# DE-ORBIT STRATEGIES WITH LOW-THRUST PROPULSION

**A. Gaudel[1], C. Hourtolle[2], J.F. Goester[3], N. Fuentes[4] and M. Ottaviani[5]**

(1-5) *CNES, 18 avenue Edouard Belin 31401 Toulouse Cedex 9, FRANCE,*

(1) *angelique.gaudel-vacaresse@cnes.fr,* (2) *catherine.hourtolle@cnes.fr,*

(3) *jean-francois.goester@cnes.fr,* (4) *nathalie.fuentes@cnes.fr,* (5) *matthieu.ottaviani@cnes.fr*

**ABSTRACT**

In the framework of the French Space Operations Act (FSOA) [1-2], it is now necessary to take into account the orbit lifetime of the satellites, in particular for the Low Earth Orbits (LEO), whose population is increasing. But after 2020, it will be mandatory to foresee a controlled re-entry, except if it is actually unfeasible. Currently, only few spacecraft, like the ATV (Automatic Transfer Vehicle), is able to perform such de-orbit manoeuvres for a controlled re-entry. For more classical satellites, such manoeuvres will imply a too important amount of propellant. Thus, it could be interesting to analyse de-orbit strategies with low-thrusts provided by an electric propulsive system. Indeed, even though these low-thrusts do not allow to bring the satellite on a directly re-entering orbit, it may be envisaged to position the spacecraft on an orbit whose altitude is low enough to be able to predict its re-entry within some hours, therefore limiting the debris fallout zone to a small number of orbit ground-tracks, chosen in order to decrease the risk on ground for human population.

## 1. INTRODUCTION

### 1.1. French Space Operation Act (FSOA) and regulations of spacecraft's re-entries

The technical regulations related to the FSOA are given in a decree [3]. The article 44 of that decree defines the quantitative objectives for ensuring the safety of human population during the re-entry of a spacecraft. In particular, the quantitative objectives of safeguard, expressed as the allowed maximum probability of having at least one victim and denoted by PD, are defined as follows:

- $PD_{max}$ = 2.10-5 for a controlled re-entry with disintegration of the spacecraft;
- $PD_{max}$ = 10-4 for an uncontrolled re-entry with disintegration of the spacecraft, in the case the impossibility of a controlled re-entry is fully justified.

Moreover, it is also specified that the risk has to be computed with a method able to take into account the phenomena leading to have a risk of catastrophic damage, the trajectories before fragmentation, the reliability of the spacecraft and the dispersions of the debris on the ground. The method must also be able to model the scenarios of fragmentation and the generation of debris corresponding to the re-entry.

### 1.2. Paper purpose

The main objective of this paper consists of finding a way of decreasing dramatically the altitude of the satellites at the end of their lifetime in order to allow a re-entry within some hours without needing a large amount of propellant or a modification of the Attitude and Orbital Control Systems (AOCS). Indeed, these two aspects are prohibitive for small satellites.

Thus, a solution may consist of reducing the altitude through low-thrust manoeuvres for the satellites which are already equipped with an electric propulsion system. In fact, electric propulsion is a technology aimed at achieving thrust with high exhaust velocities, which results in a reduction in the amount of propellant required for a given space mission or application compared to other conventional propulsion methods [4]. However, one of the drawbacks of this technology is that, nowadays, it can only provide low-thrusts (less than 1 N).

Regarding a controlled re-entry, it has been seen in [5] that the satellite should be able to target a perigee whose altitude is about 50 km. However, through the electric propulsion, because of the low-thrusts, the difference of altitude of the perigee over one orbit is at most 23 km. This means that the perigee of the previous orbit should be about 73 km, which is below the altitude of fragmentation and even lower than the altitude at which it is likely to lose the most fragile appendices, such as the solar panels.

Therefore, a controlled re-entry cannot be performed through the low thrusts provided by the electric propulsion. Nonetheless, it may be interesting to study the feasibility of performing semi-controlled re-entries with low-thrusts in order to see if it is possible to obtain smaller risks than the ones obtained with the uncontrolled re-entry within 25 years. This strategy consists of decreasing gradually the altitude of the satellite by breaking it through electric thrusters. This approach will lead the spacecraft to perform a spiral-like trajectory. On the other hand, it is not possible to indefinitely decrease the altitude. Indeed, below a given altitude, the AOCS goes out of its nominal range, i.e. the torques provided by the electric thrusters are not sufficiently intense to compensate the increasing perturbations (because of the lower altitude). Thus, the altitude is limited by the

 T. Sgobba, I. Rongier (eds.), *Space Safety is No Accident*, DOI 10.1007/978-3-319-15982-9_7 

torque capacity provided by the actuators. This altitude will be fixed to 150 km in this paper. Once the altitude at which thrust cannot be provided anymore is reached, the re-entry becomes uncontrolled. Yet, at this stage, the possible re-entry orbits are foreseeable and might be phased opportunely with the Earth in order to minimise the risk for human population at the moment of the impact of the fragments.

However, it has been seen in [5] that the target altitude for a semi-controlled re-entry should be smaller than 150 km (in order to have less orbit ground-tracks and then reducing the risk for the human population). Thus, it could seem that even this kind of strategy cannot be performed through the electric propulsion. Nevertheless, for studying the feasibility of semi-controlled re-entries through low thrusts, it is necessary to analyse in detail the variation of the risk with the position of the probable fallout zone on the ground.

The analysis of the feasibility of such a strategy has been performed thanks to a new computation method of the software ELECTRA (Launch and re-entry risk analysis tool) [6], concerning the final phases of uncontrolled re-entries [7].

