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Innovative Solutions for Satellite Conformity to Space Debris Mitigation

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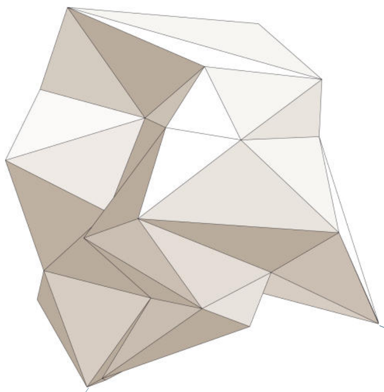

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Abstract

This thesis presents the work I have accomplished during my 6 months internship in Altran Research more precisely in the Space Innovation Unit in Cannes, France. This period as trainee was also the conclusion of the double degree program I followed in Aerospace Engineering at KTH, the Royal Institute of Technology of Stockholm, Sweden.

This report details every Space Safety related projects in which I have been involved. Every topic is related to the management of low earth orbit satellites disintegration during their atmospheric re-entry.

Nowadays orbital pollution has pushed national space agencies to take the lead on space debris mitigation. There are currently more than twenty thousand objects of more than 10 cm constantly tracked from ground to avoid collision with in-progress missions. This is implying expensive avoidance manoeuvres thus equipment and budget associated. Items shorter than 10 cm are even more numerous and they cannot be seen from ground so they are estimate by models. The debris population is threatening future missions and even launches if nothing is done to prevent/reduce the debris formation.

To avoid this catastrophic scenario, space agencies have developed and financed projects to prevent and reduce debris creation. In the meantime, risk on ground must be reduced to limit population injuries from falling object. Now satellites are designed/retro-designed to demise more, and in known ways, during uncontrolled re-entry. Software are also currently developed to simulate more precisely the complex aerothermal phenomenon of ablation during atmospheric re-entry.

Sammanfattning

Detta examensarbete presenterar mitt arbete under en sex månader lång praktik på Altran Research och deras Space Innovation Unit i Cannes, Frankrike. Rapporten beskriver de rymdsäkerhetsprojekt jag arbetade med, alla relaterade till utformning av satelliter i låg jordbana för effektivare sönderfall vid deras atmosfärsåterinträde.

Nedskräpning av jordbanor har tvingat nationella rymdstyrelser att ta initiativ för att lindra rymdskrotsproblematiken. Idag spåras banorna för fler än tjugo tusen objekt med en storlek större än 10 cm för att förhindra kollisioner med aktiva satelliter. Detta betyder att dyra undanmanövrar måste göras, vilket påverkar satelliternas utrustning och rymduppdragens budget. Objekt mindre än 10 cm är ännu fler till antalet och de kan inte spåras från jorden. Istället används modeller för att uppskatta deras antal. Rymdskrotspopulationen hotar framtida rymduppdrag och uppskjutningar om ingenting görs för att förhindra eller reducera tillväxen av rymdskrot.

För att undvika detta katastrofscenario så har rymdstyrelser utvecklat och finansierat projekt för att förhindra och reducera att nytt rymdskrot skapas. Samtidigt måste risken för att rymdskrot faller ned på marken reduceras för att begränsa skador på människor och egendom. Nuförtiden utformas satelliter för att falla isär och helt och hållet förbrännas vid ett okontrollerat återinträde i atmosfären. Mjukvaror håller på att utvecklas för att mer precist kunna simulera det komplex aerotermiska processer och ablationen av material vid atmosfäriskt återinträde.

Résumé

Cette thèse présente le travail réalisé pendant mon stage de 6 mois à Altran Research, plus précisément dans l'unité chargée des innovations aérospatiales à Cannes en France. Cette période en tant que stagiaire fût également la conclusion de mon parcours bi-diplômant en Ingénierie Aérospatiale à KTH, Institut Royal de Technologie de Stockholm en Suède.

Ce rapport relate les projets d'ingénierie en sécurité aérospatiale dans lesquels j'ai été impliqué. Chaque sujet est lié au management de la fin de vie des satellites en orbite basse lors de la rentrée atmosphérique.

De nos jours la pollution orbitale a poussé les agences aérospatiales nationales à prendre les devants sur la problématique de la réduction de la population de débris spatiaux. Il y a actuellement plus de 20 000 objets de plus de 10 cm, constamment suivis depuis le sol pour éviter les collisions avec les missions en cours. Cela implique donc des manœuvres d'évitement couteuses en conséquence de l'équipement et des budgets associés. Les déchets de moins de 10 cm sont encore plus nombreux et ne sont pas visible depuis le sol, ils sont donc représentés par des modèles. Les débris menace les futures missions et même les prochains lancements si rien n'est fait pour prévenir/réduire la génération de débris.

Pour éviter ce scénario catastrophique, les agences spatiales ont développé et financé des projets pour prévenir et réduire la création de débris. En même temps, le risque au sol doit être réduit pour éviter que des personnes soient blessées par des objets retombant. Désormais les satellites sont conçus pour se désintégrer, de manière connue, durant les rentrées non-contrôlées. Les logiciels sont actuellement développés pour simuler plus précisément les phénomènes aérothermiques complexes d'ablation pendant les rentrées atmosphériques.

Acknowledgment

I would like to express my gracefulness to the Cannes' Office and especially my company supervisor, M. HEINRICH Stephane, who gave me this opportunity to do my master degree placement. Thanks to him I had the chance to learn more about satellite design from the demise point of view. It also has been an important experience for me to discover the work of space engineer as consultant working on different research project to win development study case with other firm.

I am also grateful to have the opportunity to take part in many space safety projects involving major industrial contributors such as Thales Alenia Space, Airbus, Rockwell Collins Deutschland and the European Space Agency.

Finally I would like to express my thankfulness to the Aerospace Engineering teaching staff at KTH, especially my examiner M. TIBERT, who taught me a lot during my double degree program without whom I could not done this instructive internship.

Summary

Figure table

Glossary

AOCS	Attitude Orbit Control System			
D4D	Design for demise			
ESA	European Space Agency			
ESTEC	European Space Research and Technology Centre			
NASA	National Aeronautics and Space Administration			
ITT	Invitation To Tender			
LEO	Low Earth Orbit:	Orbit altitude below 2,000 km		
GEO	Geostationary Earth Orbit:	Orbit around 35,786 km		
HEO	High Earth Orbit:	Orbit entirely above 35,786 km		
MEO	Medium Earth Orbit:	Orbit between 2,000 and 35,786km		
R	Perfect gas constant			
$M_\oplus$	mass or Earth:		5.9722×10^{24}	kg
$R_\oplus$	radius of Earth:		6.378135×10^{6}	km
G	gravitational constant:		6.67408×10^{-11}	$m^3/kg/s^2$
$\mu_\oplus$	gravitational parameter of Earth:		3.98601×10^{14}	m^3/s^2
$\mu_\odot$	gravitational parameter of Sun:		1.32715×10^{20}	m^3/s^2
AU	Astronomical Unit		1.49599×10^{11}	m
S/C	spacecraft			
SDM	Space Debris Mitigation			
SMA	Shape Memory Alloys			
SME	Single Memory Effect			
TRL	Technology Readiness Level	From 1 (research) to 9 (commonly used)		
TWME	Two-Way Memory Effect			

Terms and Definitions

For the purposes of understanding, the terms and definitions given in European Standard ECSS-P-001B, *Glossary of terms* [RD1], and the following definitions apply.

Casualty risk : The probability of serious injury to or death of a single person due to the re-entry of a space system.

Damage: Loss of human life, personal injury or other health impairments, occupational illness, total or partial loss of property, or deterioration caused to the aforesaid property or to environment.

De-orbiting: Deliberate or forced re-entry of a space system into the Earth's atmosphere by applying a retarding force, usually via a propulsion system.

Direct re-entry: The space system performs the manoeuvres to complete its re-entry phase within a single orbit revolution.

Disposal orbit *(Synonym: Graveyard orbit)****:*** Earth orbit remaining outside the protected outer space regions even under the influence of perturbations.

Disposal phase: Begins at the end of the operational phase of a space system, and ends when either the space system has performed a direct re-entry or completed activities to enable it to reach its disposal orbit and has been passivated.

End-of-life: End of the disposal phase.

End of mission: Completion of the scheduled mission of the space system, or mission stop due to a space system failure or to a voluntary decision.

Geostationary Orbit: Earth orbit having zero inclination and zero eccentricity, whose orbital period is equal to the Earth's sidereal period; the altitude of this unique circular orbit is close to 35,786 km.

Geostationary Transfer Orbit: An Earth orbit which is or can be used to transfer space systems from lower orbits to the geosynchronous region; such orbits typically have perigees within LEO region and apogees near or above GEO.

Geosynchronous region (GEO region): The space region close to and including the Earth orbit with an orbital period equal to one sidereal day.

Hazardous event: An unplanned event or series of events resulting in damage or potential for damage.

Launch phase: Begins when the launch vehicle is no longer in physical contact with equipment and ground installations that made its preparation and ignition possible (or when the launch vehicle is dropped from the carrier-aircraft, if any), and continues up to the end of the mission assigned to the launch vehicle.
The launch phase ends when the launch vehicle has achieved an Earth orbit or an interplanetary trajectory or, if not, when it is in physical contact with the ground again.

Launch vehicle: See "Space system".

Launching state *(UN Liability Convention definition)*
(i) A State which launches or procures the launching of a space object;
(ii) A State from whose territory or facility a space object is launched.

Low Earth Orbit: Orbit with apogee altitude lower than 2,000 km.

Low Earth Orbit region (LEO region): The spherical region that extends from the Earth's surface up to an altitude of 2,000 km.

Operational phase: Part of the orbital phase of a space system starting as the orbital phase and ceasing at the end of mission.

Operator: Any citizen of a nation in which he/she is performing space activities, or any organisation existing under the laws of a nation or under an international governmental agreement in order to perform space activities or any worker of such an organisation when he/she is duly authorised by that organisation.

Orbital lifetime: The length of time a space object remains in Earth orbit.

Orbital phase: Starts when a space system is in orbit separated from the launch vehicle and ends when it re-enters the Earth's atmosphere.

Passivation: The elimination of all stored energy on a space system to reduce the chance of break-up. Typical passivation measures include venting or burning excess propellant, discharging batteries and relieving pressure vessels.

Prevention measure: Any measure which decreases the potential for generating space debris, or reduces the associated risk.

Protection measure: Any measure reducing the effective damage caused by space debris.

Re-entry phase: Begins when the space object comes into the Earth's atmosphere, and continues until either the intact object or its surviving parts come to rest on the Earth's surface or when the object and all of its parts have disintegrated.

Re-orbiting: Intentional changing of a space system's orbit.

Safety: All the arrangements intended to control safety risks stemming from activities contributing to the flight of a manned or unmanned space system, in order to ensure the protection of people, public and private property, and the environment, against any damage caused by such activities.

Space debris *(Synonym: orbital debris, debris)*
Any man made space object including fragments and elements thereof, in Earth orbit or re-entering the Earth's atmosphere, that is non-functional.

Space object: Any man-made space system and any of its components or fragments.

Space system: Spacecraft, launch vehicle, and launch vehicle orbital stage are defined as space systems within this document.
Spacecraft: an orbiting object designed to perform a specific function or mission (e.g. communications, navigation or Earth observation). A spacecraft that can no longer fulfil its intended mission is considered non-functional. (Spacecraft in reserve or standby modes awaiting possible reactivation are considered functional.)
Launch vehicle: any vehicle constructed for ascent to outer space, and for placing one or more objects in outer space, and any suborbital rocket.
Launch vehicle orbital stage: any stage of a launch vehicle left in Earth orbit.

I. The company: Altran

1. History/Creation

Engineer consulting firm Altran was created in 1982 based on two principles: use innovation as factor of differentiation for companies and have capacity to mobilize the best people as condition of success.

Altran takes the lead in its industrial sector by creating the job of engineering consultant. Native country of the company, France represent today around 50% of the global group's activities. They are divided in two branches:

- High technologies and innovations,

- Organizational consulting and information systems.

Through the pass thirty years, Altran has anticipated the increasing need of technologic innovations in key activities. The strength of this consulting firm is the multidisciplinary which permit to conduct a project entirely. Altran is also helping other companies by providing training to increase their potential of success.

In 2016, Altran is composed of more than 26,000 collaborators distributed in more than 20 countries working for around 500 major clients.

Figure 1 : Altran is present all around the world
(Credit: Altran)

2. Organization

Altran's offices are distributed in six regions in France and the group also has offices in 22 countries. It is working and giving consulting expertise in the following industries:

- Aerospace, defence and railway,
- Energy, industry and sciences of living,
- Automotive, infrastructure and locomotion,
- Telecom and medias,
- Finance and public services.

Usually each office has a specialty depending on industrial partners present in the region.

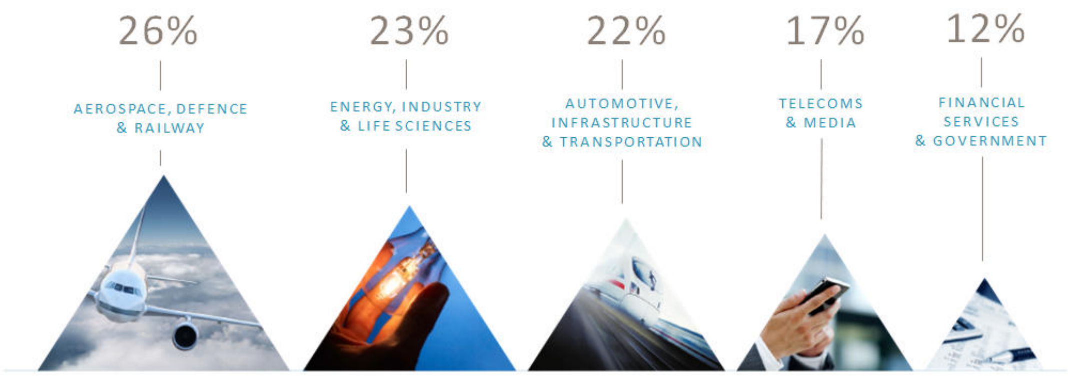


Figure 2: Altran's major field of competences
(Credit: Altran)

3. Cannes office description:

Based in the Space Camp of Cannes close to its collaborator Thales Alenia Space, Altran Cannes Office is composed of 40 engineers specialized in aerospace and working on the following thematic/areas:

- Mechanical design and analyses,
- Thermal and mechanical architecture,
- Energy and fluid calculation,
- Qualification and certification,
- Prototyping and industrialization.

Presented results and report have been made in the Space Safety unit of Altran Research based at Cannes.

II. Space debris problematic

1. Origins

50 years of space activity

Since 1957, more than 5,000 launches have placed around 7,000 satellites into Earth's orbit with more than the half remains in space. A thousand of these remaining satellites are still operational today. This population of dead satellite represent a total mass of more than 6,300 tonnes. These space objects are not intact and have been spread into thousands of space debris. Nowadays the US Space Surveillance Network is following more than 23,000 objects (in September 2012), from 5 cm in size, in order to keep up to date a catalogue of space debris and their attitudes.
[Reference Document 1: http://www.esa.int/Our_Activities/Operations/Space_Debris/About_space_debris, 04/2013]

Objects in orbit include spent upper stages

One third of the space objects orbiting around Earth are composed of missioned satellites, upper stages of rockets and mission related objects. On other hand, only six percent of space objects are operational spacecraft. Since 1961, more than 250 in-orbit fragmentation, collision and/or burst of object, have been recorded. Explosions are more frequent than collisions in orbit. [RD2]

Figure 3: Source of space debris include explosions of rocket bodies (Source: Rocket body explosions, Copyright ESA, www.esa.int)

Explosions of satellites and rocket bodies

Bursts have generated an enormous population of debris smaller than 1 cm. Objects of size under 1 mm may mainly come from meteoroids events. Most of the time explosions of decommissioned satellites are due to residual fuel that remains in tank when the spacecraft is disposed into orbit. Over time the space environmental conditions can wear parts and create weakness leading to leak which could trigger self-ignition. Explosion of a tank can spread debris of several size and mass at different velocities and on different attitude. [RD2]

Anti-satellite test: 25% more debris

Nevertheless, unexpected debris creation is not the only source of uncontrolled space object in orbit. Tactical destruction of satellite by surface-launched missile can also be a major contributor according on recent past event. US government has destructed for the first time a satellite; mission P78-1 (Solwind), thanks to a missile launched by a chaser in 1985. This intervention created 285 debris, which deorbited in 20 years. The most known example is the Chinese demonstration/test of surface-launched missile to destroy the satellite Feng-Yun 1C. It happened in January 2007 and created more than 2,000 of debris bigger than 1 cm corresponding to an increase of 25% of trackable space object population. Those new debris are situated a relatively high orbit for LEO mission which means that they will remains during decade even century (for small one) according on US calculations based on solar flux. [RD2]

Other sources of debris fragments

Solid motor rockets have also been an important source of debris in the form of µm-sized dust/particle due to aluminium oxide (Al_2O_3) contained into ergol fuel. Another source is the ejection of reactor fluid from nuclear soviet satellite (RORSAT) which released numerous droplet of reactor cooling fluid composed of low-melting sodium potassium alloy.

Atomic oxygen present at the edge of Earth atmosphere are also creating debris by eroding/abrading external panels which can leads to loss of surface coating and detachment of paint flakes with mm sizes. [RD2]

First-ever in-orbit collision

In February 2009, the first accidental collision happened between an American communication satellite Iridium 33 and an out of mission military satellite, Kosmos-2251. This event created more than 2,200 trackable fragments.

Distribution of catalogued objects in space - global view

Due to Earth atmospheric drag, solar pressure and luni-solar gravitational attraction, there is a kind of cleansings mechanism which led to spatial distribution of debris concentration. Typically, the maximum debris density is located at altitude from 800 to 1,000 km, and also near 1,400km.

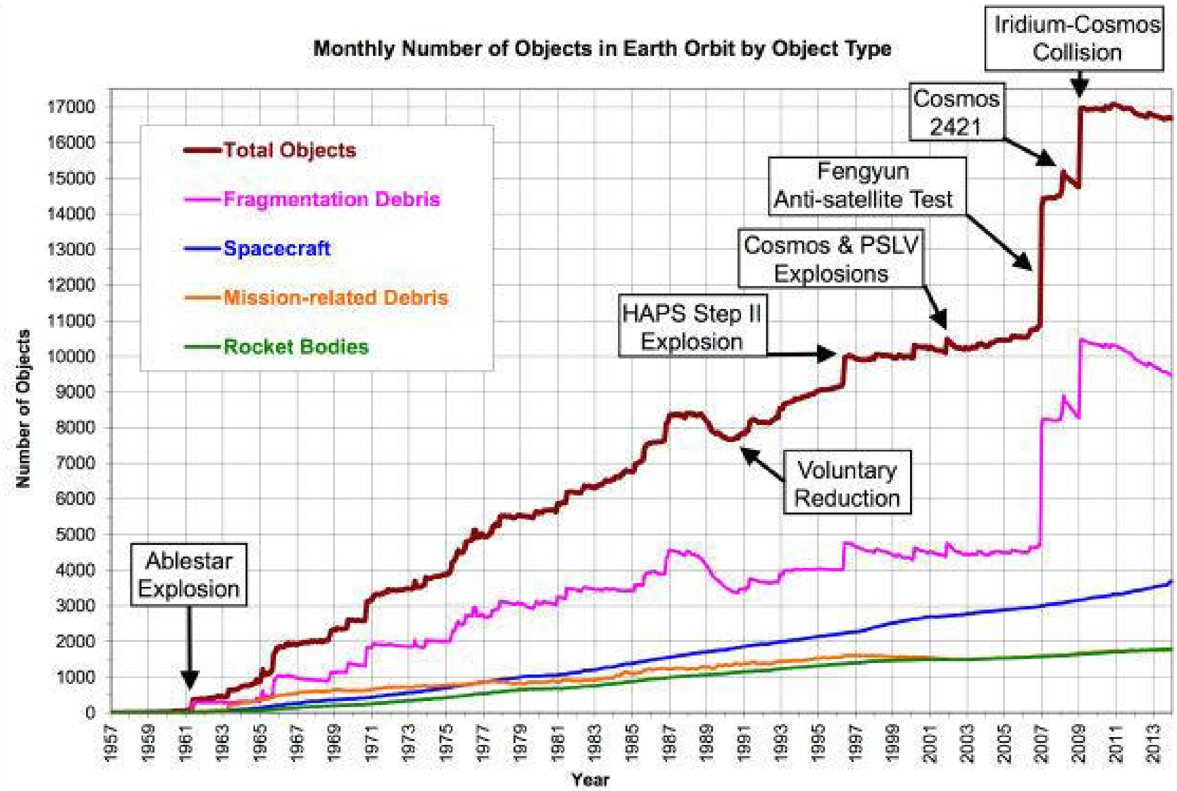


Figure 4 : Catalogued space debris over time
(Source: NASA, "Orbital Debris Quarterly News "annotated by Mika McKinnon)

2. Consequences

Forecast if 'business as usual': debris growth

> *"Business-as-usual' activities will lead to a progressive, uncontrolled increase in debris objects, with collisions becoming the primary debris source in less than 50 years."* **Donald J. Kessler**

The space debris problem is following a self-sustained process which is critical for low orbit region. This phenomenon of probabilistic increase of debris population is known as the 'Kessler syndrome'. In brief, Donald J. Kessler imagined in 1978, that a sufficient amount of space debris could possibly leads to a chain reaction of collisions increasing exponentially the debris population and consequently the probability of collisions. Such type of scenario could make impossible any space activities during decades or even century. This kind of situation must been avoided by improving application of space debris mitigation and developing remediation solutions on international scale.

The space community is barely taking the problem at its source but tries to avoid to not aggravate the situation by performing avoidance manoeuvers when it is possible. For example the International Space Station has to perform from 1 to 3 manoeuvers per year. Technical solutions are under development to reduce the debris creation rate during mission but also some methods to remove debris/reduce the population from the ground or directly in space.

