Position Paper

on

Space Debris Mitigation

Implementing Zero Debris Creation Zones

15 October 2005

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ABSTRACT

For several decades, orbital debris have been identified as a serious concern! This orbital debris potentially threatens future space missions, mainly in Low Earth Orbits and in Geostationary Earth Orbit, due to the risk of high energy collisions with valuable spacecraft. Orbital debris comprise the non-functional hardware orbiting the Earth, decommissioned spacecraft, spent upper stages, operational debris or residues from collisions; 94% of catalogued orbital objects are nowadays orbital debris. A complete presentation of the topic has been published with the year 2000 revision of the IAA Position Paper on Orbital Debris[1].

There are only very limited ways to improve the risks or effects of collisions:

- Removal of large potential colliders does not seem practically feasible today, due to operational and programmatic constraints,
- Collision avoidance is possible only with large catalogued debris, but requires access to precise orbital data for the largest debris, thanks to propagation of orbital tracks based on large observation facilities
- Shielding of critical spacecraft is possible up to a low energy limit only: debris larger than 1 or 2 cm impacting an active spacecraft may have very deadly effect
- Mitigation is by far the most efficient strategy: limiting the number of orbital debris in the critical orbital zones is the most efficient strategy for long term stability of the orbital population

The study led by an ad-hoc group of specialists from the International Academy of Astronautics (IAA) under the leadership of Commission V, has established a number of clear recommendations aiming at promoting long term orbital debris mitigation.

The study covered both the spacecraft and the launchers topics, through two independent sub-working groups, whose findings are presented separately in this document. Their major recommendations are very similar:

- **There shall be no generation of operational debris**: a space mission shall be clean, generating no long-term orbital debris such as clamp bands, fairings, optics protections, …
- **There shall be no risk of explosion following end of mission**: any spacecraft or upper stage left in orbit shall be “passivated”, i.e. its internal energy shall be eliminated: residual propellants shall be dumped, pressurants shall be depleted, batteries safed, etc. As per beginning of 2005, more than 180 in-orbit explosions have occurred, generating about 40% of the orbital debris population: it can easily be avoided.
- **Two orbital regions shall be protected**, due to their economical importance: Low Earth Region, ranging up to 2000 km altitude, and Geostationary Earth Orbit. A clear motto has been identified as a long term strategy: **there shall be no orbital debris creation within these two protected regions**. As this recommendation may not sound realistic currently, it may be replaced in the coming decade by **there shall be no long lived orbital debris creation within the two protected regions**

[1] IAA Position Paper on Orbital Debris, Revision 1, 24 November 2000

Part 1

Executive Summary

Section 1.1
BACKGROUND

Space debris are all man made objects including fragments and elements thereof, in Earth orbit or re-entering the atmosphere, that are non functional; the expression "orbital debris" is often used and bears exactly the same meaning as "space debris". It includes fragments and parts of man-made Earth-orbiting objects, such as fragments generated by satellite and upper stage break-up due to explosions and collisions. This derelict hardware is strewn across a wide range of altitudes, but is clustered around regions where space activity has been the greatest: LEO and GEO. Fewer debris currently reside in HEO.

Most of the man-made objects currently in orbit are space debris. Only about 6% of the catalogued objects are operational satellites. About one-sixth of the objects are derelict rocket bodies discarded after use, while over one-fifth are non-operational components. Pieces of hardware released during payload deployment and operation are considered operational debris and constitute about 12% of the catalogued population. Lastly, the remnants of the over 180 (at the end of April 2003) satellites and rocket stages that have been fragmented in orbit account for over 40% of the population by number. These proportions have varied only slightly over the last 25 years. Small and medium-sized space debris (smaller than 20 cm) include paint flakes, aluminium oxide particles ejected from solid motor boosters, break-up fragments and coolant liquid droplets escaped from nuclear reactors.

Space debris continually passes though space shared with functioning fragile and expensive spacecraft, manned and unmanned, performing vital navigation, communications, remote sensing, surveillance and scientific missions. This presents a variety of problems to the space faring community, from the possibility of catastrophic collision to the corruption of astronomical observations and possible interruption or degradation of RF paths.

Some space agencies are striving to generate less debris by applying debris mitigation measures. However, there will be little net benefit if only one space faring nation introduces preventative measures. Space is a public domain, and if it is to be protected so that all can continue to exploit its unique attributes, there must be concerted and cooperative action among all space faring nations. In part, this is necessary to make economic competition equitable, but it is also necessary to keep valuable operational regions technically and economically viable for the future.

Since operational lifetimes are generally much shorter than the orbital lifetime of both LEO and GEO satellites, it becomes clear that some active mitigation of debris creation in these regions of space is required. Unfortunately, because these have been the most widely used regions of space, they also have the largest population of orbiting objects. In LEO, both inadvertent and a few deliberate destructions have added significantly to this population. New developments such as constellations of communication satellites may increase the population further. To minimize collisions among objects large enough to generate substantial further debris, measures to limit the orbital lifetime of non-functional satellites will be required.

Space faring nations and space agencies have established the Inter-Agency Space Debris Coordination Committee (IADC) in order to exchange information on space debris research activities, to facilitate opportunities for cooperation in space debris research, to review the progress of ongoing cooperative activities and to identify debris mitigation options. Since 1994, space debris has been an official item on the agenda of the Scientific and Technical Subcommittee of the United Nations Committee on the Peaceful Uses of Outer Space (UNCOPUOS).

The most distressing aspect of the space debris problem is that it is getting worse in those regions most extensively used and could grow out of control at some altitudes and inclinations in the sense that collisions among the catalogued objects could become a significant debris growth factor. Because of the time and cost necessary to modify designs and operations practices, the debris problem will have a significant time lag between the recognition of the issues and the effect of changes. For this reason it is prudent to initiate action as soon as practical. The uncertainty involved in many of the present analyses highlights the need for technological developments to depict more accurately the hazard from space debris, prevent its creation, and provide protection from its impact.

The economic studies performed to date indicate that, because of significantly higher flux densities associated with small debris particles, economic impacts from collisions with small debris may be significantly greater than with large [≥ 1 cm] debris. In addition, since only limited orbital regions have significant debris flux densities, mitigation measures and policy actions may be staged so as to focus initially on those regions having relatively high flux densities.

Study objectives

This study is a follow on to the International Academy of Astronautics (IAA) *Position Paper on Orbital Debris* published in September 2001. The aim of this study is to prepare a new interdisciplinary work to promote the adoption and implementation of space debris mitigation measures. Debris mitigation addresses three broad issues: debris preventive measures, removal of existing debris, and protective measures including debris avoidance manoeuvres. This study will investigate the feasibility of and discuss cost issues associated with debris mitigation methods.

The scope of the overall IAA study is to:

- Promote the adoption and implementation of debris mitigation measures.
- Produce a follow-on Academy addendum to the position paper on Space Debris.
- Outline operational procedures for compliance with evolving space debris mitigation guidelines and standards.
- Establish a set of guidelines for space debris mitigation purposes.

When completed, these guidelines should be applicable to all space-faring nations. The study will not deal with safety issues associated with uncontrolled or controlled re-entry, those issues being the responsibility of each launching state.

This study group was tasked to:

- Review space debris minimization goals established by IADC and other bodies.
- Review current spacecraft design, deployment and operations practices that affect the creation of space debris.
- Compile and assess design and operational options to enhance compliance with evolving space debris mitigation guidelines and standards.
- Establish a set of IAA-recommended guidelines to be used by spacecraft and launchers manufacturers and operators for space debris mitigation purposes for all satellite and launcher pre-operation, operation, and end-of-mission phases and orbit regimes.
- Consider cost, benefit, and affordability issues associated with space debris mitigation.

Section 1.2

SPACE DEBRIS MITIGATION GOALS

The recognition of the need to mitigate space debris and its effects on space operations arose from observations of on-orbit break-ups, most notably those of Delta second stage explosion events (first break-ups December 1973). In 1980 the International Astronautical Federation (IAF), on behalf of the United Nations Outer Space Affairs Division, issued a study in which debris management in the geostationary orbit was addressed. The American Institute of Aeronautics and Astronautics (AIAA) position paper on space debris in 1981 was the first comprehensive public pronouncement about the problem. NASA's subsequent efforts to reduce the risk of further Delta fragmentations led to a policy of removing residual propellants from spent stages. Debris mitigation goals began to be articulated to reduce debris releases from operations and to prevent on-orbit break-ups.

Following bilateral and multilateral discussions among space agencies, the IADC was established in 1993. Founding members are ESA, Japan, NASA and ROSAVIAKOSMOS. Later members include CNSA, ISRO, CNES, BNSC, DLR, ASI and NSAU. The primary purpose of IADC is to exchange information on space debris research activities, to facilitate opportunities in debris research, and to identify debris mitigation options.

Proceeding from the principles of limiting the release of operational debris, preventing on-orbit break-ups, removing spent objects from useful orbit regions, and collision avoidance; the IADC developed mitigation guidelines for space systems. The IADC guidelines include a top-level recommendation for introducing space debris mitigation considerations early in the lifecycle of a space system and documenting their implementation measures in a Space Debris Mitigation Plan. Under the Guidelines, space systems should be designed so as not to release objects during normal operations or else an assessment should be provided that any debris release has little or acceptably low impact on the orbital environment and other users of space. Since on-orbit break-ups have contributed significantly to the current debris environment, special considerations are recommended for minimizing break-ups during operations and preventing accidental break-ups after mission operations are concluded; intentional destruction that could create long-lived space debris is strongly discouraged.

In general, there are currently no direct incentives encouraging satellite operators to follow the guidelines or penalties for a failed attempt. Some operators do follow voluntarily or are in compliance with requirements adopted by individual governments. Specific penalties and incentives will most likely emerge as governments codify the IADC guidelines.

The IADC guidelines utilize the concept of *protected regions* for Low Earth Orbit (LEO) up to 2,000 km in altitude and about the geostationary altitude in bands extending 200 km above and below 35,678 km altitude as well as plus and minus 15 degrees in latitude. These *protected regions* are depicted in Figure 1.1 from CNES. Debris mitigation for space systems passing through or operating in the protected regions is basically removal of the systems from the regions. For LEO, the options are de-orbit (direct re-entry is the preferred method), placement in an orbit in which atmospheric drag will limit the orbital lifetime after completion of operations (IADC have found 25 years is a reasonable and appropriate lifetime limit), or retrieval. Post mission disposal of space systems in the geosynchronous region requires removal from the protected region to avoid interference with other users in the region, including those engaged in orbit transfers above and below geostationary altitude.

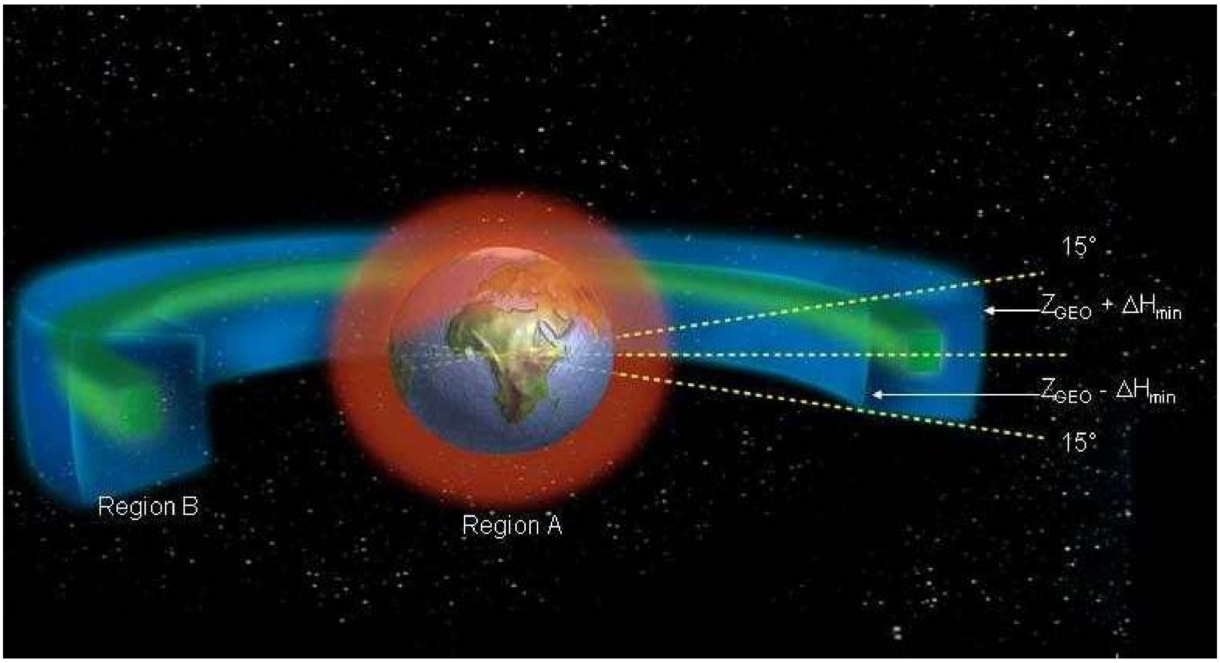

Figure 1.1 Space Debris Protected Regions

Space Debris mitigation goal for the Space Community

A purpose of this Academy Study is to initiate actions leading to agreement among space faring nations and the space industrial community that new satellite or launch systems put into space after 2012 create no debris within the protected zones. The Academy believes this goal can be achieved within the engineering environment through imposition of appropriate design requirements for new systems (System Requirements Review after 1st January 2006). While limited specific cost impact data is available for a conclusive assessment, the Academy believes that if the requirement--"zero debris creation"--is established at the Systems Requirements Review, the cost impact to a specific space vehicle will be minimized.

During some transitory phase however, the minimal guideline consists in a strong limitation of the lifetime of debris generated within the protected zones; a typical figure of 25 years following the decommissioning of the space vehicle would give excellent results.

The final goal to be established is:
<u>Zero debris creation within the protected zones</u>**.**

An interim step may be considered:
<u>Zero long lived debris creation within the protected zones</u>**.**

Section 1.3
SPACE DEBRIS MITIGATION RULES STRUCTURE

Numerous documents have been produced to date dealing with recommendations, guidelines, rules, standards, … related to space debris mitigation.
One can identify a general documentation structure based on three levels, depending on the field of applicability, as illustrated in figure 1.2[2]:

- General principles are defined at international level. These rules are guidelines, or recommendations, i.e. they are not legally binding. The most important set of guidelines has been produced by IADC, unanimously approved by its 11 members in October 2002[3]. This document has been introduced to the Scientific and Technical Sub-Committee of UNCOPUOS in February 2003[4]. Other important international organizations promote the subject through Position Papers, such as IAA through its 2001 Position Paper[5], or this document.
- National Regulations are produced at country level by National Agencies. They can be legally binding for contractors working for these Agencies, if the corresponding country decides to consider these rules as applicable. Numerous examples can be quoted from NASA, FCC, JAXA, CNES, BNSC, DLR[6]… These regulations proposed by members of IADC are generally very close to the IADC Guidelines. They are generally applied unilaterally, on a voluntary basis, which may result in inconsistencies at international level
- Norms and Standards are much more severe, imposing legally binding rules to industry and operators. They have to be developed at international level in order to guarantee an equal treatment of all through organizations such as ECSS and ISO. The process is currently on-going. These norms and standards are also very closely derived from the IADC Guidelines.

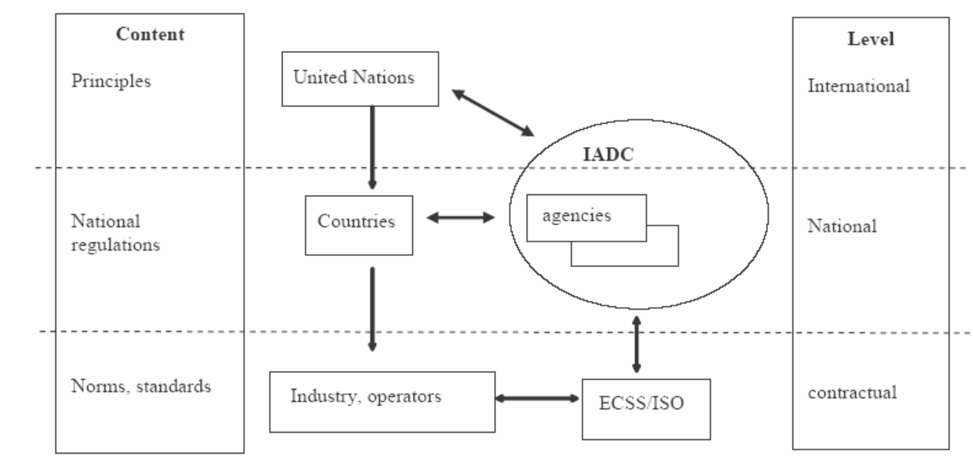

Figure 1.2 Documentation structure for Debris Mitigation

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[2] IAC-05-B6.3.08 Implementation of Space Debris Mitigation Guidelines in CNES, F. Alby, 2005
[3] IADC-02-01 IADC Space Debris Mitigation Guidelines, 15 October 2002
[4] UN A/AC.105/C.1/L.260 United Nations General Assembly – Inter-Agency Space Debris Coordination Committee space debris mitigation guidelines, 29 November 2002
[5] IAA Position Paper on Orbital Debris, September 2001
[6] European Code of Conduct for Space Debris Mitigation, Issue 1.0, 28 June 2004, approved by ASI, CNES, DLR

Part 2

Space Debris Mitigation Guidelines for Spacecraft

Section 2.1
CURRENT DESIGN AND OPERATIONS PRACTICES THAT LEAD TO DEBRIS CREATION

2.1.1 Introduction

The usual practices of satellite manufacturers and of operators responsible for in-orbit control of these vehicles may generate space debris. This section reviews the main practices giving rise to debris in three mission phases: transfer and early orbit operations (includes deployment), operations, and end of mission and disposal. This examination shall highlight the main potential sources of debris and indicate where the most urgent measures should be taken to minimize debris creation.

By way of background, break-ups (due to explosions or, possibly, collisions with untracked objects) of space objects have been the largest single source of orbiting debris. Even though historically the number of spacecraft and upper stage break-ups were comparable (52% vs. 48% of the total, respectively), approximately 75% of the fragments characterized by a long orbital lifetime resulted from the break-up of rocket bodies. Limiting attention to spacecraft, break-ups--both accidental and deliberate--were the third source of catalogued debris (~800), behind abandoned satellites (~2000) and mission-related objects (~1000). However, break-ups were the dominating source of millimetre and centimetre sized particles, while surface degradation is the leading debris generation mechanism in the sub-millimetre range, with a small addition from ejecta produced by meteoroid and orbital debris collisions with satellites. More complex is the quantification of the environmental impact of aluminium slag and aluminium oxide dust generated as combustion products by the Solid Rocket Motors (SRMs) integrated in a spacecraft. Nevertheless, the amount of slag released is still uncertain, most of the aluminium oxide particles have short orbital lifetimes, and the use of this propulsion technology is declining, at least for motors integrated in the spacecraft structure.