## 2. RISKS ASSOCIATED TO SPACECRAFT RE-ENTRIES

What makes a safe spacecraft re-entry feasible is the mastery of the knowledge of the debris fallout zone in order to minimise the risk for human population. In a utopian case in which the re-entry of a spacecraft was modelled perfectly, the debris fallout zone would be known deterministically. However, because of the lack of precise knowledge of too many parameters, the debris fallout zone can only be estimated statistically. In fact, the analytic results do not provide the desired accuracy for the latitudes and the longitudes of the impact points. This is the reason why it is necessary to talk about probable debris fallout zone.

The size of this fallout zone depends on the different re-entry strategies:

- **For a controlled re-entry**: The probable debris fallout zone is relatively small so that it can be placed in uninhabited regions of the Earth. Thus, the idea is to perform the de-orbiting manoeuvres once a given phasing between the spacecraft and the Earth is satisfied.
- **For uncontrolled re-entry**: The probable debris fallout zone is so large that in fact it can be considered equal to the Earth's region comprised between the latitudes -i and +i, i being the inclination of the spacecraft's orbit. Since de-orbiting manoeuvres cannot be performed, it is not possible to target an uninhabited region of the Earth.
- **For semi-controlled re-entry**: The probable debris fallout zone extends for a length that can be smaller than one orbit ground-track but that can reach several orbit ground-tracks. Depending on the length of this fallout zone, it could be possible to minimise the risk with respect to the uncontrolled re-entry. For a given length of the impact track, the risk will depend on the location of this impact track on the ground, i.e. with the phasing with the Earth.

The development of ELECTRA method and software [6], undertaken in 2007, meets the requirement of precise quantification of the risks induced by fragments fall back during a launch or an atmospheric re-entry specified by the FSOA [1]. In 2010, ELECTRA was implemented for internal CNES safety needs, but soon it has been provided to space operators, in the frame of the FSOA, to assess human risk associated to their operations. The tool has also been used to estimate uncontrolled re-entry risk of all CNES LEO missions and to assess the fragment impact footprint for the controlled re-entry of the Automated Transfer Vehicle (ATV).

ELECTRA assesses the risks associated to three different events that are rocket launching failure, controlled re-entry failure, and uncontrolled re-entry. For each case, ELECTRA computes two complementary estimations of the risk, i.e. the probability of causing at least one victim and the expected value of the number of victims.

A new module has been developed to estimate the risk a few hours or days before the uncontrolled re-entry of a spacecraft [7]. Indeed, a few hours before the re-entry, the possible re-entry conditions are distributed on a limited amount of orbits and the random re-entry module, for which the fragments can fall anywhere between the latitudes over which the satellite flies, is not appropriate anymore. The possible re-entry conditions can be characterised by a set of possible state vectors at a given altitude taking into account uncertainties on the spacecraft itself and on the environmental conditions. This can be done varying the ballistic coefficient of the intact spacecraft, between $Cbal_{min}$ and $Cbal_{max}$ (in m²/kg), as shown in Fig. 1. The trajectory of the intact spacecraft is extrapolated until a given geodetic altitude. This altitude is fixed to 85 km in order to avoid the situations in which a fragment is not captured by the atmosphere.

Each of these possible state vectors is then extrapolated until the altitude of fragmentation after which the trajectories of the fragments are extrapolated until their impacts on the ground. Finally, knowing the characteristics of the fragments and their speed before impacting the ground, it is possible to compute the risk for the human population in a similar fashion as for the controlled re-entry module.

This module, together with the feature of the computation of the risk by country, may be very useful to inform the national authorities of the countries in which the fragments could fall. In fact, if it is known that the uncontrolled re-entry of a spacecraft will occur soon, this module may be used to evaluate the risks associated to the countries which are directly involved by the fall of the debris, and the national authorities can decide the appropriate measures to reduce the risk of having victims. Moreover, it may be envisaged to follow the actual position of the spacecraft in order to update the possible re-entry conditions and then to study the evolution of the risks for the different countries.

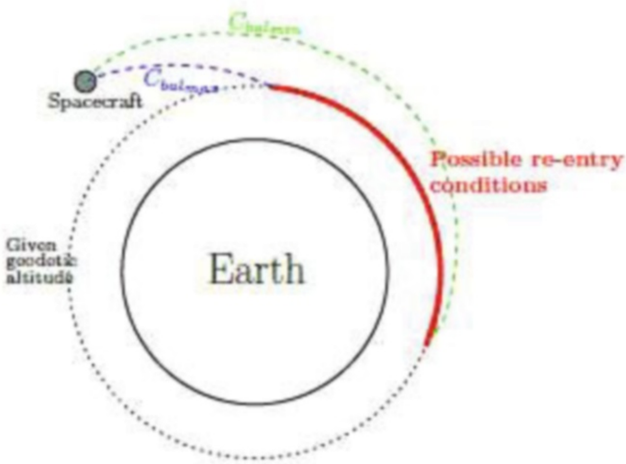



*Figure 1. Possible re-entry conditions*

## 3. FEASIBILITY STUDY OF RE-ENTRIES PERFORMED WITH LOW-THRUSTS

The short-term re-entry module, described in the previous chapter, may be used to evaluate the risks associated to a particular kind of semi-controlled re-entries, i.e. the ones performed with electric propulsion. These semi-controlled re-entries can be subdivided into two main phases, represented in Fig. 2:

