place at the end of the S/C operative phase)*

More accurate analyses led to the LEO diagram in fig.8, where the availability or not of an on-board propulsion system on the original S/C has been taken into account. In case of S/C with on board propulsion (right side of the diagram) at altitude above 1500 Km, the preferred approach is to add a "delta-propellant" quantity in the existing propulsion S/S in order to carry out the re-orbiting burns above the 2000 Km.

The same approach of using a delta propellant on the existing propulsion S/S, but with firing to de-orbit instead of re-orbit is the preferred approach between 1500 and about 600 Km.

Below such altitude down to about 500 km, two options should be traded for the specific S/C characteristics:

- De-orbiting based on M1 quantity of delta-propellant calculated as sufficient to assure the 25y residual orbital life with the existing S/C original drag surfaces
- De-orbiting based on a M2 (lower) quantity of delta-propellant calculated as sufficient to assure the 25 years of residual orbital life when the S/C drag surfaces are augmented by using the END module

The last option could be the preferred one in case significant constraints/ impacts are expected to increase the delta-propellant mass on the existing S/C propulsion subsystem.

c)Higher altitude LEO range

In this altitude range no significant use of the atmospheric drag can be expected. Consequently a chemical propulsion system shall be utilised to transfer the S/C to an altitude above the LEO protected range.

In case the S/C already houses a propellant S/S, only impact will be to increase the usable propellant budget of the quantity needed to transfer the S/C from its operative orbit to the 2000 Km + an adeguate contingencies to account for a long term no-reentry risk. In case the S/C has no propulsion S/S on board, the implementation of an Additional Propulsion Orbital module is required.

Finally, as an example, the sizing process for selecting the more appropriate LEO de/re-orbiting approach is presented below:

S/C operative Orbit:	Circular at 800 Km
S/C mass (without aero brake):	800 Kg
Aerod coeff (w/o aero brake):	2.5
Ballistic coeff (w/o aero brake):	100 Kg/m2

1) Calculation of the Ballistic coefficient of the flight assembly that assure the compliance with the Code Requirement (<25 years): From Fig 3, with 800 Km altitude → the (required) Ballistic coefficient to meet the 25 years max residual orbit lifetime is → 8 Kg/m^2

2) Calculation of the required aero brake shield to lower the Ballistic coefficient from 100 to 8 Kg/m^2 being the ballistic Coefficient ratio KBC = BC (original S/C) / BC (new, after aero brake implementation), we calculate: KBC= 8/100= 0.08; Using the diagram in fig 5, for KBC=0.08 we read a surface shield of about 30 m^2

4. GEO RANGE ANALYSIS

The GEO protected altitude range is presented in fig 7.

Within this range, since all S/C's are assumed to have their propulsion S/S compatible with the graveyard firing manoeuvre, the requirement is to leave the GEO protected range at the end of the operative service mission, with a minimum probability of having enough residual propellant of 0.9

Here the problems are:

1) who, and on the basis of which data and calculation methodology, decides when this probability has been achieved

2) who and how verifies the calculation at point 1

Note: *point 2) is particularly critical since the utilisation of design margins on the calculation in 1) is against the economical interest of the S/C Operator that has significant revenues for any additional operative day of the satellite itself.*

The methods to safe the propellant to be utilised for the S/C final graveyard firing, are basically the following two:

- To implement on board a "segregated" propellant quantity sufficient for the graveyard maneuver, and that cannot be utilized by the S/C operator for the operative life (very complex and intrusive wrt the S/C)
- To oblige the Operator to install a reliable and qualified standard measurement device of the residual propellant, and to perform measurements at specific mission time (likely around 80-90% of estimated operative life) in order to fix the mandatory graveyard maneuver date.

We believe that it is the "Country Space Authority" where the S/C has been developed that has to manage the responsibility to comply with the code requirement agreed at the level of Space Agencies. Moreover, the easiest way for national Agencies to control such responsibility is to force the utilization of a standard (i.e. Agency approved) high reliability residual propellant gauging system, on whose output base the last date at which the graveyard firing must be executed.