2.1.2 Transfer and Early-Orbit Operations

Solid Rocket Motors represent the main source of debris created during this phase in a mission. Solid Rocket Motors are used as upper stages or integrated into some payloads to perform orbital transfers, in particular between a geostationary transfer orbit (GTO) and the final geostationary orbit. These motors include up to 20% of aluminium particles in order to stabilize combustion.

During the thrust, particles of aluminium oxide (Al_2O_3) are ejected, which make up about 30% of exhaust products. For instance, in 1997 there were 24 SRM firings ejecting about 16t of aluminium oxide. The size of these particles is generally less than 10 microns and they are ejected at high velocity in the opposite direction to the velocity vector of the spacecraft, which results in a very limited in-orbit lifetime.

At the end of combustion, however, instability induces the ejection at low velocity of larger particles called slag. These particles can reach 1 to 2 cm in size and are made of a mixture of aluminium and aluminium oxide. Because the ejection occurs at low velocity, the particles remain in the vicinity of the initial orbit, which means that for a geostationary transfer orbit, the corresponding impact flux can be observed at any altitude.

These larger particles represent a significant part of the debris population at centimetre size. For instance, at the International Space Station altitude, SRM particles represent a significant part of the overall flux of particles on the order of 1 cm.

2.1.3 Deployment and Operational Debris

During the deployment phase of satellite operations, several items are deliberately released in-orbit. These include:

- Spin-up devices or spring release mechanisms released when the spacecraft is separated from the launcher
- Debris from explosive bolts and pyrotechnic devices used for separating the spacecraft from the launch vehicle stages
- Large structural elements (dispensers) left in-orbit in the event of a multiple launch
- Attach mechanisms released during deployment of antennae, solar panels, and other appendages
- Protective covers released during activation of optical, attitude, and other sensor systems

Objects may also be released by astronauts during extra vehicular activities, but their number remains limited and their orbital lifetime is low due to the altitude of such missions.

Most satellites contain deployable elements (antennae and solar panels, for example) whose movement is triggered by a pyrotechnic system that either loosens nuts or cuts shanks. Cutting a metallic shank using such a device produces debris from the shank itself or from the cutter.

Debris produced can vary in size, from a few tenths of a millimetre to a few millimetres. In all cases, the mass of debris produced remains low. However, it is possible to reduce this mass by design (cutter shape and material, nature of shank). In addition, devices designed to trap debris may be added.

2.1.4 Operations

Some missions include the deliberate release of objects as part of their mission. The worst case seems to be the so-called Westford Needles experiment, where several million copper needles were released in 1961 and 1963 between 3500 and 3800 km altitude, with 96 and 87 deg. inclination. These clouds of 2 cm needles were released as part of a communication experiment.

A large amount of debris may also be produced as an unexpected outcome of normal operations. For example, the nuclear reactor core disposal procedure adopted after the accidental re-entry of the RORSAT satellite Cosmos 954 resulted in many liquid metal (sodium potassium) droplets escaping from the primary cooling system encircling the expelled reactor core. The diameter of these liquid metal spheres, located at 850-1000 km with an inclination of about 65 degrees, can reach 5 cm or more. Unfortunately, such debris can remain a hazard for years--the orbital lifetime of a 1 cm droplet is about 100 years.

In addition to these specific cases, the production of debris during the operation phase can come from break-ups, surface degradation or collision with other objects.

2.1.4.1 Break-ups

At least 184 orbital fragmentations have occurred, with 30% of these break-ups known to have propulsion-related causes[7] Approximately 4% were due to batteries, 8% were caused by aerodynamic break-up, and 29% were the result of deliberate actions that are thought to be related to national security. Only 1% can be attributed to collisions. The remaining 33% are officially considered to have unknown causes, although many are rocket body break-ups that are believed to be propulsion-related explosions.

Of all break-up debris currently still in orbit, 73% has been associated with rocket body fragmentations. These are generally assumed to have been caused by explosions during manoeuvres, break-ups caused by post-mission mixing of propellant and oxidizer residuals, and pressure bursts.

In general, satellite break-ups produced about 1/3 as much of the long lifetime catalogued debris as generated by break-ups of upper stages. This was due, in part, to the low altitude at which satellite break-ups often occurred and the subsequent removal of short-life fragments by the atmosphere. This often occurs for break-ups induced by the significant aerodynamic forces on spacecraft undergoing catastrophic decay. The debris produced re-enters the atmosphere very rapidly. Break-ups induced by battery and propulsion failures, on the other hand, can occur at any altitude and, in fact, have been confirmed also in geostationary orbit.

In conclusion, spacecraft break-ups have historically been a significant debris source. The satellite explosion risk has significantly decreased in the last decade as measures to vent propellant tanks and discharge batteries have been implemented.

2.1.4.2 Surface Degradation

Surfaces of spacecraft are exposed to the deleterious space environment of ultraviolet radiation, atomic oxygen, thermal cycling, micro-particulates, and micrometeoroids. This can lead to degradation in the optical, thermal and structural integrity of surfaces and coatings, and subsequent shedding of material into the space environment.

Although limited analysis of this phenomenon has been conducted, such shed material has the potential to make a significant contribution to the micro-particulate population at altitudes above 1000 km, where the atmospheric density is too low to effect a timely natural decay of these characteristically high area-to-mass ratio coefficient particulates.

It is estimated that there are over 63,000 m^2 of painted surfaces currently in orbit. A painted surface tends to be multi-layered with a brittle surface covering a ductile substrate such as aluminium.

2.1.4.3 Collision Risk with Small Particles (Ejecta)

Micro-particle impact on a spacecraft surface can result in the production of secondary particulates, also known as *ejecta*, which are released from the surface material and injected at low velocity into the orbital environment. It is estimated that secondary particles represent 3% of the debris in the millimetre size range at 800 km. Secondary particulate production is much greater

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[7] "History of on-orbit satellite fragmentations", N.L.Johnson et al. NASA JSC, 12th edition, JSC-29517, July 2001

from brittle materials such as glass and paint and for multi-layer materials. Depending upon the energy of the impactor, large paint fragments (called spalls) can be generated with up to five times the diameter of the crater in the substrate.

2.1.4.4 Collision Risk with Catalogued Objects

The collision of a satellite with a catalogued object (size above 10 cm) would have catastrophic consequences. Firstly, the satellite would be seriously damaged, or even destroyed, and secondly, a large amount of debris of all sizes could be produced.

Since the beginning of the space age, there has been three verified collisions of two tracked objects - the Cerise satellite with a piece of debris from a launch vehicle stage was the first recognized one. In this case, the Cerise satellite was seriously damaged, but some operational capability was recovered. Predictions are that the frequency of collisions will increase as the number of objects in orbit increases. It may be possible to manoeuvre operating satellites to avoid collisions with tracked objects.

2.1.5 End of Mission and Disposal

After its end of mission the satellites keep on producing debris as during the operational lifetime. The debris sources are the same: explosion risk, surface degradation, ejecta, and collision risk with other objects. The production of debris being proportional to the on-orbit lifetime, one objective of the mitigation measures is to minimize the presence of the satellite in useful orbits. Therefore, the design and operation practices that reduce the probability of success of the end-of-life manoeuvres are indirect sources of debris. These include:

- Satellites without propulsion systems where no disposal is possible
- Inaccurate estimation of remaining propellant, where disposal may not be completed successfully
- Insufficient manoeuvre capability, where disposal is not included in the mission plan
- Random and wear out failures that affect disposal propulsion systems.

2.1.5.1 Estimation of Remaining Propellant, Prolongation of the Operational Mission

Until recently, the end-of-life of a satellite with no major anomalies was determined by depletion of attitude and orbit control fuel reserves. The application of re-orbiting or de-orbiting measures at end-of-life implies defining the end-of-life as the moment when the quantity of fuel in the tanks equals the quantity necessary to carry out these end-of-life measures. Now, end-of-life is no longer a fact which is noted, but a decision that operators must make.

This decision is difficult to make because it may represent the operating loss of a system that is still functioning. For example, for a geostationary satellite, the transfer into disposal orbit requires about 10.4 m/s to be compliant with the IADC recommendation. This represents a loss of about ten weeks of operations for a satellite controlled in North/South and East/West, and several years for a satellite with no North/South station keeping.

The economic impacts of moving GEO satellites to higher altitudes near the end of useful life are shown in Figure 2.1. Net Present Value (NPV) and Return on Investment (ROI) are both seen to decline dramatically as the effective lifetime of a communication satellite is reduced in order to move to a disposal orbit.

As an example, one can see on this diagram that a reduction of 3 months on the spacecraft's life would decrease its ROI by some 1% and its NPV by 5%, as a reduction of 1 year out of 10 would decrease its ROI by 4% and its NPV by 13%.

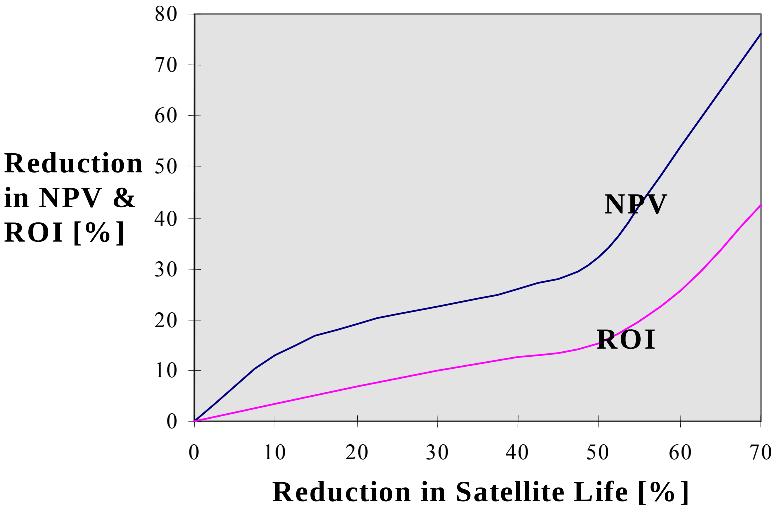


Figure 2.1 Effect of satellite useful life reduction on financial performance of GEO communication satellite business ventures.

Added to this significant cost is the problem of taking uncertainties into account. Estimating the amount of propellant available in the tanks of an in-orbit vehicle is very delicate. The methods used lead to significant uncertainties at end-of-life. These uncertainties must be taken into account in the form of margins that reduce the satellite's operational life. These margins depend also on the required success probability of the disposal manoeuvre.

In low orbit, the situation is even less favourable: for example in the case of a satellite located at an altitude of 800 km, transfer to an orbit with a lifespan of less than 25 years would signify a cost of 80 m/s (impulsive manoeuvre, 60 kg in case of a SPOT satellite), while direct de-orbiting with atmospheric re-entry would mean a cost of 190 m/s (150 kg).

2.1.5.2 Satellites without a Propulsion System

There are a number of satellites that do not require propulsion for their primary mission and therefore do not have the option of a de-orbiting or re-orbiting manoeuvre at the end of their operational lives.
The orbital lifetime of these satellites is normally limited due to their low altitude and relatively high area-to-mass ratio. Due to their size, especially in the nano-satellite (<10 kg) range, these objects are difficult to track and as a result, it may be difficult to warn other satellites with propulsion systems in time for collision avoidance manoeuvres.

Other satellites, like the Hubble Space Telescope, have no propulsion system for mission reasons. These vehicles also are unable to move to avoid collisions and cannot de-orbit on their own. Adding a de-orbit system to some spacecraft will have a major cost implication.

2.1.5.3 Passivation Process (Design and Operations)

Many satellites are simply abandoned in their operational orbit or transferred to a disposal orbit without taking other debris prevention measures. Satellites generally remain for a long time in this adverse environment where collisions with space debris or meteoroids and the high temperature changes (thermal cycling) between Sun-lit passages and eclipses may trigger break-ups. The main stored energy sources, which may explode include the following:

- Overpressure in charged batteries
- Residual propellant in the tanks
- High pressure gasses in pressure vessels (helium tanks)

Momentum devices such as gyros or wheels also represent an internal energy source, but it can be assumed that their energy level will rapidly decrease when the power is switched off.

2.1.6 Summary

The most critical sources clearly are solid rocket motor firings and on-orbit break-ups that may produce a large number of particles. Another debris source is surface degradation and ejecta. In this case the debris production rate is rather low, but is continuous during the entire orbital lifetime.

Section 2.2
OPTIONS FOR MINIMIZING DEBRIS CREATION

Several possible means of limiting the growth of orbital debris have been proposed. In broad terms, they may be classified in two categories, debris generation prevention and debris removal, even though the reality is more complicated. For instance, looking at the short-term, the satellite de-orbiting at the end of life belongs to the debris removal category. However, the fundamental motivation of such an action would be to prevent the long-term generation of new orbital debris due to collisions.

This distinction between short- and long-term perspectives is very important when the effectiveness of the techniques proposed for debris mitigation is evaluated against the additional cost of having them implemented. According to the results obtained by the models developed to investigate the future evolution of the debris environment, the mitigation measures intended to minimize the generation of new debris without constraining the total mass put into orbit (e.g., explosion prevention or curtailing mission-related objects) will be adequate only in diminishing the short-term hazard, while a long-term benefit can be uniquely achieved by reducing the amount of mass in orbit (e.g., through spacecraft removal or de-orbiting). Therefore, an appropriate "time interval" in which to assess the relative merits and drawbacks of the various mitigation techniques is needed. At present, 100 years may be considered a reasonable compromise between too short-sighted a view and the intrinsic unpredictability of future technological developments.

In order to control or reduce the debris in Earth orbit, the following strategies have been devised:

1. Minimize the release of mission-related objects
2. Avoid the occurrence of break-ups in space
3. Reduce the degradation of satellite surfaces
4. Avoid collisions between sizable space objects
5. Manoeuvre satellites to disposal orbits at the end of life
6. De-orbit satellites or reduce orbital lifetime
7. Actively remove intact space objects or debris

The goal of the first five is to minimize the creation of new artificial debris, without reducing the total amount of mass in orbit. The first three, in particular, address the most important sources of debris active today, if abandoned spacecraft and upper stages are disregarded. For this reason, their widespread application may result in significant short-term benefits for the debris environment, already evident after a few decades (~ 20 years), with respect to a "business as usual" scenario. The last two [6-7] strategies, on the other hand, are intended to reduce the mass in orbit, looking at the long-term stabilization or reduction of the collision probability among the space objects larger than 10 cm. In this case, several decades (~50 years) will be needed to clearly appreciate their differential advantage in terms of overall collision risk reduction, but such measures are deemed necessary for the long-term preservation of the environment in Earth orbit, in particular in LEO and GEO.

In order to implement the debris mitigation strategies listed above, several technological solutions exist. Some are already practicable and affordable, others could be economically advantageous but still need further investigations, and some are feasible from a technical point of view, but prohibitively expensive, at least for the foreseeable future.

2.2.1 Mission Design

One effective measure to avoid debris creation is to limit the orbital lifetime of spacecraft. In GEO or MEO the acting trajectory perturbations have only minor influence on the lifetime of the spacecraft, thus here the only option to avoid debris creation is to actively remove the spacecraft from its operational orbit at the end of its mission. In LEO, or elliptic orbits passing through LEO, mainly the residual atmospheric drag determines the orbital lifetime. Figure 2.2 shows, for a broad range of typical spacecraft, the orbital lifetimes as a function of the area-to-mass ratio and the initial circular orbit altitude.

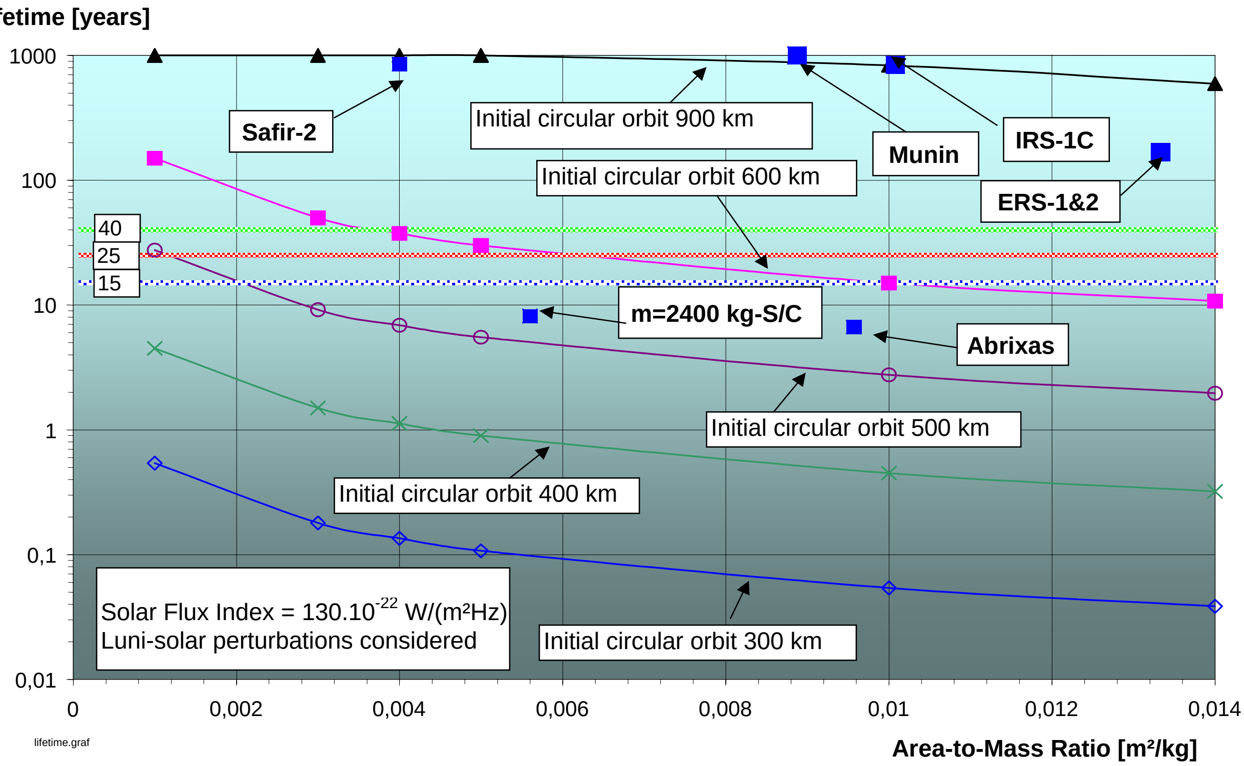


Figure 2.2: Orbital lifetime for circular orbits[8]

The initial orbital altitude has the main influence on the orbital lifetime. In the range of area-to-mass ratios shown, the lifetime varies from a few days to a few months for an initial orbital altitude of 300 km, to several hundred or even a thousand years for a 900 km initial altitude. The area-to-mass ratio has a relatively strong impact for compact satellites (area-to-mass ≤ 0.005 m^2/kg). For very large and lightweight spacecraft the importance of this factor reduces. If consistent with mission objectives, one option is to lower the orbital altitude, either initially or at the end of the operational phase, reducing the spacecraft lifetime and limiting the risk of debris creation. The remaining lifetime will be limited to 25 years if the spacecraft is in a circular orbit of about 600 km at the end of its operational life.