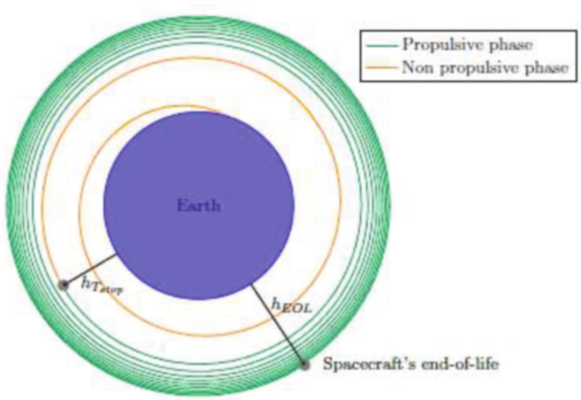



*Figure 2. Representation of a semi-controlled re-entry with low-thrusts*

- **Propulsive phase**: This phase begins when the spacecraft has reached the end of its lifetime. During this phase, the spacecraft is reducing continuously its altitude through continuous low-thrusts. Since the low thrusts provided by the electric propulsion are not able to dramatically change the eccentricity of the orbit, it is not possible to target a given fictitious perigee as in the case of a controlled re-entry. The low thrusts can be provided only if the attitude of the satellite is known and mastered, i.e. when the AOCS works properly. While decreasing the altitude, the perturbing torques start growing bigger and bigger and the AOCS may not be able to counteract them. When this happens, the attitude of the satellite cannot be mastered anymore and the corresponding altitude will be denoted by $h_{limAOCS}$ from now on. This altitude corresponds to the minimum altitude that can be reached during the propulsive phase. However, the low thrusts might be stopped at a different altitude, denoted by $h_{Tstop}$ from now on, satisfying the following condition: $h_{Tstop} \geq h_{limAOCS}$.

- **Non-propulsive phase**: During this phase, no thrust will be provided by the propulsive system. At this stage, the only responsible of the altitude decay is the aerodynamic drag, which is increasing with the decreasing altitude, being proportional to the atmospheric density. This phase is mostly characterised by uncertainties, in particular about the atmosphere, the aerodynamic coefficients, the velocity of the satellite with respect to the atmosphere, the fragmentation and the ablation of the fragments. These uncertainties affect the length of the probable debris fallout zone together with the altitude at which the continuous low-thrust is stopped $h_{Tstop}$. Actually, this phase corresponds to an uncontrolled re-entry starting from the altitude $h_{Tstop}$. If this altitude is small enough, the probable fallout zone can be foreseen and thus the risks may be controlled. Since the non-propulsive phase can be seen as an uncontrolled re-entry starting from the altitude $h_{Tstop}$, and since $h_{Tstop}$ is small enough to guarantee a re-entry within some hours, the short-term re-entry module of ELECTRA is able to assess the risks associated to this kind of semi-controlled re-entries.

Before starting to analyse in detail the de-orbit strategies with low-thrusts, it is necessary to study their feasibility. Some assumptions will be made in order to decrease the complexity of the problem and thus to obtain approximated but general results.

### 3.1. Description of the feasibility study

The whole semi-controlled re-entry analysis depends on several variables. In order to simplify the analysis, the whole problem is subdivided into different sub-problems, each one with a smaller number of depending variables. The idea is to relate one phase only with its precedent, and not with each one of the others.

**Step 1: Phasing the probable debris fallout zone with the Earth**

Firstly, the risks related to the probable impacts tracks are analysed. Indeed, if from this study it appears that it is not possible to decrease the risks with respect to the uncontrolled re-entries, it is useless to consider the strategy of a semi-controlled re-entry using electric propulsion.

This study is done varying the length of the probable impacts tracks and the position of the first point of the impact track in order to cover all the regions the satellite can fly over. When the length of the probable impacts track is very short, the problem could be similar to a controlled re-entry, for which it is possible to obtain risks dramatically smaller than the ones obtained with the uncontrolled re-entry. Therefore it can be foreseen that playing with the length of the impact track; it is possible to decrease the risks when the length is small enough. Thus, the objective of this study is to quantify how much small enough is, i.e. to compute the maximum length of the probable impacts track, $Limp_{max}$, for which the risks can be minimised with respect to an uncontrolled re-entry.

This study depends on the spacecraft, on the inclination i of the orbit, as well as on the set of debris generated by the fragmentation and on the human population. Once they are fixed, it is not necessary to vary them in the other phases.

**Step 2: Natural re-entry phase**

Secondly, the phase from $h = h_{Tstop}$ to the ground is analysed. Being inclination i fixed at this stage, the orbital parameters that are varied are ($a$, $e$, $\omega$, $\Omega$, $v$). The study is performed varying also the ballistic coefficient of the spacecraft and the geomagnetic and solar activity. In particular, it is computed the distance between the impact point associated to the minimum ballistic coefficient and the one associated to the maximum ballistic coefficient, both in the worst and in the best case, respectively for weak and strong solar activity. In a first approximation, this distance can be considered equal to the length of the probable impacts track.

The output of the first study, $Limp_{max}$, will be used as input for this study. The idea consists of finding, for a given ballistic coefficient, which is the range of geocentric altitudes $h_{Tstop}$, limited by the best and the worst case, at which the thrust can be stopped in order to have a probable impacts track which is long $Limp_{max}$. For a satellite of a given ballistic coefficient, if the whole range lies below $h_{limAOCS}$, then this strategy of re-entry is not convenient, since the risk is of the same order of magnitude as the one obtained with an uncontrolled re-entry or even bigger. Otherwise, the strategy is feasible and the analysis can proceed.

**Step 3: Propulsive phase**

Finally, knowing the satellite, and thus knowing its ballistic coefficient and its mass (and available power), and knowing which is the altitude at which thrust can be stopped, it is possible to perform an analysis of the propulsive phase, in order to compute the time and the consumption necessary to reach $h_{Tstop}$ . This study will depend on the available power on the satellite, on the final orbit of its lifetime and on the environmental conditions. It will also depend on the propulsive parameters that drive the motion of the satellite.

If the constraints about the total transfer time and the propellant consumption are respected, then the strategy is feasible and it is possible to analyse in detail the problem, i.e. to obtain the optimal re-entry points and to compute the optimal strategy to reach them.

