FAST RE-ENTRY DEORBITATION WITH ACCEPTABLE RISK LEVEL

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ABSTRACT

One important action, needed to limit the space debris population increase in the low Earth orbit region, is to deorbit all space systems after the mission lifetime. This can generally be done either by controlled direct re-entry or by moving to an orbit which will ensure a natural decay of the space object within a limited time span, as short as possible.

Direct controlled re-entry is, of course, the most efficient way to proceed with regard to lifetime reduction and human casualty risk control. However, it is demanding in terms of means and requires a well dimensioned propulsion subsystem to perform the last re-entry burst. In particular, controlled re-entry is not feasible with low thrust propulsion

Uncontrolled re-entry is less efficient, leaving the space objects uncontrolled during years before effective re-entry, but is simpler to achieve. However, the risk towards human population at the time re-entry occurs should be limited and several national or agency regulatory texts require the human casualty risk to be lower than 10^{-4}.

Between these two routes, fast re-entry deorbitation appears as an attractive solution: manoeuvres are performed until the satellite is left very close to its effective re-entry, human casualty may be limited, and this method could also be available with low thrust propulsion.

This paper will analyse several key elements regarding final orbit, casualty risk and manoeuvre strategy implementation in order to progress towards operational feasibility of fast re-entry deorbitation.

The first part will introduce fast re-entry deorbitation concept, constraints and principles. A second part will address the final decay phase: casualty risk and target re-entry orbit choice. The third part will discuss manoeuvre strategies to reach this target orbit with chemical or electrical propulsion.

1 INTRODUCTION

1.1 Concept

For an uncontrolled re-entry, the effective casualty area at the time the re-entry occurs may be everywhere on the globe, inside a latitude bandwidth determined by the satellite inclination.

Fast re-entry deorbitation is a good improvement to debris mitigation regarding two aspects:

- It drastically reduces the total residual lifetime and, thus, the risk of explosion or collision which creates new debris before effective re-entry.
- It may reduce the casualty risk to an acceptable value, by limiting the length of the possible on-ground casualty area and targeting favourable geographic conditions (see example in Figure 1).

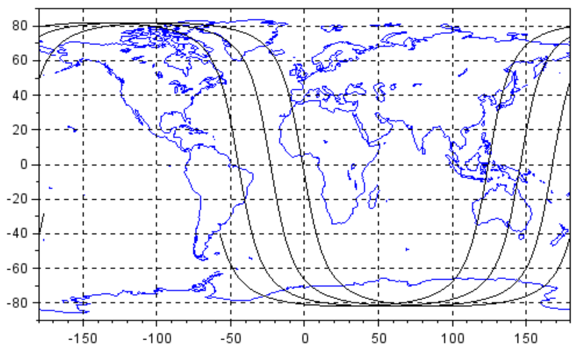

Figure 1. Fast re-entry deorbitation ground track length

1.2 Constraints

We will call "ground track" the estimated area where debris are likely to reach the ground: it is not really a ground track as it corresponds to a set of possible re-entry points.

Several elements with uncertainties contribute to this pseudo ground track, among which:

- ballistic coefficient of intact object,
- surviving fragments and their aerodynamic characteristics,
- atmospheric density model,

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- Solar activity level.

These elements are not well known and vary with time. It appears very difficult to predict with good precision the possible impact zone of an object which is due to re-enter soon. Therefore, the duration of natural decay at the end of deorbitation must be as short as possible in order to reduce the final ground track length: typically one day or so.

Before natural decay, the satellite will execute manoeuvres in order to reduce its altitude. At the beginning, manoeuvre will not be a problem but, with the altitude decreasing, the satellite will have difficulty to maintain its nominal attitude control capability due to the increase of aerodynamic forces and torques: this limit in altitude depends on the satellite geometry and inertia. It may typically happen below 250 km.

For electric propulsion there is another important constraint regarding the electrical power budget: long thrusts will discharge the battery and some amount of time will have to be devoted to battery charge. Available thrust time will, thus, be limited. This limit will depend on various parameters such as: local time, inclination, eccentricity, apogee/perigee orientation, eclipse duration, season, Solar panel orientation during thrusts or altitude.