However international and national space agencies are now slowly taking in consideration space debris problematic and try to develop legal text, principles in order to improve the situation and keep the space as a peaceful shared zone. [RD2]

3. Space debris Mitigation and laws

a. Principles, laws

There is no firm limit between Earth's atmosphere and outer space but it has been internationally admitted that outer space start above Karman line: the "Edge of space". This limit is situated at an altitude of 100 km and corresponds, by definition, to the lowest perigee attainable by an orbiting space vehicle. Even if, in reality, the lowest perigee at which an object in an orbit can complete at least one full revolution is about 130 km for elliptical orbit and 150km for circular one. [RD3: https://en.wikipedia.org/wiki/Outer_space, 07/2016]

Historically, at the very beginning of space activities, when the first spacecraft (the soviet satellite Sputnik 1 in 1957) was sent to outer space there was not any laws or regulations concerning this medium. Instead of apply existing laws such as aviation or marine laws adapted to outer space, nations began discussing systems in order to guarantee a peaceful use of outer space. [RD3]

In 1959, the United Nations created the Committee on the Peaceful Uses of Outer Space (COPUOS) which wrote five international principles and declarations:
- The Legal Governing principles of Outer Space activities (1963),
- The Principles Governing the of Satellites for Television Broadcasting (1982),
- The Principles Relating to Remote Sensing of the Earth (1986),
- The Principles Relevant to the Use of Nuclear Power Sources (1992),
- The Declaration on International Cooperation in the Exploration and Use of Outer Space for the Benefit and in the Interest of All States (1996).

[RD4: http://www.unoosa.org/oosa/en/ourwork/spacelaw/treaties/status/index.html, 05/2016] and
[RD5: United Nations, « Treaties and principles on Outer space », 2002]

However, previous principles are just giving guidance but are not binding any state legally. Consequently COPUOS wrote the following existing international treaties concerning outer space, which have been endorsed by many states:
- **The Outer Space Treaty (1967):** nobody can claim ownership of outer space or celestial body, peaceful use of outer space and space activities.
- **The Rescue Agreement (1968):** duty to provide assistance in case of non-expected landing.
- **The Liability Convention (1972):** define international responsibility for space object.
- **The Registration Convention (1975):** a census of space objects, information and their launching state.
- **The Moon Treaty (1979):** use of celestial bodies for peaceful purposes only, no ownership possible.

[RD4]&[RD5]

In the space debris topic, the most interesting U.N. legal aspect is the liability and it is defined by the term "Launching state": who is responsible of a space object.

Article VII of the **Treaty on Principles Governing the Activities of States in the Exploration and Use of Outer Space**, adopted in 1967 [RD5] under the auspices of the United Nations provides that:

"Each State Party…that launches or procures the launching of an object into outer space…and each State Party from whose territory or facility an object is launched, is internationally liable for damage to another State…or to its natural or juridical persons by such object or its component parts on the Earth…"

A State is qualified as "Launching State" and so liable of a space object if:

- the State has launched the object,
- the State has procured the launching (the rocket) used to launch the object,
- the space object is launched from State's territory or facility

Consequently a space object can have three or more distinct launching states.

A good hypothetical example would be a Russian-built Soyuz rocket bought and launched by an American company to orbit its satellite from French Guiana. Consequently USA, Russia and France will be considered as "Launching State" and thus could be prosecuted in case of any accident related to the launched satellite. Obviously in case of compensation owed to an aggrieved entity/person, the three States in this hypothetical example would not have the same amount of fine.

Currently space agencies cannot condemn a state for a space issue even with another state involved, because international agencies do not have any jurisdiction in space field. Every space international issue is managed under "pour parler" between states. Condemnations can only come from a national minister against a company or an actor of the same country. However space agencies are playing a main role of counsellor for national agencies by giving guidance and harmonizing laws.

Concerning space debris mitigation, U.N. has not provided guidance but only general principles of equal and durable peaceful activities in outer space indirectly implying an eco-friendly behaviour regarding to space debris. However NASA and ESA have published some requirement on space mitigation for their own projects. Those requirements have role of guidelines and principles for national ministers and then has to be followed if a States choose to incorporate those principles as national law.

In following sections, an overview of guidelines, principle and space debris mitigation is presented.

b. Protected regions

Nowadays, distinction is made between types of missions based on their orbital location during their mission. Several orbital regions have been defined by spaceagencies with the following terms [RD7: “IADC Space Debris Mitigation Guidelines”, 2007]:

Low Earth Orbit region (LEO region): spherical shell region starting from 100 km height from Earth’s surface up to 2,000 km altitude. This region is situated between atmosphere and Van Allen belt.

Geosynchronous Earth Orbit (GEO): is a circular orbit above Earth’s Equator, following Earth’s direction of rotation with a period of a Earth’s sidereal day (23 hours, 56 minutes, 4.0916 seconds). That means every day the satellite will be at the same position in the sky with respect to the ground. The corresponding altitude is 35,786 km. The orbit can be qualified geostationary if it is in the equatorial plane consequently it has always the same position in the sky because rotation axes and rotational rate are identical.

Medium Earth Orbit region (MEO region) is the region between LEO region and GEO thus the sphere shell region that extends from 2,000 km altitude up to 35,786 km. This region contains the high energy radiation zone called Van Allen belt therefor it is not commonly used for satellite mission.

High Earth orbits/Highly Elliptical Orbits region (HEO region): A High Earth Orbit is any orbit above geosynchronous (above 35,786 km). A Highly Elliptical Orbit is an orbit of low perigee (about 1,000 km) and a high apogee over 35,786 km). These orbits have an inclination between 50 and 70 degrees. Highly elliptical orbits are mainly perturbed by the Earth’s oblateness and by gravitational attraction of the Sun and Moon. HEOs are popular orbits for Earth magnetosphere measurements and astronomical observatories.

The “IADC Space Debris Mitigation Guidelines” written by Inter-Agency Space Debris Coordination Committee, in 2007, define orbits and protected regions as follows:

- Region A, **Low Earth Orbit** (or LEO) Region – spherical region that extends from the Earth’s surface up to an altitude (Z) of 2,000 km
- Region B, the **Geosynchronous Region** - a segment of the spherical shell defined by the following:
  - lower altitude = geostationary altitude minus 200 km
  - upper altitude = geostationary altitude plus 200 km
  - -15 degrees ≤ latitude ≤ +15 degrees
  - geostationary altitude (Z GEO) = 35,786 km (the altitude of the geostationary Earth orbit)

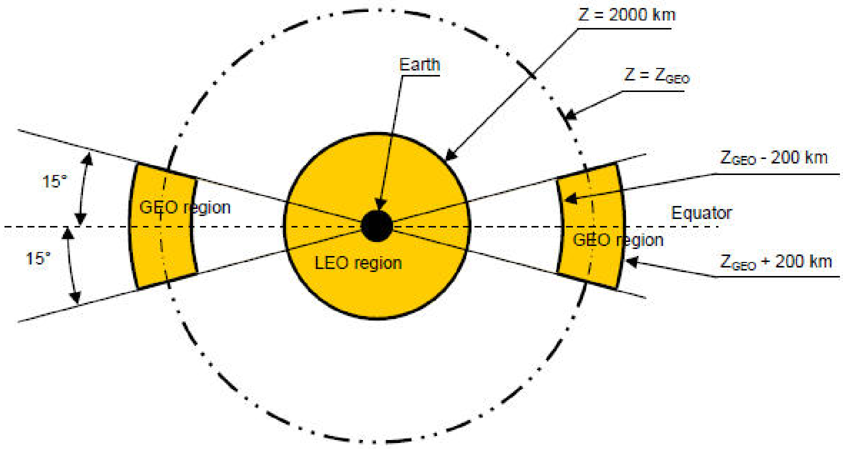


Figure 5: Representation of protected regions A and B defined in "IADC Space Debris" 2007 [RD7]

c. Passivation rules

In order to prevent any additional debris in the space, due to collision and even potential explosion (burst of pressure vessels, explosion of battery cells, etc.), space agencies have agreed on a passivation requirement at the end of a satellite mission. Passivation means to permanently deplete or make safe all on-board sources of stored energy in a controlled way in order to prevent break-ups. For ESA project, passivation has to be done 2 months after the end of the mission. [RD7]

This requirement does not apply to spacecraft which will perform a controlled re-entry because it is admitted that risk aspects are covered by the requirements on disposal reliability and re-entry safety.

Current list of passivation measures:

- **Guidance and Navigation control subsystem :**
  - Disconnect attitude control sensors and actuators (cold gas thrusters for example) from any power or chemical supply sources;
  - De-spin/stop rotating part of control moment gyros then disconnect it from power sources.
- **Mechanism Subsystem**
  - Fix and block any relative movements of rotating or movable parts;
  - De-activation of electro-explosive and pyrotechnic devices if there are not useful any more then disconnect it from power supply sources.
- **Power Subsystem**
  - Batteries and fuel cells have to be discharged, disconnected and depressurized if necessary;
  - Power Conditioning and Distribution Unit (PCDU) has to be disconnected and all possible circuits have to be switch-off;
  - Solar array has to be disconnected of power bus and batteries then short-circuited.
- **Propulsion Subsystem**
  - Pipelines have to be vented and scavenged though pressurization or by slow evaporation;
  - Gas/Propellant tanks must been vented, depleted by burn(s) for propellant and depressurized at least down to a level which can guarantee that no burst can happened due to over-pressure/temperature or collision.
- **Telecommunication Subsystem:**
  - Telemetry and communication must been switch-off with a monitoring of radio frequencies signal.

Obviously all of these required operations have to be performed in a logical order of passivation procedure appears to be:

1. Propulsion Subsystem
2. Mechanism subsystem
3. Guidance and Navigation control subsystem
4. Power Subsystem
5. Telecommunication Subsystem

This order allows the ground control segment to follow the passivation process and even correct potential impact of propulsion passivation for example by using for the last time the AOCS equipment before switching it off.

d. 25 years decay rule

Artificial satellites are present in every types of orbit around Earth but the characteristics of every orbit (altitude, space environment, etc.) lead to a distribution of the mission depending on their purposes.

Nowadays there are around 1,400 spacecraft orbiting around Earth and the half of this population is situated in the Low Earth Orbit. Major reasons are:

- Easy communication with on ground stations
- High precision measurement possibility of Earth surface
- Very short orbital period which can be useful for tactical operations
- Reduce launch cost since it requires less energy to dispose the satellite in orbit

The LEO region has a lot of advantages but due to its popularity a major drawback came with time: space debris. In 2016, the number of debris larger than 10 cm is estimated around 30,000. (NASA, «History of on-orbit satellite fragmentation» 14th edition, 2008 [RD8])

Due to this overcrowding of LEO region, both for satellites and debris, space agencies decided to write a rule imposing that LEO satellite must been designed such that after mission the spacecraft will fall into Earth atmosphere within 25 years.

This rule will stabilize and with time reduce the number of dead satellites orbiting in LEO region. In addition, on ground risk for population has to be guaranteed, this topic will be detailed in next chapters.

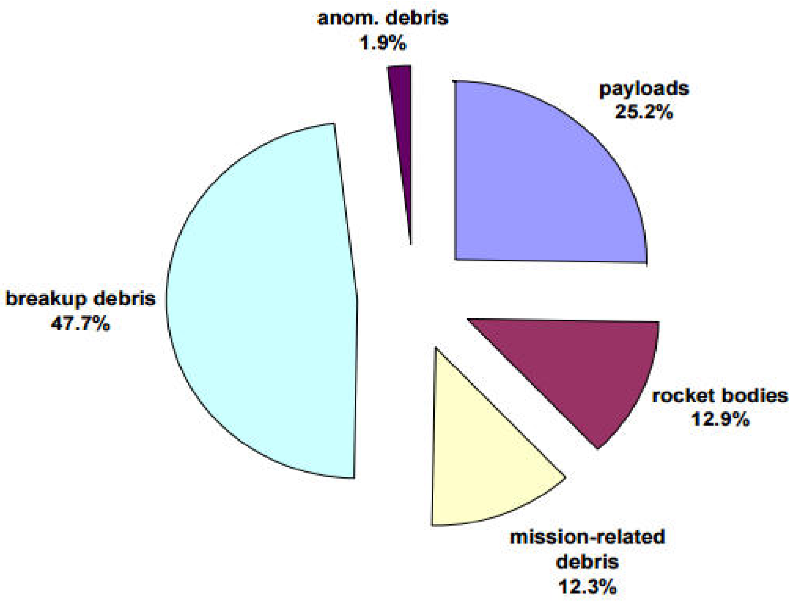


**Figure 6: Relative segments of the cataloged *in orbit* Earth satellite population.
(Source: History of on-orbit satellite fragmentation 14th edition (2008) [RD8])**

e. Graveyard rule

Concerning GEO missions it is not possible to consider a return into Earth atmosphere because it implies an enormous need of propellant. Consequently, it has been decide that GEO satellites will have to park in a graveyard mission after the end of their mission and a passivation process.

The Inter-Agency Space Debris Coordination Committee (IADC) is defining the disposal orbit by the minimum perigee altitude ΔH (in kilometer) above the geostationary orbit with the following formula:

$$\Delta H = 235 + (1000\ C_R \frac{A}{m}) \qquad (1)$$

Where C_R correspond to the coefficient of solar radiation pressure (between 1.2 to 1.5 N/m^2) and $\frac{A}{m}$ the surface ratio per mass unit of the satellite. [RD7]

This formula define the distance between geostationary orbit and graveyard orbit taking into account the solar pressure which is constantly acting on the spacecraft surface (like a solar sail) but also the gravity perturbation of the Sun and the Moon (35 km precaution) and a security distance of 200 km.

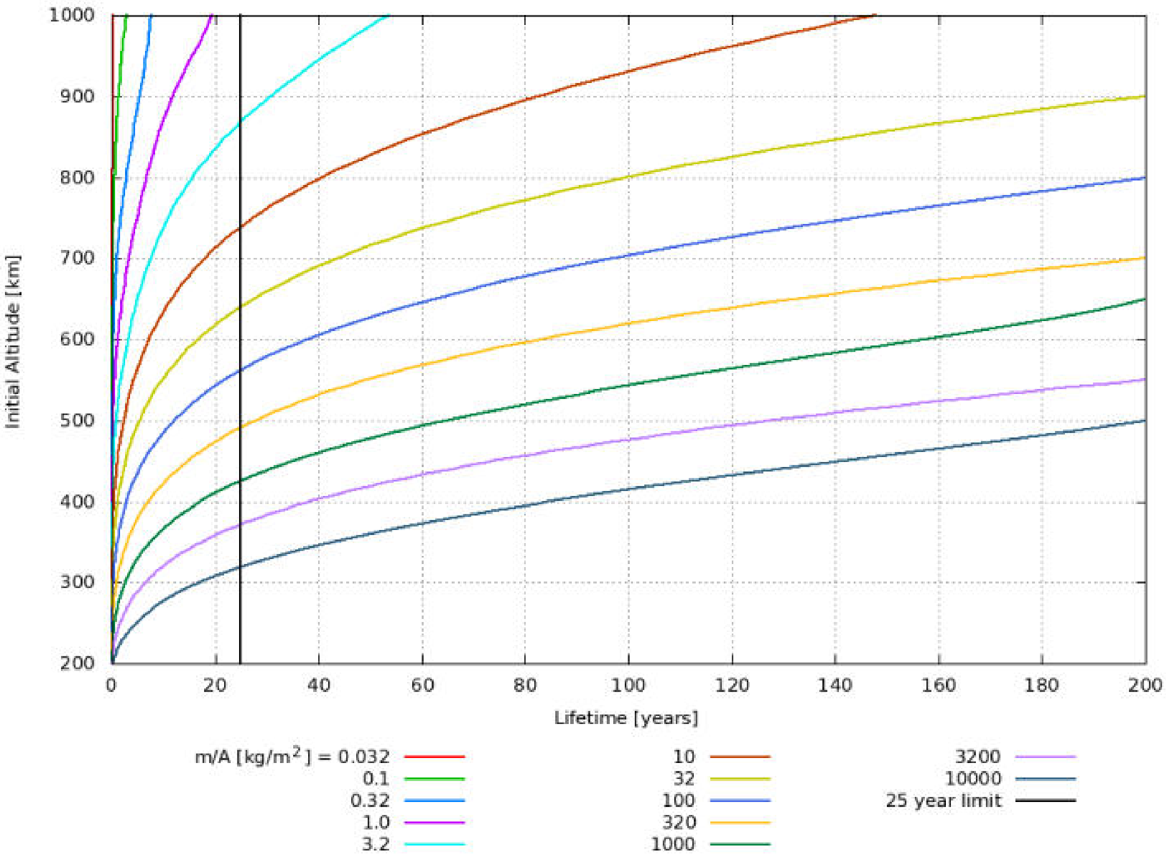


Figure 7: Orbital lifetime for an object decaying from a circular orbit as function of mass-to-area ration.
(Source: ESA, "Space Debris Mitigation Compliance Verifications Guidelines", 2015 [RD9])

f. Risk on ground limitation

Space agencies agreed on defining an on ground risk limitation for spacecraft doing uncontrolled re-entry into Earth atmosphere in order to guarantee the safety of population. The limit defines a risk limit of 10^{-4} of on ground casualty odds resulting from the fall of a space vehicle or a fragment of a space object.(CNES, “French Space Operations Act - Technical regulations”, 03/2011 [RD10])

Space debris which has a kinetic energy under 15 J at ground level will be neglected. A fragment is considered harmful (means require medical attention within 48 hrs) if its kinetic energy is between 15 and 100 J. Over 100 J the impact is defined as mortal. [RD10]

To calculate the on ground human casualty risk, European Space Agency has defined a formula [RD9]:

$$Hc = DCA \times P_D \tag{2}$$

Where P_D is the average population density, for the particular orbital inclination and year of re-entry given by prediction and surface swept by satellite ground track. The DCA is the debris casualty area:

$$DCA = \sum_N \pi \left(\frac{d_h}{2} + R_i\right)^2 \approx \sum_N \left(0.6 + \sqrt{A_i}\right)^2 \tag{3}$$

Where N is the number of objects that survive re-entry and A_i is the area of one surviving piece. The term 0.6 represents the square root of the average cross-sectional area of a standing person, as viewed from above (considering 66.7 cm shoulder width).

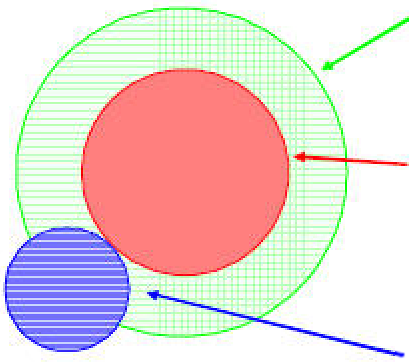


Figure 8 : Geometric explanation of DCA formula

The formula (3) is the most used even if it can lead to a maximum error of 1% compare to the circle formula (for very small area close to 0). But it is easier to use 0.6 corrections for human area consideration than the exact value with lot of digits.

The on ground human casualty risk limitation of 10^{-4} will be applicable from 2020 (“French Space Operations Act – Technical regulations” 03/2011 [RD10]. If a spacecraft does not satisfy this requirement then uncontrolled re-entry will not be allowed. Instead, a controlled re-entry will have to be performed such that the impact foot-print can be ensured over an ocean area, with sufficient clearance of landmasses and traffic routes (depending on the state space laws).

III. Atmospheric re-entry theory

This chapter has been written in order to give an overview of physical principles ruling re-entry theory, it does not have the pretention to explain it completely and detail everything but at least give a short explanation of parameters and assumptions used, mainly in re-entry software.

1. Atmospheric and Earth related models

a. Isothermal-barotropic model of atmosphere

Assuming a constant temperature of the atmosphere it is possible to estimate the density in function of height using an exponential function:

$$\rho(H) = \rho_0 e^{-\left(\frac{H}{H_0}\right)} \quad (4)$$

with H_0 the isothermal scale height: $H_0 = \frac{RT}{Mg} \sim 8.4\ km$

- Temperature at ground level: T=288K,
- Perfect gaz constant: $R = 8.314$ J/mole/K
- Acceleration due to gravity at ground level: $g = 9.81$ m/s^{-2}
- Mean molecular weight of air: $M = 29$ g/mole

(assuming a gas made up of 78%N, 21%O, and 1%Ar)

- Reference density at sea level: $\rho_0 = 1.225$ kg/m^3

(Willam E. Wiesel, "Spaceflight Dynamics" Third Edition, 2010 [RD11])

There are several numerical models to represent atmospheric parameters such as U.S. Standard 1976 Model, MSISE-90 Model, GRAM-99 Model, etc. In reality, it is known that atmosphere temperature is not constant and varies with altitude and composition (Figure 10) but for a first approximation this model can be sufficient.

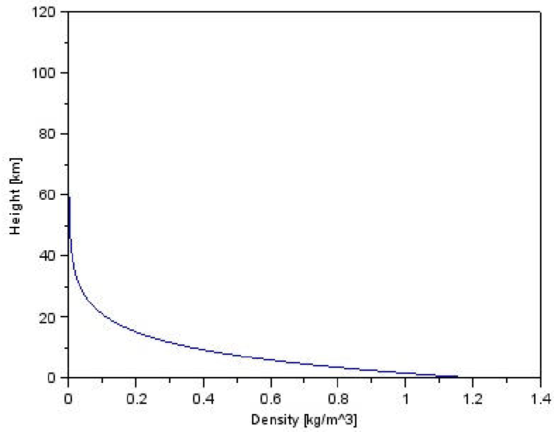


Figure 9: Isothermal-barotropic model of density

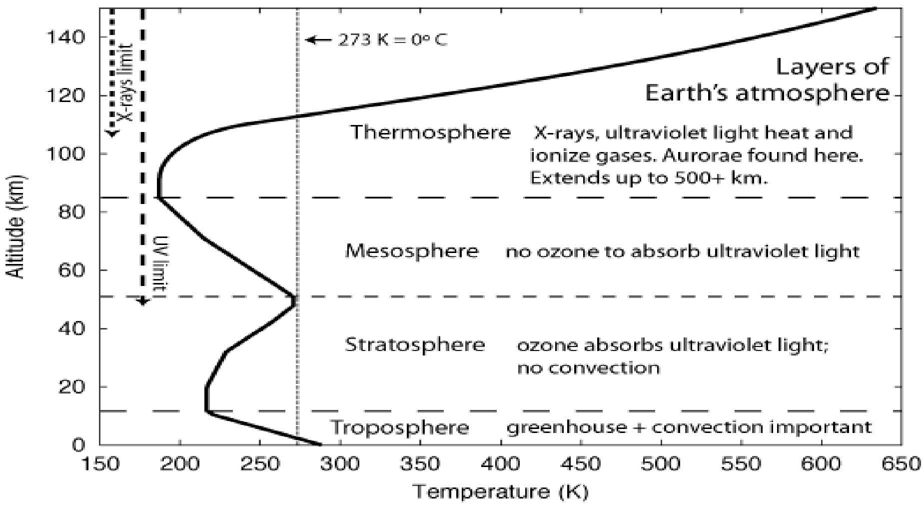


Figure 10 : Earth atmosphere layers description
(Source : http://www.astronomynotes.com/solarsys/s3c, 02/2014 [RD12])

b. Gravity model

Based on Earth characteristics it is possible to estimate acceleration due to gravity as function of height to take in account the small variation which appears with distance, by neglecting oblateness effect, non-uniform ground composition and no Sun and Moon gravity participation [RD11].

$$g(H) = G * \frac{M_{\oplus}}{(R_{\oplus}+H)^2} \tag{5}$$

- Earth gravitational constant $G = 6.67408 \text{ x } 10^{-11} \text{ N.m}^2/\text{kg}$
- Earth mass $M_{\oplus} = 5.9722 \text{ x } 10^{24} \text{ kg}$
- Earth radius $R_{\oplus} = 6.378 \text{ x } 10^{6} \text{ m}$

In reality the 3D model of Earth gravity is not spherical and uniform, it consists of a geoid (3D shape representing equipotential energy level) with some static local variations corresponding to the ground composition (some element are denser than others) but also there are some dynamic variations such as the Moon participation (which lead to ocean tides) other celestial bodies (Sun, planets, etc.). Such model is pretty complicated and it is also time dependent.