The orbital lifetime of spacecraft with elliptical orbits is also determined by aerodynamic deceleration forces, having most impact close to the perigee. As for circular orbits, the size of the

[8] "End-of-Life De-Orbit Strategies", Final Report, Doc. EOLL-OHB-FR-001, Iss. 1, 3rd July 2002, Study within ESA's, GSTP ESA-Contract 15316/01/NL/CK

deceleration depends on the mass and geometric properties of the spacecraft. In case of highly eccentric orbits, the gravitational effects of Sun and Moon considerably perturb the perigee of the orbit, resulting in a periodic oscillation of this altitude. The effect on the orbital lifetime depends very much on the initial launch date and the position of Sun and Moon relative to the spacecraft in its eccentric orbit. Their impact must be determined, taking into account the initial orbital parameters of the location of the launch site and the launch date.

As an example, 100 Monte Carlo simulation runs to determine the orbital lifetime of a spacecraft in an initial 36,000 × 250 km orbit (inclination i = 50°, area-to-mass ratio = 0.01 m^2/kg) were performed. The right ascension of the ascending node was varied stochastically between 0° and 40°, the right ascension angle to Sun was stochastically varied between 45° and 55°. Depending on the initial conditions, the orbital lifetime varies between about 8 years, with a relative frequency of 5%, to about 70 years, with a relative frequency also of 5%. The orbital lifetime distribution shows no regular trend or distinctive maximum. The effect of lunar and solar perturbations on the orbital lifetime is highly non-linear and dependent on the spacecraft-Sun-Moon geometry at the beginning of the mission. A small variation in the initial parameters, (e.g., a launch delay by a few minutes) results in significant changes of the orbital lifetime, which may vary by orders of magnitude.

2.2.2 Satellite and Hardware Design

2.2.2.1 End of Life Passivation

To prevent the on-orbit break-ups (including break-ups caused by chemical reactions and rupture by mechanical energy), all forms of energy storage should be made passive or quiescent at end of life. In this subsection, the following energy sources are addressed:

a) Residual propellants
b) Pressure vessels and other high pressure devices
c) Batteries

2.2.2.1.1 Residual Propellants

Usually, propellants will be spent during operations and disposal manoeuvres, but the following items should be taken into consideration:

(1) Bi-propellant systems should be designed to avoid common shaft valves, a common propellant tank separated by a bulk head, and lines which would induce mixing of propellants by single point failure, unless these systems are properly passivated;
(2) A combined propulsion system (apogee engine and AOCS) should be so designed that any line which is specifically used only for the apogee engine would be shut-off after the apogee boosting, and residual propellants and other gases trapped in the apogee related lines would be vented off at an appropriate time;
(3) The vent lines should be so designed as not to cause freezing;
(4) If it is impossible to vent, enough safety margin to avoid break-up should be adopted, or a pressure relief mechanism should be incorporated in the design.

2.2.2.1.2 Pressure Vessels and Other High Pressure Devices

A blow down system would not have pressurant that would cause break-up at the end of operation. A propellant tank in which fuel and pressurant are separated by bladder cannot be passivated completely. In such case, enough safety margins to avoid break-up under expected solar heating should be adopted or a relief valve implemented.

Leak-before-burst designs are beneficial, but are not sufficient to meet all passivation recommendations of propulsion and pressurization systems. They are effective only when the rise in pressure is gradual. Heat pipes may certainly be high-pressure devices, but they may usually be left pressurized if the probability of rupture can be demonstrated to be very low. Namely they should be designed with a sufficient safety margin not to be ruptured by heating after completion of the mission.

2.2.2.1.3 Batteries

There have been eight break-up events attributed to batteries. Some types of battery cells (Ag-Zn) have pressure relief valves, but they are used on launch vehicles and only a small number of satellites. Usually, for high-pressure battery cells such as Ni-H_2 batteries, attachment of pressure relief valves or diaphragms should be done considering potential decrease of mission reliability.

A fundamental cause of break-ups is inadequate structural or electrical design. Well-designed battery cases should have enough strength to withstand the increase of inner pressure so that normal situations will not cause break-ups. Switching off the charging lines and discharging the battery at end-of-mission will surely reduce the risk of break-up.

The usual sequence should be (1) shut-off charging lines and (2) provide for positive or natural discharging. This operation can be conducted by ordinary devices and operations, or additional small number of relays and command lines, if required, so that additional cost should be negligible.

2.2.2.1.4 Command Destruct Charges

Some satellites may be equipped with command destruct charges for safe re-entry, security of data, or other purposes. There has been one documented case of accidental break-up events caused by these devices. Such events can lead to high intensity break-ups with large amounts of debris. To prevent explosions caused by erroneous commands, the command receivers should be disconnected at an appropriate time, preferably early in the mission. Moreover, explosive charges should be protected from thermal heating or other external sources that could lead to accidental activation.

2.2.2.1.5 Momentum Wheels

Flywheels and momentum wheels have kinetic energy so that they may be treated as an energy source for break-ups. Fortunately, they will stop shortly after cutting off the power supply, as was confirmed by the ground test of two momentum wheels, whose angular momentum was 30 Nms and the rotating speed was 4600 rpm: the coast-down time was about 60 min.

2.2.2.2 Propulsion

Propulsion is required at end of mission either to send the spacecraft to a disposal orbit, an orbit with limited lifetime, to perform a controlled Earth-atmospheric re-entry or to avoid a collision. Options include the use of either chemical or solar-electric propulsion. In Figure 2.3, an overview of the thrust-ranges and the specific impulse I_{sp} of typical spacecraft propulsion systems is shown. It should be noted that the thrust range varies over 10 orders of magnitude, while the specific impulse varies over 2 orders of magnitude.

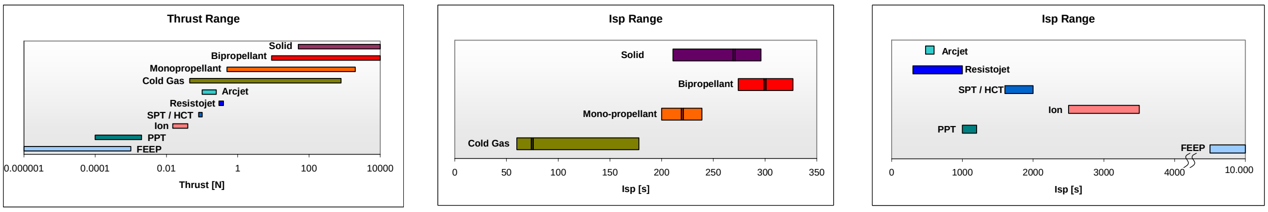


Figure 2.3: Thrust range and I_{sp} of spacecraft propulsion systems

In most cases, the addition of a de-orbit function on a spacecraft has a significant effect on satellite design due to the fact that the ΔV requirements for a de-orbit can rise up to 450 m/s and above. The impact on small spacecraft may be even greater, since the nominal mission usually does not require an AOCS. The mass of micro- and nano-satellites can double due to the addition of a de-orbit capability, substantially increasing the cost of such missions. Miniaturization efforts and ongoing developments in this domain could improve the situation.

- Solid propulsion appears as an attractive solution in many cases due to its compactness and a relative small mass increase.
- Monopropellant thrusters seem to be a good alternative in most cases, having the advantage of being readily available.
- Bi-propellant thrusters seem to be competitive only on heavy satellites.
- Cold gas systems are tremendously impaired by their low I_{sp}, and may only have use for de-orbiting small satellites.
- Electric propulsion was not identified as an optimal solution if the spacecraft is not equipped with this kind of propulsion for nominal mission life, since in that case the available electric power is in general too low. It can be expected that the outcome is considerably different for sufficiently heavy satellites with an appropriately large power/mass ratio and a relatively high initial orbit.

2.2.3 Satellite and Hardware Deployment

Separation from the launch vehicle, injection into provisional orbit, firing of the apogee propulsion system, deployment (and occasionally, retraction and re-deployment) of paddles and antennae, and transfer to operational orbit will be conducted in the early operations phase. During this phase, the possibility of debris generation should be minimized by the implementation of the following design requirements:

1. If fasteners or de-spinners would have long orbital lifetimes, they shall not be separated;
2. The apogee propulsion system shall not be separated;
3. Propellants for apogee engines shall be vented once they are no longer required.

The objects released during nominal operations shall be minimized as follows:

1. Aft-end throat plug type igniters and nozzle closures for solid motors shall be limited to the minimum;
2. Use of paddle clamp wire, antenna release mechanism parts, heat shields of apogee motor case and protective covers, etc. shall be minimized;
3. Yo-yo de-spinners shall not be adopted unless there are no feasible alternatives.

The cost to implement these measures is difficult to estimate, but costs should be low as long as they are implemented early in development. Of course, there are additional costs related to changes in mass and/or performance that must also be considered.

2.2.4 Collision Avoidance Manoeuvres

The collision of an operational satellite with a catalogued object could be a significant source of debris. In theory, these collisions can be avoided by monitoring of close flyby passes and by performing avoidance manoeuvres if necessary. In practice, monitoring collision risks is complex and delicate. Data contained in the catalogue supplied by US Space Surveillance Network is relatively imprecise: it is therefore necessary to use significant margins concerning the safety distance, which can increase the rate of false alarms.

The principle of collision risk monitoring consists in performing a processing operation with several steps. The first step, which is a rough sorting, uses information given by the Two Line Elements in the catalogue to highlight possible risks while taking significant margins into account. When a potentially dangerous object is thus highlighted, tracking measurements are performed using radar or optical telescopes to gain more knowledge about the object's trajectory. Since the precise orbit of the satellite is known to its control centre, it is then possible to determine the closest flyby distance and the probability of collision[9].

When the probability of collision is greater than the accepted risk, an avoidance manoeuvre allowing a safety distance between the two objects can be performed. To limit the amount of fuel necessary, the manoeuvre must be carried out a few orbits before the encounter event. In addition, carrying out such a manoeuvre modifies the operational orbit of the vehicle and can thus disrupt its mission. Because of these drawbacks, the collision risk monitoring procedure in practice applies only to manned vehicles (Shuttle, Space Station) or to high-cost, high-value vehicles.

Difficulties associated with collision avoidance include the following:

- Computer processing operations are heavy and complex. The database used for the evaluation contains around 14000 objects whose trajectories must be extrapolated over several days and compared with that of the operational satellite;
- Flight dynamics experts are often necessary to interpret the results and develop mitigation strategies;
- Executing a manoeuvre can modify the nominal orbit and interrupt the mission, representing a cost for the operator. A second manoeuvre may be necessary in order to return to nominal orbit;
- These operations can be complex and represent a risk for the satellite due to thruster malfunction or other potential failure associated with the additional manoeuvre.

[9] IAC-04-IAA.5.12.3.01 Collision avoidance as a space debris mitigation measure, Ailor & Peterson, 2004

Due to all these difficulties, most satellite operators refrain from carrying out this monitoring, particularly since the associated risks are still low. In addition, catalogued objects only represent a small part of the overall risk of collision, given that there are many more non-catalogued objects.

Despite these difficulties and complexities, some operators are beginning to be interested in knowing when threat objects may be flying near their satellites and prototype warning capabilities are being tested. It may be that the increasing risk of collision, or a high visibility collision of two tracked objects, may lead to the rapid evolution and acceptance of collision avoidance and space traffic control services.

The additional cost of these services would be an increase to the overall cost of operating a satellite or constellation of satellites. Assuming that tracking capabilities and data quality improve and that operators are not required to pay for tracking sensors (currently, many of these are maintained and paid for by governments), the additional cost per satellite would be minor. Additional satellite costs due to mission life reduction as a result of fuel use for collision avoidance or due to increased anomalies caused by more frequent thrusters firings may depend on orbit altitude (GEO vs. LEO, for example) and specific strategies used to implement avoidance manoeuvres.

2.2.5 Satellite Disposal

Simulations have shown that a real reduction of the debris population can only be achieved with very far-reaching measures. The only effective way to limit the growth of the orbiting debris is to remove satellites and rocket upper stages at the end of their mission from the near Earth space. This can be done either by de-orbiting or re-orbiting of the spacecraft. In the first case, a deceleration manoeuvre is performed, resulting either in an immediate atmospheric re-entry or in an orbit with limited residual lifetime. If the de-orbit manoeuvre is excessively large (e.g. for GEO spacecraft), the spacecraft orbit can be raised or lowered (re-orbited) to an altitude having no more interference with the orbits of operational spacecraft.

Due to the large required ΔV, de-orbiting is practically feasible only for spacecraft in LEO (below 2000 km altitude), or passing through LEO. For spacecraft expected to be completely destroyed during atmospheric re-entry with a negligible risk for the ground population, an uncontrolled de-orbit manoeuvre is permissible. This is the case, if the expected number of human casualties does not exceed a specified limit (for example 1 in 10,000 per re-entry event[10]) and the spacecraft does not contain hazardous objects with large masses and/or radioactive or poisonous materials. In these cases it is also possible to perform a braking manoeuvre, resulting in a new spacecraft orbit with a limited lifetime leading to an uncontrolled re-entry.

A remaining lifetime in the order of 15-40 years is assessed to be sufficient to provide the above-mentioned atmospheric cleaning effect. For a broad range of typical spacecraft having initial circular orbital altitudes below about 600 km, no specific end of life manoeuvre is required, because their remaining lifetime is below 15 years, the lowest value presently discussed with regard to debris mitigation.

In the case where the atmospheric destruction process is expected to be incomplete, a controlled re-entry should be considered. Simulations with typical spacecraft have shown that small spacecraft below approximately 20 kg mass do not require a controlled de-orbit, since they are

[10] NASA Safety Standard 1740.14 , August 1995

completely melted during atmospheric re-entry or the risk of residuals impacting on ground is negligible. Larger spacecraft having a mass of about 500 kg or more require with a high probability a controlled de-orbit, since the residual risk for human casualties on ground is assessed to be too high in case of uncontrolled re-entry. Detailed analyses for the intermediate mass range are necessary to predict their destruction behaviour during re-entry.

Controlled as well as uncontrolled de- or re- orbits require a manoeuvre generating a ΔV, which could exceed 100 m/s. Thus, the strategy and technical means to perform this manoeuvre have to be selected carefully, in order to limit the impact on the design of the spacecraft. Various technical options to generate the ΔV can be considered[11]:

- Active manoeuvres, using chemical or electric propulsion;
- Passive manoeuvres, using devices to increase aerodynamic drag;
- Tethers (e.g., dynamic or electro-dynamic tethers).

The first option can be used for de- or re-orbit and also for a controlled de-orbit if the generated thrust-to-mass ratio is sufficient. This is an important point. If the spacecraft is to be targeted to an ocean or other disposal area, the propulsion system must be sufficient to impart the required ΔV without subjecting the vehicle to subsequent perigee passes where atmospheric interactions could cause the spacecraft to go unstable. Such instability could prevent the final de-orbit burn from being implemented and result in a random re-entry.

Devices increasing the aerodynamic drag (e.g., by inflatable structures) can be used only for an uncontrolled de-orbit by lowering the semi-major axis and are limited to altitudes with sufficient aerodynamic drag (≤ 1000 km). Tethers can provide an impulse either dynamically or by generating a thrust through interaction with the Earth's magnetic field. Due to their wide application range, their flexibility and experiences gained with them, chemical or electric propulsion systems are considered the preferred near-term option to perform the manoeuvres.

The overall impact of the manoeuvre on the spacecraft design in terms of mass and cost increase is very much dependent on the size of the spacecraft. A significant increase of both terms by more than 50% is probable for very small (nano) spacecraft, whereas it can be on the order of less than 5% for larger spacecraft having already a propulsion function, which has to be adapted to the additional end-of-life manoeuvre. An issue that must be considered is the effect of these changes on other mission parameters, such as the launch vehicle (heavier vehicle might require a more expensive launch vehicle) or mission lifetime (reserving propellant for de-orbit might require early mission termination).

The situation in the geostationary ring deserves special attention because about half of all operational satellites (340 controlled satellites in total as of January 2004) are located in this narrow orbital region. International recommendations to re-orbit defunct satellites after end-of-life to orbits some 300 km above GEO were issued from various organizations since the early eighties. International consensus was finally reached in 1997 when the IADC recommended the following minimum altitude increase (in km):

$$\Delta H = 235 + 1000 \cdot C_R \cdot A/m$$

where C_R is the solar radiation pressure coefficient (usually with a value between 1 and 2), A is the average cross-sectional area and m is the mass of the satellite.
ITU adopted the same formula in 2004.

[11] IAC-04-IAA.5.12.3.05 Orbital SpaceCraft Active Removal OSCAR, Cheese, Martin, Klinkrad, 2004

Section 2.3
MITIGATION GUIDELINES

2.3.1 Operations

The overall objective of mitigation techniques is to reduce the growth of the threat space debris poses to operational spacecraft and to space operations. The previous section provided a good overview of steps that can be taken to accomplish this goal. For currently operating satellites, options to reduce debris creation are limited to:

1. Satellite and constellation operators should coordinate launch stage and hardware manoeuvre and disposal plans with other operators to minimize the possibility of future interference.
2. If data of sufficient quality is available, conduct collision avoidance analyses and move satellites if threat of collision is intolerable.
3. Properly dispose of spacecraft by moving to a disposal orbit or de-orbiting in accordance with guidelines.
4. Vent tanks, discharge batteries, and passivate spacecraft at end of mission.