1.3 Main principles

The deorbitation will consist in a variety of tangential negative manoeuvres that will decrease the altitude until reaching a final re-entry orbit. This final orbit must ensure a rapid decay ending in a targeted zone on Earth that will guarantee a low casualty risk.

If a circular orbit is kept during the descent, the satellite will be on a quasi-circular orbit when manoeuvres are stopped before decaying naturally. The decay time starting from a 250 or 300 km circular orbit is one week to a few weeks (for a 1 ton / 10 m^2 satellite), which is too long to ensure a short ground track length.

Therefore it is necessary to adopt an elliptical descent strategy rather than a pseudo circular one: this strategy may allow continuing manoeuvres at apogee, even if perigee is very low (see Figure 2). This will be possible as long as attitude stability can be maintained at these very low altitudes, which may need a dedicated attitude pointing in order to minimize the atmospheric torque. Nominal attitude control must be reacquired after each perigee pass and before next apogee manoeuvre.

Figure 2. Elliptic deorbitation

2 TARGET RE-ENTRY ORBIT

Once the satellite is in the re-entry orbit, the manoeuvres are no longer possible. Thus, this re-entry phase should be as short as possible in order to reduce the uncertainties and reduce the impact zone.

2.1 Influence of some parameters

Let us examine several elements that impact the re-entry duration.

2.1.1 Altitude

Of course the altitude is a major element that impacts re-entry duration: lower the altitude, faster the re-entry.

2.1.2 Solar activity

Depending on the solar activity, the satellite decays faster or slower. A faster decay takes place when the solar activity is high, which also reduces the possible impact zone dimension. Thus, high solar activity is favourable to improve the final re-entry, as it can be seen in Figure 3.

2.1.3 Perigee argument

The position of the initial perigee argument in the final re-entry orbit has a huge impact in the duration of this phase and in the impact zone dimension. Indeed, due to the Earth potential term J3, re-entry orbits with arguments of perigee in the North hemisphere are more prone to a faster descent than the orbits with South perigee arguments, as shown in Figure 3. The possible impact zone is also smaller for the cases with the argument of perigee in the North than the ones in the South. However, the best position is to be as close as possible to the Equator, where the Earth potential term J2 produces minimum decay duration.

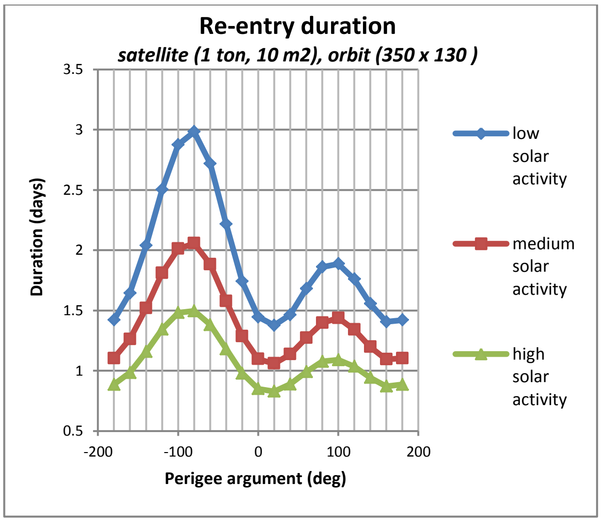

Figure 3. Re-entry duration depending on perigee argument (ω) and solar activity.

This difference between the North and the South hemisphere is independent from the season because all the tests performed in both equinoxes and solstices offered this exactly same behaviour. Indeed, the difference between North and South is only due to Earth potential and independent from the Sun position.

2.1.4 Low impacting parameters

Some of the input parameters have small or negligible effect.

2.1.4.1 Season

Some dates are slightly more favourable because the atmospheric density around the Earth depends on the season. Figure 4 shows the atmospheric density at equator for very low altitude (130 km) given by MSIS2000 model where a re-entry in the months of September-October is faster.