### 3.2. Study cases

The interest of semi-controlled re-entries with low-thrusts addresses mainly for small LEO satellites, because their size does not allow them to carry enough propellant to perform controlled re-entries. Thus, this strategy will be studied for three different satellites which are on a Sun-Synchronous Orbit, namely PARASOL, SMOS, and SPOT-5. Although these satellites are not equipped with an electric propulsive system, it could be interesting to pretend that they are in order to compare the risks associated to this strategy with the ones associated to an uncontrolled re-entry within 25 years. Moreover, in order to take into account the case in which the inclination of the End-Of-Life (EOL) orbit is equal to i=51deg, a satellite having the same characteristics of SPOT-5 is studied.

The main characteristics of the EOL orbits of the studied satellites are given in Tab. 1, together with their re-entry periods, their masses and the ballistic coefficients corresponding to their tumbling motions.

*Table 1. Re-entry characteristics*

| SATELLITE | $i_{LEO}$ [deg] | $h_{LEO}$ [km] | Re-entry year | Cbal [$m^2$/kg] | Mass [kg] |
|---|---|---|---|---|---|
| PARASOL | 98.28 | 699.6 | 2045-2053 | 0.01833 | 120 |
| SMOS | 98.445 | 719.1 | 2040 | 0.03515 | 630 |
| SPOT-5 | 98.6 | 822 | 2055-2060 | 0.015 | 3000 |
| SPOT-5-like | 51 | 822 | 2055-2060 | 0.015 | 3000 |

Concerning the propulsive phase, it is necessary to know the main parameters, such as the thrust $F_T$ and the specific impulse Isp. There exist different technologies of electric propulsion, such as Arcjet, Pulsed Plasma Thruster (PPT), Field Emission Electric Propulsion (FEEP), Ionic and Stationary Plasma Thruster (SPT). The possible technology that may be used for mini satellites, such as PARASOL and SMOS, is SPT. Although SPOT-5 is bigger, it is supposed that it can be equipped with the SPT technology as well. Then, it is possible to estimate $F_T$ and Isp for the imaginary electric propulsive system of the studied satellites. These values are given in Tab. 2.

*Table 2. Imaginary propulsive characteristics*

| SATELLITE | POWER [W] | THRUSTER | $F_T$ [mN] | Isp [s] |
|---|---|---|---|---|
| PARASOL | 150 | SPT | 8 | 1500 |
| SMOS | 560 | SPT | 30 | 1500 |
| SPOT-5 | 2400 | SPT | 140 | 1500 |
| SPOT-5-like | 2400 | SPT | 140 | 1500 |

Concerning the non-propulsive phase, i.e. the natural re-entry phase, it is necessary to model the fragmentation in order to have an idea of the fragments that can reach the ground. The list of these fragments, for the studied satellites, has been obtained by the projects data. In order to be coherent, the same fragments will be considered both for the uncontrolled re-entry and the semi-controlled re-entry.

### 3.3. Uncontrolled re-entries

Before starting the feasibility study of the semi-controlled re-entries with low-thrusts, it is necessary to compute the risk associated to an uncontrolled re-entry, $PD_{random}$. For each considered satellite, the risks are computed with the random re-entry module of ELECTRA using as inputs the inclination of the EOL orbit, given in Tab. 1, and the list of fragments. The results are represented in Tab. 3.

*Table 3. Risks associated to the uncontrolled re-entries*

| SATELLITE | Re-entry year | $PD_{random}$ |
|---|---|---|
| PARASOL | 2045-2053 | 5.20E-05 |
| SMOS | 2040 | 5.26E-05 |
| SPOT-5 | 2055-2060 | 3.11E-04 |
| SPOT-5-like (i=51deg) | 2055-2060 | 4.59E-04 |

From the results, it is possible to notice that, for PARASOL and SMOS, the probability of having at least one victim is smaller than the limit imposed by the FSOA, i.e. $PD_{max}$ = 10-4. Instead, for SPOT-5 and a SPOT-5-like satellite (i = 51 deg), the same probability is bigger than the limit imposed by the FSOA, whatever the year of re-entry. Anyway at the time of the launch of SPOT-5, the FSOA did not exist. This means that in the future, for satellites like SPOT-5, it is necessary to consider a given de-orbit strategy in order to meet the constraints imposed by the FSOA. On the other hand, even though PARASOL and SMOS meet the safety requirements, it could be interesting to study whether the risks could be further decreased with a semi-controlled re-entry with low-thrusts.

While the uncontrolled re-entries of the considered satellites will occur later than 2040, the semi-controlled re-entries with low-thrusts can take place immediately. In particular, the re-entry year will be fixed to 2010, for which the distribution of the human population is known. Since the population is supposed to increase in the future, and thus also within 25 years, a semi controlled re-entry with low-thrusts may further decrease the risks with respect to an uncontrolled re-entry.

### 3.4. Step 1: Phasing of the probable debris fallout zone

For a given set of fragments, the length of the probable fallout zone depends mainly on the altitude at which the low-thrust is stopped, $h_{Tstop}$. More precisely, the higher $h_{Tstop}$, the longer the probable impacts track. If the probable impacts tracks were very short, the situation would be similar to a controlled re-entry. However, since the altitude at which the low-thrust is stopped is limited by the minimum altitude at which the AOCS is supposed to work properly, the length of the probable impacts tracks cannot be minimised indefinitely. Thus, the idea consists of finding the maximum length of the probable impacts track for which a given phasing with the Earth can provide smaller risks than the ones obtained with the uncontrolled re-entry. In particular, it is desired to obtain risks of one order of magnitude smaller, because of all the assumptions that will be made along the feasibility study, allowing having some margins.

In the limit case in which the probable debris fallout track is infinitesimal, it is possible to place it in an uninhabited region of the Earth, so that the risk is null. In the other limit case in which the probable debris fallout track is infinite, it is useless to translate it because all the points of the Earth are covered. This corresponds to the random (uncontrolled) re-entry for which the risk is equal to $PD_{random}$.