2.3.2 Design

For spacecraft in the design stage, there are a number of measures that can be taken to minimize the possibility of debris creation. These include:

1. Assure quality and reliability of critical satellite systems. Failures that cause break-ups or loss of control are potentially large sources of debris.
2. Assure adequate design robustness. For example, assure that the design applies leak-before-burst design as well as structural and electrical design robustness for batteries.
3. Design hardware and control systems to vent residual propellants, shut off the battery charging lines, and minimize the onboard energy.
4. Design for periodic monitoring of critical parameters and take immediate action for debris mitigation should critical failures be experienced.
5. Design propulsion system for the disposal phase to comply with mitigation requirements (lifetime reduction or controlled re-entry for LEO satellites and re-orbit manoeuvre for GEO satellites). Include accurate measuring systems and algorithms to estimate the residual propellant to assure planned disposal manoeuvres. Size propulsion to target de-orbiting hardware into safe disposal location on final burn. Include an overall system for tracking, control and monitoring during the de-orbit in the mission design.
6. Design to avoid release of fasteners, nozzle closure, lens caps, and other materials from satellites during payload injection and initial operation phase.
7. Any program, project or experiment that will release objects in orbit should not be planned unless an adequate assessment can verify that the effect on the orbital environment, and the hazard to other operating space systems, is acceptably low in the long-term.
8. Minimize solid by-products (slag) production when using solid propellant. Since the actual phenomenon for slag production is still not fully understood and technology to minimize it has not been developed, liquid propellant engines or gas jet systems are preferred for apogee propulsion or attitude control systems.

9. Assess the collision risk with small debris on systems critical to operation of disposal systems. If the risk cannot be ignored, critical parts should be shielded by protection screens, hidden behind the structural elements, or include redundancy.
10. A bi-propellant system could be designed to avoid common shaft valves, a common propellant tank separated by a bulkhead, and lines where a single point failure would induce mixing of propellants and subsequent explosion.
11. A combined propulsion system (apogee engine and AOCS) should be designed so that any line of the system which is specifically used only for the apogee engine (i.e., oxidizer tank, some of the helium pressure bottles, etc.) would be shut-off after the apogee boosting (pyrotechnic devices may be used to shut-off the lines by physically deforming them), and residual propellants and other gases trapped in the apogee related lines should be vented off at an appropriate time.
12. Vent lines should be designed so that freezing will not hinder the venting operation.
13. If it is impossible to vent, sufficient safety margin should be adopted to assure no tank rupture under increased pressures due to solar heating, or to assure controlled pressure relief should limits be exceeded.
14. An apogee propulsion system that is to be separated after boosting should be designed to allow venting of the residual propellants before or shortly after separation. In general, the release of propulsion hardware with long orbit lifetimes should be avoided.
15. Decomposition of fuel remaining in a closed system will result in pressure build-up. In an adiabatic system, temperature will increase and thermal runaway can result in an explosion. Fuel lines should be vented to prevent explosions from this source.
16. High-pressure vessels should be vented to a level guaranteeing that no break-up can occur.
17. Use of aft-end throat plug type igniters and nozzle closures for solid motors should be avoided.
18. Use of paddle clamp wire, antenna release mechanism parts, heat shields of apogee motor case and protective covers, etc. should be avoided;
19. Yo-yo de-spinners should not be adopted unless there are no feasible alternatives.
20. Surface materials and coatings that are likely to lead to significant shedding should be avoided. Consideration should be given not just to the response to individual environments but also to the synergistic effects of combined environments observed in space and simulated in the laboratory.
21. Small satellites that do not include the capability to be de-orbited or moved to a disposal orbit should be launched into orbits with lifetimes consistent with the stated disposal guideline.

2.3.3 Orbital debris remediation: cost, benefit and affordability

Taking actions such as those suggested in the previous section may maintain satellite life, but lead to increased satellite cost and mass, increased transportation cost, or reduced satellite capability and loss of benefits. Requiring propulsion modules to re-enter the Earth’s atmosphere and burn up may affect payload delivery capability and lead to increased transportation costs. Requiring satellites to move to other orbits or re-enter at near end of life will increase costs and/or reduce satellite useful life. However, not taking actions to reduce growth of the orbital debris environment will, in the long term, result in reduced satellite useful life and higher costs or other financial and/or availability consequences. The direct effect of either alternative would be a reduction in satellite performance, increased transportation costs, or possibly both. Moving satellites to higher altitudes requires mass in the form of propellant and/or propellant and thrusters. In either case it would amount to shortening the satellite’s on-orbit station keeping life (the amount of life reduction being heavily dependent upon the utilized technology) or utilizing a launch vehicle with greater payload delivery capability and higher cost, both of which have financial impacts.

Debris mitigation has been an ongoing activity for GEO communication satellites. As per international agreements, GEO communication satellites are normally moved to higher altitudes near end of life in order to reduce the probability of collisions and other forms of interference. At least two mitigation measures have been considered that would affect the financial performance of GEO communication satellite business ventures. One would require GEO satellites to be moved to higher altitudes upon reaching the end of their useful life; the other would place constraints on transfer orbits by requiring, for example, that transfer stages re-enter the Earth's atmosphere and burn up rather than remain in orbit for an appreciable length of time. Both of these solutions could have a negative impact on the near to mid-term financial performance of communication satellite business ventures but may have beneficial impacts in the long-term.

Decisions regarding orbital debris mitigation policies are quite similar to most investment decisions (i.e., spend now for future rewards) except that the time frame is considerably greater (measured in terms of perhaps 50 to 100 years or more) than that encountered in most investment decisions. Therefore, simple discounting models are not likely to effectively represent the economic impacts of alternative orbital debris mitigation policies since the cost savings occur well into the future. To avoid the long-term discounting problem, an alternative approach is to utilize discounting over the relatively short mission horizon and then to consider different mission start dates. This allows the present value of mission costs to be developed based upon different debris mitigation policies (including the no debris mitigation alternative) and relative costs established for missions starting at different points in time and under the influence of different debris mitigation practices initiated at different points in time.[12]

Crucial to the analysis of economic impacts is the long-term forecast of the orbital debris environment. This forecast should be described in terms of the probability of impact with debris per unit spacecraft surface projected area per unit time as a function of debris size, models of debris sources, projected space traffic and debris mitigation practices, and satellite orbit and design characteristics. The probability of impact, along with satellite subsystem failure rates, can be used to establish satellite failure and replacement rates and to establish mission life cycle cost impacts. Since economic impacts resulting from orbital debris occur in the long-term, forecasting the debris environment in excess of 50 (perhaps to 100) years is necessary.

Evaluation methodology can be specifically developed so that both the impacts of debris scenarios as well as the combination of debris scenarios and mitigation policies (i.e., requirements imposed upon satellite configurations and/or launch vehicles) can be evaluated utilizing a common set of equations or models. Thus, the basic approach is to specify a set of sensors (or transponders) that then impose requirements upon satellite bus subsystems that are then configured and costed. Since mitigation policies can be specified in terms of changes to bus subsystem requirements, the same set of equations can then be used to reconfigure and re-cost the subsystems. This approach minimizes errors by basing calculations and costs on a common set of equations and related data and assumptions. Relative cost and availability are of primary concern with absolute values being of secondary importance.

The question will be asked: With mitigation efforts having only small effects well into the future, how can expenditures be justified today that will reduce the economic consequences of debris that

[12] Greenberg, J.S., "Economic Implications of Orbital Debris Mitigation [LEO Missions]," IAA-97-IAA-6.5.08, 48th International Astronautical Congress, October 1997

may not become consequential for 50 to 100 years or more? For multi-satellite missions, if it is assumed that the primary effect of debris remediation will be on unit recurring cost (not nonrecurring cost), then for Landsat type missions in the altitude and inclination angle regime of Landsat, and for small debris, it may be appropriate to increase satellite costs (with many caveats) by up to 3 to 4 percent starting now in order to eliminate the future growth of orbital debris – in other words, remediation actions may be economically attractive (for the high flux density Landsat regime) if they result in less than 3 to 4 percent increase in satellite cost and have but little effect on other satellite attributes. Of course, this would be less, and could approach zero, for satellites in orbits with lower levels of debris flux.

In summary, there is a need for models that can be used to assess more completely the economic impacts of mitigation strategies. To address this need, it is recommended that:

- Long-term debris forecasting models be developed that explicitly allow the effects of specific debris remediation schema implemented at different points in time to be observed in terms of probability of impact per unit cross sectional area as a function of time. This would allow a relationship to be developed between the cost of the remediation schema and the long-term savings that may result from their implementation. If this is not accomplished, economic and policy analyses relating to orbital debris will be significantly adversely affected.

- Integrated satellite performance, cost and life cycle cost models (such as SMALLSAT[13]) be developed and used to evaluate the economic impacts of orbital debris and remediation schema.

- Orbital debris remediation schema be considered in terms of satellite altitude, inclination angle and the time that remediation commences. Since the debris environment varies with space and time, so should the implementation of orbital debris remediation schema.

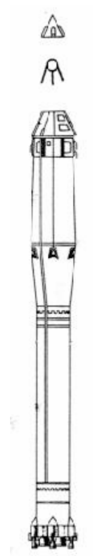
- Additional analyses be performed to establish mechanisms for modelling the relationship between satellite damage and debris size. Particular emphasis should be placed on small debris particles since they appear to dominate the debris environment when characterized in terms of likely impacts per unit cross sectional area.

13

Part 3

Space Debris Mitigation Guidelines for Launchers

Section 3.1
CURRENT DESIGN AND OPERATION PRACTICES



3.1.1 Launchers are significant contributors to debris population

Since the first flights to orbit in 1957, launch systems have contributed heavily to the growth of the orbital debris population. Almost all launches have left the upper stage in orbit, often in the same orbit as its payload. The first flight to orbit with Sputnik 1, for instance, schematized on the figure 3.1, had a payload of 84 kg, but the orbital stage of Zemiorka weighing 6,500 kg was left on the same orbit, as was also left the small fairing covering the satellite. Ever since this time, upper stages and secondary structures have been abandoned in orbit and have become the source of the largest population of integral heavy debris in orbit. Today, more than 1,430 integral stages represent 17% of the orbital objects population.

Figure 3.1: Zemiorka orbital stage with Sputnik 1 and the protective fairing

3.1.2 Upper stages cover a very wide range of definitions

Very diverse techniques have been used historically, depending on the size of the payloads and the definition of the planned orbits. Propulsion technologies range from small solid propellant stages to huge cryotechnic ones. The problems they generate in orbit often depend on the propulsion scheme which is adopted.

3.1.2.1 Solid propulsion

Solid propulsion tends to generate slag at the end of combustion. Large bits of alumina are ejected from the engine with very low velocities, on the same final orbit as the satellite. Very light alumina dust is also ejected during the operation of such engines, but with velocities opposed to the orbital one, which leads to a very short lifetime in orbit. Small bits of the nozzle may also be ejected due to erosion; lastly, the pressure cap of the motor may also be released in orbit at the firing of the engine.

3.1.2.2 Liquid propulsion

The main problems caused by liquid propulsion come from the propellants remaining at the end of the mission, either un-burnable or performance reserve. If not properly dumped, they may leak and explode (hypergolic propellants separated by a single wall), or become over pressurized in case the thermal insulation is inadequate, even a very long time after the end of the mission. The pressurization system may also be a source of debris should for instance a leak occur over time in the pressure regulator separating the High Pressure from the Low Pressure.

3.1.2.3 Electrical equipment bays

The upper stages are also associated with vehicle equipment bays that may also be source of orbital debris generation: electrical batteries often produce gaseous Hydrogen, which may lead to overpressure and explosion of the cell.

3.1.2.4 Paints

These stages are usually covered by a thin layer of paint, mainly used for electrical conductivity and to adjust the solar reflectivity coefficients of the material used in the manufacturing; unfortunately, these coatings are meant to be functional during the operation of the launcher, typically 1 hour maximum, and can peel as paint flakes with time under the thermal cycling in vacuum.

3.1.2.5 Thermal protection

Thermal protection is often applied on the propulsion and pressurization tanks, either simple sheets of Kapton-like material or more complex cellular foams in case of cryotechnic propulsion; these materials also tend to break down over time and generate significant orbital debris populations.

3.1.2.6 Secondary structures

In some cases, launches leave much more than the upper stage in orbit. Depending on the architecture of the launch system, lower stages may be orbited and operational debris may be created from structures dedicated to dual payload launches, modules dedicated to propellant settling or attitude control, or auxiliary tanks.

3.1.2.7 Destruction devices

Although there have been no actual accidents in orbit so far related to inadvertent triggering of destruction devices, these systems could be another source of explosive energy. Solar heating would probably not be sufficient to cause explosion, but a erroneous order could lead to an un-intended triggering of the destruction system in orbit.

3.1.2.8 Operational debris

Launch vehicle may release and leave in orbit a wide variety of operational debris such as fasteners, inter-stage structures, separation mechanisms, attitude control assemblies, protective shrouds. These operational debris represent some 13% of the total orbital population.

3.1.3 The range of missions covered by the stages is very wide

The full set of imaginable missions have been experienced, ranging from very low Earth orbits, to Geostationary Earth Orbit (GEO), as well as Sun synchronous or polar orbits, navigation constellations orbits and all the transfer orbits used either towards GEO or further to the Moon or escape missions.

Even though the debris generation mechanisms only depend on the manufacturing choices for any given stage, the importance of such a production is a function of the orbital region in which it occurs: generating debris in very low Earth orbits may not be as critical as generating them in the highly inclined relatively crowded regions from 800 to 1,500 km altitude or the GEO region. The main parameter is the lifetime in orbit of the debris, which lifetime varies strongly with altitude.

Under the current yearly traffic, one can expect some 80 to 100 new stages or major structures to be left in orbit every year.

Section 3.2
OPTIONS FOR MINIMIZING DEBRIS CREATION

3.2.1 High level considerations

3.2.1.1. Recommendations have been published aiming at mitigating space debris generation

3.2.1.1.1 Existing recommendations and requirements

For more than a decade, the major Space Agencies have been working on recommendations and requirements aiming at limiting orbital debris generation, and a few Agencies have developed their own standards. Based on these standards, the IADC coordinated efforts towards worldwide common guidelines and released them as the “IADC Mitigation Guidelines” after being officially approved by the 11 agencies composing IADC. These guidelines have been presented to UNCOPUOS and are currently under review.

In a limited number of cases, these guidelines have been translated into requirements through Standards published by some Space Agencies. In parallel, one can also note the effort at the European level through ECSS and at international level through an ISO ad-hoc working group to translate the IADC guidelines into practical requirements. However, this process is very long, and as of today there are practically no globally applicable requirements.

3.2.1.1.2 General rules

The high level principles are clear and can be summarized in three sentences:

- avoid voluntary debris generation,
- avoid break-ups in orbit,
- avoid long lived debris including stages themselves in the protected zones.

- The first rule tends to limit the generation of operational debris: the use of a launcher shall not lead to long lived debris left in orbit as a result of the nominal launch operations. Operators shall be as clean as possible, trying to leave only useful objects in orbit.

- The second rule aims at minimizing the number and the effect of break-ups in orbit by requiring systematic passivation of every stage left in orbit, i.e. the minimization or cancellation of any internal energy, either mechanical (pressure in the tanks) or chemical (residual propellants, or self-pressurization of the batteries at end of life).

- The third rule is based on the consideration that two protected zones can be defined with maximal allowable lifetime of 25 years. The first is the Low Earth Region ranging from ground to an altitude of 2,000 km at any latitude. The second is around the GEO arc, taking into account some margins for orbital manoeuvring. Depending on authors, and pending confirmation following studies, there may also be a need to protect the Navigation Satellites.

3.2.1.2 These rules are very rarely applied

3.2.1.2.1 Operational debris

Operational debris are always generated during launch missions. It is indeed very difficult in most of the missions not to leave at least the upper stage in orbit following the payload injection: the energy required for its controlled re-entry de-orbitation is normally too large to be feasible, and the effect of such a manoeuvre would hamper severely the competitiveness of systems applying these measures unilaterally.

In some cases, during multiple payloads launches, specific structures may also be left in orbit. Even though such inert structures may be more benign than upper stages and present shorter orbital lifetime, they contribute significantly to the orbital debris population.

Depending on launch systems, some other subsystems may be left in orbit voluntarily, such as inter-stages structures, separation mechanisms, attitude control assemblies, protective shrouds, and secondary propellant tanks.

3.2.1.2.2 Passivation

Upper stages are not always passivated. Most of the modern launchers nowadays do passivate their upper stages at the end of mission; however, it is not yet a general rule: some stages derived from propulsion modules of space probes are not modified to dump propellant and pressurant after use. It is also the case for some newly developed stages on small launchers, developed through non-governmental initiatives.

3.2.1.2.3 Orbital lifetime duration constraints

The 25 year rule is not yet universally applied in the Low Earth Orbit (LEO) region: some missions are fulfilling naturally this requirement (typically circular orbits below 600 km altitude, or Geostationary Transfer Orbits (GTO) with perigees lower than 250 km), but in most of the cases the operators do not yet take this requirement into account. There are only very few cases where the upper stage is lowered at the end of mission to cope with this rule. In a similar way, numerous upper stages have been left in the past in the vicinity of GEO.

3.2.2 Impact on design and operations

3.2.2.1. Releasing operational debris should be avoided

The limitation of operational debris may be easily achieved if considered early in the design of the launcher. Nevertheless, it is often difficult to have no operational debris at all.

3.2.2.1.1 Upper stage end of life

It is difficult not to leave an upper stage in orbit at the end of its nominal mission. One would think that upper stages could theoretically easily be de-orbited: they have a propulsion system, propellant tanks, an electrical equipment bay with all guidance, navigation and control functions, so it seems simple to just reignite the stage after payload separation and drive it to a destructive re-entry in the atmosphere.

3.2.2.1.2 Direct re-entry

The reality is unfortunately much more complex: when aiming for a direct re-entry, one should establish precisely the impact zone for the surviving debris in order to limit the probability of ground casualty; it is necessary to have a very reliable "de-orbit function" to achieve a re-entry with a steep entry angle and a well-controlled ground footprint, with redundant propulsion equipment and a guaranteed quantity of propellant. As an example, a 3 tons upper stage with storable propulsion (Specific Impulse (Isp) = 260 s) performing a direct re-entry from a 800 km circular orbit would require roughly 300 kg propellant, leading roughly to a 350 kg payload mass loss. Practically, such a de-orbitation has been applied in a very limited number of cases; one can quote an excellent example of what to do: the de-orbitation of the Russian Fregat upper stage of Soyuz used for the ESA Cluster-2 demonstration mission.

3.2.2.1.3 Controlled de-orbits

Controlled de-orbits can lead to a safe re-entry of spacecraft that are designed to withstand high deceleration levels and heat loads (e.g. Space Shuttle Orbiter or Soyuz capsules), or they may alternatively lead to an intentionally destructive re-entry (general case for any expendable launcher at End Of Life (EOL)), with a minimized risk of damage due to ground impacts of surviving fragments. In order to reduce the risk of casualties and damage on ground, the re-entry footprint should be chosen such that it is located over ocean areas, with sufficient clearance of land masses and traffic routes. Under such conditions even large stages can be de-orbited with acceptable residual risk.