5. SUMMARY OF SYSTEMS TO SUPPORT THE COMPLIANCE WITH THE CODE REQ'S

Three products, presently under development at ELV (an ASI and Finmeccanica Company), have been defined in order to assist the S/C Designers:

- to design the newly S/C's to comply with the space debris Code requirement relevant to the protected altitude ranges, in the more economical way
- to design the lower cost modifications to be introduced on an already designed S/C's, to comply with the space debris Code requirement relevant to the protected altitude ranges

They are:

Name	Acronim defin.	Use
END	End-of-life Natural Deorbiting	to create additional drag
CAOS	Compact Active Orbiting System	to house a Prop S/S
Spot Gauging		to carry-out prop residuals accurate & reliable measure.

and are briefly described below.

Note: *it is not the scope of this paper to present the technical & economical baseline of the specified product; this will be the subject of dedicated papers, based on the already obtained phase A results.*

A typical trade-off diagram to guide the designer in the selection of the more appropriate Debris / deorbiting control approach, is presented on fig.8

<u>END</u>

END is a module designed for S/C's that need to increase the atmospheric drag force at their operational orbit, in order to comply with the Code requirement to limit the S/C residual presence in the LEO protected altitude range below 25 years.

It is schematically represented in fig 9.

The END module is positioned between the S/C and the Launch Vehicle interface flange in order to:

- allow one, well localised and simple interface S/C – END
- allow a late and simple integration of the END module on the S/C
- consent to the S/C the dis-embarkment of the separation flange, and relevant interface connectors, with the Launcher (it is transferred to the END module)
- consent the damping of the separation loads (shock) acting on the S/C

The END module mission starts at the launch with the transferring of the relevant loads from the Launcher to the S/C and then for implementing the separation of the flight assembly (i.e S/C + END module) from the last stage of the Launcher. At this point the module starts the so called "flight storage" part of its mission that ends at the completion of the operative life of the S/C.

At this moment the S/C sends to the END module the command for deployment and stiffening of the aero-brake shield. The END module baseline mission continues in passive mode with the function of maintaining the structural integrity of the shield itself and of mechanical interface with the S/C, up to the final part of the re-entry into the (dense) Earth atmosphere (namely 90-150 km of altitude) where the loss of the structural integrity is accepted.

As option, the aero brake shield can be equipped with solar cells, magneto torque coils and magnetometer in order to allow powering of low consumption equipment during the long orbit lowering phase (position beeper, accelerometers for debris impact detection, low consumption sensors etc..)

The END Module is constituted by the following S/Ss:

- END Structure S/S:

In charge of housing the separation flange with the Launcher (standard are Ariane 937 and 1194 mm), to provide fixation and proper stiffness to all attached equipment and brackets, to properly damp the shock load resulting from the separation flange activation, to consent axial protrusion of S/C equipment (i.e. main engine), if needed

- END Inflatable shield S/S:

Constituted by a shield storage assembly (to house the folded aero brake shield during pre-operative ground and flight storage) and by a shield deployment and stiffening assembly (to deploy the aero brake by using a pneumatic system, and provide it with the necessary stiffness even when the pneumatic pressure will be lost)

- END Command & electrical interface S/S:

Whose major constituents are the main and redundant interface connectors toward the S/C and the Launcher, an electronic command unit to receive the deployment command from the S/C and to activate the shield deployment and stiffening assembly

- END thermal control S/S:

Constituted by the passive thermal control H/W and sensors needed to maintain the equipment operational temperatures and to protect the module from thermal impingement caused by the S/C during its pre-operative and operative life.

<u>CAOS</u>

CAOS is a module designed for S/C's that need a propulsion S/S in order to leave the LEO protected altitude range at the end of their operative life.