For a given length of the probable debris fallout track, it is necessary to translate the probable impacts track such that its central point can span over all the regions comprised in the range of latitude [-i; +i], i being the inclination of the satellite, and such that both ascending and descending tracks (with respect to the central point) are considered.

In order to compute the risks for a given portion of the long probable impacts track, it is necessary to know the probability distribution of the re-entry points (at a geodesic altitude equal to 85 km), which can be computed with the ELECTRA tool. Both Gaussian and uniform distributions will be considered for the impacts corresponding to the given possible re-entry points.

**Preliminary considerations**

Firstly, it is analysed the variation of the risk with respect to the length of the probable debris fallout zone. In Fig. 3, the location of the central point of the probable debris fallout zone spans all the regions over which the satellite can fly and a risk is associated to each of the probable impacts tracks. It is possible to notice that the extension of the regions where the risk can be minimised decreases with the increasing length of the probable debris fallout zone, as it was expected.

**Computation of $Limp_{max}$**

It is computed the maximum length of the probable impacts track, noted as $Limp_{max}$, for which it is possible to minimise the risk of at least one order of magnitude with respect to an uncontrolled re-entry. Since Fig. 3 shows that the minimum risk increases with the increasing length of the probable impacts track, the search is performed starting from a value of length equal to $L_{imp}(0) = 1/4$ orbit. Once the value of the length of the probable impacts track is fixed, the position of its central points is varied in order to cover all the regions over which the satellite can fly. For each of these positions, the risk PD is computed, considering both Gaussian and uniform distributions. The variation is stopped once the following condition is satisfied $PD \leq PD_{random}/10$ and the length of the probable impacts track is increased of $\Delta Limp = 1/4$ orbit. At the i-th iteration, if the risk PD never satisfies these conditions, then $Limp_{max} = L_{imp}(i-1)$. The obtained results are given in Tab. 4.

*Table 4. Values of $Limp_{max}$*

| | $Limp_{max}$ [orbits] | |
|---|---|---|
| DISTRIBUTION | Gaussian | Uniform |
| PARASOL | 4.5 | 3.25 |
| SMOS | 4.5 | 3.25 |
| SPOT-5 | 5 | 3.25 |
| SPOT-5-like (i=51deg) | 4.25 | 3.5 |

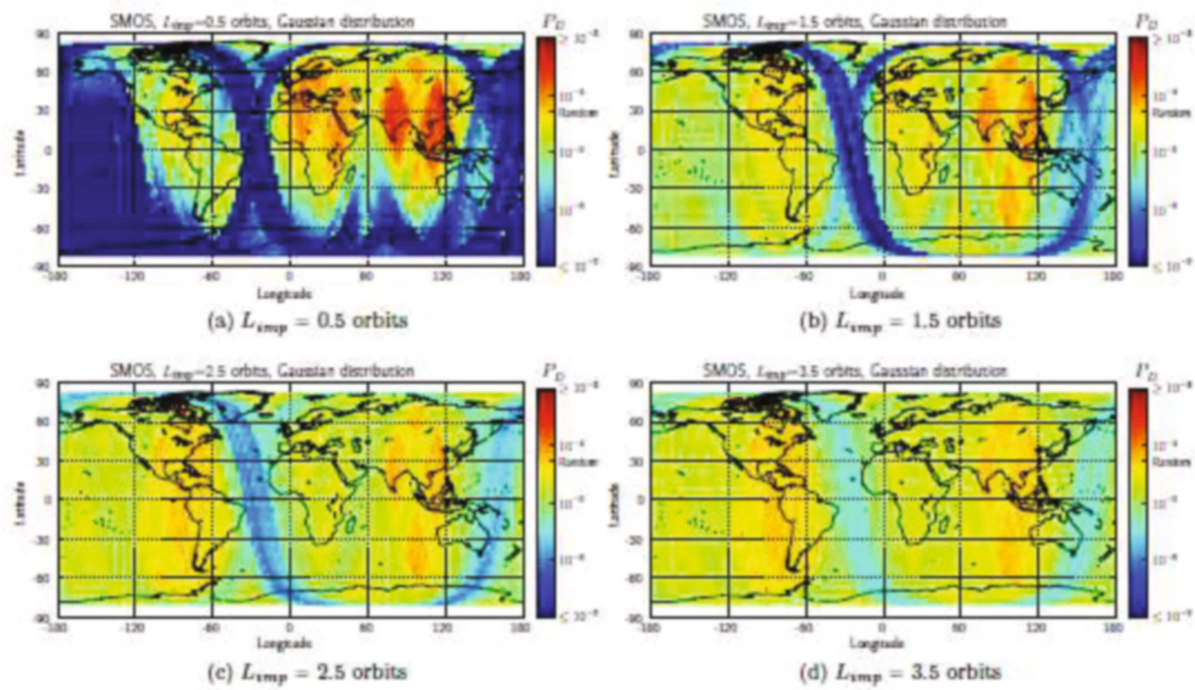



*Figure 3. Variation of the risks with the length of the probable impacts track*

**Computation of the minimum risk**

For $Limp = Limp_{max}$, the minimum risk is computed together with the risks associated to all the impacts tracks, with their central point spanning all the regions over which the spacecraft can fly. These computations are done considering both Gaussian and uniform distributions. As example, in the plots concerning SMOS satellite in Fig. 3, the colour map is associated to the value of the risks, going from blue (minimum value) to red (maximum value). A particular colour, corresponding to the value of $PD_{random}$ given in Tab. 3, is associated to the uncontrolled re-entry.