In order to initiate a controlled re-entry, a sufficiently large ΔV manoeuvre at the orbit apogee must be applied to lower the perigee altitude well into the denser atmosphere. For a non-lifting body, a perigee altitude of about 60 km ascertains a safe atmospheric capture, and an immediate, steep re-entry with a well controlled, relatively small impact footprint of fragments which survive the deceleration and heating. The selection of the de-orbit perigee altitude depends on the mass-to-area ratio and on the drag and lift coefficient of the re-entry object. Heavy, compact objects, or objects with a high lift-to-drag ratio may require a lower perigee.

Figure 3.2 extracted from the ESA Handbook[14] shows the ΔV magnitude required for a controlled de-orbit of an object from a near-circular orbit in LEO region of given initial altitude into a direct re-entry orbit with a perigee height of 60km.

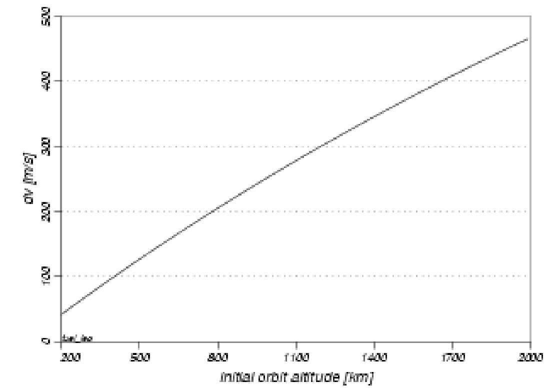


Figure 3.2: Required ΔV for a controlled, direct de-orbit of LEO objects from a near-circular orbit, as a function of the initial orbital altitude .

In figure 3.3 extracted from the ESA Handbook, the equivalent propellant mass fraction is provided for exhaust velocities V_e ~ 2,000 m/s (mono-propellant hydrazine system of specific impulse I_{sp} ~ 220 s) and 3,000 m/s (bi-propellant system of specific impulse I_{sp} ~ 310 s). The ΔV-related propellant mass fraction m_p/m_o can be derived from $m_p/m_o = 1 - \exp[-\Delta V/V_e]$, where m_p is the required fuel mass, and m_o is the spacecraft or upper stage mass at EOL.

[14] ESA Space Debris Mitigation Handbook, Second Edition, Issue 1.0, March 3, 2003

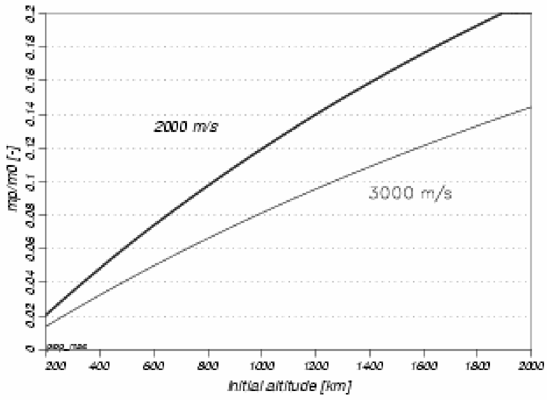


Figure 3.3: Required fuel mass fraction for a controlled de-orbit of LEO objects from circular orbits, as a function of initial orbit altitude, for mono- and bi-propellant systems with exhaust velocities of 2,000 and 3,000 m/s, respectively.

3.2.2.1.4 Final altitude decrease

An interesting variant consists of decreasing the final altitude of the upper stage as much as possible without achieving re-entry, by consuming the statistical propellant reserve, some dedicated additional propellants or even a dedicated propulsion system. The orbit may be lowered to achieve atmospheric capture and vehicle re-entry within a given post-mission lifetime limit, such as 25 years.

- Chemical propulsion systems usually have a high thrust and low specific impulse (200 to 300 seconds), and therefore their most fuel efficient way of manoeuvring to an orbit with a limited post-mission lifetime is to perform a small number of short-arc burns close to the apogee of the final mission orbit. These apogee burns lower the perigee into the denser parts of the atmosphere, thereby ensuring sufficient atmospheric drag and a corresponding altitude decay. Due to the high thrust levels over short time intervals, chemical propulsion is applicable to the lowering of perigees for highly eccentric as well as circular orbits.

- Electric propulsion systems usually have a low thrust and very high specific impulse (2,000 to 4,000 seconds), and therefore they perform a long duration, low thrust burn opposite to the velocity vector. This causes a near-circular orbit to slowly spiral inwards, gradually reducing the altitude, until a sufficient atmospheric drag level is reached. Such systems are not applicable for launch systems due to their cost and complexity, and they will hence not be considered hereafter.

- The duration of the orbit-lowering manoeuvre, the required change in orbital velocity, and the fuel consumed by typical chemical and electric propulsion systems can be derived for different post-mission lifetime limits, mission altitudes, and space system characteristics (e.g. mass-to-area ratios).

Figures 3.4 and 3.5 extracted from the ESA Handbook show the final perigee altitude and fuel mass fraction requirements as a function of the maximum post-mission lifetime for four reference scenarios, using a chemical propulsion system. The reference scenarios were chosen to reflect de-orbit manoeuvre requirements for moderate and high altitude LEO circular orbits (800 km and 1,400 km), and for low and high mass-to-area ratios (20 kg/m^2 and 200 kg/m^2). A specific impulse of 260 seconds was assumed for the chemical propulsion system (hydrazine/N_2O_4). The fuel mass fraction is defined as the proportion of the end-of-life space system mass occupied by the propellant needed to execute the de-orbit manoeuvre(s).

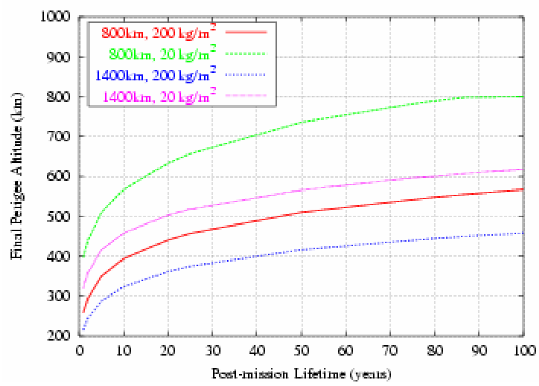


Figure 3.4: Required final perigee altitude to achieve a given post-mission lifetime limit after single-impulse de-orbit from a circular orbit at 800 km and 1,400 km, for an object with a mass-to-area ratio of 20 kg/m^2 and 200 kg/m^2.

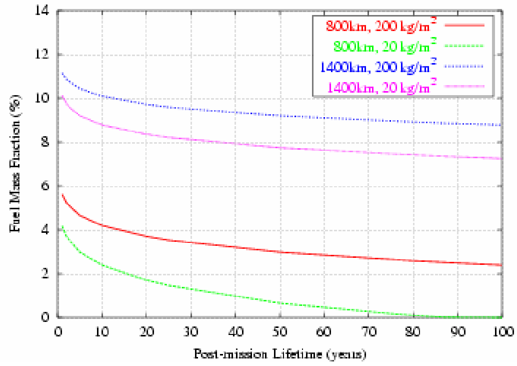


Figure 3.5: Required fuel mass fraction (for a chemical propulsion system with a specific impulse of 260 s) to achieve a given post-mission lifetime limit after impulsive de-orbit from a circular orbit at 800 km and 1,400 km, for an object with a mass-to-area ratio of 20 kg/m² and 200 kg/m².

Figure 3.6 from the ESA Handbook shows the initial circular orbit altitude and fuel mass fraction requirement for manoeuvring to a 25-year lifetime orbit, for five different mass-to-area ratios (10 to 350 kg/m²) of a space system. A chemical propulsion system is assumed (with I_{sp} = 260s). One can note that the magnitude of the perigee-lowering manoeuvres is strongly influenced by the mass-to-area ratio of the space system. As an example, considering a 3 tons empty upper stage left in a circular orbit at 800 km, with a mass-to-area ratio of 100 kg/m², the perigee has to be lowered to 520 km for a 25-year post-mission lifetime, consuming roughly 90 kg propellant. This figure can easily be compared to the 300 kg computed previously for a direct re-entry.

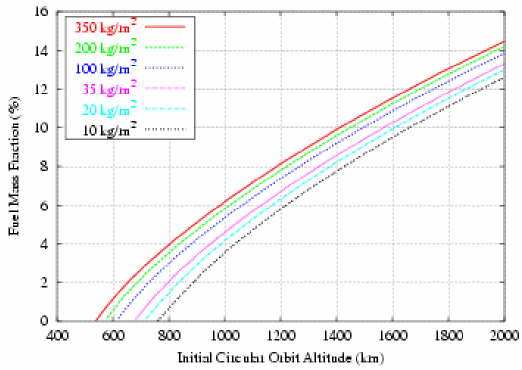


Figure 3.6: Required fuel mass fraction (for a chemical propulsion system with a specific impulse of 260 s) to achieve a post-mission lifetime of less than 25 years after impulsive de-orbit from a circular orbit, for an object with a given mass-to-area ratio.

- In the case of upper stages, which deliver spacecraft into GTO or into highly eccentric orbits, design constraints on the launchers (e.g. short battery lifetimes, TM visibility, thermal equilibrium) may prevent the option of an apogee burn to lower the orbit perigee and to reduce the orbit lifetime. In this case, idle burns or directed fuel venting after payload injection can be a non-optimal alternative. The availability of the system will normally be constrained to the first part of the initial ascent to the GEO destination altitude (in case of a GTO). In order to reduce the orbital lifetime of an upper stage, a reduction of the perigee altitude to increase air-drag or to intersect the Earth is the preferred option. This can be most efficiently accomplished by an in-plane manoeuvre, with an optimal orientation angle β (β = 0° in transversal, in-flight direction, β = 90° in radial, outward direction) a function of the orbit position (either in terms of true anomaly f, or time since perigee pass t_p).
  Table 3.1 shows the possible perigee reduction for GTO orbits of perigee heights 200km, 400km, and 600km, as a function of the thrust magnitude and the orbit position.

  The perigee reduction manoeuvre on highly eccentric orbits increases in its efficiency as a spacecraft or upper stage approaches its apogee. The optimal manoeuvre time is hence at the maximum possible time distance from the perigee. For an upper stage which delivers a payload into GTO, the perigee-reduction manoeuvre time should hence be at the maximum permissible time distance from the injection perigee which meets the criteria of system availability.

		True anomaly: f(°)	20°	40°	60°	80°	120°	150°	180°
Initial Perigee	200km	t_p (min)	(4′)	(8′)	(13′)	(25′)	(52′)	(128′)	(316′)
		ΔV =25 m/s	188km	174km	154km	107km	18km	-142km	-276km
		ΔV =50 m/s	177km	148km	108km	13km	-163km	-485km	-752km
		ΔV =100 m/s	153km	96km	17km	-174km	-526km	-1169km	-1704km
	400km	t_p (min)	(4′)	(8′)	(14′)	(27′)	(54′)	(131′)	(317′)
		ΔV =25 m/s	394km	386km	376km	351km	305km	223km	157km
		ΔV =50 m/s	388km	373km	352km	302km	211km	47km	-86km
		ΔV 100 m/s	376km	345km	304km	205km	22km	-306km	-572km
	600km	t_p (min)	(4′)	(9′)	(13′)	(28′)	(56′)	(134′)	(319′)
		ΔV =25 m/s	594km	586km	575km	549km	502km	418km	352km
		ΔV =50 m/s	587km	571km	550km	498km	403km	237km	104km
		ΔV =100 m/s	574km	543km	450km	396km	207km	-126km	-393km

Table 3.1: Maximum perigee reduction capability of a GTO mission

Table 3.1 shows maximum perigee reduction capability of a GTO mission for different initial perigee heights (200, 400, and 600km), and for given ΔV magnitudes (25, 50, and 100 m/s) as a function of the orbit position where the impulsive manoeuvre is applied. The orbit position is indicated in terms of true anomaly f(°) and time since perigee pass t_p(minutes). Final perigees lower than some 70 km, including negative values, indicate a direct atmospheric re-entry of the stage.

As a practical example, considering the Ariane 5 storable propellants upper stage (EPS) in a typical GTO mission with a perigee of 600 km, the payload separation occurs at an altitude of roughly 1,600 km. Waiting 5 more minutes for payload distancing, reorientation of the stage, propellant settling, and consuming part of the propellant statistical reserve (100 kg for instance), the final perigee can be decreased by 150 km, ending at 450 km, thus reducing the lifetime by thousands of years.

- This strategy can unfortunately not always be applied: the re-ignition of an upper stage requires, naturally, to have a re-ignitable engine. If such engines are easy to conceive with hypergolic propellants, it is much more complex with cryotechnic ones, requiring a specific cooling procedure prior to re-ignition and a dedicated igniter. Re-ignition also requires having the propellant located at the proper place in the tanks, in the adequate thermo dynamical state; to that extent, long propellant settling phases may be required in order to “de-bubble” the propellant and to stabilize the pressure in the tanks. Furthermore, stages have to be controllable without payload, which may be complex for “flat compact” stages. Numerous stages are already conceived to be re-ignitable after short coast phases, lasting some 20 minutes for instance; it however is much more complex to do so after hours of ballistic phase such as required for GTO missions where either the re-ignition has no efficiency whatsoever or up to 5 hours ballistic phase are required.

- Practically, if the stage is not conceived to be re-ignitable for its main mission, it will be hard to modify a posteriori. A solution may then consist in using the engine in a much lower thrust mode called idle mode, by-passing the turbo-pumps; this mode is used for instance by the Japanese H-IIA launcher.

3.2.2.1.5 Additional systems

Additional systems may help to decrease the lifetime in orbit and fulfil the 25 year rule for the protected region. Several solutions have been proposed, some of them demonstrated, but none of them applied yet:

- Tethers can be very useful to limit lifetime of stages in orbit.

- Passive tethers, acting by momentum transfer, require a heavy mass to counterbalance the mass of the empty stage.

  One could therefore imagine using such a tether between the upper stage and its satellite to help the latter to reach its final orbit, leading simultaneously to circularizing the satellite and de-orbiting the upper stage; this concept is currently studied in CNES as an option for micro-satellite launchers.

  Another possibility would be a cascading strategy such as the "mailman"[15]: this theoretical solution considers one single large "hunter satellite" visiting progressively a large population of defunct upper stages, performing a rendezvous with them, then de-orbiting them thanks to a passive tether; using optimally the re-boost due to the tether, an overall optimization enables de-orbiting a large number of upper stages. Preliminary results show for instance that with a "hunter" launched by Ariane 5 with full performance, a ten years mission in the crowded orbits between 500 and 1200 km, highly inclined, would enable the de-orbitation of some 50 defunct upper stages.

  However, these concepts are only theoretical and a lot of effort is required prior to concrete implementation. Passive tethers have been and are still extensively studied and tested: to mention just a few examples, NASA performed several flights of the SEDS experiments, and Delta-UTEC is planning a complex flight demonstration dubbed YES-2 under ESA funding.

- Electro-dynamic tethers, when exposed to an electrical current, generate a Laplace-Lorentz force which can be used to increase the drag for an upper-stage

  Numerous concepts have been proposed following this idea, such as the Terminator Tether© for instance (Forward & Hoyt), object of numerous publications[16] (see figure 3.7). Such a system can be added even on a old upper stage left on orbit, either with a visiting satellite or small dedicated interceptor, such as the Remora© concept for instance.

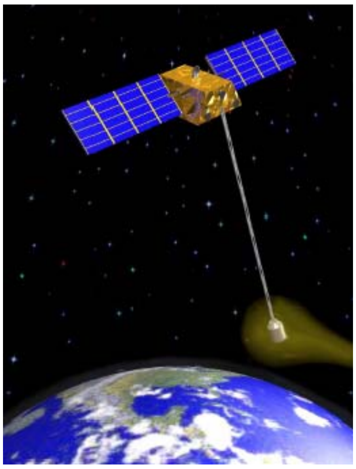


Figure 3.7: De-orbitation of a spacecraft using the Terminator© electro-dynamic tether (Forward & Hoyt)

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[15] SPACE-2005-1-00038, Optimization of tethered de-orbitation of spent upper stages: the "Mailman" process, Bonnal, Missionier, Malnar, Bullock, 4th European conference on Space Debris, Darmstadt, Germany, 18-20 April 2005

[16] AIAA-00-0329, The Terminator™ tether: Autonomous deorbit of LEO spacecraft for space debris mitigation, Hoyt, Forward, 38th Aerospace and science meeting & exhibit, 10-13 January 2000, Reno, Nevada, USA

These techniques nevertheless suffer from several technical difficulties: the variation of magnetic field on highly inclined orbits limits their practical use near polar inclinations; for the same reason, their efficiency is also limited to Low Earth Orbits where the magnetic field is significant; the dynamic behaviour of the tether may also be critical, its main eigenfrequency being commensurate with the orbital period.

They have not been tested in flight yet, if one excepts the two partly successful experiments performed from the Space Shuttle with the TSS. Last, the defunct stage, with its tether, will follow a spiralling decreasing trajectory during a very long time during which the tether will present a risk of collision with other spacecraft; indeed, the overall collision risk seems to be very comparable considering either a natural decay of the stage alone, or the enhanced decay with the tether, despite the much shorter mission time.

- Passive Drag augmentation devices may also be used.

  A ballute can be added to the defunct upper stage in order to increase its surface, increase its drag and reduce its lifetime in orbit. Russians from Roskosmos for instance, have proposed such a concept for their upper stages. Figure 3.8 describes such a Passive Deceleration System, and table 3.2 gives the associated performances[17].

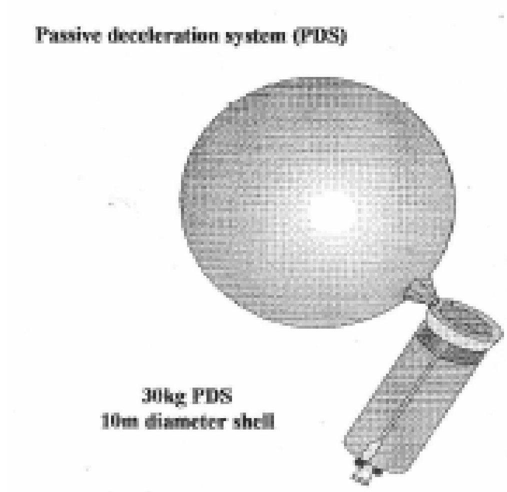


Figure 3.8: Passive Deceleration System PDS; Roskosmos

The overall efficiency of such systems is however not yet demonstrated, since once has to consider the overall collision probability: upper stage lifetime is indeed shorter, but the cross section of the assembly is much larger. Furthermore, such systems must remain inflated during very long periods despite numerous small impacts from debris or micrometeoroids: practical implementation is not yet demonstrated.