The CAOS module is positioned between the S/C and the Launch Vehicle interface flange in order to:

- allow one, well localised and simple interface S/C – CAOS

- allow a late and simple integration of the CAOS module on the S/C
- consent to the S/C the dis-embarkment of the separation flange, and relevant interface connectors, with the Launcher (it is transferred to the CAOS module)
- consent the damping of the separation loads (shock) acting on the S/C

The CAOS module mission starts at the launch with the transferring of the relevant loads from the Launcher to the S/C and then for implementing the separation of the flight assembly (i.e. S/C + CAOS module) from the last stage of the Launcher. At this point the module starts the so called "flight storage" part of its mission that ends at the completion of the operative life of the S/C.

When disposal phase begins the S/C will start the de-orbiting or re-orbiting burn to clear the LEO protected altitude range as requested by the Debris Code of Conduct.

In its baseline version, the S/C sends the CAOS activation command after having assumed a pre-defined attitude on the (known) starting orbit. From this time on, the module executes one or two burns (de- or re-orbiting) controlling the flight assembly attitude by OFF modulation strategy. At the end of the last burn, the CAOS module carry-out the passivation operation, also requested by the Code Of Conduct, in order to avoid any risk of later explosion on the flight assembly. This ends the CAOS mission.

The CAOS Module is constituted by the following S/Ss (level of redundancy can be sized depending on mission characteristics):

- CAOS Structure S/S:

  In charge of housing the separation flange with the Launcher (standard are Ariane 937 and 1194 mm), to provide fixation and proper stiffness to all attached equipment and brackets, to properly damp the shock load resulting from the separation flange activation, to consent axial protrusion of S/C equipment (i.e. main engine), if needed
- CAOS propulsion S/S

  It is a blow-down monopropellant S/S with 3+1 reaction control thrusters
- CAOS Avionic & electrical interface S/S:

  Whose major constituents are the main and redundant interface connectors toward the S/C and the Launcher, a gyro package and a control and driver unit for the thrusters and isolation valves operations.
- CAOS thermal control S/S:

  Constituted by the passive thermal control H/W and sensors needed to maintain the equipment operational temperatures and to protect the module from thermal impingement caused by the S/C during its pre-operative and operative life.

SPOT GAUGING

Spot Gauging system is a set of H/W to be integrated on board GEO S/C's in order to carry out a discrete number of accurate measurements of the (storable) residual propellant on board of each of the S/C propulsion system tank(s).

The System architecture is shown on fig.11.

The two tubing ends of the assembly are connected to the ullage side of the propellant tank whose residual shall be measured. During "no measurement mode operation" the latching valve is open, each of the measurement cartridges are sealed by individual isolation zero-leak valves, and both sides of the differential pressure transducer are sensing the same ullage pressure of the propellant tank.

When a measurement of the propellant residual is requested, and no propellant flow is leaving the tank, the latching valve is closed, trapping inside the branch from the valve itself and the differential pressure transducer, the actual tank ullage pressure. Few seconds later, one measurement cartridge is activated, injecting a known quantity of inert gas in the pressure envelope of the tank ullage. This cause a (small) increase of the ullage pressure that is function of the residual propellant volume inside the tank. The differential pressure read on the sensor will be finally transformed in the requested residual kilograms of propellant on board.

The typical measurement scenario for the management of a communication Satellite placed in GEO through a GTO to GEO transfer burn, is the following:

- First measurement (mandatory):

At the end of the GTO to GEO orbit transfer, since more than 60-80% of propellant has been already utilised (with a not perfectly known specific impulse) and this measurement can provide the S/C Operator with the first accurate estimate of the S/C expected operative life.

- Second measurement (optional):

A second measurement can be performed at 50-60% of the expected S/C operative life defined with the first measurement, in order to obtain a better estimate of the date at which the graveyard manoeuvre must be performed not to risk a propellant depletion before the completion of the manoeuvre.