It is possible to take some picture of this phenomenon by measuring the sea level and determining the equipotential geoid, which give the altitude corresponding to an equal gravity.

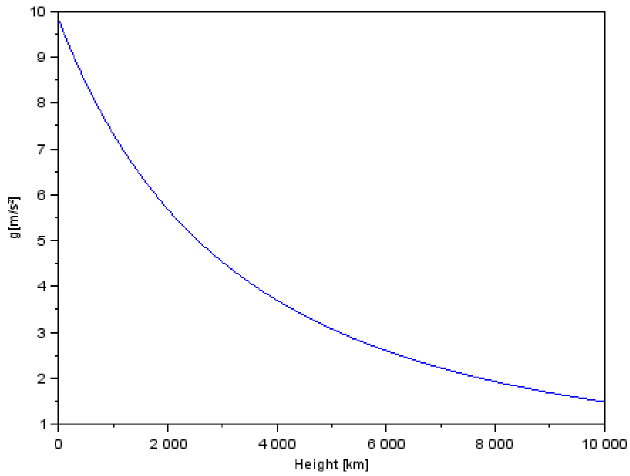


Figure 11 : Plot of gravity decreasing as function of height

The Gravity Recovery and Climate Experiment (GRACE), a satellite launch in 2002 by the NASA, has made lot of measurement to analyse the gravity variation on Earth (Figure 12).

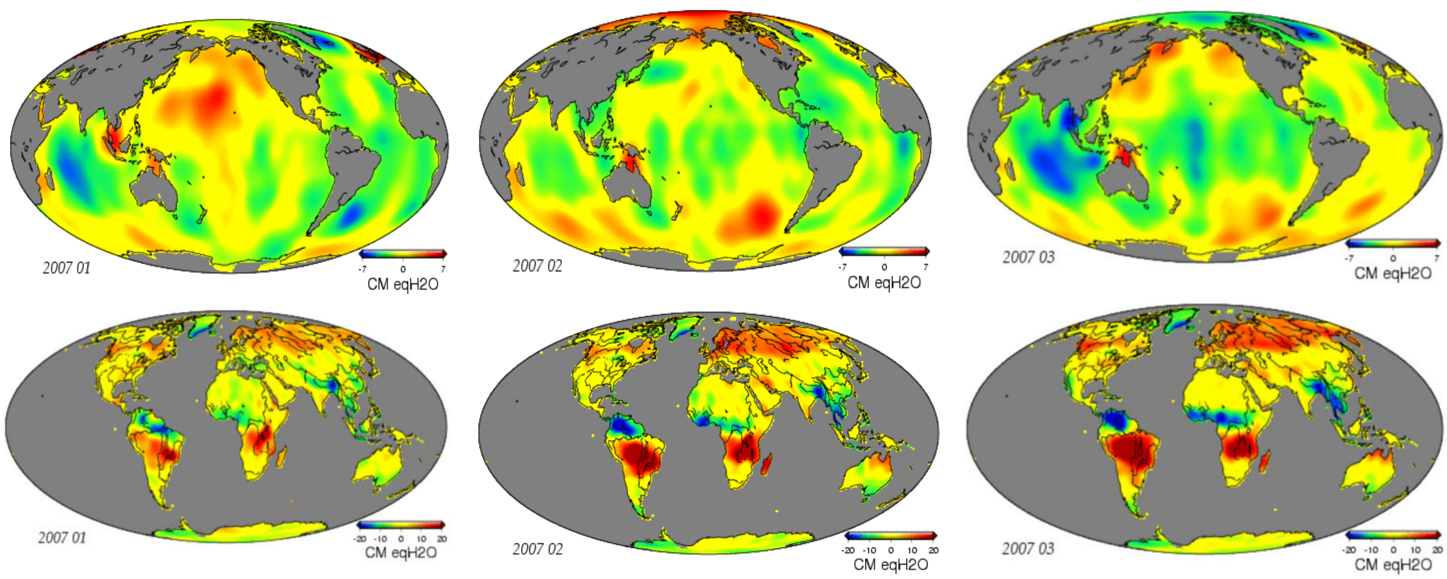


Figure 12 : Gravity anomaly variation between January and March in 2007
(Source : http://podaac.jpl.nasa.gov/,05/2016 [RD13])

c. Earth population distribution model

Considering that satellites have to be designed in order to guarantee an on ground casualty risk under 10^{-4}, that implies to know the global population density in order to multiply with debris casualty area taking into account orbital inclination.

Actual global population distribution is more or less accurately known by U.N. agency, but the re-entry of spacecraft is a time dependent problem which implies an orbit decay less than 25 years according on space agency recommendations. Thus it appears necessary to model evolution of population distribution through time taking into account birth rate, local expectancy and migration flow, etc. (Table 1 and Figure 13 for examples). This work is conducted by U.N. which publishes a study called "World Population Prospects", 2015 [RD14].

Year	Low variation	Mean variation	High variation
2020	7 688 595 000	7 758 157 000	7 827 607 000
2030	8 179 515 000	8 500 766 000	8 821 836 000
2040	8 532 257 000	9 157 234 000	9 789 249 000
2050	8 710 042 000	9 725 148 000	10 801 105 000

Table 1: Demographic prediction
(Source: « World Population Prospects: The 2015 Revision» [RD14])

With the geographic distribution of human population is possible to calculate a mean population density for a given orbit inclination considering the on ground surface which can potentially be hit by a satellite fragment.

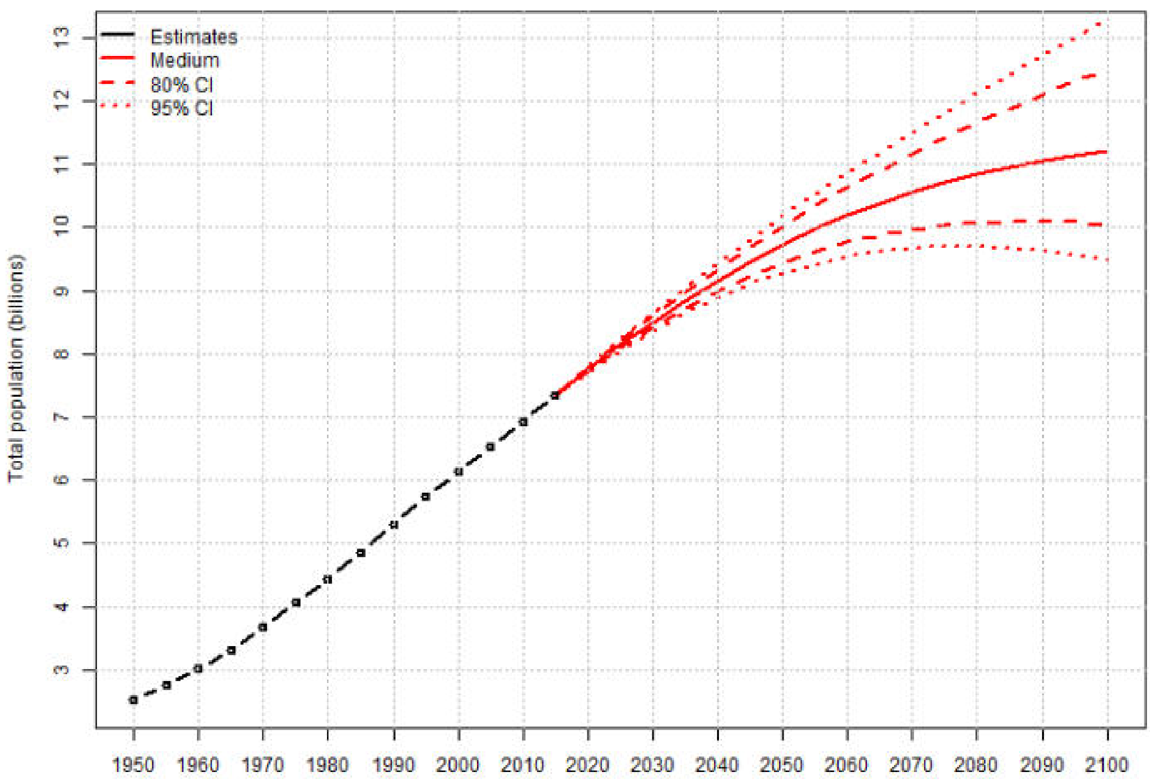


Figure 13 : Global demographic prediction, projection 80 and 95 percent condidence intervals, 2015-2100
(Source: «World Population Prospects: The 2012 Revision» [RD14])

The global population is currently growing and will continue to grow until stabilization that implies that for constant satellite configuration with a given DCA and inclination, the on ground casualty risk will inevitably grow. For example if we consider the population in 2015 (7.350 billion), it gave a DCA of around 7 m², assuming uniform distribution. Then for a 9.25 billion population (in 2050), the maximum tolerated DCA associate to a casualty risk will be of 5.5 m². Consequently improvements will have to be done constantly through time for demise ability of spacecraft doing uncontrolled re-entry.

2. Re-entry models

The following models are based on the book «Spaceflight Dynamics» Third Edition by William E. Wiesel [RD11], especially the chapters 3 "Earth satellite operations" and 8 "Re-entry dynamics". They have been used for next chapter and can be necessary to understand physical principles governing high altitude thermal and dynamic behaviour.

a. Decay lifetime

When a satellite is orbiting around Earth, in low Earth orbit, it inevitably will fall, even very slowly, to the ground, because of the drag created by the Earth atmosphere.

Drag deceleration:

$$a_d = \frac{1}{2}\frac{C_d A}{m}\rho v^2 \quad (6)$$

With : C_d drag coefficient, A the area in meter square and m the mass in kilogram,

With equations of motions, of energy and Kepler's laws, it becomes:

$$H(t) = h \ln\left[e^{\frac{H_0}{h}} - \frac{\sqrt{\mu R_\oplus} B^* \rho_0 (t-t_0)}{h}\right] \quad (7)$$

This formula gives height as function of time from an initial altitude, it is also possible to estimate the time needed to decay and to hit the ground:

$$t_d = t - t_0 = \frac{h}{\sqrt{\mu R_\oplus} B^* \rho_0}\left(e^{\frac{H(t_0)}{h}} - 1\right) \quad (8)$$

This formula give the decay life time for a satellite for a given initial height but it is not including solar pressure participation. Sun radiation is the first contributor in deorbit process for LEO satellite, because it is the main force slowing down the S/C and thus decreasing orbit altitude until it finally reaches the atmosphere. Around 170 km height satellites are falling faster due to atmospheric break. The length of atmospheric fall duration is about a day. Consequently this phase is not driving the time constraint of 25 years to fall back into Earth atmosphere, solar pressure is the key driver

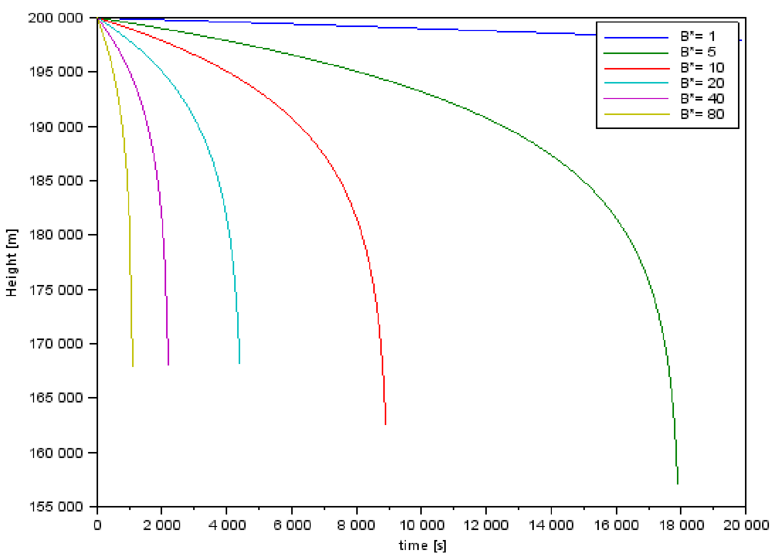


Figure 15 : Decay lifetime in function of ballistic coefficient for 200km of initial altitude

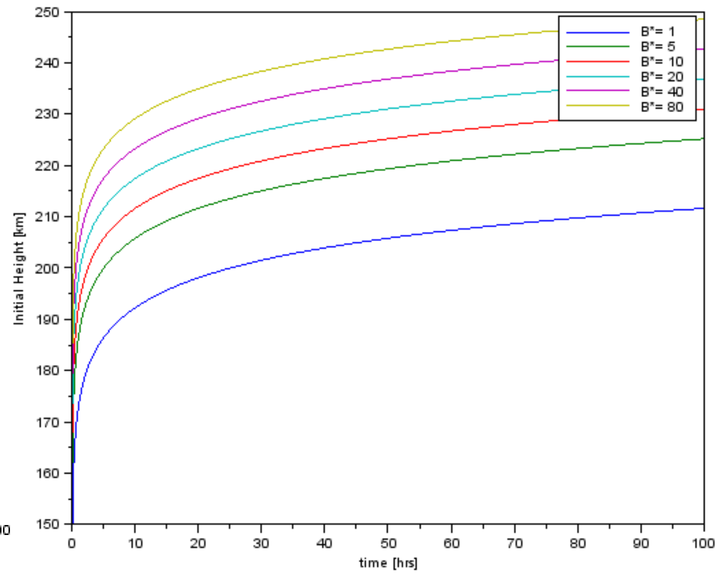


Figure 14 : Decay life time for several ballistic coefficients in function of initial altitude

b. Steep ballistic re-entry

In this section the vehicle is considered without lift for the case of a steep angle. From kinematics relations, Newton's second law and differential equation resolution, it is possible to estimates the evolution of the flight path angle of a vehicle during re-entry. In this section only the terminal velocity will be discuss because it can be a criteria of design for some type of mission.

$$V_{terminal} = \sqrt{\frac{2mg}{\rho A C_d}} \quad (9)$$

Based on the formulas present in chapter 3 [RD11], it is possible to design spacecraft in order to promote maximum deceleration at high altitude and also decrease as much as possible the terminal velocity. This kind of design is common for atmospheric probes which can acquire more measurements if it goes slower. This terminal velocity is also important to know from which height a parachute can be deployed from a re-entry object.

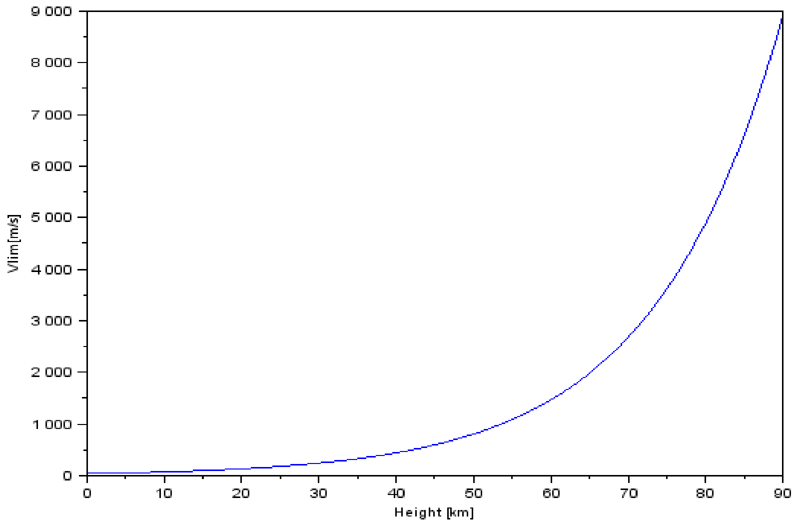


Figure 16 : Terminal velocity vs Height for a given ballistic coefficient (~75 like Sentinel-3 satellite)

c. Ballistic orbital re-entry

Based on kinematic relation, centripetal acceleration, Newton's laws and with some exchange with independent variables, it is possible to write a formula expressing deceleration due to drag during a ballistic re-entry:

Using those expressions, aerodynamic deceleration can be written:

$$\frac{dV}{dt} = -\sqrt{\frac{H_0}{R_\oplus}}\, y e^{-2x} = -\sqrt{\frac{8H_0}{3R_\oplus}}\, x^{\frac{3}{2}}\left(1+\frac{1}{6}x+\frac{1}{24}x^2\right)e^{-2x} \quad (10)$$

With x an independent variable: $x = -\ln(\bar{V})$ with $\bar{V} = \frac{V}{\sqrt{gR_\oplus}}$

This deceleration rate has a maximum when $\frac{d}{dx}\left(\frac{dV}{dt}\right) = 0$. Some calculation shows that the maximum deceleration will be: $\frac{dV}{dt}|_{max} = 81.42\ \mathrm{m/s^2}$

For a speed of $V \approx 3.3\ \mathrm{km/s}$ independent of the drag coefficient of the vehicle, that mean every vehicle will experience the same maximum deceleration at the same speed during re-entry. This maximum deceleration is equivalent to more than 8 times the Earth's gravity.

This result is very important because it gives the maximum deceleration load that a spacecraft will endure during a re-entry. This value of 8 gee's corresponds to the upper limit what a human body can "tolerate".

Hopefully this maximum of deceleration appears in the special case of ballistic re-entry, by adding some lift it is possible to reduce the maximum deceleration by doing a longer re-entry trajectory.

3. Aerodynamics, aerothermodynamics and ablation model

Information and formulas written in the following sections have been found in user manuals explaining how the software Object Re-entry Survival Analysis Tool (ORSAT 6.0) created by the NASA [RD15] works. Major principles and formulas are presented with their assumption and hypothesis in a short and simplified way.

a. Aerodynamics

In the re-entry software, principal acting forces are the drag and the lift. They are defined based on coefficients measured during experimentations in wind tunnels.

Drag coefficient: $$C_D = \frac{D}{\bar{q}_\infty A_{ref}} \quad (11)$$

Lift coefficient: $$C_L = \frac{L}{\bar{q}_\infty A_{ref}} = \frac{LC_D}{D} \quad (12)$$

Where the dynamic pressure, $\bar{q}_\infty$, is : $$\bar{q}_\infty = \frac{1}{2}\rho_\infty {V_\infty}^2 \quad (13)$$

Through experimentations, it has been found that drag and lift coefficients are dependent on object's geometry and if it is tumbling but also flow regimes and flow properties (density, viscosity, pressure, etc). In the re-entry software objects are discretized in spheres, boxes, plates or cylinders to estimate their drag and lift coefficient as functions of the flow regime. The atmosphere is considered compressible and consequently several types of regimes can be distinguished.

Reminder:

The **mean free path (λ)** is the average distance that a moving particle has to travel between two successive impacts (collisions), which can modify its direction/energy/particle properties.

Vacuum range	Pressure [hPa]	Molecular density [m^{-3}]	Mean free path [m]
Ambient pressure	1013	2.7×10^{25}	68×10^{-9}
Low vacuum	300 – 1	10^{22} - 10^{25}	10^{-7} – 100 $\times 10^{-4}$
Medium vacuum	1 – 10^{-3}	10^{19} - 10^{22}	10^{-4} –10^{-1}
High Vacuum	10^{-3} – 10^{-7}	10^{19} - 10^{22}	10^{-1}–10^{3}
Ultra-High Vacuum	10^{-7} – 10^{-12}	10^{19} - 10^{22}	10^{3} – 10^{8}
Extremely High Vacuum	< 10^{-12}	< 10^{10}	>10^{8}

Table 2: Mean free path typical value for different type of vacuum
(Source: https://en.wikipedia.org/wiki/Mean_free_path, 07/07/2016 [RD16])

Knudsen number (Kn) is a dimensionless number corresponding to a ratio of molecular mean free path length to a representative length scale (for example the radius of a body in a fluid). It permits to determine the type of flow based on continuity of the medium and not on turbulences (such as Reynolds numbers)

$$K_n = \frac{\lambda}{L} \tag{14}$$

For Boltzmann gases, the Knudsen number can be calculated with: $K_n = \frac{k_B T}{\sqrt{2}\pi d^2 p L}$ (15)

with k_B, Boltzmann's constant, T, temperature, d,diameter of particle/molecule, p,pressure and L, characteristic length. Knudsen number can be used to divide a gas into three regimes:

Free molecular regime ($K_n > 10$)

It corresponds to the case when pressure is so low that the mean free path of the molecules is larger than the size of the chamber or the tested object that means that continuum assumption are not a good approximation. Instead of fluid mechanics, statistical methods should be used.

Continuum regime ($K_n < 0.001$)

This regime is considered when the mean free path is lot smaller than the characteristic length of the problem/object then continuum assumption of fluid mechanics can be used.