The obtained results do not suggest anything about the feasibility of this de-orbit strategy. In fact, they just prove that the phasing is needed in order to decrease the risks associated to such a strategy. Nonetheless, some other general considerations can be deduced. Concerning the probable debris fallout zones associated to the minimum risks, it is possible to notice that they mainly interest the oceans and some poorly inhabited regions, as expected. In addition, other solutions, whose risks are slightly bigger than the minimum one, can be found in the neighbourhood of the optimal one, increasing the likelihood of the feasibility. Furthermore, the main difference between the Gaussian and the uniform distribution is that in the second case the overall risks may be higher. Indeed, for a uniform distribution, all the points have the same importance, whereas for a Gaussian distribution only the points closer to the central point of the impacts track are more important. The other differences in the results are due to the different fragments and to the different values of the risk associated to the uncontrolled re-entry.

### 3.5. Step 2: Natural re-entry phase

The probable re-entry points (at a geodetic altitude equal to 85 km) are obtained computing the natural re-entry from a geocentric altitude $h_{Tstop}$, dispersing the ballistic coefficient of the intact spacecraft. During the natural re-entry, it is assumed that the attitude of the spacecraft is not controlled, because of the limitation of the AOCS, and that the spacecraft is thus characterised by a tumbling motion. Depending on the shape of the satellite, this can lead to very complicated expressions of the instantaneous reference surface. Since the aerodynamic characteristics change with respect to the attitude of the spacecraft, it is not possible to know exactly the actual value of the ballistic coefficient. Thus, in order to perform the computations, it is assumed that a given spacecraft, in its tumbling phase, can be modelled as a sphere whose surface is equal to the geometric average surface, $S_{av}$. Indeed, a sphere can be considered as an approximation of a tumbling spacecraft, whatever its shape. Therefore, considering the drag coefficient of a sphere, the ballistic coefficient of a tumbling satellite is computed.

In a first approximation, it is considered that the length of the probable impacts track is equal to the distance between the impact of the intact spacecraft with $Cbal_{max}$ and the impact of the intact spacecraft with $Cbal_{min}$. Indeed, the maximum distance between the impact of the intact spacecraft for a given Cbal and one of the debris, for the re-entry point, is of the order of hundredths of kilometres (negligible with respect to the length of an orbit ground-track).