- Third measurement (mandatory):

when the residual propellant is approaching the maximum needed for the graveyard firing (with a consistent safety margin to be subject of approval by Agency) the last measurement is carried-out in order to define the latest date for starting the graveyard manoeuvre.

The spot gauging system is compatible with any common liquid storable propellant (MMH, NTO, UDMH, N_2O_4 and with both bladder type or surface extension propellant management devices, inside the propellant tanks.

Layout of the spot gauging equipment shall be defined on case by case basis.
Differential pressure transducer output can be directly routed to the S/C OBC for processing, or a dedicated command and processing unit can be provided.

6. REFERENCES

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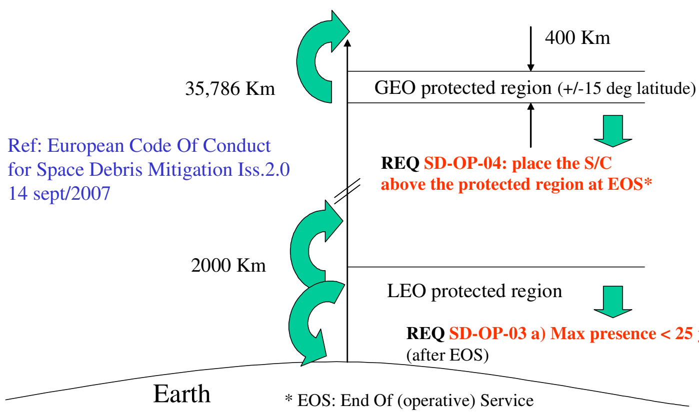