Transitional regime ($0.001 < K_n < 10$)

Principally composed of bridging functions, this regime has to ensure continuity between free molecular and continuum regime. Considerations are taken into account about aerodynamics coefficient as function of spinning and tumbling.

For example:

The drag coefficient in transitional regime is calculated with the following bridging formula for sphere with random tumbling:

$$C_{D_{trans}} = C_{D_{cont}} + \left(C_{D_{fm}} - C_{D_{cont}}\right)\ (\sin[\pi(0.5 + 0.25\log_{10}(K_n)])^3 \tag{16}$$

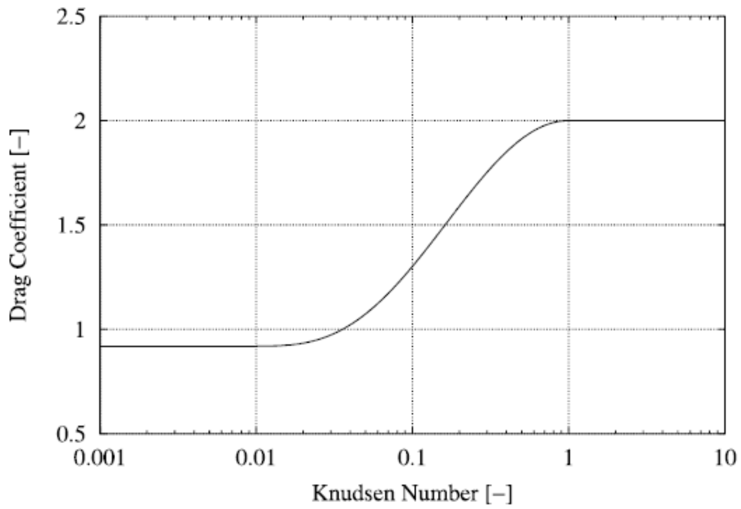


Figure 17: Drag coefficient of a sphere
(Source : Hyperschall Technologie Göttinge, "SESAM Final Report", 2004 [RD17])

Reynolds number (Re):

It is a number corresponding to a ratio of inertial force to viscous force in order to characterize the dynamic aspect of the flow.

$$Re = \frac{\text{Inertial forces}}{\text{Viscous forces}} = \frac{\rho v L}{\mu} \tag{17}$$

With ρ, the density in kg/m^3, v, the velocity of the flow in m/s, L, the characteristic length of the problem and μ, the dynamic viscosity in Pa·s

Laminar flows:
They occur when the Re is low, which means viscous forces are dominant so fluid motion is constant and smooth

Turbulent flows:
They are seen when Re is high, then inertial forces are leading the flow producing chaotic eddies, vortices and flow instabilities.

Mach number (Ma):

Is a dimensionless number representing the ratio of flow velocity to the local speed of sound:

$$\boldsymbol{Ma} = \frac{\boldsymbol{v}}{\boldsymbol{c}} \tag{18}$$

With c the sound velocity in the medium.

Regime	Subsonic	Transonic	Sonic	Supersonic	Hypersonic	High-hypersonic	Reentry speeds
Mach	<0.8	0.8–1.2	1.0	1.2–5.0	5.0–10.0	10.0-25.0	>25.0

Table 3 : Velocity regime in function of Mach number
(Source : https://en.wikipedia.org/wiki/Mach_number, 07/2016 [RD18])

Depending on the geometry shape and Mach number it is possible to determine fluid flow or even shock wave shapes for high Mach number velocity.

b. Aerothermodynamics and ablation model

Stanton number (St): is a dimensionless number representing the ratio of heat transferred into a fluid to the thermal capacity of fluid. It can be used to measure the rate of change of the thermal energy deficit (or excess) in the boundary layer due to heat transfer from a planar surface during re-entry.

$$St = \frac{h}{GC_p} = \frac{h}{\rho u C_p} \tag{19}$$

with h , the convection heat transfer coefficient, ρ, the density, v, the speed of the fluid and C_p the specific heat of the fluid.

This number also can be expressed in terms of fluid's Nusselt, Reynolds and Prandtl numbers:

$$St = \frac{Nu}{RePr} \tag{20}$$

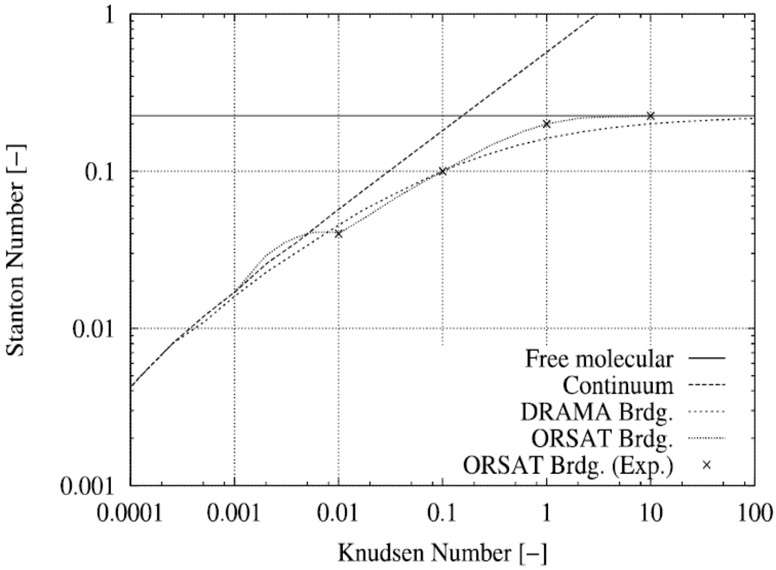


Figure 18 : Stanton number versus Knudsen number for re-entry object
(Source : Hyperschall Technologie Göttinge, “SESAM Final Report”, 2004 [RD17])

Nusselt number (Nu):

It characterizes the nature of heat transfer at a boundary within fluid with a ratio of total heat transfer to conductive transfer into the fluid.

$$Nu = \frac{\text{Total heat transfer}}{\text{Conductive heat transfer}} = \frac{hL}{k} \tag{21}$$

With h the coefficient of convectional heat transfer in W/(m²·K) and k, the thermal conductivity in W/(m·K).

Prandtl number (Pr):

It is a dimensionless number characterizing the nature of thermal diffusion in a fluid, it consist of the ratio of the viscous diffusion rate to thermal diffusion rate

$$Pr = \frac{\text{viscous diffusion rate}}{\text{thermal diffusion rate}} = \frac{C_p \mu}{k} = \frac{v}{\alpha} \tag{22}$$

Cp is the specific heat of a material in W/kg.

Small values, Pr <<1, means that thermal conduction is dominant regardless of the velocity of the fluid. A high Prandtl number means that temperature profile is highly dependent of the fluid velocity.

Assuming compressible flow dynamics during re-entry implies existence of stagnation point where the pressure is maximum and the local velocity of the fluid is zero. To calculate the temperature through time into a part, it is necessary to evaluate the heating rate at the stagnation point. The works of Klett and Cropp contained in [RD19] and [RD20] made in 1965 are still used to estimate heating rate in many software. For example for a spherical component:

Free Molecular regime ($K_n > 10$)

$$\dot{q}_{st_{fm}} = \frac{\alpha_T \rho_\infty {V_\infty}^3}{2} \tag{23}$$

With α_T thermal accommodation coefficient equal to 0.9

Then average heating rate can be evaluated: $\dot{q}_{st} = f_{fm} * \dot{q}_{st_{fm}}$

With f the averaging factor equal to 0.25 for free molecular flow.

Transitional regime ($0.001 < K_n < 10$)

For $0.001 < K_n < 10$:

$$\dot{q}_{st_{trans}} = St \frac{\rho_\infty {V_\infty}^3}{2} \tag{24}$$

It is possible to use more complicated formulas to bridge more precisely the free molecular and continuum flow but differences are minor.

Continuum regime ($K_n < 0.001$) :

Detra-Kemp-Riddell (DKR) equation gives the stagnation point cold wall heating rate

$$\dot{q}_{st_{cont,cold\,wall}} = \frac{11028.5}{\sqrt{R}} \left(\frac{\rho_\infty}{\rho_{sl}}\right)^{0.5} \left(\frac{V_\infty}{V_{circ}}\right)^{3.15} \tag{25}$$

ρ_{sl}, density at sea level 1.225 kg/m^3, V_{circ}, circular orbit velocity equal to 7,803 m/s and R the radius of the sphere in m.

Then an enthalpy ratio correction has to be applied to obtain hot wall heating rate:

$\dot{q}_{st_{cont,hotwall}} = \dot{q}_{st_{cont,cold\,wall}} \left(\frac{h_{st} - C_{p,air} T_{hw}}{h_{st} - C_{p,air} T_{cw}}\right)$ with $h_{st} = \frac{{V_\infty}^2}{2} + C_{p,air} T_\infty$

$C_{p,air}$ is normally set for 300K temperature but it can be taken as function of wall temperature

Then average heating rate can be evaluated: $\dot{q}_{st} = f_{fm} \dot{q}_{st_{cont,hotwall}}$

With f the averaging factor equal to 0.275 for continuum regime.

At this point, after all these complicated formulas and principles, a simple conclusion can be made by highlighting the fact that heating phenomena during re-entry due to stagnation point is governed mainly by velocity which appears with to the power 3, and also by the density.

4. Re-entry simulation software

Re-entry software can be sorted into two categories: high and low fidelity simulations.

For high fidelity simulation software, tumbling and exposure are considered because the S/C is fully modelled and simulated with 6 degrees of freedom. Consequently thermal distribution and mechanical stress are simulated on a mesh. Thus the simulation is taking a lot of resource and time to perform all the step by step integration.

The low fidelity software are oriented on objects, it only considers 3 degrees of freedom which means it neglects the tumbling and exposure relationship between parts. In addition, it takes into account only a simple tree structure which means that a part can only has one parent object, giving a single breakup event for every part. Thus, multiple exposures and warming by conduction are neglected, parts are exposed since their only parent is completely ablated.

a. Pampero Software

PAMPERO is a new CNES spacecraft-oriented tool. Its development allows a better understanding of the various physical phenomena during the re-entry and to find new ways to improve the DEBRISK software.

To summarise:
• 6 DOF calculations can be performed,
• Aerodynamic forces are only due to pressure effects,
• The pressure coefficient can be calculated for the 3 regimes,
• The convective heat flux is generally estimate by empirical laws or correlations with CFD,
• A 3D thermal module has been implemented,
• A preliminary ablation module has been also implemented.

PAMPERO is still in development where improvements of some modules are necessary.

b. Scarab Software

"The SCARAB ("SpaceCraft Atmospheric Re-entry and Aerothermal Break-up") software has been developed to be able to simulate the re-entry of a satellite in detail. In general, the re-entry analysis starts with the modelling of the spacecraft, to generate a geometric model, which reflects the structural composition and physical properties of the satellite. After defining the initial orbit, attitude and attitude motion, as well as certain environmental parameters such as the atmospheric model to be used, the simulation is started. After the simulation is finished, the results are analysed.

In SCARAB the modelling of a spacecraft can be very complex. The particular spacecraft components are modelled using geometric primitives, such as sphere, cylinder, box and cone, which are positioned in a way to reflect the real composition of the satellite. By defining the wall thickness and assigning a material, the thermal and mechanical properties of the primitives are obtained. Properties like mass, center of mass and moments of inertia of the finished spacecraft model are determined automatically from those of the primitives.

The re-entry simulation is then started using the complete model of the satellite. During the simulation several model properties are calculated and stored. These properties can be subdivided into three basic types. 1. properties of the whole satellite model, such as maximum surface temperature, aerodynamic coefficients and environmental characteristics, 2. properties of particular primitives, like mean temperature or particular heat loads, and 3. local surface properties. The latter are determined by the physical properties of "panels". For the simulation, the surface of the geometric primitives is divided into small triangles, called panels. For each panel the aerodynamic and thermal loads are calculated and the local heat budget is determined.

The panel based analysis provides the possibility of local destruction analysis. If a panel is molten completely, it leaves a hole inside the primitive. These holes may enable the flow to penetrate into the spacecraft, increasing the heat load on the inside. If the connecting panels between particular components are molten, the satellite or parts of it break apart. This is part of the SCARAB simulation and the generated fragments are simulated separately until they demise or reach the ground." (http://www.htg-hst.de/1/htg-gmbh/software/scarab/, 05/2016 [RD21])

c. DRAMA / SESAM software

"The software SESAM ("Spacecraft Entry Survival Analysis Module") was developed by HTG (Hypershall Technologie Göttingen GmbH ©) to determine those fragments of a satellite that survive the atmospheric re-entry and may pose a risk to the world population. SESAM is part of the DRAMA ("Debris Risk Assessment and Mitigation Analysis") software suite to handle the many aspects of space debris. For the on-ground risk assessment the software SERAM ("Spacecraft Entry Risk Analysis Module") analyses the results of the SESAM simulation.

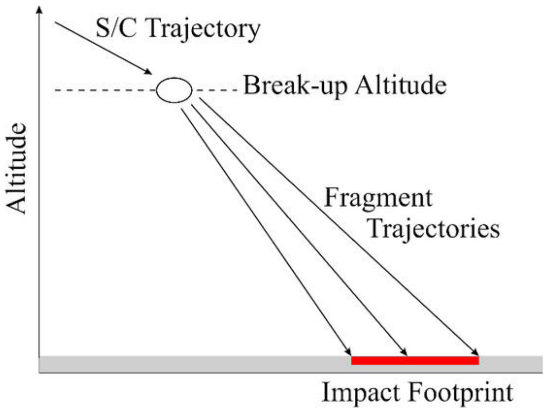


Figure 19: Schema of principle of Drama/Sesam software [RD22]

For the calculation of the satellite's re-entry a model of the satellite is created and its initial state (position, epoch and velocity) is defined. The satellite's motion inside the Earth's atmosphere is determined by numerical integration over time. SESAM uses a simplified satellite model, consisting of a compartment which represents the body of the spacecraft. The compartment contains simple geometric objects like spheres, cylinders, boxes or plates, which are characterized by their measures, mass and material properties. The typical assumption for object oriented tools like SESAM is the release of the particular simple objects from the compartment at a certain altitude. Each object is then simulated separately. Basically the release altitude is a free parameter, but due to experience from re-entry observations, usually an altitude of 78 km is chosen.

The motion of the particular objects is determined by the occurring forces. The simulation takes into account the Earth's gravity and the aerodynamic drag. The temperature of the objects is calculated using approximation formulas for the heat flow as a function of local atmospheric density, velocity and ballistic coefficient. If the temperature reaches the melting temperature of the object, it stays constant and the mass starts to deplete by melting. Thus the re-entering objects can demise complete or partially or they may survive to the ground without any demise.

The concept of SESAM is very simple. The modelling of the objects determines which kind of fragments can be generated at which altitude. The advantage of the method is that it is very fast. Thus, in relatively short time, a high number of calculations can be processed. This is advantageous, if a first quick assessment for the on-ground risk is needed, e.g. during the conceptual phase of spacecraft design or at an earlier time before the expected re-entry. For a detailed analysis, the use of more complex software tools like SCARAB, which are more time-consuming, have to be used." (http://www.htg-hst.de/1/htg-gmbh/software/dramasesam/, 05/2016 [RD22])

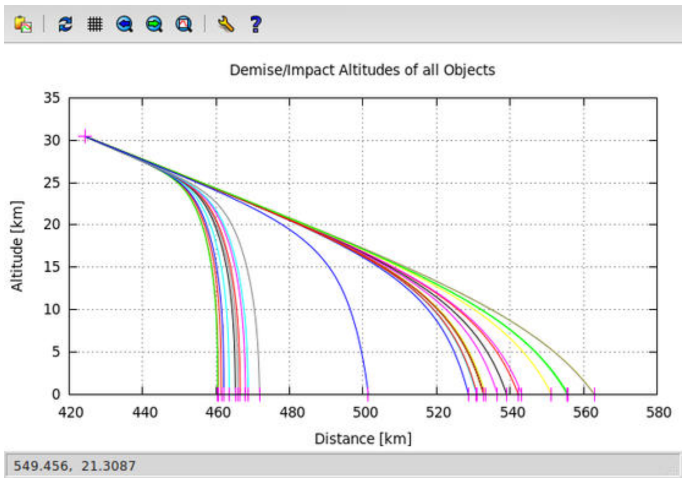


Figure 20: example of results obtained with Drama/Sesam [RD22]

d. Debrisk Software

Presentation:

Debrisk is a tool developed by the CNES (French National Center for Space Studies) which is at disposal for industrial company and space operators for qualification according on the «French Space Operations Act» [RD10].

This software is used to analyse the thermal ablation during the re-entry of a spacecraft in the Earth atmosphere. The spacecraft is represented by a group of basic object such as box, cylinder or sphere. Each object is defined by its form, size, mass and material. Components are sorted following a tree structure in order to give an ablation order. The aim of Debrisk is to estimate the number of fragments surviving the ablation, the remaining mass, kinetic energy and casualty area. (https://logiciels.cnes.fr/content/debrisk, 05/2016 [RD23])

Entry of input data

In the first tab (Figure 21), entry conditions have to be specified:

- UTC date
- Fragmentation altitude
- Orbital configuration :
  - Apogee + Perigee
    (or Semi major axis + Eccentricity)
  - Argument of the Perigee
  - Inclination angle
  - Right ascension of ascending node
  - Anomaly (true, mean or eccentric)

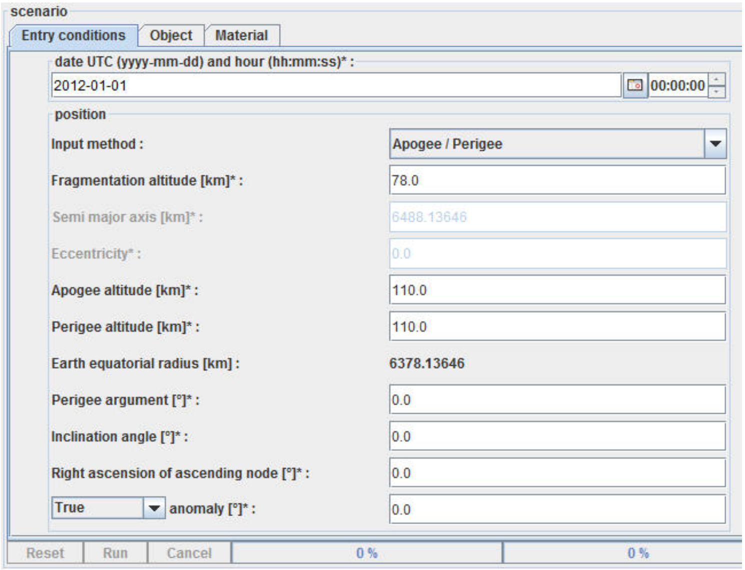


Figure 21: Mission tab

The second tab (Figure 22) is dedicated to system's geometries. In a first time, the spacecraft has to be defined with a cylinder or a box object. It will be the father of all subsystems. Then it is possible to define component by adding element like box, cylinder, sphere or plate. Kinship is defined in the mean time when the object is created. The material has to be declared for each element because the software is verifying the consistency of input. For example it is impossible to have a mass bigger than volume time density.

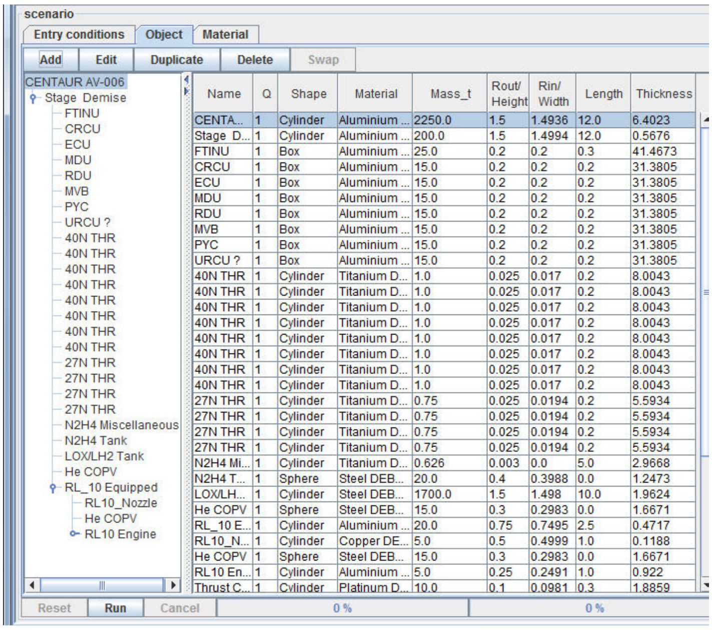


Figure 22: Geometry tab

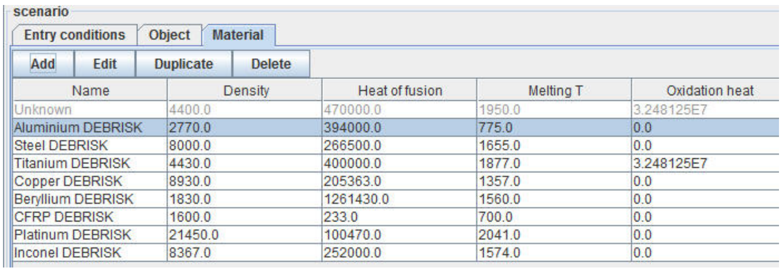


Figure 23: Material tab

The last tab (Figure 23) is the material tab where it is possible to add, edit or delete material. Each material card is containing the following information:

- Name
- Density
- Heat of fusion
- Melting temperature
- Oxidation heat (for material having an oxide layer)
- Emissivity (constant or function of temperature)
- Heat capacity (constant or function of temperature)

The process can be launched by clicking on run button on the lower left corner. The software is not calculating the solution if the spacecraft does not reach ground under 10,000 s.

Results and post-treatment

If the calculation converges, a graph of altitude versus time will appear in the system/geometry tab (Figure 24). After running the case it is possible to see the evolution of altitude, thermal mass and the wall temperature as function of time. It is also possible to show a chart of the thermal mass versus altitude and finally altitude as function of downrange.

Debrisk can export results under KLM extension file to view the results in 3 dimensions thanks to Google Earth. It is also possible to get all the steps results under a DAT extension file, and then with a macro it is possible to visualize data on Excel.