Injection orbits	Stage ballistic life without PDS	With PDS
1. Low circular 200-550km	upto 80 days	upto 15 days
2. Elliptical 300/900km 250/2500km	upto 3 years	upto 0,6 years
3. High circular 700km, 900km	upto 20 and more years	upto 3 years

Table 3.2: Performance of the Passive Deceleration System (Roskosmos)

- Electrical propulsion may be used to exert a continuous low thrust on the stage, with a very high efficiency. However, such concepts require high levels of electrical power, in the range of several kW, to feed the engines. This seems hardly realistic on upper stages which are never equipped with the same large solar generators found on spacecraft. It could be conceived for some cryogenic stages to use a fuel cell fed from the main tanks to power the electrical engines, but it would probably not be realistic economically.

[17] Conception of Russian launchers buildup and foremost mitigation measures, Chekalin, Yakovlev, Blagun, Kulik, ESA-SP-473 pp723-732, 3rd European conference on Space Debris, Darmsdtadt, Germany, 19-21 March 2001

- Other variants can also be considered to accelerate the de-orbitation of launcher stages:

- Solar sails for instance could possibly be used, but since their thrust to area ratio is extremely low (9 N/km² in the best case), it is hardly conceivable without drastically increasing the collision risk with the rest of the orbital population

- Slowing the stage with lasers or directed energy beams, either to feed electrical engines, through light pressure or vaporizing a target surface on the stage to generate a de-boost, can be imagined but feasibility is far from demonstrated even theoretically.

3.2.2.1.6 Time of launch

The time of launch for a mission may also greatly influence the lifetime duration of the stage in orbit after mission completion for highly eccentric missions passing through LEO, such as GTO. Depending on the exact time of lift-off, combined attractions of Moon and Sun may lead to a long term increase or decrease in the final perigee of the upper stage.

- Figure 3.9 and figure 3.10 extracted from the ESA Handbook show expected orbital lifetimes of highly eccentric orbits which have their perigees in the LEO regime.
  The two sample cases are GEO transfer orbits (GTOs) for launches from Kennedy Space Centre (KSC) into an orbit of $i=28.5^{O}$ (Fig.3.9), and from the Centre Spatial Guyannais of Kourou (CSG) into an orbit of $i=7.0^{O}$ (Fig.3.10). For both GTOs an initial perigee altitude of H_{pe}=300 km, and an initial apogee altitude at the GEO ring is assumed. The level lines indicate what excursions of the perigee altitude can be expected as a function of the right ascension of the ascending node (RAAN) of the orbit, and as a function of the time of year (solar aspect angle with respect to the orbit plane). Only solar and lunar attraction were considered in this analysis (no air-drag).

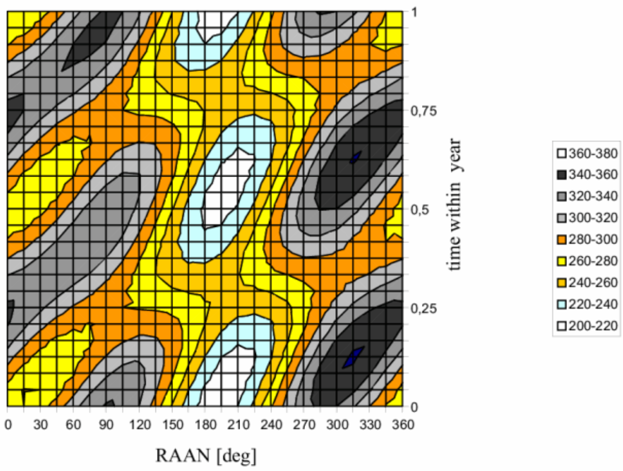


Figure 3.9: Resulting long-term mean perigee altitude of a GTO as a function of RAAN position and season, for an initial perigee altitude of 300 km, and for an inclination of 28.5^{O}(KSC launch).

Since the perigee variations may reach amplitudes of a few hundred kilometres, a proper choice of the initial orbit, or of the launch epoch can be important. Depending on the initial conditions, this may lead to an inward motion of the perigee into the Earth's atmosphere, with subsequent re-entry, or to an outbound perigee motion, leading to an extended orbital lifetime.

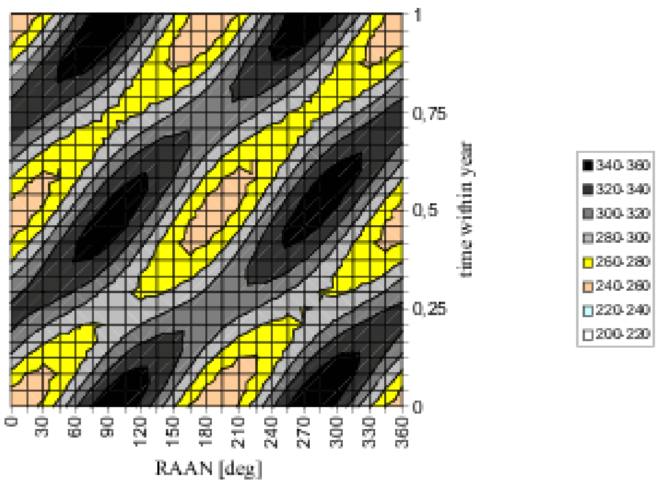


Figure 3.10: Resulting long-term mean perigee altitude of a GTO as a function of RAAN position

and season, for an initial perigee altitude of 300 km, and for an inclination of 7^{O} (CSG launch from Kourou).

From a comparison of figures 3.9 and 3.10, it becomes evident that the amplitude of the perigee variation due to luni-solar attraction, and hence the resulting long-term mean perigee altitude, increases with the inclination of the GTO plane. Lowest mean perigees, and correspondingly lowest orbit lifetimes, are obtained for RAAN positions close to 180^{O} and 0^{O}, and for launches in January and July ($\Delta H_{pe} \approx$ -100 km for i=28.5^{O}, and $\Delta H_{pe} \approx$ -60 km for i=7^{O}). For launches in April or October, into RAAN positions of 90^{O} or 270^{O}, the mean perigee altitude will increase by about +60 km. The density scale height at 300 km is such that the air density increases by a factor of 2 for a descent by $\Delta H_{pe} \approx$ - 40 km, and it reduces to about 1/2 for an ascent by $\Delta H_{pe} \approx$ + 40 km. The orbital lifetime is thus affected by the air density at perigee pass, and by the resulting air-drag, which due to the exponential density profile acts like an impulsive braking manoeuvre.
Selecting correctly the launch date may change drastically the orbit lifetime for the upper stage and operational debris: for the example mentioned above with a mission perigee of 300 km, the final perigee may vary between 200 and 400 km, corresponding to an average orbit lifetime ranging from 3 to 300 years.

- This mitigation technique appears attractive but unfortunately may not be effective due to the potential impact on the probability of launching at a specific time. The constraints to be fulfilled to launch a mission are already so severe that operators may not be willing to take another one in addition.
The point in time of a launch within the 11-year solar cycle, and the initial position of the orbital plane with respect to Sun and Moon, is in most cases dictated by mission objectives, and by spacecraft design and operational constraints. To give an example, the optimal launch slot for the TDRSS mission was studied by NASA and published during the 21st IADC.
Results (figure 3.11) were expressed as "probability that orbit lifetime will exceed 25 years", and were computed for a very wide range of launch dates. They showed that choosing among the possible launch dates and launch windows would lead to a very low probability and that choosing the best time within one given launch slot could reduce it by a factor 3.

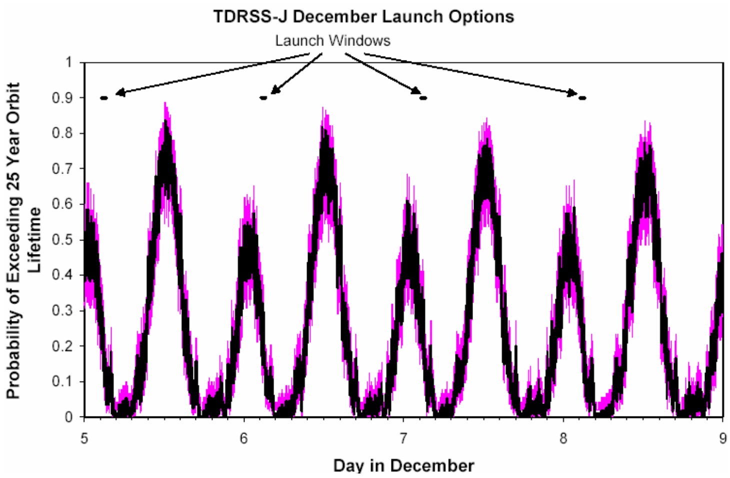
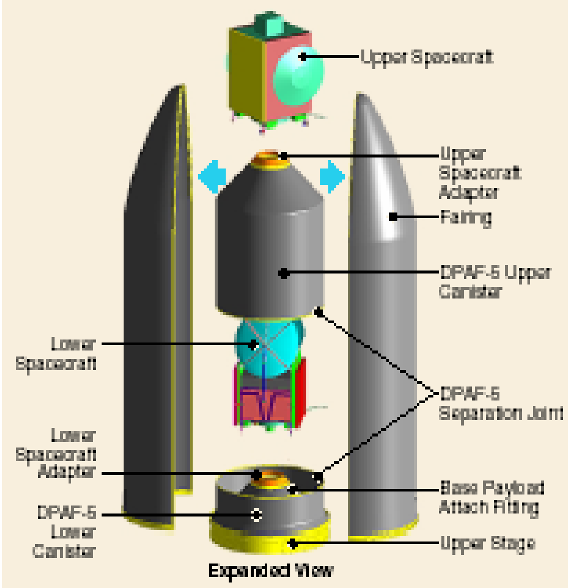


Figure 3.11: Evaluation of the optimal launch slot for the TDRSS-J mission – Probability of exceeding 25 years orbit

3.2.2.1.7 Secondary structures

Separately from the upper stage, large structures can also be left in orbit following launch operations.

- Most of the powerful modern launchers offer a capability of multiple payload delivery, generally performed thanks to a dedicated structure left on an orbit close to the satellites at



separation. For instance, the DPAF-5 structure expected to fly on the Boeing Delta-IV (figure 3.12) or the Speltra on Ariane 5 are separated after the upper payload, and before the lower one. There is however no risk to the payloads since the orbit at the end of the mission is generally a Transfer one, and not the final commercial one. Other designs could possibly prevent the release of such a large structure in space, such as petaled payload adaptors or concepts with hinges and actuators, enabling the opening of the upper part without releasing it into space, but such concepts are awfully complex, unreliable and expensive. Debris-wise, these structures are somehow beneficial: they replace an upper stage that would be much more complex to handle in orbit and would require passivation, they are totally inert and offer generally a very high area-to-mass ratio, leading to a lifetime duration in orbit significantly shorter than upper stages.

Figure 3.12: Delta IV Heavy Dual-Payload Attach Fitting DPAF-5 (Boeing)

- Some launchers are also nominally separating auxiliary tanks in orbit in order to optimize the overall performance. The Russian Proton upper stage Breeze-M, operated by ILS, has an additional external tank (see figure 3.13) enabling an important performance gain with respect to the initial Breeze upper stage. Nevertheless, this external tank is left in an intermediate orbit as an operational debris.

Figure 3.13: Breeze-M Upper Stage assembly (ILS)

- It is therefore quite unrealistic to imagine the use of launchers without associated operational debris. However, these debris are very large, and if properly passivated and in compliance with the 25 year lifetime restriction rule may not represent a severe danger. Nevertheless, one has to limit their generation as much as possible. As an example, one can quote the French CNES standard requirement on that subject: for launches with a single payload, one single operational debris is allowed (i.e. the upper stage); for multiple payload launches, two operational debris are allowed (the upper stage and the dual payload structure or payload dispenser).

3.2.2.1.8 Smaller operational debris

Operational debris may also be found at smaller scale: the separation of stages or payloads can generate debris if not properly designed. Pyrotechnical separations have to be leak-tight in order not to release lead bits in space; a dedicated test has been performed, for instance, on the Speltra of Ariane to check that the result of separation was only the release of a very small quantity of dust with a diameter less than 10 microns. Pyrotechnical bolts shall be trapped, as well as payload separation clamp bands.

When designing the components of segments connecting stage, lower fairing, payload adapter and engine, cables or other means can be used to catch the operational debris (such as: clamp band, sensor protective shield or explosive bolt).In the course of de-spinning an upper stage, the jet Reaction Control System should be adopted and a yo-yo de-spinner should be avoided. Auxiliary propulsion system, such as SOZ engine on the Russian Proton, shall not be separated after use, and should be passivated. All the fasteners should be hitched on their parents. For the parts that can be broken in the middle, if they cannot be hitched at both sides, there should be containers designed to hold them. Separation system should take measures to prevent explosive bolts, extended shaped charges, push rods and springs from ejecting debris at the time of separation. Propulsion system should not use a nozzle closure unless it will be fastened, and solid propellant upper stage engine should not use aft-end throat plug type igniters.

3.2.3 On-orbit break-ups should be avoided, and their effects minimized

3.2.3.1 On-orbit break-ups

Since the beginning of astronautics, on-orbit break-ups have been a major source of debris generation. Some 184 break-ups have occurred so far, involving every space faring nation in the world. The source of a break-up is not always easy to determine:

- In numerous cases, break-ups have been triggered voluntarily. This procedure, highly condemnable when generating long lived debris, is however exclusively associated to payloads, not launchers. It is therefore not detailed here.

- Break-up may occur following a pressure build up in isolated tanks. Typically, at the end of mission, the tanks of an upper stage may be left as they were at engine cut-off with remaining propellant and pressurant. After some time, for instance following the loss of function of external thermal protection or relief valves, the propellants start to boil-off and to vaporize in the tank. When the pressure gets higher than the burst pressure, a major explosion occurs. Some tanks are specifically designed to avoid this, mainly composite pressurization tanks, but these Leak Before Burst (LBB) concepts are not easily applicable to large low pressure propellant tanks. This pressure build up process has happened many times. For instance, one can cite the explosion of the Ariane 1 upper stage H8, mission V16, 9 months after the launch, generating some 767 large fragments of which 11 are still in orbit. It is believed today that the thermal protection of the cryogenic tank was partially torn off after thermal cycling, and that the incoming heat flux vaporized Hydrogen up to a pressure higher than the burst pressure.

- Break-up may also follow an unwanted mixture of hypergolic propellants, either through a leak in common bulkheads or through faulty isolation valves. This was typically the case with the upper stage of the second Chinese LM-4 flight in September 90, long after the launch. The Chinese delegates presented during the 2003 IADC meeting in details how a leak on the common bulkhead led to the mixture of the propellants and the explosion of the stage, and how they have since modified the stage to passivate it. The reader should bear in mind that such “anomalies” are far from being abnormal since they occur generally after mission durations completely out of the qualified regime: if an upper stage is normally supposed to work during 1 hour for instance, it is obvious that it is not going to work for 1 year or more.

- A break-up may also be triggered by electrical batteries: if not properly shorted at the end of nominal mission, there may be an increase of pressure inside the container (Hydrogen for instance) potentially leading to an explosion. Such a mishap has been witnessed at least once.

- Some break-ups may follow hypervelocity collisions, although it is very difficult to determine clearly a cause for such events. When incoming debris or a micro-meteoroid hits a stage, it may trigger a large number of debris when some pressure remains, if the remaining propellant is shock sensitive or else by a cascading effect on a large structure.

- Separation of hardware due to aging effect is generally much less energetic and correspond simply to the no-velocity separation of aged structures, thermal protection, stickers, peeling paint… Some debris may be visible and tracked, but most just corresponds to the aging of an upper stage. These events are registered under “cause unknown” but are very significant generators of large debris.

3.2.3.2 Reduction of potential energy

Solutions aiming at reducing the potential energy are relatively easy to implement.

- The **passivation of a stage** consists in removing all stored energy - chemical, mechanical, electrical - at the end of nominal mission. The removal of propellant can normally be performed through the main engine, keeping it functioning until full depletion. However, this is hardly practical due to asymmetry in mixture ratio and potential stability problems at the end of combustion. It is preferred to add dedicated hardware in order to flush the tanks after end of mission. This procedure is widely applied today. For instance, such a system has been implemented recently on the upper stage of the Indian PSLV launcher. A dedicated branching of the pressurization lines has been added on each tank, with a separate pyrotechnical valve connected to reaction-less vent nozzles[18].

- **Control of the pressure inside the batteries**. Dedicated electrical shorting procedures may be used to ensure that the amount of gaseous Hydrogen produced at the end of life is within the natural leak domain of the cell, or even more simply, relief valves may be added on the casing to guarantee that even in the worst case no explosion can happen.

- **Self-destruction safety pyrotechnics.** It is relatively easy to avoid problems stemming from the presence of pyrotechnical safety devices on board upper stages. First, their initiation chain can generally be mechanically cut at the end of the safety critical mission; then, no unexpected command can trigger the explosion of the stage. Second, even if the safety pyrotechnics could inadvertently be triggered long after the end of mission, by Hyper Velocity Impact (HVI) for instance, their action on empty tanks (no propellant, no pressure) would be very benign. Last, the probability of self triggering in orbit from heat cycles, radiation, or heavy ion is extremely remote. There has not been so far any published case of an on-orbit explosion due to inadvertent triggering of self-destruction pyrotechnics for an upper-stage.

[18] IAC-03-IAA.5.4.09 Space debris mitigation measures in India, Adimurthy, Ganeshan, 2003

3.2.3.3 Constraints linked to passivation of upper stages

Even though it may seem simple to just punch a hole in the propellant and pressurant tanks at end of mission to passivate them, numerous problems have been observed during the development of such systems.

- **Generation of droplets, freezing of the outlets.** The passivation process could possibly turn out to be a very good debris generator: during passivation, the associated pressure drop may cause a drop in temperature leading to the freezing of propellants and the release in orbit of small ice particles. This was, for instance, what was feared during the development of the passivation system of the EPS of Ariane 5, on the MMH side. An experiment aimed at qualifying the system was led by CNES, ESA and Onera, based on Mie-Rayleigh scattering, to characterize the size of the ejecta. The experiments confirmed that droplets were largely sub-micron in size.

- **Contamination of payload.** During passivation, propellant droplets are ejected from the stage at relatively high velocities, presenting a significant risk of contamination for the payload. A thorough computation is therefore needed to characterize precisely the environment around the stage during the passivation process. Once this is known, GNC manoeuvres following satellite separation can be used for minimizing potential contamination. A good example of such a process has been published by ISRO for PSLV during the 21st IADC.