Figure 1: Requirement for protected altitude ranges

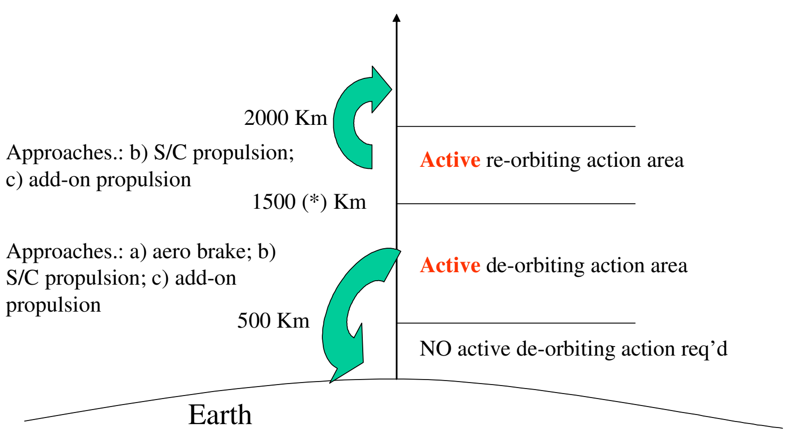


Fig 2: Technical approaches for LEO protected altitude range

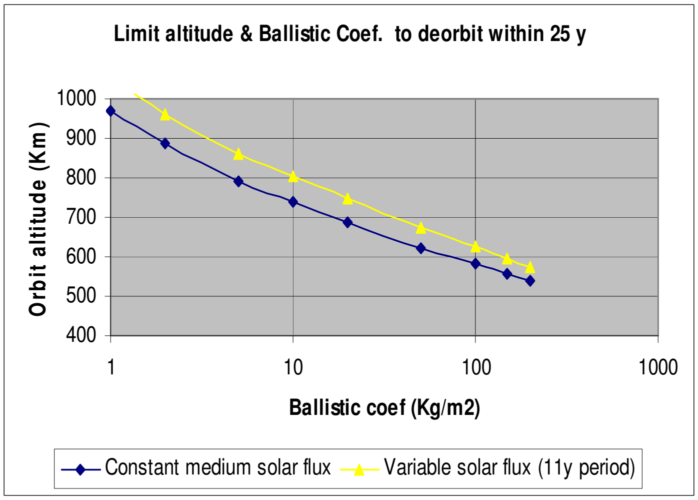


Fig 3: Limit altitude & ballistic coeff for 25y re-entry time

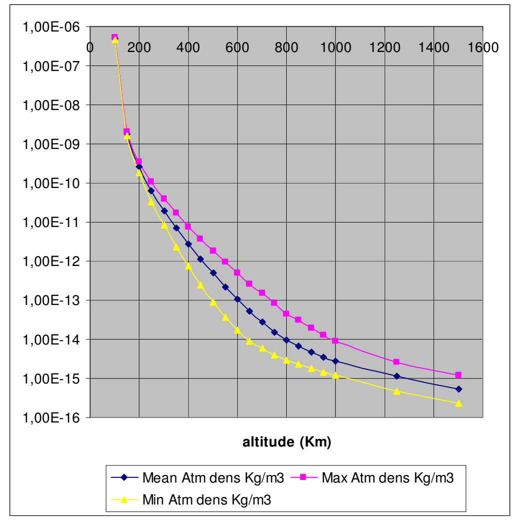


Fig.4: Equiv atmospheric densities vs altitude

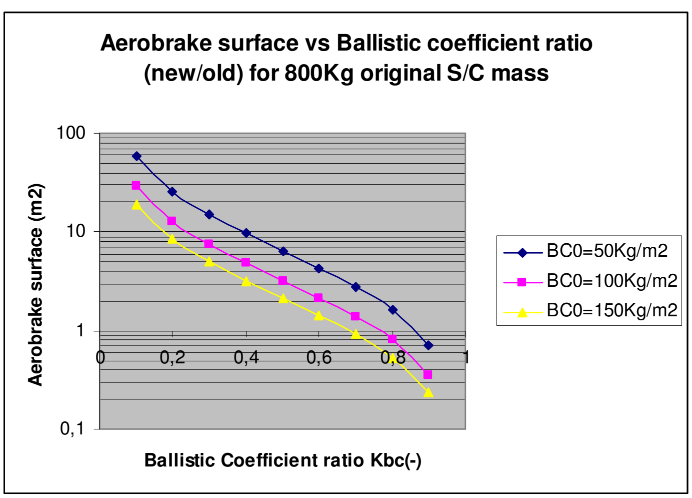


Fig 5:Ex. of calc diagram for the aero-brake surf. of a 800 Kg S/C

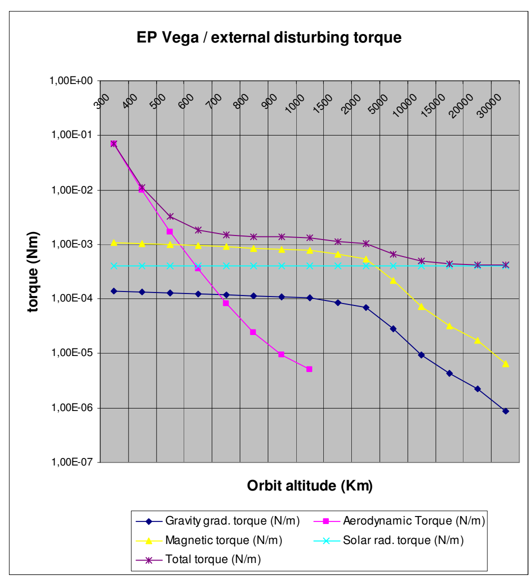
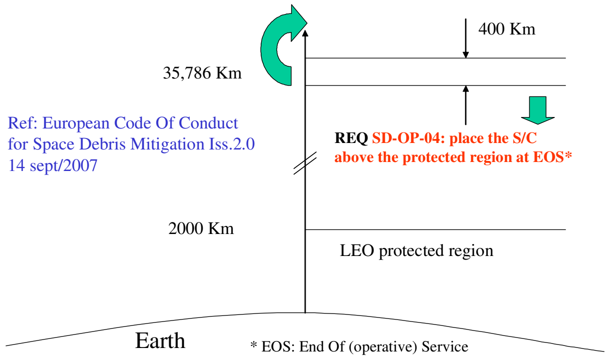