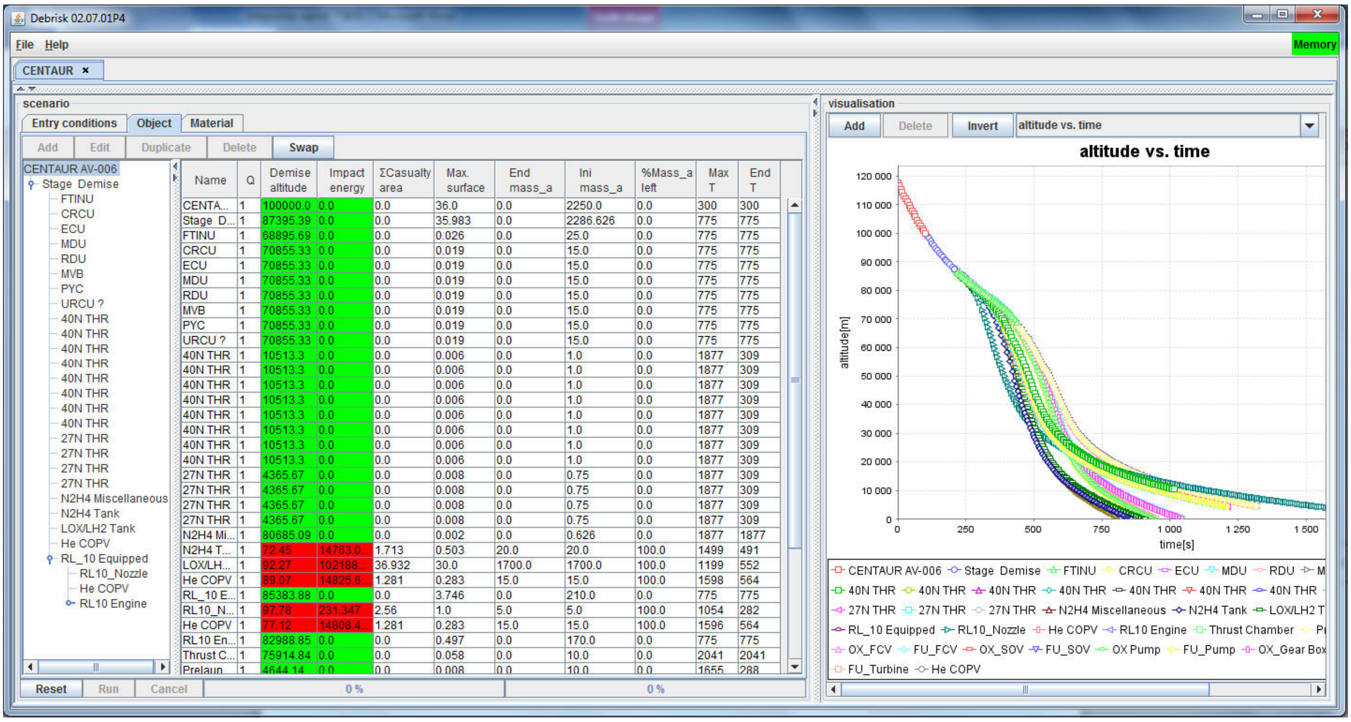


Figure 24: Result overview

IV. Design for Demise methodology

1. Context

In the 60's satellites were designed only to fulfil their missions regardless of what they will become when their work is completed. Nowadays the numbers of dead satellites, debris from space projects and collisions have created a new issue: they are compromising the missions of other spacecraft. In addition, their presences are making it compulsory to monitor debris orbit and to make avoidance manoeuvres which implies additional ground monitoring and propellant budget. Statistically, if nothing change and space activities keep going on, the debris population will increase drastically making any future launch impossible, even if atmospheric drag is breaking down low orbit debris into atmosphere where they will disappear.

Design for Demise is the intentional design of space system hardware such that it will ablate (demise) upon atmospheric re-entry. This design methodology is especially important for uncontrolled re-entries, which could pose a hazard to public safety, both on ground, air and sea.

The goal of design for demise, in aerospace industry, is to reduce the ground risk for Earth's population. In this chapter, the design for demise methodology presented will be focused on re-entry aspect.

2. Theoretical aspects

There are not a lot of parameters which can be modified to increase the demise of a system or component, ablation drivers are listed hereafter:

- Trajectory
- Geometry
- Material
- Assembly technology

Trajectory

Some trajectories can increase the demise by using better flightpath angle which will optimize the deceleration thus aerothermal flux and consequently thermal ablation.

The velocity is directly related to orbital parameter and the orientation of the velocity vector can increase the relative speed if the orbit is retrograde (because of the wind). Inclination of the orbit is also important because the Earth atmosphere has an ellipsoid shape (due to oblateness of the Earth) thus aero-breaking is happening at higher altitude close to the equatorial plane.

Designing a re-entry trajectory can also reduce drastically the casualty risk by landing the spacecraft into an ocean. In this case the procedure is called semi-controlled re-entry but it implies an additional thrust budget. This type of solution will not be described in this report.

Geometry

Mass, drag coefficient and cross section are functions of the geometry; consequently it is possible to adapt the shape of a component to increase ablation. Geometry and trajectory are linked by the ballistic coefficient. Lighter structure gives less mass to ablate or less kinetic energy on ground. In addition an airy structure permits to expose to ablation elements behind the part.

Material

Material properties impact a lot the demise of a component because the geometry and ablation depends on it. Ablation is a thermal phenomenon which appears when part receives enough energy to switch from solid to liquid state.

$$Q_{demise} = \int_{T_{initial}}^{T_m} C_p dT + Q_m \tag{26}$$

Consequently the following parameters are involved in this thermal phenomenon:

- **Melting temperature, T_m** , temperature at which a material turns abruptly from solid to liquid.
- **Specific heat capacity, C_p** , energy required to raise by one degree the temperature of 1 kg of material.
- **Specific heat of fusion, Q_m** , energy absorbed by solid on melting at melting temperature per mass unit.
- **Emissivity, ε** , define the efficiency of a surface emitting as thermal radiation thus the radiative flux
- **Thermal conductivity, k,** drive the distribution of heat inside the part

Except thermal conductivity, the lower the parameters are the better the ablation will be. A good thermal conductivity permits to warm up connected part, without having a big exposure.

Assembly technology

Usually mechanical parts are attached together with some bolt and nuts or fastener or even soldered. But those assembly parts are most of time made with high strength material such as steel making the disintegration difficult. It is possible to act directly on the breakup of a satellite during the re-entry phase by using special devices or technologies to separate part from each other at a desired time.

This type of devices can be remote controlled, electrically controlled by the on-board computer or self-activated by temperature, pressure or any other ambient condition.

It is possible to summarize design for demise methodology with following chart:

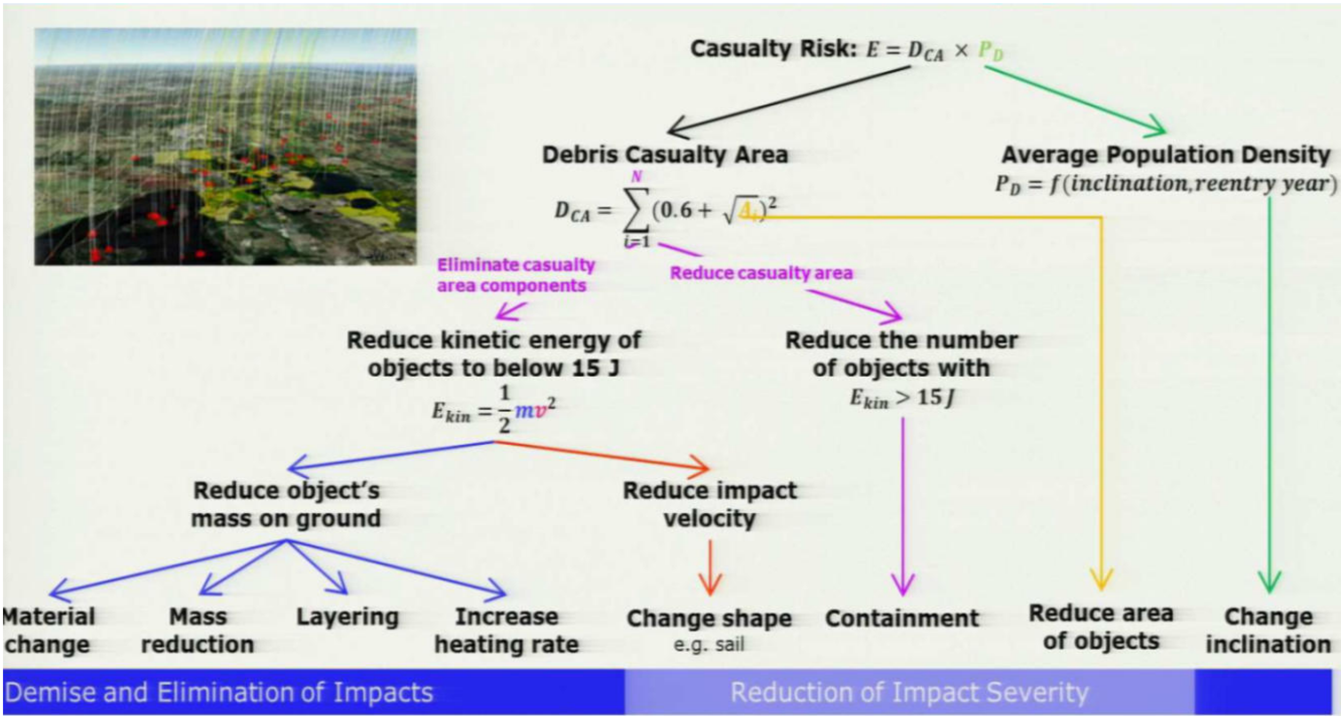


Figure 25: Diagram of design for demise methodology
(Source: ESA, D4D MiCRA CDF study", CDF team, ESA, 2013)

3. Technical solutions: industrial examples

Example of geometry solution:

- **3D printing for structural aluminium support of antenna:**

  A simple structure has been lightened to save mass (by optimizing stiffness) and increase ablation exposure.

Figure 26: 3D printed antenna bracket for Sentinel-1-satellite. (www.ruag.com, 2016)

- **Multilayer balance mass in tungsten:**

  It is common to put counter weight boxes on a satellite to balance it and often the material used is tungsten or steel (because of high density). Those two metals have high melting temperatures making them difficult to demise/melt. Instead of using one block of metal, the counter weight has been made in numerous extra thin sheet of metal. It allows reducing the kinetic energy of this part during re-entry under the threshold of 15 J. Consequently it is not anymore a problem if this part reaches the ground after re-entry because it will not injure someone.

Example of material swaps solution:

- Aluminium harness: Usually harnesses are in copper because of good electrical properties of the copper. But there are major drawbacks to use copper because its melting temperature and its density are high. In addition insulation is retarding the ablation. Consequently it has been studied to replace copper harness by aluminium to save mass and increase ablation rate.

	SP Aluminium	SP Copper
Dimensions	AWG 40 (solid) to AWG 6 (stranded)	AWG 50 to 4
Conductivity (% IACS)	63	100
Tensile strength (MPa)	150	240
Temperature rating (°C)	150	200
Density	2.7	8.89
Termination techniques	soldering / crimping	soldering / crimping
Weight saving	50% to 60%	

Figure 27 : Table of properties comparison between Aluminium and Copper
(source Axalon©, "ESA wires & cables and Awalu®" aluminium wires", 12/2014)

Example mechanical assembly part:

Mechanical assembly can be designed to demise under special condition such as temperature or electrical passive or active actuation. Dismantlement and release devices are detailed in chapter 7.

V. Demise improvement of reaction wheels

1. ITT Context

Preliminary analyses at ESA have shown that space objects in LEO with masses above 500 kg might already imply an on-ground casualty risk higher than 10^{-4} in case of uncontrolled re-entry. Compliance to the 10^{-4} casualty risk requirement may be achieved through controlled re-entry, but this solution has a major impact at system level, sometimes requiring the full re-design of the spacecraft and may involve switching to a completely different launcher (consequently a significant mission cost impact).

The need for a controlled re-entry may be avoided through design options aimed at achieving complete ablation (demise) of the spacecraft upon uncontrolled atmospheric re-entry. Design-for-Demise (D4D) is a highly multidisciplinary approach that can bring significant benefits in the future missions in the medium to long term.

At NASA, Global Precipitation Measurement (GPM) mission launched early 2015 was decided to be the first fully designed for demise LEO mission. The core spacecraft is similar to the highly successful Tropical Rainfall Measuring Mission (TRMM) spacecraft with a mass approximately of 3,850 kg. Reaction wheels where redesigned with a new demise-able material for the wheel rim replacing the original stainless steel.

ESA created in 2012 the “CleanSpace” initiative and team to promote actions on green aspects and debris remediation. Recently ALTRAN was involved inside THALES ALENIA Space consortium in ESA activities on S/C D4D techniques and proposed successfully several D4D concepts. This ESA D4D study had the objective to find D4D solutions for the Sentinel-1 study case (around 2 tons). This objective were about to be achieved.

The outcomes of those D4D studies performed in parallel at 3 LSI level (Large Single Integrator) using mainly ESA Sentinels as study cases , demonstrated all that Reaction Wheels are a major contributor to a S/C Debris Casualty Area (DCA) and an item to be managed in a D4D approach with high priority.

2. Study problematic and objectives:

a. Function and composition:

A flywheel is an attitude control actuator which is used to orient the spacecraft using third law of Newton: an action gives an opposite reaction. By spinning the wheel the motor creates a momentum on the wheel, equal to the product of the rotation rate time the inertia of the rotating mass, consequently an angular momentum is created by the wheel on the motor thus on the satellite. The acceleration/deceleration of a reaction wheel creates torque acting on the momentum vector of the satellite. For three axes of torques, three wheels are necessary, but usually there are four reaction wheels for redundancy.

Figure 28 : Reaction wheel RSI68-75/60 (top) and RSI45-75/60 (bottom) made by RCD - GmBH (Source: www.rockwellcollins.com, 2016)

In Attitude and Orbit Control System, momentum is used to stabilize a satellite and torque to orientate/change properly the spacecraft attitude.

A flywheel is commonly composed of a spinning flywheel rim driven by a motor and some electronic sheet which control the speed. It is often sealed in an airtight housing to protect the device from electromagnetic disturbance and from space radiations.

The bearing unit is one of the most significant and critical assembly in the momentum wheel and cannot be designed with redundancy. Two angular contact ball bearings are mounted in a back-to-back arrangement and are solidly preloaded by the inner and outer spacer sleeves.

b. Demise problem:

The principle of this actuator is based on inertia to avoid using high speed motor it is more convenient to have a heavy wheel. Consequently steel is one of the best candidates for the compromise between weight, volume and stiffness. However it causes a major problem for the demise of the component because the melting temperature of steel is high (1,750 K). According to simulations made by space integrators and agencies, it has been admitted that this component has to be improved on demise aspect because it represents a major contributor on debris casualty area. Consequently ESA are financing studies to encourage improvements on design from manufacturers.

c. Study objectives

The presented study had to investigate the D4D techniques identified by ALTRAN and their suitability to be implemented in current Rockwell Collins Deutschland (RCD) designs (on RSI68 model) to render the larger end of wheel family more demiseable on satellite re-entry.

Baselined options initially presented by the consortium and fully substantiated are:

Option 1: Mechanical Upgrade with intention to use iso-inertia flywheel with demise material

Changing/modifying flywheel only

a. Material change
b. Shape change (with holes to increase internal heat exposure, etc.)

Option 2: Electronics upgrade with intention to use down-sized design with increased speed and torque capability

Additive Options requested by Customer and addressed by the consortium in a best effort approach:

Option 3: Additional solutions

c. Changing/working on the core
    i. Replacement of materials in the core
    ii. Options to dismantle early the internal parts
    iii. Smaller core with flywheel clamping system for launch loads
    iv. Separation of RW electronics
d. Magnetic bearings

Altran was in charge of studying option 1 and propose possible additional solutions to early dismantle of internal part of RCD reaction wheel. In the meantime RCD has to investigate electronic and ball bearing unit upgrade.

d. Design constraints

- **Functional constraints:**

To fulfil its function the flywheel has to have certain momentum of inertia of rotating mass to produce torque when the motor is spinning the rim. It has to produce 20 Nms of momentum and 200 mNm of torque. According on RSI68 datasheet of RCD [RD25] it is obtainable at around 1800-2000 RPM (Inertia around 100 000 kg/mm²).

- **Consortium constraints**

In addition the proposal has to compromise the expectations of a consortium of clients made of Airbus Defence and Space, OHB, Thales Alenia Space and ESA. Altran and RCD proposed to try to redesign a reaction wheel in aluminium. The consortium agreed on trying to design a wheel with a maximum weight not higher than current version with, if it is possible, an equal momentum of inertia while not changing other relative part composing the reaction wheel (ball bearing unit and housing case unchanged) in order to limit the impact on the system.

- **Qualification constraints:**

To ensure that our aluminium flywheel design proposal can replace the steel original wheel, we have to demonstrate that it can sustain the same static and dynamic loads.

The first eigen-frequency of the rotating mass has to be slightly above 100 Hz to avoid any combined effect during the launch. The basic design rule is to take a safety margin by taking a first eigen-frequency $\sqrt{2}$ times bigger than the launcher one (around 80 Hz).

Concerning the load solicitation, the maximum admissible stress is defined by a safety margin of 1.5 compared to the yield limit. The flywheel has to be qualified for a maximum rotational spin rate of 6,000 rpm including safety factor and quasistatic/design loads of 20g at launch with safety margin [RD25].

3. Altran participation

a. Mechanical and Finite Element Method analyses

CAD models

Based on the manufacturer data, 3D models have been made to explore shape possibilities and consequences on mechanical properties.

- In a first step, a flywheel has been made with an external diameter comparable to RSI68 with a larger height for the same weight.
  According to NASA models and RSI45 designs, another model have been made with a lightweight web to save mass from the web and use it to increase inertia in the rim. Inertia has been increased with a larger external diameter.
- After concertation with the consortium of Large Single Integrator, it has been decided to limit the external diameter and the height of the flywheel to keep the same housing case in order to avoid designing and qualifying a new one. Consequently the thickness of the rim has been increased to tend to the same inertia with the same weight in the same volume. Thus if there is an acceptable loss of inertia, it will be balanced with a little increase of the rotational speed.

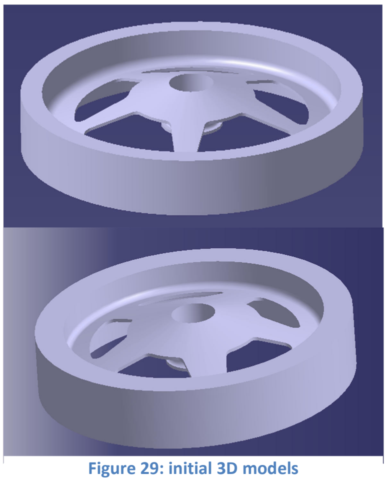


Figure 29: initial 3D models

After few FEM simulations it has been conclude that centrifugal test at 7,500 rpm is giving the designing maximum stress. For modal analysis, first mode appears to be rocking mode around radial axis, perpendicular to rotation axis.

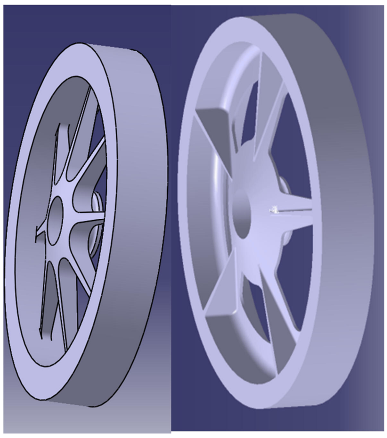


Figure 30 : 3D model optimized for reduction of mechanical stresses.

To reduce the maximum stress during centrifugal test it is necessary to increase section area of web/spoke (radial section). Eigen-frequencies are proportional to specific stiffness (E/ρ) and second moment of area of radial section (especially height). Consequently, models geometries have been improved following those tendencies. Models with optimize ratio of second moment of area /section area have been investigated.

FEM model and modal analysis

The design requires that equipment has a lowest eigen-frequency above 100 Hz in order to avoid any coupling effect during the launch phase. Eigen-frequencies are difficult to predict, however it is possible to estimate them by modelling the part in 3D by making a modal analysis with Catia.

Based on manufacturer data about component qualification and usual space test, the following analyses have been made:

- Mapping 1g axial (Z)
- Mapping 1g radial (X or Y)
- Centrifugal at 7,500 rpm
- Modal analysis

Mappings with uniform 1g (9.81m/s²) loads have been made to found stress maximums inside the model and discover the proportional stress for a 44g test for example.

Boundary condition:

The flywheel has been simulated with a pivot on the hub, the rest of the wheel is free.