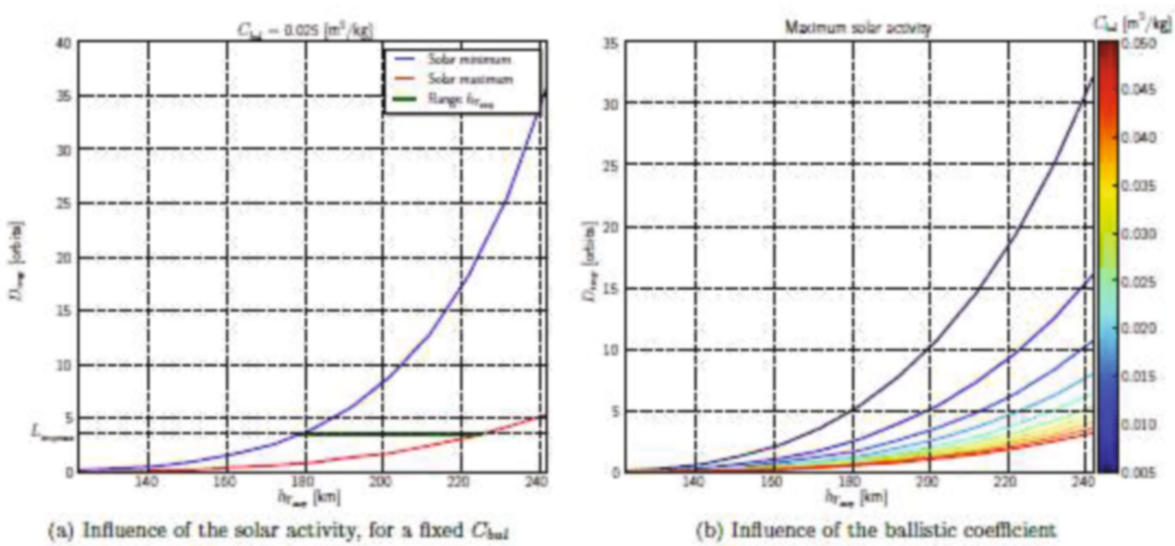


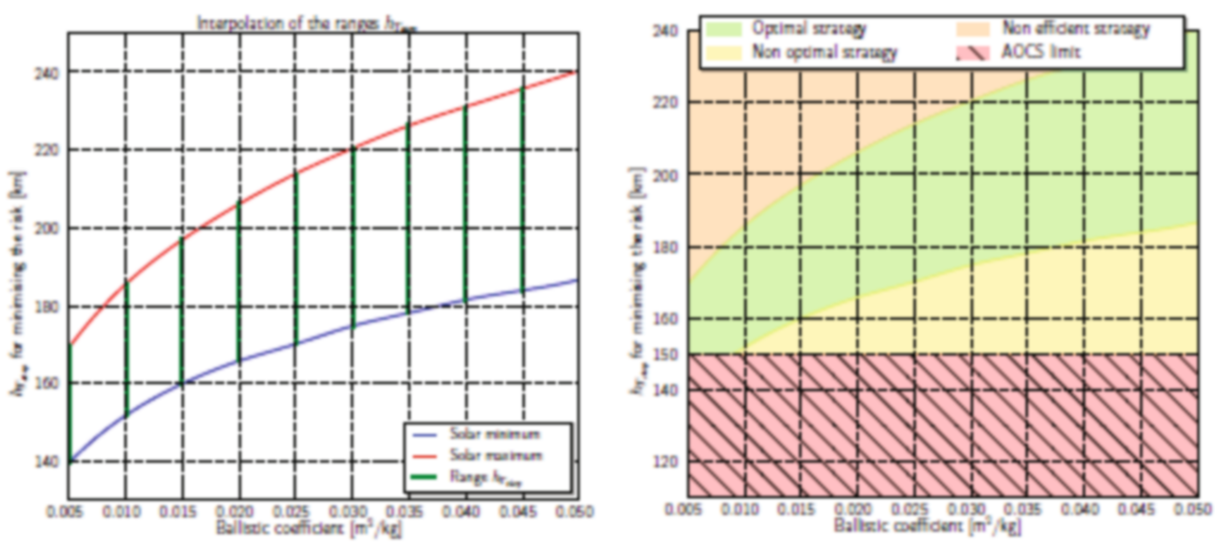

*Figure 4. Dimp function of $h_{Tstop}$*

For a given ballistic coefficient, starting from a given geocentric altitude, it is computed the distance between the impact of the intact spacecraft with $Cbal_{max}$ and the impact of the intact spacecraft with $Cbal_{min}$, denoted by Dimp, for different values of ω, Ω and v and for different environmental conditions (weak and intense solar activity). In particular, there are computed the worst case (maximum length, in correspondence of the weak solar activity) and the best case (minimum length, in correspondence of the intense solar activity). The results for a given ballistic coefficient are plotted in Fig. 4a.

In Fig. 4b it is represented the influence of the ballistic coefficient on the value of Dimp. For simplicity reasons, it is only shown the effect in the case of minimum solar activity. Anyway, the same considerations are valid for the maximum solar activity. From Fig. 4b, it is possible to infer that the higher the ballistic coefficient (in m²/kg), the smaller the value of Dimp. This happens because the effect of the dispersion is higher when the ballistic coefficient is smaller.

For a given ballistic coefficient, fixing Dimp = $Limp_{max}$, it is possible to obtain the range of altitudes $h_{Tstop}$ at which the low-thrusts can be stopped for minimising the risks with respect to an uncontrolled re-entry. This fact is shown in Fig. 4a, where the range of altitudes $h_{Tstop}$ is represented in green.

Then, in order to obtain the results for any ballistic coefficient, the ranges of altitudes $h_{Tstop}$ are interpolated, as shown in Fig. 5a. Focusing on the same figure, four different regions, concerning the de-orbit strategy with low-thrusts, can be identified and they are represented in Fig. 5b.




(a) Interpolation of the ranges $h_{T_{stop}}$ (b) Representation of the different regions

*Figure 5. $h_{Tstop}$ function of the ballistic coefficient*

- **Green region**: Optimal strategy, for which it is possible to obtain PD = $PD_{random}/10$. It is upper-delimited by the best case (intense solar activity) and lower-delimited by the worst case (weak solar activity).
- **Yellow region**: Non-optimal strategy, for which PD < $PD_{random}/10$. The strategy is non-optimal because, for obtaining a smaller risk than necessary, the low-thrusts have to be provided until a smaller altitude.
- **Orange region**: Non-efficient strategy, for which PD > $PD_{random}/10$. The strategy is non-efficient because the obtained risk is of the same order of magnitude than the one obtained with an uncontrolled re-entry. Thus, it would be more convenient an uncontrolled re-entry, maintaining the same level of risk.
- **Red region**: Prohibited area, upper-delimited by the constraint of the minimum altitude of correct behaviour of the AOCS. In fact, below this altitude, it is not possible to make manoeuvres because the attitude of the satellite is not perfectly mastered.

It is possible to observe that, below a given $Limp_{max}$, the semi-controlled re-entry is not feasible for satellites with a small ballistic coefficient (in m²/kg), not even in the best conditions (maximum solar activity). Furthermore, when $Limp_{max}$ increases, the green region expands. In particular, both upper and lower limit increase, but the first more. Because of this, satellites with lower ballistic coefficient may not be limited by the AOCS anymore and the strategy may be performed even in the worst conditions (minimum solar activity). As a consequence, the yellow region expands as well.

The semi-controlled re-entry is feasible and convenient only if, in correspondence of the ballistic coefficient of the satellite, it exist a green region. Two cases have to be considered:

- The green region is lower-delimited by the worst conditions (minimum solar activity): the semi-controlled re-entry strategy is feasible and convenient no matter the

environmental conditions. The study of the altitude to target will be performed later on.
- The green region is lower-delimited by the AOCS constraint: the semi-controlled re-entry strategy is feasible and convenient only for some environmental conditions (especially in maximum solar activity). Depending on the current year of the solar cycle, the strategy might be convenient or not. Indeed, from an operational point of view, it might not be possible to wait for the maximum of the solar cycle to come, especially if the end of lifetime occurs during its minimum.

The value of the maximum length of the probable debris fallout zone, $Limp_{max}$, coming from the study of the phasing with the Earth, is used in order to evaluate the feasibility and the convenience of the semi-controlled re-entry with low thrusts, for both Gaussian and uniform distributions. The obtained ranges $h_{Tstop}$, synthetized in Tab. 5 for each considered satellite, show that there are margins with respect to the AOCS constraint, fixed at $hlim_{AOCS}$ = 150 km. Thus, the strategy is feasible.

*Table 5. Obtained ranges $h_{Tstop}$*

| | RANGE $h_{Tstop}$ [km] | |
|---|---|---|
| DISTRIBUTION | Gaussian | Uniform |
| PARASOL | 173 – 216 | 166 – 205 |
| SMOS | 188 – 241 | 180 – 228 |
| SPOT-5 | 171 – 213 | 162 – 198 |
| SPOT-5-like (i=51deg) | 177 - 210 | 173 - 204 |

### 3.6. Step 3: Propulsive phase

Once the range of $h_{Tstop}$ is known, it is possible to estimate the time and the mass of propellant required by the satellite to reach it, starting from its end of lifetime orbit. Since it is not required to do precise computations at this stage of the analysis, some preliminary assumptions can be made for simplifying the study:
- The EOL orbit is supposed to be circular.
- The thrust is considered to be the only perturbation.
- The thrust is continuous and constant.