Fig 6: Environmental torques / Forces vs altitude



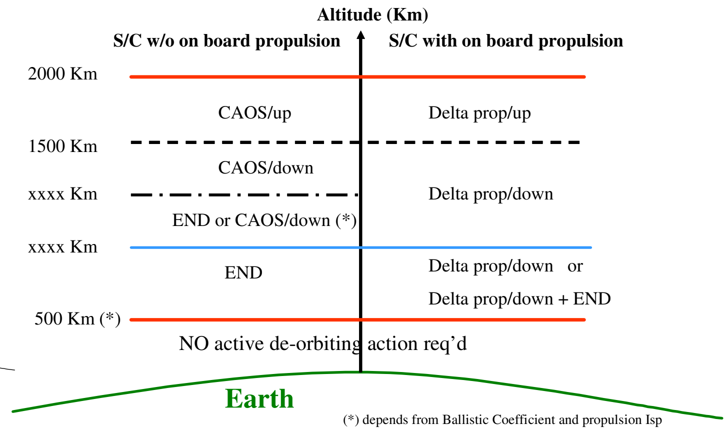


Fig 7: Requirement for GEO protected altitude range

Fig 8: trade.off diagram for de/re-orbiting approach from LEO

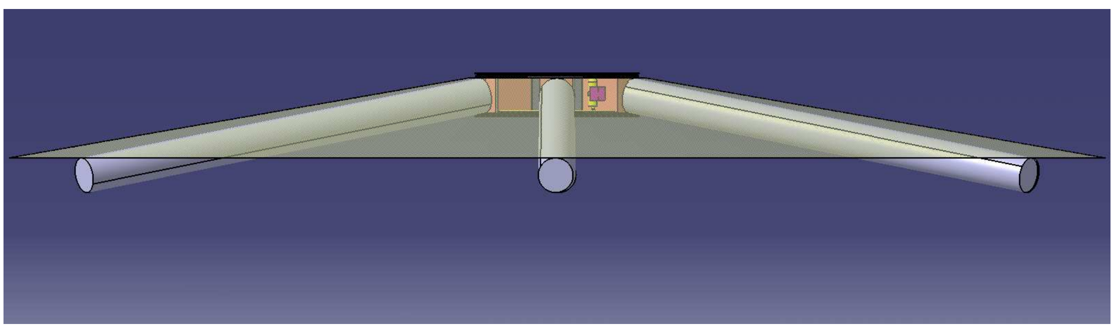


Fig 9a: detail of the aero-braking (deployed) shield