- Example for a flywheel with a lightweight web:

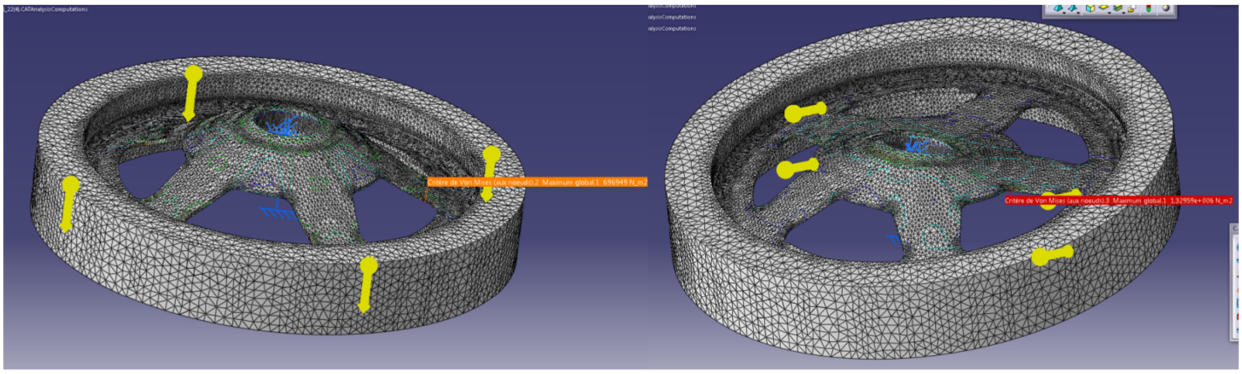


Figure 31: Mapping 1g axial and radial

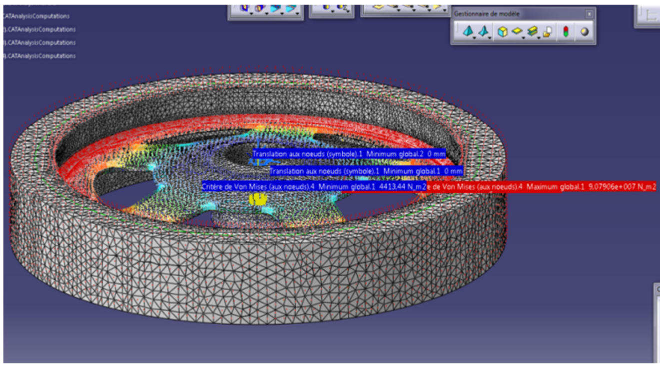


Figure 32: Centrifugal test at 7500rpm

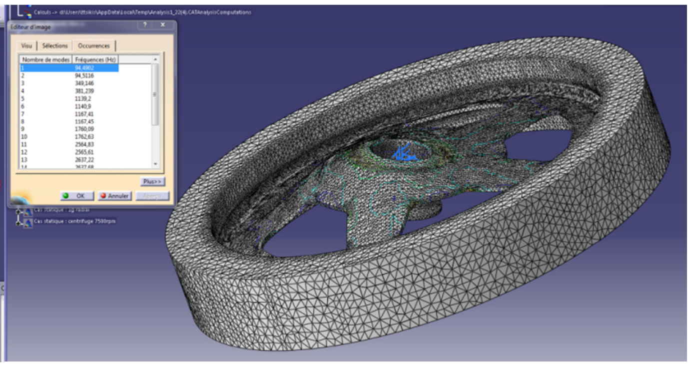


Figure 33: Modal analyze

First outcomes of FEM analysis:

The first eigen-mode (rocking mode: radial rotation) appears to be designing if the first frequency needs to be largely above 100 Hz. The centrifugal stress is significantly above stress maximum of a 30 or even 40g test, consequently the radial section has to be design in function of it.

Eigen-frequencies basics:

It is convenient to know that there are some equations giving eigen-frequency as function of material, geometry and type of load: ω is the angular rate $\omega = 2\pi\, f$

- In traction/compression for a clamped/free beam eigen-frequency is founded with :

$$\boxed{f = \frac{\omega}{2\pi} = \frac{1}{2\pi}\frac{(2n-1)\pi}{2L}\sqrt{\frac{E}{\rho}}}\text{ , n=1, 2, 3 ...} \tag{27}$$

Thus f is proportional to the traction wave propagation velocity $\sqrt{\frac{E}{\rho}}$ and to $\frac{1}{L}$ (geometry)

- In flexion for a clamped/free beam the eigen-frequencies are:

$$\boxed{f = \frac{\omega}{2\pi} = \frac{1}{2\pi}n^2\pi^2\sqrt{\frac{E}{\rho}\frac{1}{L^2}}\sqrt{\frac{I}{A}} = \frac{1}{2}n^2\pi\sqrt{\frac{E}{\rho}\frac{1}{L^2}}\sqrt{\frac{I}{A}}} \tag{28}$$

(n^2 is an order term because there are several solutions)

For a rectangular section: $I = \frac{bh^3}{12}$ then $f = \frac{1}{2}n^2\pi\sqrt{\frac{E}{\rho}\frac{1}{L^2}}\sqrt{\frac{bh^3}{12bh}} = \frac{1}{2}n^2\pi\sqrt{\frac{E}{\rho}\frac{1}{L^2}}\sqrt{\frac{h^2}{12}}$

For a circular section: $I = \frac{\pi D^4}{64}$ then $f = \frac{1}{2}n^2\pi\sqrt{\frac{E}{\rho}\frac{1}{L^2}}\sqrt{\frac{\pi D^4}{64\pi D^2}} = \frac{1}{2}n^2\pi\sqrt{\frac{E}{\rho}\frac{1}{L^2}}\sqrt{\frac{D^2}{64}}$

Thus f is proportional to $\sqrt{\frac{E}{\rho}}$ (material) and to $\frac{h}{L^2}$ or $\frac{D}{L^2}$ (geometry)

- In torsion for a clamped/free beam the eigen-frequencies are:

$$\boxed{f = \frac{1}{2\pi}\frac{(2n-1)\pi}{L}\pi\sqrt{\frac{G}{\rho}} = \frac{(2n-1)}{4L}\sqrt{\frac{E}{\rho(1-\upsilon)}}}\ , \tag{29}$$

($n$is an order term because there are several)

$\sqrt{\frac{G}{\rho}}$ is the torsion wave propagation velocity, with G the shear modulus. Thus f is proportional to $\sqrt{\frac{E}{\rho}}$ (material), to $\frac{1}{L}$ (geometry) and to $\sqrt{\frac{1}{1-\upsilon}}$ (material).

Thus it is possible to increase eigen-frequencies by choosing a material having a higher specific stiffness (E/ρ) or/and adapt a cross section area in order to increase second moment of area (for flexion).

b. Material selection

CES Edupack software

The software editor Granta design Ltd is commercializing one of the reference software in the domain of material database and material selection. The software is called CES Selector in its pay version, but it exist an education version which can be used by universities with free use rright for students: CES Edupack.

This software works simply like an enormous database of material cards containing all the properties and also all the manufacturing process data. It can be used to search for a specific material based on its name or by making a process of selection by making a list of criteria.

There are three types of criteria:

- Tree criteria: selection is made based on a folder corresponding to a family of materials (metal, ceramic, polymer,…) or process, etc. Many folders can be chosen, for example metals, and ceramic materials.
- Limit criteria: with this option it is possible to set minimum or/and maximum on properties values such as Young's modulus for example. It is like a range selection.
- Graphic criteria: this function allows you to graphically trace properties or function of properties. Then it is possible to select material as function of their position on the graph. Results can be excluded by tracing line representing criteria. For example you show Young's modulus versus density and trace a line corresponding to a certain ratio (specific stiffness)

Material constraints and selection process

To upgrade the demise of the reaction wheel RSI68, initially in stainless steel, by changing of material a material selection/analysis has to be conducted.

As seen in the chapter on demise methodology, material can be changed to lower the required energy to demise the flywheel. Consequently, melting temperature, heat capacity and melting heat can be reduced. Experience has shown that the demise heat has to be less than 1000 kJ/kg with a melting point of at most 1,500K.

$$Q_{demise} = \int_{T_{initial}}^{T_m} C_p dT + Q_m \qquad (30)$$

In the meantime, mechanical properties and geometry have to be chosen to ensure the function of the part while passing the qualification requirement (cf chapter V section 2)

The material has to allow machining process to ensure a perfect balancing of the wheel on a lathe.

Heavy metals, radioactive and toxic materials have been excluded due to atmospheric hazard which are prohibited by ESA. The list of constraint conducting to possible solutions can be sum up:

1) Machining compatibility
2) 0% of As,Be, Cd, Hg, Pb, PbO, U and V
3) Metal and alloys material
4) Density > 2500 kg/m^3
5) Young's Modulus > 60GPa
6) Yield Strength > 150MPa
7) Graphic $T_m = f(E)$

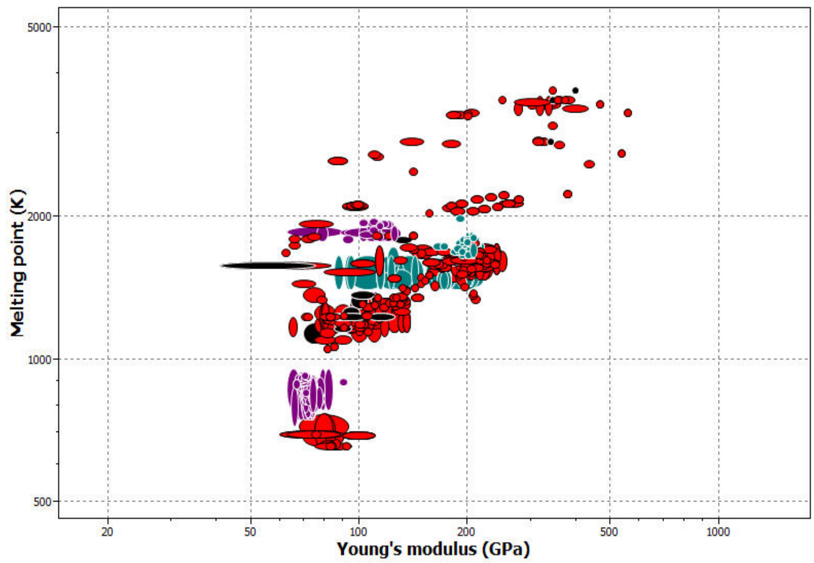
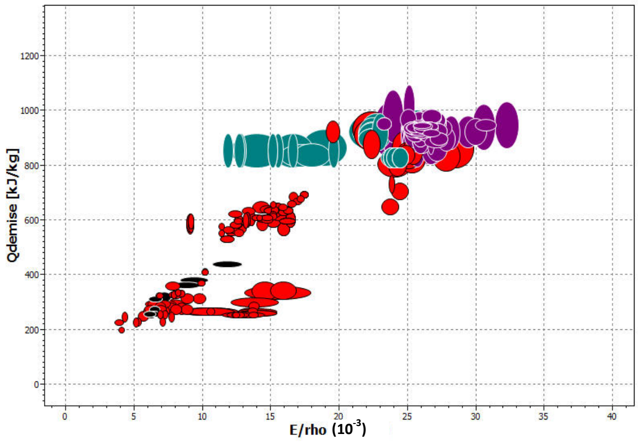


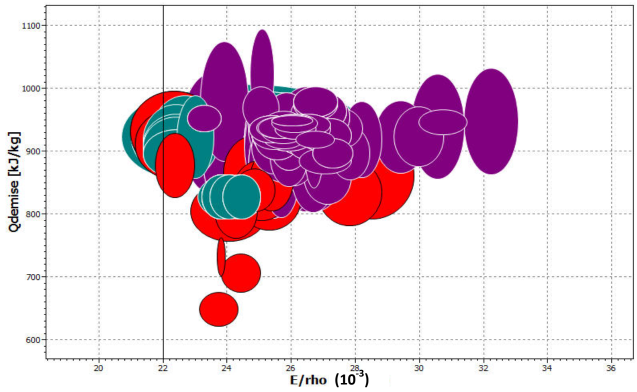
Figure 34: Material compliant with criterion until 6)
(Source: CES Edupack 2013)

8) Melting temperature < 1,500K



9) Graphic $Q_{demise} = f(\frac{E}{\rho} * 1000)$

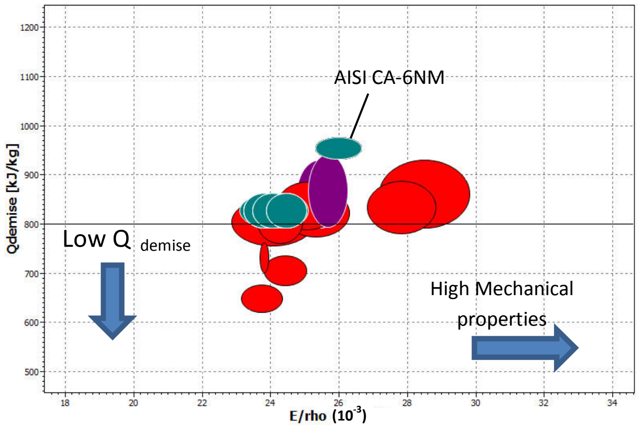
Figure 35: Material compliant with criterion until 9)
(Source : CES Edupack 2013)



10) Graphic $Q_{demise} = f(\frac{E}{\rho} * 1000)$

+ limit line $\frac{E}{\rho} \geq 22 \text{ x } 10^{-3}$ (NB: $\frac{E}{\rho_{steel}} = 26 \text{ x } 10^{-3}$)

Figure 36: Material compliant with criterion until 10)
(Source: CES Edupack 2013)



11) Graphic $Q_{demise} = f(\frac{E}{\rho} * 1000)$

+ limit line $Q_{demise} \leq 800$ kJ/kg

(-15% compared to AISI CA-6NM)

Figure 37: Material compliant with criterion until 11)
(Source: CES Edupack 2013)

12) Graphic $T_m = f\left(\frac{E}{\rho} * 1000\right)$

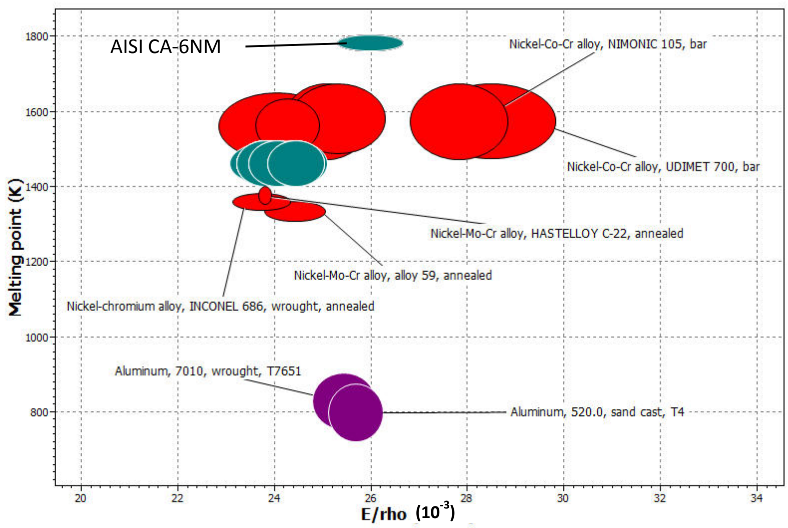


Figure 38: Material compliant with criterion until 12)
(Source: CES Edupack 2013)

	Stainless Steel (ASTM CA-6NM)	Aluminium 7010 T7651	Aluminium 520.0 T4	NiMoCr, alloy 59	HASTELLOY C-22	INCONEL 686
Density [kg/m^3]	7450 - 7600	2810-2840	2550-2600	8600	8630 - 8710	8720 - 8730
Young's Modulus [GPa]	196 - 204	70-73.6	64.7-67.3	205-215	206 - 207	202 - 212
Yield strength [MPa]	370 - 460	365-455	176-193	323-383	310 - 382	382 - 438
Melting point [K]	1763 - 1783	753-903	723-873	973-1070	1350 - 1400	1340 - 1380
Specific Heat [J/kg.K]	450 – 470	879-915	944-982	406-422	407 - 421	366 - 380
Heat of fusion [kJ/kg]	260 - 285	384-393	384-393	252-305	259 - 314	230 - 278

Table 4: Synthesis of potential material solutions
(Source : CES Edupack 2013)

Selected materials synthesis

Consequences on the flywheel:

	Stainless Steel (ASTM CA-6NM)	Aluminum 7010 T7651	Aluminum 520.0 T4	NiMoCr, alloy 59	HASTELLOY C-22	INCONEL 686
Qdemise [kJ/kg] (min)	918.350	694.287	783.312	679.238	649.52	566.74
E/ρ *10^{-3} (min)	26,3	24,9-	25,3-	23,8	23,7	23,1
Rim mass min	3.54	3.97	4.07	3.51	3.51	3.51

Table 5 : Synthesis of material swap impact
(Source : CES Edupack 2013 and calculation)

High density give a lighter rim but for the same stiffness (quadratic moment) the web will be heavier. To compromise a high first eigen-frequency and a low total mass, a quite high specific stiffness has to be chosen

Aluminum alloys :

Present good specific stiffness with almost no changes from steel, but low melting point and medium high heat capacity give a lower Q_{demise} which lead to a potential gain of 15% of energy required to melt 1 kg of matter.

Nickel alloys:

Have a sufficient specific stiffness to be considered as potential solution, even with a loss of 10%, but have a great thermal behaviour : very low Q_{demise} giving a energy gain of 30%.

c. Reentry analysis

To confirm the coherency of demise design, a re-entry analysis has been conducted with the software Debrisk.

Height	Housing case RSI 68 Steel Mass nom	Housing case RSI 68 Alu Mass nom	Housing case RSI 45 Alu	Housing case RSI 12 Alu	Housing case RSI 68 rim 345 house 360	Housing case RSI 68 rim 355 house 370	Housing case RSI 68 Cu Mass nom	Housing case RSI 68 Inconel Mass nom
100 000	22.290	0	0	0	0	0	0	0
95 000	22.512	0	0	0	0	0	0	0
90 000	22.025	0	0	0	0	0	0	0
85 000	19.729	0	0	0	0	0	0	0
80 000	15.551	0	0	0	0	0	0	0
78 000	16.020	0	0	0	0	0	0	0
75 000	18.513	0	0	0	0	0	0	0
74 000	19.742	0	0	0	0	0	0	0
73 000	21.328	0.376	0	0	0.394	0.412	0	0.376
72 000	23.154	0.664	0.360	0	0.699	0.723	0.436	0.652
71 000	25.240	1.184	0.638	0	1.238	1.274	0.753	1.154
70 000	27.548	1.992	1.153	0	2.057	2.117	1.292	1.974

Table 6: Percentage of remaining mass (non-ablated mass in kg) for several breaking altitude (Analysis conducted with Debrisk version 2015)

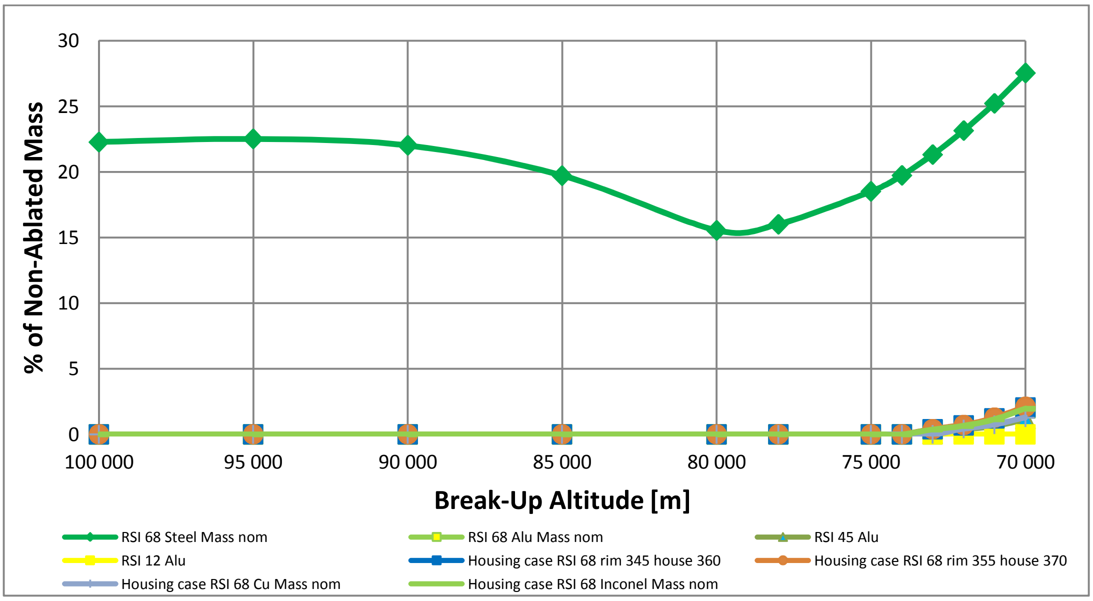


Figure 39 : Reaction wheel percentage of remaining mass in function of fragmentation altitude

	RSI 68 Alu Mass nom	RSI 45 Alu	RSI 12 Alu	RSI 68 rim 345mm house 360mm	RSI 68 rim 355mm house 370mm	RSI 68 Cu Mass nom	RSI 68 Inconel Mass nom
100 000	94 140	82 538	84 490	82 918	83 255	75 903	79 410
95 000	81 941	82 176	84 023	82 409	82 711	76 166	79 461
90 000	80 377	80 665	82 342	80 774	81 028	76 498	78 784
85 000	77 538	77 854	79 359	77 868	78 081	75 946	76 738
80 000	73 655	73 994	75 403	73 943	74 129	73 404	73 316
78 000	71 887	72 238	73 633	72 167	72 347	71 920	71 659
75 000	69 048	69 424	70 831	69 321	69 496	69 390	68 939
74 000	68 050	68 436	69 861	68 326	68 499	68 460	67 962
73 000	67 028	67 427	68 872	67 305	67 478	67 504	66 965
72 000	65 979	66 392	67 867	66 257	66 432	66 515	65 934
71 000	64 902	65 332	66 846	65 179	65 359	65 491	64 872
70 000	63 789	64 252	65 805	64 079	64 256	64 441	63 753

Table 7 : Demise altitude (complete disintegration) for several breaking altitude (Analysis conducted with Debrisk version 2015)

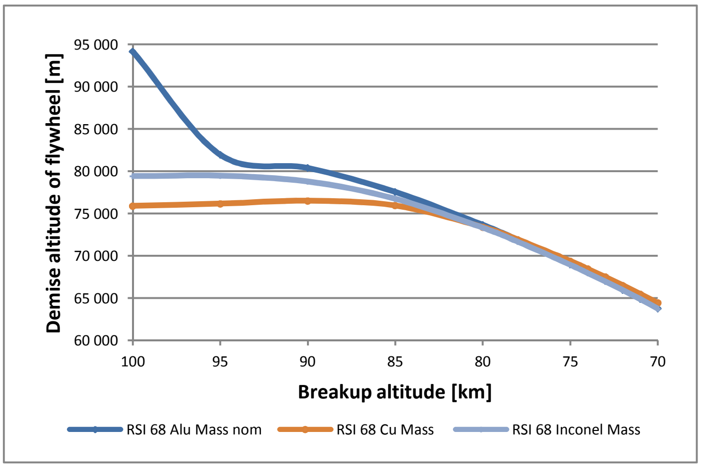


Figure 40: Fly wheel rim demise altitude as function of fragmentation altitude

Height [m]	RSI 68 Alu Mass nom	RSI 45 Alu	RSI 12 Alu	RSI 68 rim 345mm house 360mm	RSI 68 rim 355mm house 370mm	RSI 68 Cu Mass nom	RSI 68 Inconel Mass nom
100 000	61 491	61 766	64 267	61 518	61 532	57 320	59 893
95 000	61 532	61 879	64 491	61 555	61 566	57 548	60 082
90 000	61 174	61 601	64 323	61 193	61 204	58 011	60 055
85 000	59 990	60 550	63 385	60 001	60 009	58 237	59 262
80 000	57 420	58 239	61 482	57 410	57 404	56 916	57 011
78 000	29 314	56 833	60 394	29 048	28 876	28 772	28 017
75 000	14 388	21 133	58 348	13 925	13 587	17 079	13 992
74 000	7 497	15 682	57 501	6 779	6 221	11 790	7 111
73 000	0	9 410	27 600	0	0	4 483	0
72 000	0	0	23 420	0	0	0	0
71 000	0	0	18 829	0	0	0	0
70 000	0	0	13 474	0	0	0	0

Table 8: Ball Bearing Unit demise altitude for several breakup altitudes
(Source: Analysis conducted with Debrisk version 2015)

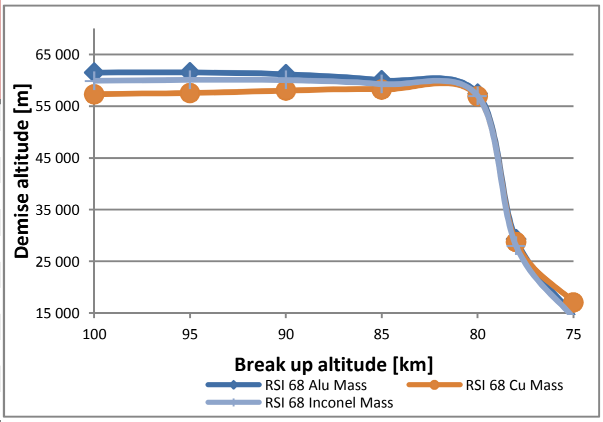


Figure 41: Ball bearing unit demise altitude in function of fragmentation altitude

Material conclusion:

A stainless steel flywheel does not manage to demise for every break up altitude when other models (RSI45 and RSI12 in aluminium) demise greatly. As expected from material selection, aluminium, copper and nickel alloys, chosen for replacement on flywheel, has almost the same thermal behaviour during re-entry. With a lower melting point aluminium appears to be the best choice between those three materials. It confirms that aluminium is a relevant choice of material swap to improve disintegration of reaction wheel.

d. Conclusion:

After having presented Altran's and RCD solutions during the Concurrent Design Facility number 2 at ESTEC (Technical center of ESA) the following conclusions have been drawn. Aluminium flywheel design appears to be a valuable solution but coupled simulation with ball bearing unit still has to be done to ensure no impact on other component of the reaction wheel. However practical test will have to be performed to verify that the high speed and life duration of ball bearing unit is not affected by this change.