The second assumption is not true at all, especially at low altitudes, where the atmospheric drag becomes stronger and stronger, even more than the thrust itself. However, since the atmospheric drag has the same direction and verse of the low-thrusts, it can be imagined itself as an additional thrust, helping the propulsive system to decrease the semi-major axis of the satellite. Thus, the results obtained with these assumptions will be pessimistic.

However, using the third assumption, it is not taken into account the fact that the satellite may be in eclipse for 35% of the re-entry duration, during which the low thrusts cannot be provided because of the lack of electric energy. Thus, with this assumption, the re-entry durations will be smaller than they would actually be, but the consumption of the propellant mass will be roughly the same.

Depending on the type of distribution, two different ranges of $h_{Tstop}$ were obtained. The total thrust time and the required propellant mass are computed, considering both limits of the range of $h_{Tstop}$ as final altitudes. The results for the studied satellites are given in Tab. 6.

From the results, it is possible to observe that the total transfer time is in the order of 1.5 to 3 months while the consumption of propellant mass is in the order of 1 to 2% of the satellites mass. Therefore, both of the parameters are acceptable, the first from an operational point of view, the second from a mass budget point of view. Moreover, the differences in the results between Gaussian and uniform distributions are very small, i.e. 1 to 2 days for the transfer time and 1 to 3% for the mass consumption. Thus, the type of distribution does not significantly affect the propulsive phase. However, it is necessary to remind that these results have been obtained considering constant thrust and neglecting the aerodynamic drag. The actual propulsive strategy will be the results of an optimisation study of the transfer trajectory, for which a compromise can be made between the total transfer time and the mass consumption.

*Table 6. Propulsive phase results*

| | RANGE $Dt_T$ [days] | | RANGE $Dm_P$ [kg] | |
|---|---|---|---|---|
| DISTRIBUTION | Gaussian | Uniform | Gaussian | Uniform |
| PARASOL | 45 – 49 | 46 – 50 | 2.10 – 2.30 | 2.15 – 2.33 |
| SMOS | 62 – 69 | 63 – 70 | 10.85 – 12.11 | 11.16 – 12.31 |
| SPOT-5 | 79 – 85 | 81 – 86 | 65.17 – 69.94 | 66.86 – 70.96 |
| SPOT-5-like (i=51deg) | 80 - 84 | 81 - 85 | 65.50 – 69.25 | 66.18 – 69.71 |

## 4. CONCLUSIONS AND PERSPECTIVES

A de-orbit strategy using electric propulsion has been envisaged in order to reduce the risks for human population with respect to uncontrolled re-entries within 25 years. It consists of continuously decreasing the altitude through low-thrusts until it is reached an altitude from which the probable impact points associated to the natural re-entry are foreseeable and placeable over mostly uninhabited regions.
Before analysing the de-orbit strategy in detail, it was necessary to evaluate its feasibility. In particular, it has been considered actual LEO satellites that are destined to naturally re-enter within 25 years. Indeed, this de-orbit strategy mainly addresses to this kind of satellites, which are not able to perform controlled re-entries, and whose uncontrolled re-entries could lead to high risks for human population.
This feasibility study has been divided in three steps, in order to deal with the risk assessment, the limitation imposed by the AOCS constraint and the propulsive phase.
The main conclusion that can be drawn from this paper is that a de-orbit strategy with low-thrusts seems to be feasible. Indeed, the feasibility study of this kind of re-entries has shown that it is possible to decrease the risks with respect to an uncontrolled re-entry within 25 years by a factor of 10. In addition, the same study has shown that the results present some margins with respect to the limitation imposed by the minimum altitude at which the AOCS is supposed to properly work, whatever the environmental conditions.
Concerning the propulsive phase, for the studied satellites, it has been obtained a total transfer time is in the order of 1.5 to 3 months while the consumption of propellant mass is in the order of 1 to 2% of the satellite's mass. Therefore, both of the parameters are acceptable, the first from an operational point of view, the second from a mass budget point of view.
However, being a feasibility study, several assumptions have been done in order to simplify the problem. Thus, more accurate studies have to be carried out, such as the analysis of the probable re-entry points leading to the minimum risk and the optimisation study of the whole re-entry trajectory, starting from the EOL orbit of the spacecraft and arriving to the targeted probable re-entry points. Other points could also be analysed such as collision risks or failure events during the propulsive phase, as well as eclipse constraints for instance.
Finally, if also these studies provide positive results, this de-orbit strategy may be used in actual missions, for satellites which are equipped with an electric propulsive system, because it would meet the requirements of the FSOA while minimising the consumption of propellant.

## 5. REFERENCES

1. LOI n° 2008-518 du 3 juin 2008 relative aux opérations spatiales (1), NOR: ESRX0700048L, JORF n° 0129 du 4 juin 2008, Texte n° 1.
2. A space act and a new foundation for a forward-looking space sector, http://www.cnes.fr/web/CNES-en/7406-french-space-act
3. Arrêté du 31 mars 2011 relatif la réglementation technique en application du décret n° 2009-643 du 9 juin 2009 relatif aux autorisations délivrées en application de la loi n° 2008-518 du 3 juin 2008 relative aux opérations spatiales, NOR: ESRR1103737A, JORF n° 0126 du 31 mai 2011, Texte n° 38.
4. Dan M. Goebel and Ira Katz, "Fundamentals of Electric Propulsion: Ion and Hall Thrusters", JPL Space Science & Technology Book Series.
5. Optimisation de rentrées contrôlées, GNC T.TCN.767456.ASTR 01/00, 12/12/2012.
6. Hourtolle C., Gaudel A., Blazquez A., "ELECTRA: Launch and re-entry risk analysis tool", ICATT 2012.
7. A. Gaudel, C. Hourtolle, J.F. Goester, N. Fuentes, "Risk assessment during the final phase of an uncontrolled re-entry", IAASS 2013.