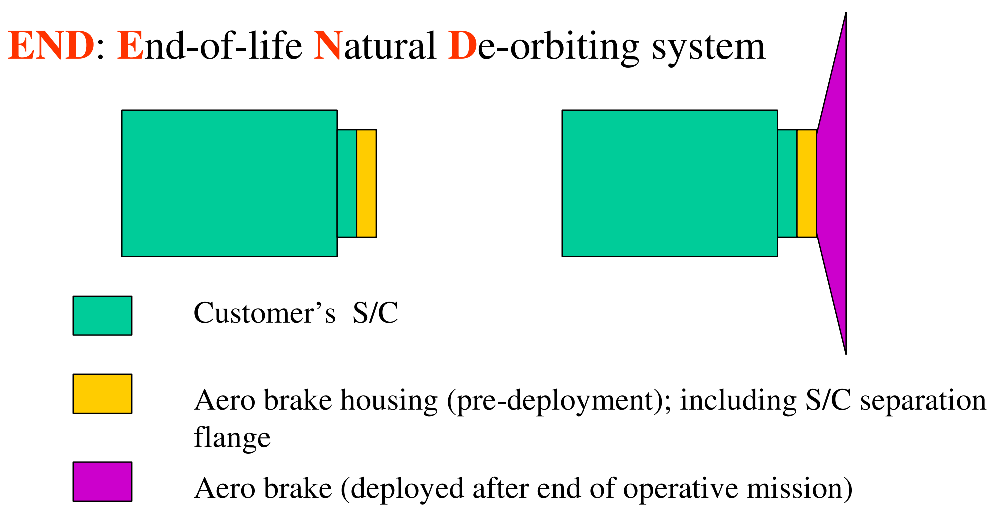


Fig 9: END concept

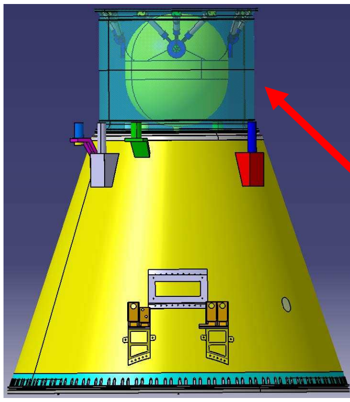


Fig 10a: Detail of the CAOS propellant tank

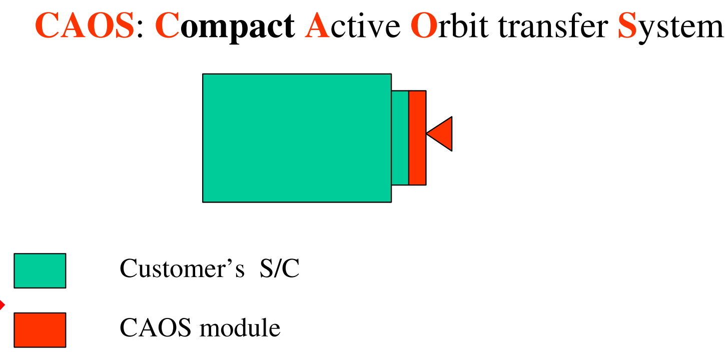


Fig 10: CAOS concept

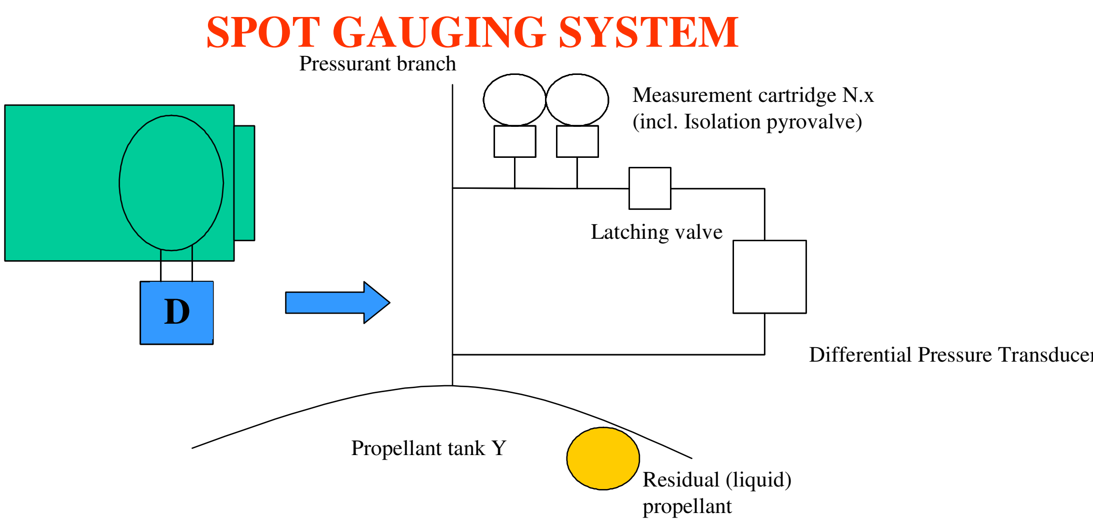


Fig 11: Spot Gauging System schematic