Concerning RCD solution, of changing the motor to increase rotational velocity, the same conclusion applies: real test will have to be conducted to verify the consequences on the ball bearing unit (BBU).

The BBU is the key part of reaction wheel because it allows the flywheel to spin at high rates for a very long time. Unfortunately the design of this part is really complicated because it involved material, surface quality and thermomechanical cycling/fatigue. Consequently it takes many years even decades to design and test a ball bearing for such applications.

VI. Study on demise ability of optical payloads

1. Introduction on the studied component

This section is about demise survey on optical payload ordered by Thales. The purpose is to analysis demise behaviour of components and identify problematic materials. Then, several payload configurations have been studied in order to propose material to improve the demise ability of those satellites.

In space engineering, optical equipment is requiring very exigent positioning to ensure right focal point and accurate measurements. Consequently component materials must have good thermal behaviour especially very low coefficients of thermal expansion (CTE). Optical payload parts can be sorted into two big categories: optical part and structural component.

The role of structural part is to put and maintain in position optical part, consequently they need a low coefficient of thermal expansion coupled with high mechanical properties such as Young's modulus and yield strength. It is common to investigate material on their specific stiffness which is their Young's modulus divided by their density.

For optical parts, mechanical properties are the last problem but CTE is critically important, close to zero even sometimes negative to compensate a positive expansion. A good thermal conductivity is a plus to reduce strain due to thermal gradient in a part. In fact, if the mirror substrate bend or deform itself, due to temperature or a thermal gradient, huge error can be made on measurement.

2. Material and demise survey

a. Material list

Before getting into details, it can be useful to make a short reminder about material denomination, usually materials are sort in few categories:

- **Metallic alloys:** bi-compound mixture composed of a metal and another element (metallic or not), the mixture may be a solid solution of both compound (single phase alloy) or a mixture of phase with different composition and properties.
- **Ceramic:** an article having a vitreous body or not, crystalline or partly crystalline structure, or of glass. The body is formed of essentially non-metallic inorganic substances, which are formed by a melt which solidifies on cooling, or formed and brought to maturity at the same time or subsequently, by the action of heat.
- **Composite material** (also called a composition material or shortened to composite): is a material made from two or more constituent materials with significantly different physical or chemical properties that, when combined, produce a material with characteristics different from the individual component. The individual components remain separate and distinct within the finished structure.

Few subgroups can be distinguished:

Carbon Reinforced Polymer (CFRP): is a polymer matrix which contains carbon fiber to increase greatly the mechanical properties.

Metallic Matrix Composite (MMC): is a composite with at least two materials, one being the matrix and the other may be a different metal (alloy) or another material such ceramic (cermet). If there are more than two materials then the proper designation is **hybrid composite**.

Ceramic Matrix Composite (CMC): like MMC it is a bi-compound material with a ceramic matrix and another material as inclusion or fiber. If the second material is a ceramic fiber then the material enters in the category of ceramic fiber reinforced ceramic (CFRC).

Usually mirror substrates and lenses are made of technical ceramic glass based on oxide, carbide or nitride of silica. Those derivate have impressive thermal properties, such as very high melting temperature and low CTE. Few companies have specialized in those typical material, they have created brands like Zerodur® and Cesic®. Zerodur® (registered trade mark of Schott AG©) is a lithium alumino-silicate glass-ceramic. It is composed of ternary system made of lithium oxyde (LiO_2), alumina (Al_2O_3) and silicate (SiO_2).

Cesic®(registered trade mark of ECM is a carbon-fiber reinforced silicon carbide, which consists of a matrix of silicon carbide (SiC) reinforced with microscopic carbon fibers of various compositions and lengths.

b. Potential unfriendly to demise materials

Titanium is known for having poor demise behaviour due to is high Q_{demise}, in addition titanium parts are often massive because this material is used for very high mechanical stress applications. Consequently it is most of time difficult to compromise mass efficiency with design for demise.

Most of ceramics are expected to have a very high heat of demise cause of high temperature resistance of such compound. This type of material is also complicated to replace due to their special properties.

3. Conclusion: recommendations

a. Material constraints

For optical applications, it is common to look after material with low coefficient of thermal expansion in order to reduce important deformation on component. Beside for demise capability, low demise heat and low melting point are appreciated.

b. Potential replacement material

A possible solution to replace ceramic benches could be a metallic satellite structure composed of nickel alloy continuously cooled by a cryogenic cooler system but it implies extra mass, vibrations and power budget.

VII. Release mechanisms and shape memory alloy for early breakup

1. Background of the activity

"*There is an increased attention to safeguarding Earth's orbital environment, as reflected by the number of relevant regulations that are being set forward. European Space Agency (ESA) has published an update of the Space Debris Mitigation (SDM) Policy for Agency Projects in March 2014. It is expected that in the near future more countries will adopt mandatory SDM requirements. ESA's Clean Space initiative has been promoting several engineering activities and technology developments to support the compliance with the SDM requirements, hence improving the competitiveness of European industry in the global market.*

Satellites and rocket bodies operating in low Earth orbit (LEO) are the most impacted by the SDM requirements due to the de-orbit and re-entry requirements. Future ESA satellites and launcher upper stages which will be disposed of by atmospheric re-entry at the end of their operational life must demonstrate that the risk from fragments surviving the re-entry and causing casualties on ground is less than 1 in 10,000 [AD3].

Design for Demise is the intentional design of space system hardware such that it will ablate (demise) upon atmospheric re-entry. This design methodology is especially important for uncontrolled re-entries, which could pose a hazard to public safety, both on ground, air and sea.

CleanSat is an initiative between ESA, the European Large System Integrators (LSIs) and subcontractors to coordinate European technology developments for space debris mitigation. During the CleanSat Preparation Phase the European LSIs (Airbus, Thales and OHB) have revealed that increased spacecraft demise ability is a key solution for LEO platforms to ensure compliance to the SDM requirements.

Simulations with spacecraft oriented re-entry analysis tools have shown that an early break-up of the spacecraft main structure improves the overall demise ability and reduces the casualty risk on ground. This improvement is mainly achieved because of the early exposure of demise-critical components to the aero thermodynamic forces during re-entry.

Even if a re-entry is not controlled regarding its trajectory however it is possible to include some devices to modify the dismantlement behaviour. Thus dismantlement can partially modify just by opening a structure in order to expose some part earlier and then increase the demise ability of the total system.

The demise-critical components that benefit from a structural break-up are either located inside or attached to the main structure.

Previous studies (cf. RD 2 and RD 3) have indicated that mechanisms and joints of spacecraft play a significant role in initializing the break-up event of the main structure. As a consequence, the following technologies shall be addressed in particular in the frame of this activity:

- *Passive Technologies (e.g. structural elements and mechanisms that are actuated by heat and/or mechanical loads (e.g. shape memory alloys))*
- *Active Technologies (e.g. launch locks, active release mechanisms)*
- *Demisable Joining Technologies (e.g. brackets, cleats, adhesives, bolted connections, riveted joints, welds, thermal expansion joints).*

While previous activities have provided a system level overview of the potential impact and use of the different Design for Demise techniques and technologies, this activity shall focus on these three specific technologies that are considered key technologies to increase the spacecraft demise ability by initiating an early break-up."

ESA's Invitation To Tender N°AO8632, « Multi-Disciplinary Design and Bread boarding of Technologies for Early Break-up of Spacecraft During Re-entry », 04/2016

Definitions:

Joining Technologies are components, parts or materials which connect structural elements or equipment of the spacecraft. Joining Technologies include but are not limited to brackets, cleats, bolted joints, riveted joints, bonded joints, soldered, brazed, welded and wired joints.

Demisable Joining Technologies are Joining Technologies that are designed for early demise during atmospheric re-entry to initiate the break-up of the external spacecraft structure.

Active Technologies are components which are actuated by command (during operational lifetime) before atmospheric re-entry.

Passive Technologies are components which are triggered by the environment they are exposed to during the atmospheric re-entry. Passive technologies include but are not limited to structural elements and mechanisms that are actuated by heat and/or mechanical loads.

Atmospheric re-entry refers to the re-entry of a spacecraft through the Earth's atmosphere.

Launch lock is a mechanism which intentionally locks two or more parts together providing a structural load path during launch which is released in orbit. Removing this structural path facilitates the break-up during re-entry.

Structure break-up refers to the segmentation of the spacecraft structure during atmospheric re-entry.

Structure opening refers to the partial separation of structural elements with the objective to increase the exposure of demise-critical components located inside the spacecraft to the re-entry environment.

Breadboard models are used in the development areas of new design or where substantial redesign is performed. They are applicable to every type of product and can be subjected to functional and/or environmental testing. The breadboard model is sometimes also called Development Model.

As it is mentioned in the ITT N°AO8632 on « Multi-Disciplinary Design and Bread boarding of Technologies for Early Break-up of Spacecraft During Re-entry » and in the previous chapter dedicated to design for demise, it appears obvious that breakup and dismantlement chronology can contribute a lot in ablation process during atmospheric re-entry phase. By controlling separation, it is possible to control exposure and to act directly on the dynamic of the spacecraft re-entering. In this chapter, a non-exhaustive state of the art of dismantlement and release devices is presented. Systems are sorted into two categories the active and the passive devices.

2. State of the art of active dismantlement devices

Active dismantlement/release device is the category of subsystems regrouping all the assembly which can be included in a spacecraft to performed controlled separation or/and disconnection. This type of actuator is commanded during operation lifetime either electrically either by pressure. An overview of existing technologies is made in this chapter.

a. Thermal cutter [CubeSat release mechanism]: (500°C)

Adam Thurn developed a release device for the American Naval Research Laboratory, his system is based on a NiChrome wire mounted on a pair of springs. This wire is electrically warmed up to 500°C to cut through a polymer part by being pulled by the springs thus the wire is directly in contact with the matter. This system can be adapted be modified to cut polymer pins, wire or maybe even nuts or bolts.

The system impact of such device is pretty small, it is pretty light, really cheap and required around 2 Amps. Such system has a low risk of involuntary trigger because of the need to warm up the NiCr wire to cut the polymer part.

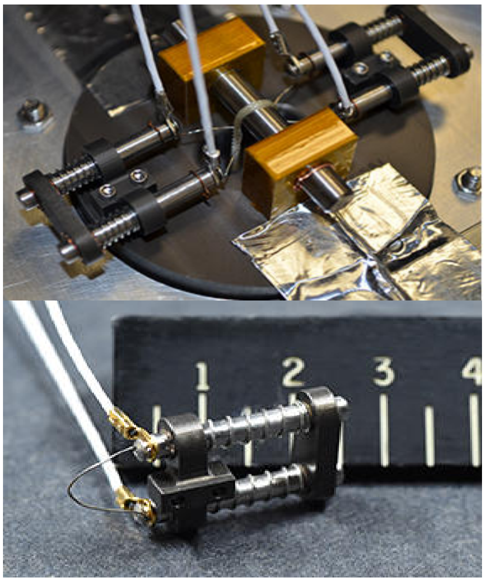


Figure 42 : Pictures of thermal cutter
(Credit : Adam Thurn, NRL,« A Nichrome Burn Wire Release Mechanism for CubeSats», 2014)

b. Fusible link (Boeing©)

The space and defense mechanisms division of Boeing© has developed a fusible link based on a Nitinol® restraint strap which fuse when the wire is powered with 30 A and 2 V. Then a leaf spring is releasing two lock arms which were retaining the mobile part of the fusible link.

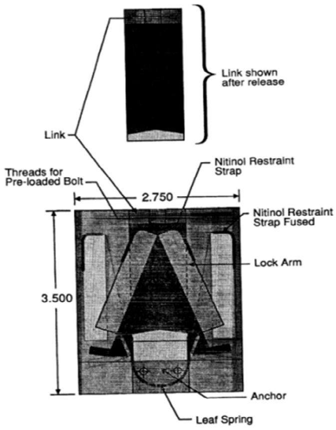


Figure 43 : Cross section view of fusible link mechanism.
(Source: Bill Purdy, NRL, «Advanced release technologies program», 1994)

c. Active Frangibolts

Thanks to shape memory alloys, it is possible to design expanding washers/braces which can break a bolt due to the axial stress induced.

Expansion comes from thermal shape memory phenomenon; initially the washer is compressed then assembly and put under tightening constraint.

When it is is warmed up, shape memory effect expand the washer to take its shape back (before compression step).

Commonly, frangibolts are sold with a thermal sleeve containing a coil to activate the SMA effect on command and an insulation layer.

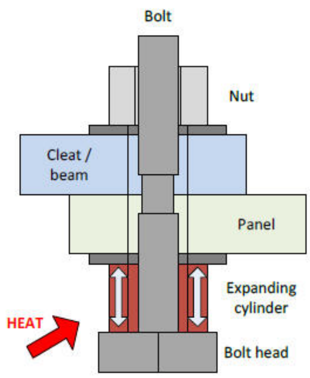


Figure 44 Frangibolt® working principle

d. Bolt release mechanisms based on SMA

A cartridge of shape memory alloy is pushing a piston against a sliding part which releases the threaded part allowing it to expand and then release the bolt screwed in it.

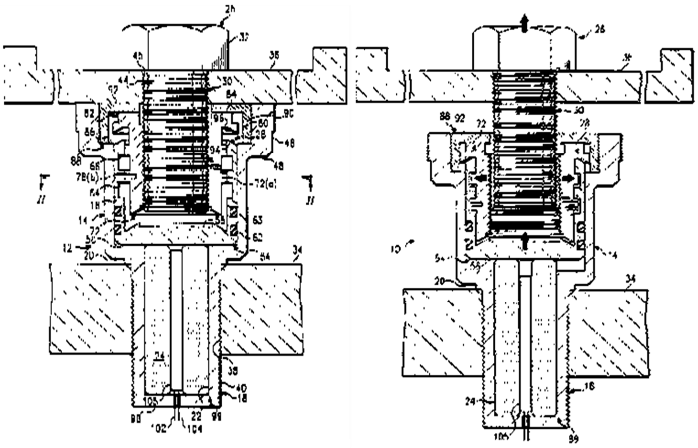


Figure 45 : Bolt release system cross section views before and after actuation
(Source: U.S. Patent 5,160,233, 11/2012)

e. Clamping-ring separation system based on SMA completed by mechanical actuator

Historically pyrotechnic devices were used to perform the separation of a spacecraft from the launcher, by breaking the mechanic bond holding the payload to the launcher structure. Unfortunately this pyrotechnic operation generate a lot of shock, consequently other solutions have been investigated to replace this operation by a low-shock release system. Nowadays the release of a satellite is performed by the loosening of a clamp band containing SMA in the buckle of the belt. The opening of the band is controlled thank to pin puller distributed all around the belt in order to pull off the band.

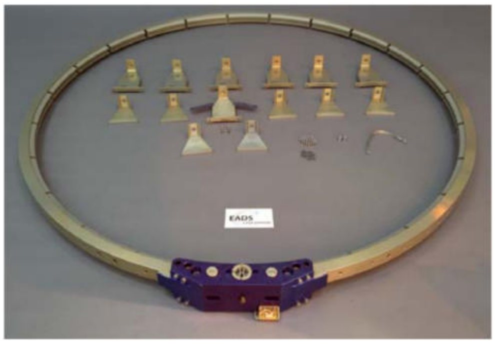


Figure 46 : Clamp band holding payload to the launcher structure
(Source: Arianespace,«Ariane 5 User's Manual Issue 5 revision 1»,July 2011)

f. Release device based on split spool technology

Many release systems are based on a split spool system composes of a plunger blocked between two halves spool which are released by electrically fusing a link wire. Then the spring rolled around the two halves spool deploys and release the plunger allowing translation displacements.

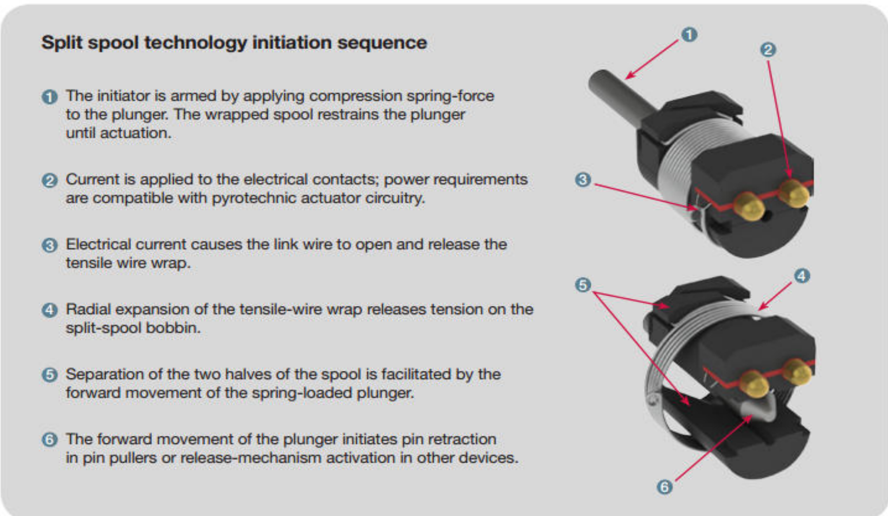


Figure 48 : Split spool working sequence explanation
(Source: Cooper Interconnect « Non-Explosive Actuators », 2012

Figure 47: Stowed TacSat-4 UHF Antenna retained by a belt split spool system
(Image credit: Naval Research Laboratory)

Belt system:

A spool device has been use to build a releasable belt to deployed a metallic flexible antenna on TacSaT-4 American military satellite.

Nuts release system:

The spool mechanism can be used in a numerous/redundant way, for example, to create a nut release system. Another system has been designed base on the same principle with a spring release by a fusible wire.

g. Pyro-mechanisms

Historically, a lot of release mechanisms were composed/based on pyrotechnic actuation in order to split/cut some parts. The Lunar Module of the Apollo mission is a good example of many various ways to integrate pyrotechnic actuation in a spacecraft [RD24].

Guillotine device: explosive cutter

Compose of two blades, propelled by the blast of high explosive charge, which have to cut the wire or rope on which the device is mounted.

It has already been used on lunar module to cut wires and umbilical link between the lander and the module. Consequently the Technology Readiness Level of this device is of 8.

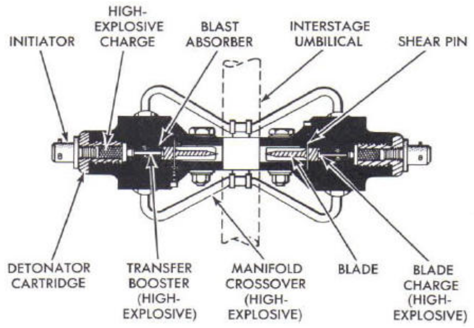


Figure 49 : Explosive guillotine cross section view. (Source: NASA Apollo Lunar Module News Reference, Grumman Aircraft Engineering Corporation, 1968 [RD24])

Explosive nuts and bolts

Pyrotechnic cartridges can be integrated to nuts and bolts assembly in order to make them releasable on command. The Lunar Module of the Apollo mission integrated a lot of this type of mechanisms to disassemble umbilical connection. It is possible to adapt this principle for structural assembly, however the shock generation can be a problem for integration with delicate component.

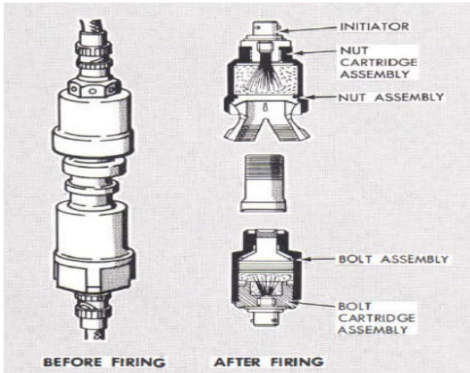


Figure 50: View of assembly composing an explosive nut and bolt link. (Source: NASA [RD24])

Frangible and explosive mechanical joint

Some assembly can be made by integrating pyrotechnic charges directly inside the joint in order to have the possibility to trigger a structure explosion. However, this frangible structural part has to be design to suffer the stress loads of the launch which lead to a conflict to compromise mechanical strength during launch and intended mechanical weakness.