- **Unwanted torque, parasite ΔV.** The propellant and pressurant expelled during the passivation of a stage may have very good propulsive efficiencies and generate a perturbing torque, long after the cut-off of any Attitude Control System. To avoid undesirable perturbations (ΔV and torque) on the stage, it is recommended to design reaction-less passivation nozzles. The example of such a layout, as applied by CNSA on Long-March 4 upper stage, is given on figure 3.14. It has been completely qualified through flight tests twice in 2002.

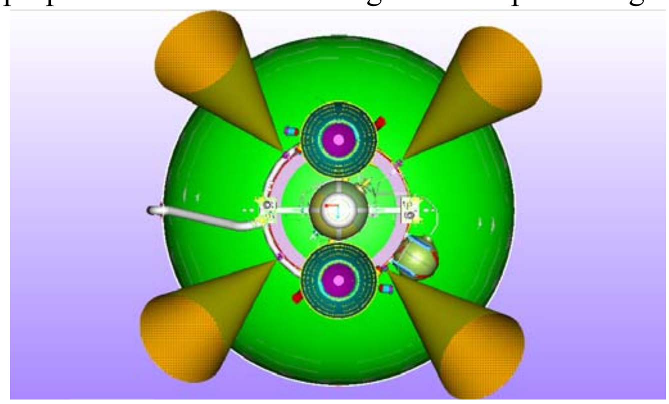


Figure 3.14: layout of the passivation nozzles on the Long-March 4 upper stage

- **Problems associated with the passivation process.** Several other phenomena also have to be taken into account when designing a passivation system.
  - When the propellants are hypergolic, the distance between the exhaust nozzles has to be large enough in order to prevent any risk of mixing and ignition; this is however easy to fulfil due to the very low ambient temperature and jet pressures.
  - There may be some interaction between exhaust plumes and structure, generating an unwanted torque despite ad-hoc nozzles. The proper selection of geometry of the system can reduce such effects.
  - The potential freezing of the propellants during the passivation may clog the passivation nozzles if they are too small. Coatings to prevent icing and proper geometry choices may solve these concerns.

- **Impacts of passivation on operations.** Passivating an upper stage may generate some additional operational work at the end of mission. Since passivation generally happens several minutes after payload release, one has to count on additional power to feed the on-board electrical chains and transducers, additional GNC propellants, increased telemetry window durations with associated operators in the ground stations, and even increased post flight analysis.

- **Decrease of reliability; safety.** The passivation systems on an upper stage may also lead to a decrease in reliability. Use of dedicated pyrotechnic valves raise questions of inadvertent triggering, either on ground or during the propelled phase, or on the opposite non-functioning leading to an explosion. Playing with some sensitive propellants, such as Hydrazine derived propellants for instance, may also be critical, juggling between shock sensitivity and vapour flash explosions risk.

- **Hardware cost.** These systems may turn out to be very expensive. One generally has to add some specific piping, for instance between the tanks and the outer surface of the stage, together with pyrotechnical valves, burst disks (for safety during ground phase), structural reinforcements, dedicated nozzles, …

- **Impact of passivation on global mission costs.** These three last points can turn out to be a significant burden in the highly competitive commercial market. Some operators have deliberately chosen to apply such measures unilaterally, i.e. without any international standard imposed upon them. Now, if they are not applied on an equal basis by all the competitors, it may be necessary to generate some kind of international ruling to check that the competition is fair, proposing for instance to restrict the commercial satellites launches to the “clean” operators.

3.2.3.4 Avoidance of short lived technological solutions

An upper stage is generally meant to operate for one hour or so, very seldom half a day. The rules which are proposed at international level nowadays ask on one hand for a minimization of debris generation, and for a lifetime limitation to 25 years in critical zones on the other hand; this clearly raises the problem of the long term qualification of some hardware widely utilized on upper stages such as thermal protection and paints.

- **Thermal protection may be of very differing types.** Cryogenic tanks’ cold protection is usually made of PVC foam, either glued in complete panels or sprayed on the tanks. It has been shown, in the very early missions of Ariane 1 for instance, that the aging of glued thermal protection panels may lead, after years of thermal cycling, to the release of large pieces of debris in orbit. It was corrected early in the Ariane 4 program, and the ESC-A stage uses a sprayed foam, much more robust to this kind of problem. Some large thermal panels may be used in the vicinity of engines to protect equipment located in the thrust frame for instance. It is important to assess, even very roughly, the durability of such assemblies in time in order to avoid the release of large panels in orbit, through an easily identifiable process.

- In a very similar way, **the paints applied to upper stages** have to show some robustness to aging. It is well known and demonstrated that after several years in orbit, exposed to thermal cycling and radiations, paints tend to peel into small flakes, so small in terms of size, but so deadly in terms of effect to other spacecraft during collisions.

- It is therefore highly recommended to qualify as much as possible all thermal protection or paints to lifetimes as long as possible, even though lifetime of a stage may be measured in minutes. Ideally, their selection should be done considering a robustness criteria, giving priority to the ones less susceptible to release debris after long orbital durations.

3.2.4 Some sensitive orbital regions have to be protected

The international community should work towards the aim of **"no long lived debris generation in protected regions"**. There are currently proposals to consider two protected regions, one in the vicinity of the GEO ring, one in the vicinity of Earth at altitudes lower than 2000 km. It is proposed to limit the lifetime of space objects to 25 years after the end of their nominal mission duration. Practically, it means for the upper stages for which the mission duration is very short, to leave them in orbits with orbital lifetime duration lower than 25 years.

3.2.4.1 Definition of the protected regions

The protected regions have been proposed considering the current spatial density. The justification

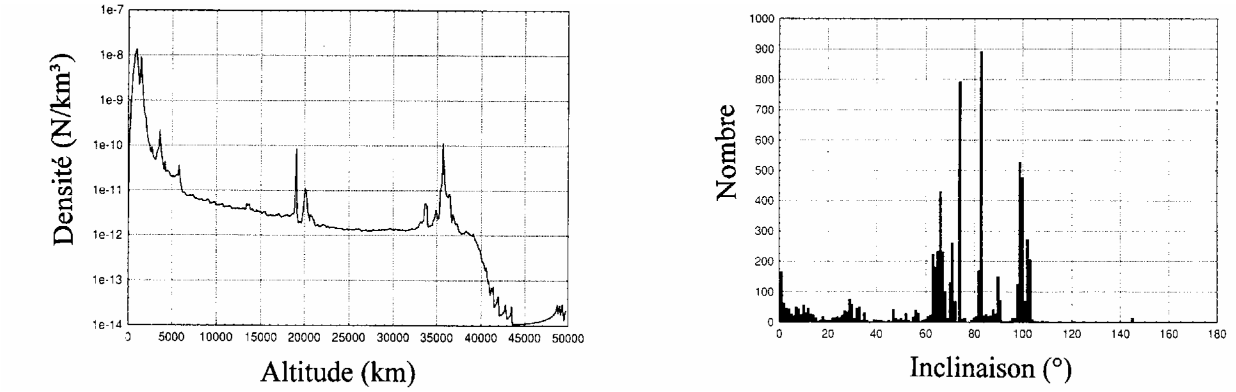


Figure 3.15: Density of orbital debris versus altitude (Left) and inclination (Right)

of these zones can be found in figure 3.15 which presents the density versus altitude and the number versus orbit inclinations[19].

A major concentration of debris can be noted for altitudes lower than 2,000 km with high inclinations; in this region, a few highly dense altitude regimes in which a kind of chain reaction may have started, the cascading effect of debris regeneration through collisions being more important than the decay through atmospheric drag sweeping. The specific case of the GEO ring, even though lower in density, is also considered due to its spatial restriction leading to a significant concentration of spacecraft.

The case of Medium Earth Orbits (MEO), where navigation satellites are located, is currently under study.

3.2.4.2 Orbit lifetime limitation

Choosing 25 years as a limit for orbit duration may not appear very ambitious. However, it was demonstrated through several studies that there is no significant difference on long term orbital debris population between 10, 15 or 20 years, being almost comparable to a direct re-entry strategy, whereas continuing Business As Usual with lifetimes of several hundreds or thousands of years leads to a very fast increase in orbital debris densities in the protected zones. Therefore, this

[19] NSS 1740-14 NASA Space Debris Mitigation Guidelines, October 1995

choice of 25 years is the careful result of years of discussions within IADC and has been finally approved by consensus.

3.2.4.3 Impacts on launchers

- **Orbital lifetimes**

The orbital lifetimes of spacecraft and upper stages, both of which are characterized by relatively large mass-to-area ratios, are determined by orbit perturbations due to atmospheric drag, luni-solar attraction, and solar radiation pressure. Of these perturbations, only atmospheric drag is an energy dissipating force, which reduces the semi-major axis and hence the mean altitude of an orbit. Due to the exponential decrease of air density with altitude, air drag is mainly effective below 500 km. Luni-solar perturbations are mainly effective on eccentric orbits with high apogees (e.g. GTOs). They generate long-periodic variations, primarily of the perigee altitude, with amplitudes which depend on the orbit shape, and on the orbit orientation relative to Sun and Moon. Solar radiation pressure is smaller than air drag below altitudes of about 500 to 600 km (depending on solar activity). Moreover, radiation pressure only generates long-periodic perturbations on the orbit shape, with no net effect on orbit energy, except if the orbit motion has resonances with Earth shadow transits, and/or if periodic perigee variations go along with increased air drag.

- Lifetimes of near circular Low Earth Orbits

Figure 3.16 shows expected orbital lifetimes for near-circular LEO orbits up to 800 km altitude. For the lifetime predictions a constant mean solar activity of $F_{10.7} = 125 \cdot 10^{-22}$ W m^{-2} Hz^{-1}, and a mass-to-area ratio of $m/A = 100$ kg/m^2 was assumed (for a drag coefficient of $c_D = 2.2$). Results are provided as normalized lifetimes ($t_L = t_{L,ref}/(m/A)$ in units of years per kg/m^2). They can be used with sufficient accuracy for standard upper stages, provided that their lifetimes extend over more than one solar cycle. Else, more refined prediction techniques with realistic solar activity forecasts must be applied.

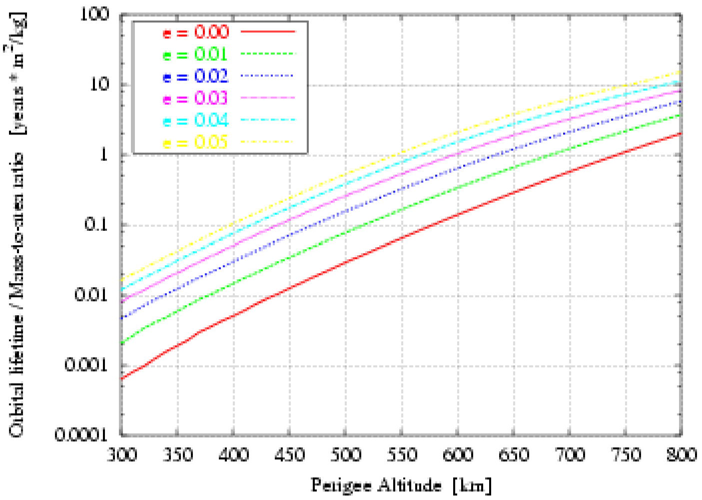


Figure 3.16: Normalized orbital lifetimes of LEO objects (in years per kg/m^2).

- Lifetimes of highly eccentric orbits, including Geostationary Transfer Orbits

In a similar way, expected orbital lifetimes of highly eccentric stages can be determined, knowing their area-to-mass ratio and initial orbital parameters. Figure 3.17 from the NASA draft safety standard displays, for one given area-to-mass ratio, the time spent by objects in the LEO protected region as a function of their initial apogee and perigee.
One should however note that since the duration may be very important, perturbations effects such as solar activity (11 years cycle mainly) and luni-solar influences lead to very unreliable figures. A dedicated computation is highly recommended for each specific case.

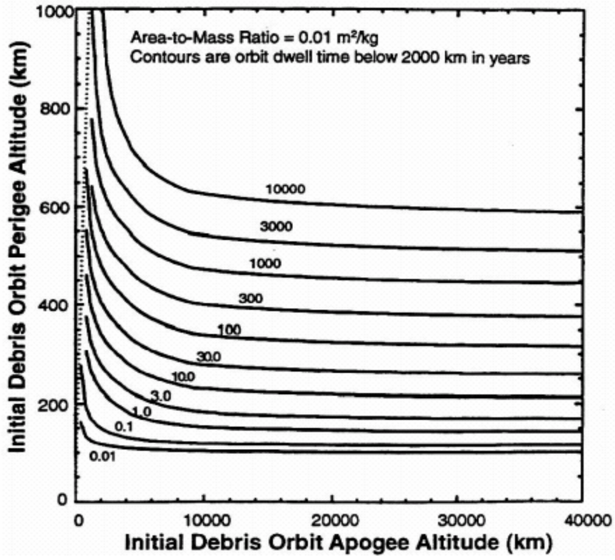


Figure 3.17: Orbital lifetime function of Apogee and Perigee – orbit dwell time below 2000 km in years – Area-to-mass ratio = 0.01 m²/kg

- **Special case of GTO stages**

Stages used to launch satellites into Geostationary orbit can be left in a very wide range of orbits:

- the most used up to now have been the GTO, with an apogee close to GEO altitude and a low altitude perigee, generally in the range of 200 to 300 km, with an orbital lifetime of a few decades,
- however, several modern launchers, such as Zenit Sea-Launch or Ariane 5 in its EPS version, have optimal perigees at much higher altitudes, in the range of 600 to 1,500 km, leading to lifetimes of thousands of years in orbit,
- optimal injection procedures often lead to much higher apogees, in the range of 42,000 km for instance, through super-synchronous strategies,
- the re-ignitability of modern upper stages, such as the H-IIA LE-5B or the Centaur open way to a wide range of other potential orbits, for instance GTO+ missions (acronym for orbits with very high perigees in the range of 10,000 km and any apogee below or above the GEO ring),
- these upper stages may even perform direct GEO injection, circularizing the satellite directly in the final orbit; however, so far, this procedure has been done for some Centaur and Block-DM missions,
- last, very similar concerns can be considered for other highly eccentric missions such as the Molniya ones or the Navigation satellites' transfer orbits.

Two main concerns have then to be dealt with: the protection of the GEO region, and the protection of the LEO region.

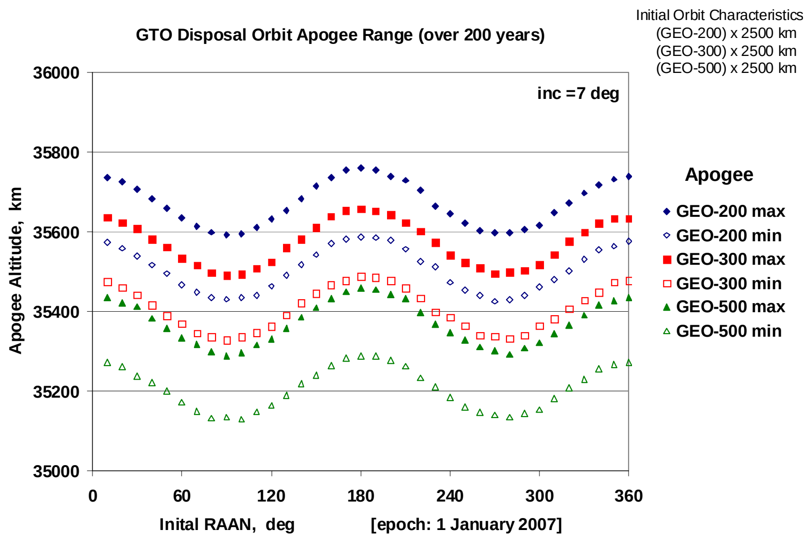
- Protection of the GEO region

Various options can be chosen to protect the GEO region: they all have the same goal which is to remove the stage from the protected region after at most a short delay.

Select apogees below GEO altitude. The most intuitive process is to select an apogee below GEO altitude and to let the spacecraft proceed to GEO. The initial apogee shall be such that taking into account the effective date of launch, long term orbit propagation shows that there is no risk of intrusion within 200 km from GEO. Figure 3.18 from Aerospace[20] gives the typical amplitudes of apogee, for one given inclination, for various perigees, as a function of the initial RAAN. A more complete set of similar figures, for a wide range of altitudes, inclinations and RAAN has been published. It is recommended either to perform a complete calculation for each GTO mission, or to stay well below GEO, for instance at 500 km below. If, in addition, the final perigee is low enough, the apogee decay will be quick; this is by far the preferred option.

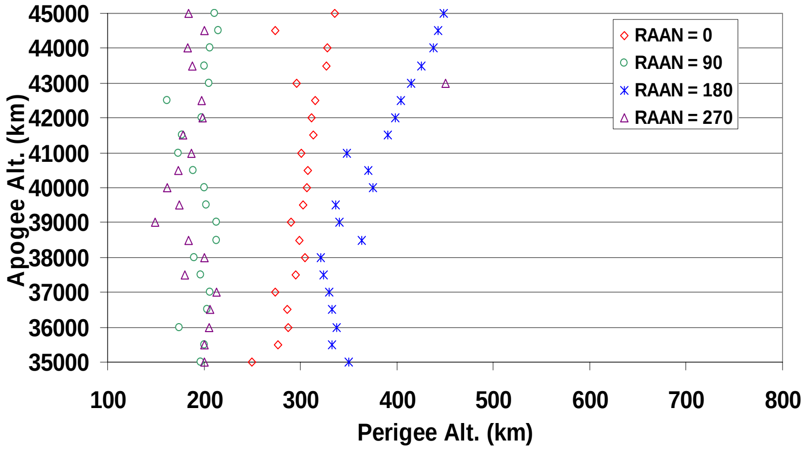
[20] Geosynchronous Transfer Orbit Stability and Disposal Options, 2Oth IADC, April 2002, Campbell, Chao, Gick

Figure 3.18: evolution of the apogee in time, for various initial apogees, versus RAAN – case of an inclination of 7°



Lower the perigee for the super-synchronous missions. For the cases where the final apogee is higher than GEO altitude, the most efficient option is to choose a final perigee low enough to guarantee a quick drift through the GEO region; selecting a final perigee lower than 200 km, for instance, will lead to a re-entry within less than 25 years, so as to have a short stay in the GEO region. The figure 3.19 presented by the NASA delegation during the 21th IADC[21] show how to select a final perigee, as function of the apogee and the RAAN, in order to ensure an atmospheric re-entry within 25 years, fulfilling the protection rules for both GEO and LEO regions.

Figure 3.19: perigee altitude enabling the fulfilment of the 25 years rule in LEO, as function of Apogee altitude and RAAN



Select a disposal orbit. In some specific cases where the final mission perigee is very high, it may not be reasonable economically to lower it significantly. It may then be preferable to lower the apogee well below the GEO arc and to choose a perigee high enough to protect the LEO region. Final disposal orbit is acceptable if perigee is higher than 2,500 km and apogee lower than GEO-500 km. However, such a procedure should only be chosen in the few cases where a low perigee is not selectable.

Upper stages in Geostationary orbit. Upper stages injecting directly their satellites should follow the same rules as the GEO satellites at end of life. It is recommended to raise the orbit following the IADC rule, e.g. by some 300 km or so as a minimum, and to leave the stage in a nearly circular orbit; the final eccentricity should be lower than 0.005. Should the mission impose a final orbit below GEO instead of above, similar values should be applied. One should of course avoid leaving secondary structures, in case of multiple launch, in the GEO region; should such a mission be planned, these structures should be left in circular orbits at least 500 km from the GEO arc.

[21] Debris mitigation considerations for GTO with Supersynchronous Apogees, 21st IADC, March 2003, Campbell et al

- Protection of the LEO region from the GTO upper stages and structures. Protecting the LEO region means leaving upper stages and secondary structures on an orbit with a lifetime lower than 25 years.

  Direct atmospheric re-entry. The most obvious procedure is to perform a direct re-entry of the upper stage at the end of its mission. It is a complex process since the necessary ΔV may be relatively high, depending on the anomaly of the stage at the time of the de-orbiting burn, and since it may raise some safety concerns linked to the debris dispersion zone on land-masses.

  Lower the perigee after satellite release. The lowering of the perigee to a value such that the final lifetime is less than 25 years is a very acceptable procedure, should the direct de-orbitation not be selected. The corresponding options and numerical values are addressed in Section 3, relative to the limitation of operational debris. The final perigee selection may be complex and require detailed specific computations, since it is highly dependent on RAAN and argument of perigee. A example of such variation is given in figure 3.20 produced by Aerospace in the frame of IADC[22]: it gives the maximal allowable perigee altitude, apogee being at GEO altitude, for a 25° inclination, versus the value of RAAN, for two values of argument of perigee. As a rule of the thumb, it is always acceptable to select a perigee lower than 200 km, and never acceptable to select it higher than 300 km.

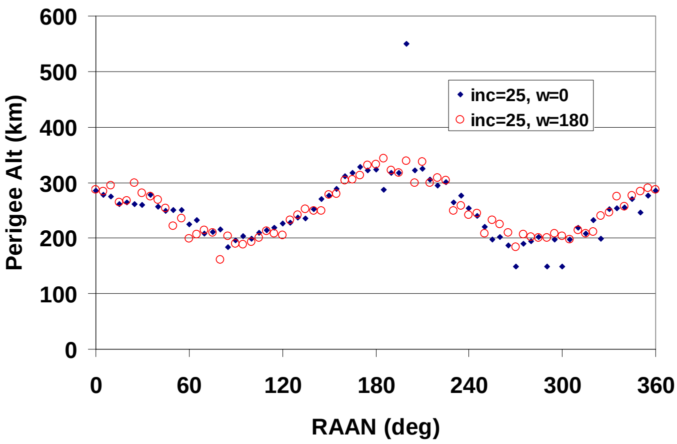


Figure 3.20 : maximal acceptable perigee to fulfill the 25 year rule protecting the LEO region, as a function of RAAN and Argument of Perigee – Inclination 25°

  Select a disposal orbit. As described in the paragraph 3.3.4.1.3., a disposal orbit with a perigee higher than 2,500 km may be selected, although it is not recommended. The margin of 500 km should be confirmed depending on the specifics of the mission in order to guarantee an efficient long term protection of the LEO region.

3.2.4.4 Protection of Semi-Synchronous Orbits, GPS-Glonass-Galileo orbits

The injection of navigation satellites can be performed following two kinds of procedures:

- In a GTO-like process, the upper stage and associated upper structures may be left in a transfer orbit, with the apogee at the final satellite altitude (or slightly below to avoid interfering with the useful zone) and a perigee selected at a low altitude. The corresponding rules are very similar to those applied for GTO, and the recommendations described in the previous chapter are applicable.
- In a GEO-like process, the upper stage is left in a circular orbit close to the final one. This situation, unfortunately, may not be acceptable at long term: whatever the final orbit selected in the vicinity of Navigation satellites, it is instable in time if not perfectly

[22] Debris mitigation consideration for GTO with Supersynchronous Apogees, 21st IADC, March 2003, Campbell et al

circular. Excursion of the apogee may reach several thousands of kilometres, and the abandoned upper stage may even cross GEO one day.

- **Long term evolution**

The following figure 3.21 from NASA-JSC[23] is one of many showing the long term evolution of a representative GPS disposal orbit.

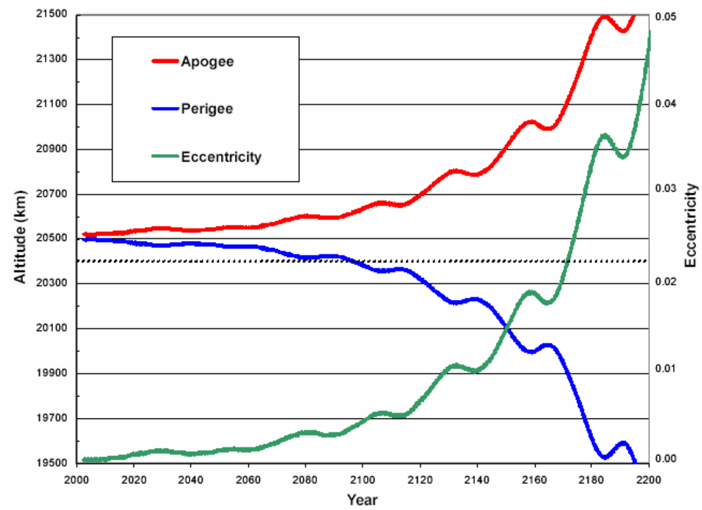


Figure 3.21: typical long-term evolution of altitude (apogee, perigee and eccentricity) for a GPS spacecraft

Numerous studies have been performed in that sense in the US, Russia, Japan and Europe; all of them concur in finding that the situation may become critical if nothing is done:

- the collision velocities are comparable to those encountered in LEO, 5 km/s in average; there is not the relative smoothness observed in GEO where all spacecraft are moving roughly in the same direction with roughly the same speed,
- as in GEO, there is no “cleansing” effect of atmosphere; the natural decay observed in LEO does not exist in such altitude regimes.
- It means that collisions or ruptures in this band will be catastrophic: the secondary debris proliferation will be important due to the energies in place, and will remain stable for thousands of years, offering ever renewed risk of new collisions.

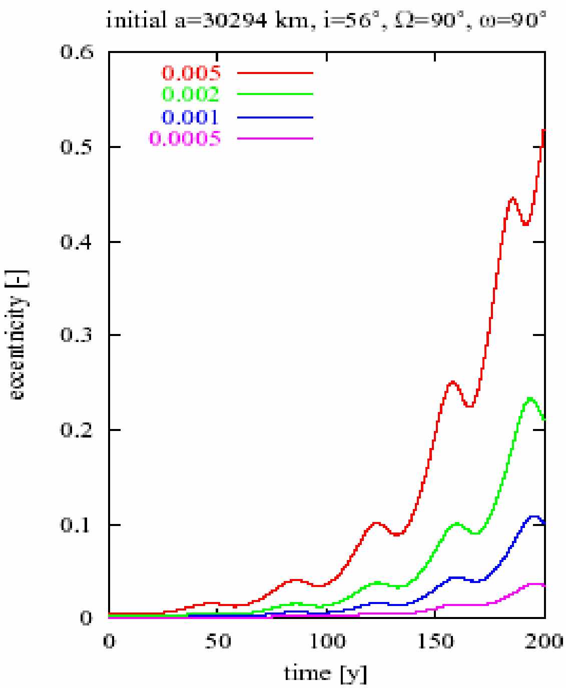


Figure 3.22: typical long-term eccentricity evolution for a Galileo spacecraft

The example shown in figure 3.22, extracted from an ESA simulation presented during the 22nd IADC meeting, corresponds to a long term evolution of a Galileo satellite as a function of its eccentricity at end of life.

This apparently critical situation may however never become a problem: densities in this region remain small, at least two orders of magnitude less than in the LEO region. Furthermore, the various satellite constellations considered in this zone do not have the same inclinations. The adequate risk criteria is therefore the probability of collision risk. It is recommended to examine carefully the situation in the navigation satellites zone to evaluate whether the long term evolution may lead to a severe collision risk.

[23] Opiela et al., NASA, Orbital instability in the GPS disposal orbits, 21st IADC, March 2003

- **Potential improvements**

Unfortunately, there are only very few improvement possibilities:

- Direct re-entry is an extremely expensive solution requiring some 1,500 m/s and is associated with significant operational problems to control the impact point on ground. Since there would be associated safety requirements linked to potential risk to over flown populations, the de-orbiting burn has to present an extremely high reliability, probability of success of the manoeuvre being in the range of 99.99 %. The corresponding amount of propellant for a typical modern cryogenic upper stage would be in the range of 1.5 tons of propellant,

- Reducing drastically the altitude of perigee in order to re-enter within 25 years is also a very costly solution, which can only be considered when significant performance margins are available. The required ΔV, 1,400 m/s, is almost as high as for a direct re-entry, but the precision required for the manoeuvre does not need to be as great: a probability of success of 99 %, for instance, may be enough for the mission planners to consider using the statistical performance reserve to do the de-orbiting burn. The corresponding dedicated propellant mass then becomes limited to 600 or 700 kg for a modern cryogenic stage, but more importantly the associated operations are much less stringent. It may be noted that this kind of manoeuvre is nominally considered for the upper stage of Ariane 5 in the Galileo missions planned before 2010.

- The last solution consists in finding a disposal zone for the upper stage in the vicinity of the operational zone; this solution is similar in principle to what can be considered for the navigation satellites themselves. During the 22nd IADC, the ASI delegation recommended to consider a disposal zone for the upper stages starting 500 km above the highest constellation foreseen today, Galileo. This measure shall nevertheless be accompanied by recommendations about final orbital elements in order to minimize the dispersion in time. Several studies have shown that the driving parameter for long term stability can be expressed as a function of the sine of an angle of twice the argument of perigee plus right ascension of the ascending node ($2\omega + \Omega$). As an example, recommendations for GPS disposal were that the orbit should be raised by at least 500 km with eccentricity no greater than 0.005 and initial argument of perigee inside the windows determined for each of the six GPS planes. The corresponding estimated ΔV is 50 to 70 m/sec. Similar evaluations have been performed for the Galileo satellites confirming the instability of final orbits and the influence of the term ($2\omega + \Omega$).

Part 4

Recommendations

Section 4.1
GENERAL RECOMMENDATIONS

⇨ **This study recommends a goal of "zero debris creation within the protected zones"** and looks at space debris mitigation guidelines from the perspective of effectiveness in reducing the overall risk and the costs associated with implementation.

There are a number of actions that can be taken to reduce the growth of the debris environment which appear to be cost effective.
⇨ **For operating satellites, upper stages and operational launch structures , it is recommended that operators:**

- **Safe the orbital objects at the end of mission.** This action will help assure that only one large, trackable object will be a threat, rather than debris from an explosion.
- **Dispose of operating spacecraft or upper stage by de-orbiting or moving to a disposal orbit.**
- **Avoid collisions with other objects by coordinating moves with other operators who might be affected and, if available, utilizing satellite collision avoidance services.**

Each of these options is seen as a viable means of limiting the growth of debris, while requiring some additional costs. Costs associated with changes of mass and/or performance must be taken into account.

The collision avoidance option requires accurate tracking data and utilization of that data to identify objects approaching the spacecraft.
⇨ **It is recommended that options for providing accurate tracking data be explored and, if warranted, implemented.**

For new spacecraft and orbital stages, there are a number of recommendations for mitigating debris. These include design of both spacecraft and missions to cover safing, disposal, and collision avoidance options, utilizing designs which don't produce non-trackable debris such as solid propellant slag and paint flakes, and assuring reliability of the disposal systems. The cost/benefit of these options has not been addressed in detail.

The de-orbiting option may add large cost and complexity to some spacecraft and stages. Waivers may be used for such systems when it is shown that other measures have been taken to limit debris. Continued development of systems such as tethers that can speed the de-orbit process is encouraged.
⇨ **It is recommended that very small satellites such as picosats, too small to be tracked and not currently capable of de-orbiting, be encouraged to use orbits whose lifetime is less than 25 years.**

While there has been considerable work on modelling the future debris environment and predicting the number of collisions given various mitigation scenarios, there has been relatively little work to assess the overall economic impact and cost effectiveness of proposed space debris mitigation policies and approaches.
⇨ **Future work in the area of modelling and prediction is recommended.**

Section 4.2

<u>SHORT TERM ACTIONS</u>

4.2.1 Know what is written

- The highest level rules are the IADC guidelines. They are short and present simply the main principles to apply: avoid operational debris, prevent explosions and protect the two sensitive regions with the GEO re-orbiting rule and the 25 year rule for LEO.

- These guidelines have been presented to UNCOPUOS and are currently under review. Once this phase is passed, the corresponding document will have to be widely distributed internationally, explained and justified.

- Several standards are existing or currently being drafted within several organizations, often derived from the IADC guidelines. Some are already applicable, some will become so as soon as an international standard, ISO type for instance, will be approved, and some only have a Code-of-conduct status, aimed at giving recommendations but not requirements. These documents present the fundamental advantage of being practically applicable to any designer or operator. Therefore, even if they are still under draft form, they should be widely distributed and explained.

⇨ **It is recommended to organize workshops with all the concerned entities, mainly the operators, to describe and explain the existing rules**. This is very well done within agencies, but has to be spread to the private and commercial world.

- The general recommendation "zero debris creation within the protected zones" may not be practically implementable before a long time; an interim short term recommendation may be promoted, recognizing the importance of orbital lifetime in the global collision risk:

⇨ **It is recommended to promote internationally a call for "<u>no long lived orbital debris in protected regions</u>"**. Such a simple and clear motto can enhance a worldwide awareness in the space community.

4.2.2 Do what is written

Several rules approved today were drafted almost 10 years ago, but are still not applied. The difference between best effort and effective application of the rules is obvious and has to be corrected. In 2003, only 40% of the decommissioned satellites in GEO did respect the IADC rule which was drafted in 1994.

⇨ **It is recommended to control the effective application of IADC guidelines.**

- To that extent, IAA should publish and distribute widely on a regular basis a figure of merit concerning the main IADC guidelines.

- Comparisons between operators should be established, enabling a fair competition with everyone playing under the same rules. Respecting the high level debris mitigation rules shall become a discriminating criteria for choosing between operators.

- Any new launcher or spacecraft family shall be examined by an ad-hoc committee formed of IAA experts to evaluate the appropriateness of the selected options: lack of operational debris, passivation systems, respect for the 25 year rule.

- These rules shall become systematically applied to new programs, and it is recommended to modify the existing ones to improve the situation.

⇨ It is recommended to increase the awareness of all IAA members to the increasingly problematic situation and to report on a yearly basis in an independent forum such as IAA figures of merit for each of the high level guidelines.

4.2.3 Promote work on what is missing

If not protected, the navigation satellites zone (GPS-Glonass-Galileo) could become very heavily polluted after the first accidental break-up. There is no atmospheric cleansing process in this altitude range, these orbits are unstable, and there are several new projects aimed at this very sensitive zone. Nevertheless, there has not yet been a correct evaluation of the potential collision risk and effects in the long term.

⇨ It is therefore recommended to promote at the IADC or the IAA level the study of the potential evolution of this critical zone over time, taking into account spacecraft and upper stages.

⇨ It is recommended to ask all manufacturers to characterize their hardware as to aging effects. A typical launch operation lasts less than one hour, but IADC guidelines allow to have orbital lifetimes up to 25 years.

- The behaviour of thermal protection, paints, stickers, … shall be characterized for these long durations.

- It is recommended to have an exchange on such R&D experiments at an international level in order to improve globally the orbital debris situation.

GLOSSARY

ABM	Apogee Boost Motor, used to insert spacecraft into the geostationary orbit.
AIAA	American Institute of Aeronautics and Astronautics
AOCS	Attitude and Orbital Control System
Apogee	Point in an orbit that is furthest from the Earth
ASI	Agenzia Spaziale Italiana (Italian Space Agency)
BNSC	British National Space Centre
CCIR	Consultative Committee on Radiocommunication
CNES	Centre National d'Etudes Spatiales (French Space Agency)
CNSA	China National Space Administration
COSPAR	Committee on Space Research
CSG	Centre Spatial Guyannais (Guiana Space Centre)
De-orbit	Forced entry into the Earth's atmosphere; includes also the case where the entry into the Earth's atmosphere occurs with some delay
DLR	Deutsches Zentrum fuer Luft-und Raumfahrt (German Aerospace Centre)
ΔV	Velocity increment; equals the time integral of acceleration during the functioning of an engine
ECSS	European Cooperation for Space Standardization
EOL	End Of Life
ESA	European Space Agency
FCC	Federal Communications Commission
GEO	Geostationary Earth Orbit
GNC	Guidance Navigation and Control
GPS	Global Positioning System
GTO	Geostationary Transfer Orbit
HEO	High Earth Orbit
HVI	Hyper Velocity Impact
IAA	International Academy of Astronautics
IADC	Inter-Agency Space Debris Coordination Committee
IAF	International Astronautical Federation
ISO	International Organization for Standardization
Isp	Specific Impulse
ISS	International Space Station
ISRO	Indian Space Research Organisation
ITU	International Telecommunication Union. A specialized organization of the UN
JAXA	Japan Aerospace Exploration Agency
LBB	Leak Before Burst
LEO	Low-Earth Orbit. Orbital region below 2000 km altitude
MEO	Medium Earth Orbit close to 20000 km altitude
MMH	Mono-Methyl Hydrazine
NASA	National Aeronautics and Space Administration
NASDA	National Space Development Agency of Japan, now JAXA
NORAD	North American Air Defense Command
NPV	Net Present Value
NSAU	National Space Agency of the Ukraine
PDS	Passive Deceleration System
Perigee	Point in an orbit that is closest to the Earth

RAAN	Right Ascension of Ascending Node. Angular position of the ascending leg of an orbital track in the equatorial plane
Removal	To remove from orbit
Reorbit	Transfer to another orbit
RF	Radio Frequency
ROI	Return Over Investment
ROSKOSMOS	Russian Space Agency
R&D	Research and Development
SRM	Solid Rocket Motor
SSO	Sun-Synchronous Orbit
TM	Tele-Metry
TSS	Tethered Satellite System
UN	United Nations
UNCOPUOS	United Nations Committee on the Peaceful Uses of Outer Space

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