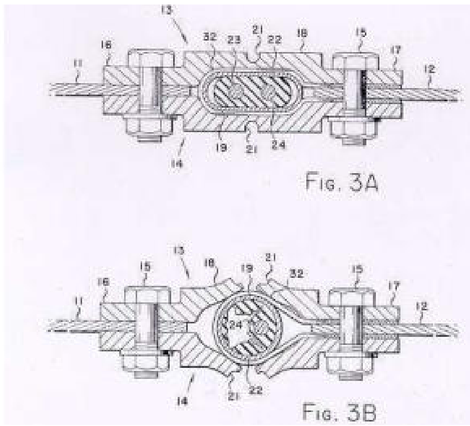


Figure 51 : Cross section of Lockheed's "Super*Zip" separation joint before (top) and after function. (Source: US Patent 3,698,281 "Explosive System", 1972)

Nowadays, pyrotechnic devices are less used than before because of potential debris or particles release after actuation. In addition, such devices generate some shock which is not appreciated in space design. Another drawback is that pyrotechnic behaviour is not properly known after long exposure (more than 20 years) to space radiation which can lead to a problem of reliability and safety to implement explosives charges actuated at the end of life of a spacecraft. Today others types of devices are under design in order to make separation devices with low-shock level, sometimes integrating shape memory alloys.

There is a lot of active release devices in aerospace engineering, it would be pretentious and difficult, to present an exhaustive list of different type existing. Lot of active dismantlement systems can be designed based on classic component such as pin pullers or split spool system. However it is important to highlight that active devices are appreciated and designed to ensure a high level of reliability and stability even under very constraining conditions like in space.

References and further reading:

Others systems or examples of active release devices can be found on following websites or articles:

- http://www.neaelectronics.com
- NASA Apollo Lunar Module News Reference, Grumman Aircraft Engineering Corporation, 1968

3. Overview of passive release mechanisms

Passive devices are regrouping all components which are triggered by the environment they are exposed to during the atmospheric re-entry. Passive technologies include but are not limited to structural elements and mechanisms that are actuated by heat and/or mechanical loads.

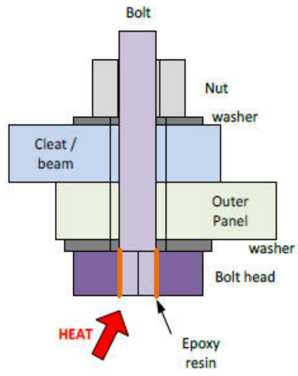


Figure 52 : Cross section view of an assembly composed of glued-head

a. Glued-head bolts

Instead of being monolithic, bolts are made of threaded axle/shaft which is completed by a bolt head glued on it thanks to epoxy resin.

This assembly design introduces a thermal weakness because epoxy resins lose mechanical strength with temperature.

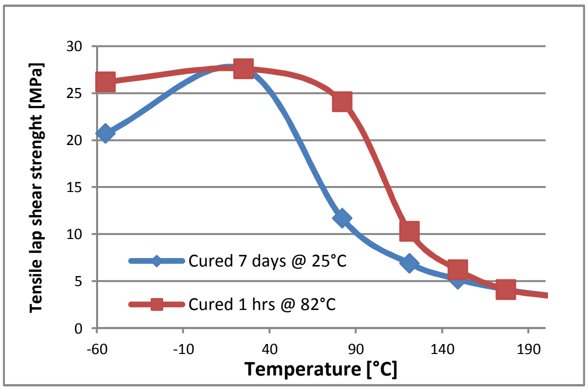


Figure 53 : Tensile lap shear strength evolution with temperature for EA HYSOL 9321. 9321.
(Source : «Hysol® EA 9321, Epoxy Paste Adhesive»)

For example in aerospace manufacturing the epoxy resin EA HYSOL 9321® is often used for part joining or glued assembly. As it is shown in Figure 53, it appears that above 200°C mechanical strength become very low. For an early break up it is better to orientate the glued-head to expose it to the thermal flux before the other side. A spring can complete the system by helping to push the bolt head to overcome the residual stickiness of epoxy.

b. Passive Frangibolts®

They use the same principle as active Frangibolts® without the thermal command sleeve. Phenomenon is passively activated by temperature. Consequently a good knowledge of thermal load of the system is required to avoid unintentional triggering.

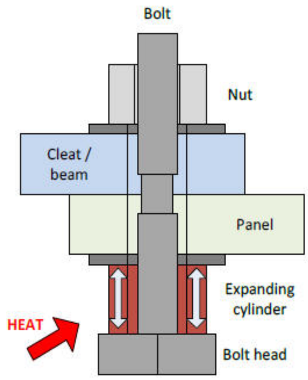


Figure 54 : Frangibolt® working principle

c. Low strength burst bracket

Structural weakness is made voluntary inside a bracket with a hollow shape. The hollow shape can be filled with an incompressible fluid to increase, locally, a structural weakness with a sufficient strength. It will also create a burst when the bracket will break up.

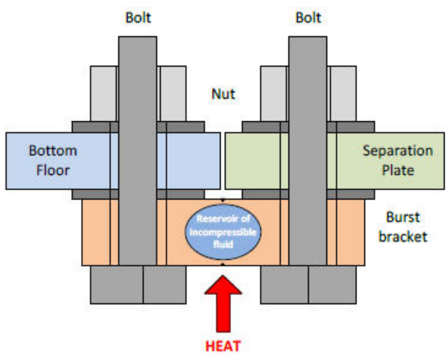


Figure 55: Low strength bracket crass section view

d. Demise-able washer

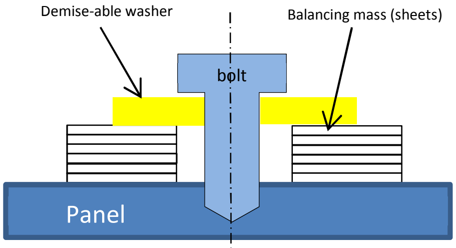


Figure 56: Demise able washer cross section view

Thales Alenia Space has designed a specific washer to release a counter weight on a satellite. Usually the balancing mass on satellite is made of heavy metal like steel or tungsten, whose do not have a low melting enthalpy. The design for demise methodology applied to this case leads to an interesting solution. The balancing mass has been decomposed in many thin metal sheets stacked together and held by a bolt. A washer has been design based on materials with low melting enthalpy (zinc alloy in this case) in order to create a passive release of the balancing mass sheet once the demise-able washer has melt.

e. Shaped charge explosives with self-igniting initiator

Figure 57 : Shaped explosive charges Sabrex®
(Credit: http://www.chemringenergetics.co.uk)

It is possible to design shaped explosive charges which can be self-ignited during re-entry by the thermal flux. This type of charges can be dispatched strategically in order to weaken a structure or even cut literally through structural parts. The main problem of this type of solution is the reliability because the risk of unwanted triggering could be fatal for mission.

In the same way, thermite application has been considered and studied in order to melt metallic structure or tanks by using the very exothermic oxidation reaction of this kind of material.

In aerospace engineering, it is not common to use passive devices because satellites are monitored and controlled from ground. Consequently it is appreciated to have the possibility/the choice to trigger or not a release system according on the situation and the required conditions. In addition, passive devices have to ensure a very high degree of reliability to be considered for a space application. Nowadays, the context of space debris regulation, especially the passivation constraint, coupled to the design for demise approach lead to a reconsideration of passive devices eventually self-triggered during atmospheric re-entry thanks to aerothermodynamics heat flux. However, research have still to be done to estimates properly the mechanical and thermal loads during re-entry (thermal mapping) in order to design passive systems which will trigger at the right time.

A promising type of release system could be dismantlement devices composed of shape memory alloy which will activate them self when a specific temperature is reached. Shape memory alloy and effect are detailed in the following section.

4. Shape memory alloy principle

a. Single memory effect

Thanks to properties called pseudo elasticity, shape memory alloy can recover from apparent permanent strains with enough heat. This property comes from the fact that certain kind of metallic alloys has two stable phases: austenitic high temperature phase and martensitic low temperature phase.

The transformation from strained martensitic to austenite by heating is accompanied by a displacement of the crystallographic net. This displacement comes from properties of crystallographic net, martensitic net is less organized and less stiff with chaotic orientation of crystal mesh. For austenite it is completely different, the web is more compact (more stiff) and meshes are oriented in the same direction. Consequently when a disordered martensitic structure is heated enough to turn into an austenitic net then the mesh is self-redistributing into ordered structure releasing internal constraints.

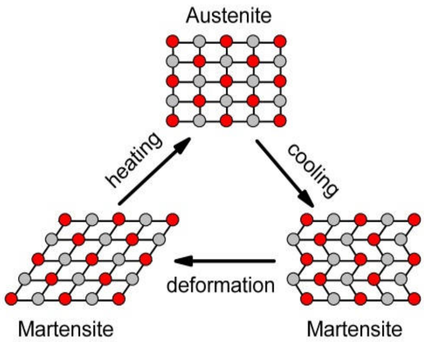


Figure 58 : Crystallographic changes of the metallic structure in a SMA
(Source : wikipedia.org/wiki/Nickel_titanium , 06/2016)

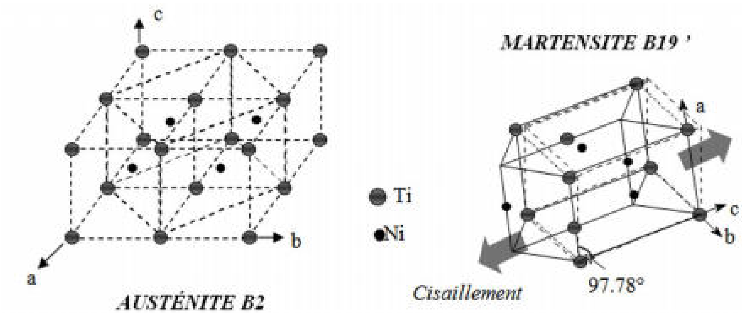


Figure 59 : Austenite and martensite metallic mesh
(Source : L. Jordan, P. Rocher, «Les alliages NIckel-Titane (NiTi)», 2009)

This phenomenon is called single memory effect because it happened in only one way from martensite to austenite. This structural phenomenon come from composition of alloy, for example nickel titanium alloy has this property because of the size of atoms allowing such crystallographic organization. This property is often accompanied with super-elasticity, thanks to martensitic net, allowing relative strain up to 6-8% when classic metallic alloys rarely overcome 0.5%.

The most known alloys presenting shape memory effect are NiTi alloy, CuZnAl, CuAlNi and CuAlBe (Beryllium use is nowadays regulated due to its carcinogenic aspect)

b. Two-way memory effect

With some thermomechanical treatment it is possible to educate a shape memory alloy in order to make it learn a second stable position at a low temperature with martensitic phase. Repeated thermomechanical loads create an internal constraints field inside martensitic meshes (on their orientation precisely) thus when the part is cooled without load this constraint field give a second shape memory.

Two-way memory effect generates less force than single memory effect and this phenomenon decrease with the number of thermal cycles because of crystallographic relaxation.

c. Space applications

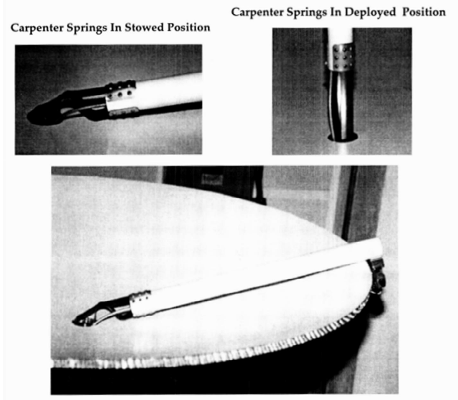


Figure 60 : Clementine high gain antenna system
(Credit: ESA, Clementine satellite, 1999))

Shape memory antenna deployment:

In order to reduce the volume of the satellite during the launch phase, the Clementine's antenna has been design with an articulation composed of SMA to fold the stick. When the satellite is in orbit the antenna is fully deployed thank to 2 SMA strips which will unbend thank to temperature increase.

Shape memory alloy hinge

When a shape memory alloy is heated above its activation/transition temperature, a memory effect appears helping the part to take back its original shape. This principle is called single memory effect.

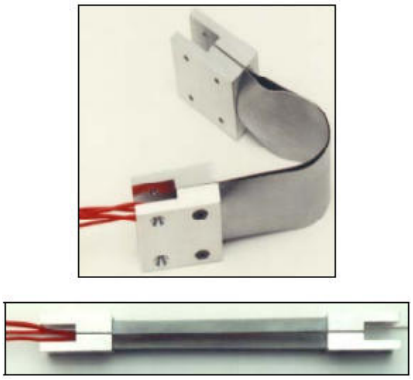


Figure 61 : Shape memory hinge used for solar panel deployment
(Source: Bernie Carpenter, «Lightweight Flexible Solar Array Experiment Summary»,2009)

Torque limiter composed of shape memory device

Thales Alenia Space-France (TAS-F) has developed a system limiting the torque during the deployment of solar array on a satellite. It consists of a shaft with several SMA rings mounted in series. If the measured torque is too high then shape memory ring are heated by resistors which lead to ring contraction. Consequently the friction is increasing and then a resisting torque is created.

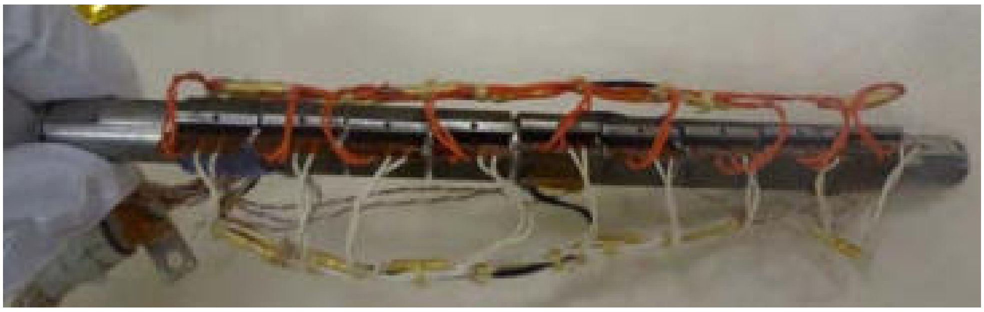


Figure 62 : Torque limiter implement on a solar panel root hinge
(Source: TAS-F ESMAT2013 conference paper, Solar Array Root Hinge Based On Shape Memory Alloy (Sma))

d. References and further reading

Shape memory alloys are complex materials and are still investigated in order to understand more properly their mechanical and thermal behaviour. For more details and explanations, following articles can be read:

- «Effect of Heat Treatment on Stress-Induced Martensitic Transformation and Associated Pseudoelasticity in Monocrystalline Cu-Al-Ni Alloys», H. Sakamoto, Transactions of the Japan Institute of Metals, Vol 28, 1987
- «Effect of Heat Treatments on Thermally Formed Martensite Phases in Monocrystalline Cu-Al-Ni Shape Memory Alloy», H. Sakamoto, ISIJ International, Vol 29, 1989
- «Etude de la fatigue thermique et thermomécanique d'un alliage à mémoire de forme haute température type Cu-Al-Ni», P. Rodriguez, Thèse INSA Lyon, 1989
- «Pseudoelastic Deformation and Generation of Reactive Stresses in a Cu–Al–Ni Shape-Memory Alloy in the Temperature Range 4.2–293 K», V.I. Nikolaiev, Physics of the solid state, Vol 49, 2007

5. Insert enhanced on demise release

The major techniques used in satellite assembly are screwed and glued parts, consequently to maximize the impact of an innovative dismantle device it would be preferable to improve connection design/demise ability. A lot of inserts are used in satellite architecture, commonly they are added into honeycomb or full composite panels. They are glued with polymer resin that lose their mechanical properties and start to demise around 200°C. Concerning screw assembly, most of time it is composed of a bolt constraining a bracket or a plate on an insert or another threated bracket. Then bolts and inserts are major components of connection in satellite assembly. Improvement on those types of parts can be globally applied to a satellite and thus make an early dismantlement of a satellite during atmospheric re-entry.

a. Geometry:

Basically there are three types of insert on a composite panel composing a satellite structural panel: central insert (Figure 63), side insert (Figure 64) and structural insert (Figure 65).

Figure 63 : Central insert (Source: Sentinel-3 3D model)

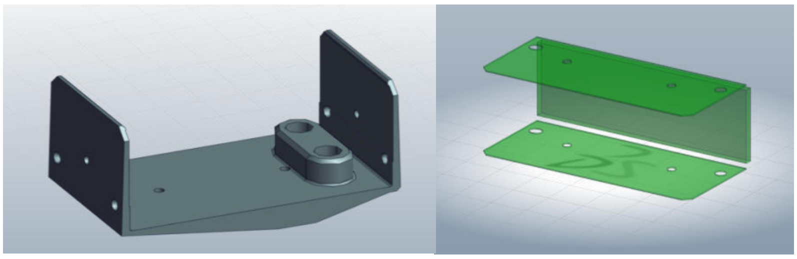


Figure 64: Side inserts examples (Source: Sentinel-3 3D model)

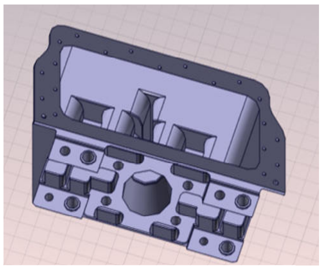


Figure 65 : Structural insert (Source: Sentinel-3 3D model)

b. Demise improvement

Basically the only way to separate an insert from a panel is by axial extraction. This phenomenon happened when the traction force is above the glue resistance. Mechanical characteristics of polymer resin are largely decreasing above 100°C then mechanical force is required to pull out the insert. A way to help this phenomenon would be to add an axial force which can be made by adding an axial spring between the top skin of the composite panel and the top surface of an insert. Thus the spring has to be designed in order that normal working plus the spring force has to be under the nominal working mode resistance of the insert. It is also possible to use an expanding washer made of shape memory alloy instead of a spring which will expand axially with temperature elevation.

Bread boarding tests still have to be conducted because thermomechanical behaviour of composite and insert's glue have to be understood more accurately.

6. Bolt improvement for demise release

a. Common geometry

The second component which is composing a screw connection on a satellite assembly is the bolt. The portion of a bolt limiting the demise of a connection is the bolt head which is keeping a part close to on other.

b. Glued bolt-head

According to geometry and requirement of mechanical links, it is possible to design a bolt head in order to make it demise friendly. Bolt heads can be glued onto threaded axle in order to start to demise when temperature is lowering the mechanical properties of the glue. To enhance bolt head separation it is possible to add a spring or a shape memory alloy washer to push out axially the bolt head.

c. Multipart bolt-head

Bolt head can also be mechanically linked to threaded axle, for example bolt head can be made of two half heads held against threaded axle thanks to an SMA ring or a SMA spring. With elevation of temperature the SMA spring will release the half bolt-head consequently releasing the threaded axle, then the mechanical link.

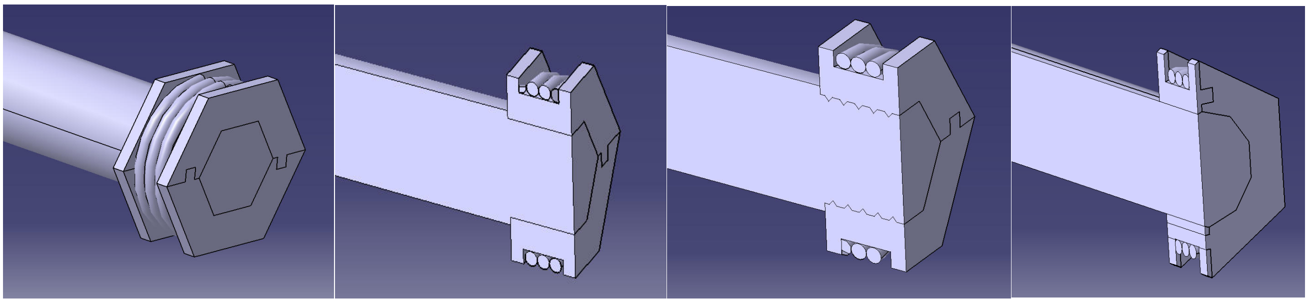


Figure 66 : Various concepts of multipart bolt-head inspired of split spool device.

As mentioned before, improvements have to be made to enhanced structure fragmentation and thus increase the ablation of spacecraft during atmospheric re-entry. Mechanical links and assembly are promising candidates according on their numerous presences in structure.

VIII. Conclusion

1. Design for demise an space debris mitigation

Since 10 years, design for demise has become an indispensable discipline in spacecraft design. Principles and laws evolved quickly during last decades but it is complicated to follow and understand properly which rules are applying to whom. In addition, design for demise is pretty complicated to council with mission performances. It became compulsory to develop some techniques to improve demise ability in order to anticipate new regulation constraints but also to anticipate evolution of actual constraints. Like on-ground casualty risk for human regarding re-entry debris, the risk regulation will be constant through time but human population will increase which will make this risk limitation more difficult to respect.

Space debris problematics has been mediatized the last years and coupled with space regulation principles and laws it has became a major point in space industry. This new problematic is forcing all the major integrator and companies working in the space field to review their standards in order to integrate new constraints. Unfortunately design of spacecraft is becoming more complicated and new solutions have to be found but it also synonym with work and research to do to overpass this new engineer challenge.

2. Internship

This working experience in a research structure collaborating with major integrators has been a great opportunity to learn a lot about satellite architecture, space regulations and laws but also on human aspect of the work as engineer. I had the chance to work on several projects during one or two months and they were for me opportunities to discover more specifically some components such as reaction wheels or optical payloads. Those projects made me discover new problematics and stakes about components but also research and investment in space industry. This internship confirmed my ambition to work in aerospace to learn more and use the knowledge I have collected so far at university and during my several projects.

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