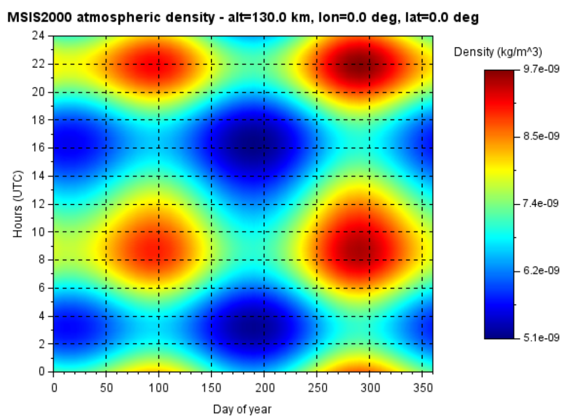

Figure 4. Density depending on the local hour and day of the year.

2.1.4.2 Local hour

The mean local hour (angle between the Sun direction and the orbital plane) have a small effect during the re-entry orbit.

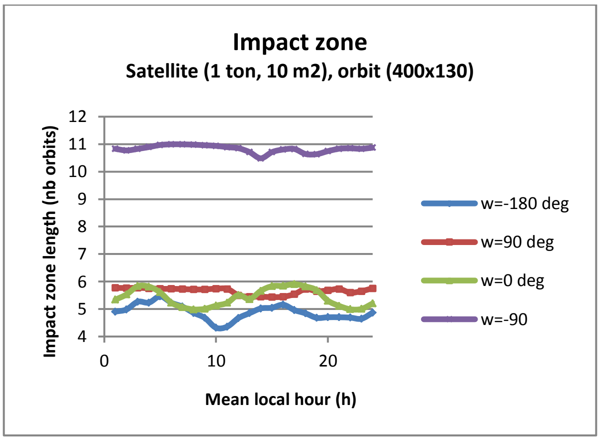

Figure 5. Negligible differences in the impact zone length depending on the local hour.

2.2 Casualty risk

The use of a fast re-entry deorbitation is mainly justified if there is a reduction of the risk compared to an uncontrolled re-entry. Thus, the gain in terms of risk must be demonstrated.

2.2.1 ELECTRA final re-entry orbit mode

The ELECTRA program computes the casualty risk at ground. It has four modes:

1) Risk at launch (RL)
2) Risk for controlled re-entry (RC)
3) Risk for final re-entry orbit (RF)
4) Random risk (RA)

The third mode was used to compute casualty risk and ground track length.

The principle is to consider a dispersion on ballistic coefficient, which represents the uncertainty on atmospheric density model, solar activity and effective attitude during re-entry. In this mode, numerous points are provided for the intact vehicle before fragmentation, which are geographically regularly spaced to fit with the population grid spacing, in order not to miss a small but populated area. Each point has its associated probability of occurrence.

The second part consists in continuing the propagation starting from these regularly spaced points and splitting the satellite into fragments. At this point the satellite is no longer considered as a single object but as a group of independent debris. The survival debris are propagated down to the Earth where total casualty risk is computed for each debris trajectory. Finally, global casualty risk is obtained by summing casualty risks for each trajectory weighted with its probability.

2.2.2 Perigee argument impact

In the current revision of ELECTRA, the RF mode runs with a fixed solar activity model. The following hypotheses have been taken, corresponding to a typical Pleiades-like observation satellite:

- 1 ton and 8.5 m2,
- local time at ascending node: 22h30,
- final re-entry orbit altitude: 130 km / 350 km,
- ballistic coefficient: 52 kg/m^2, with dispersion : ±10% uniform.

2.2.3 Perigee argument impact

The first results confirm a different behaviour depending on the initial perigee argument, as presented in paragraph 2.1.3. The minimum impact zone length is around 3 orbits for a perigee near equator, higher for a perigee near the poles. Moreover, a perigee in South hemisphere will lead to a greater impact zone length than a perigee in the North hemisphere (see Figure 6).

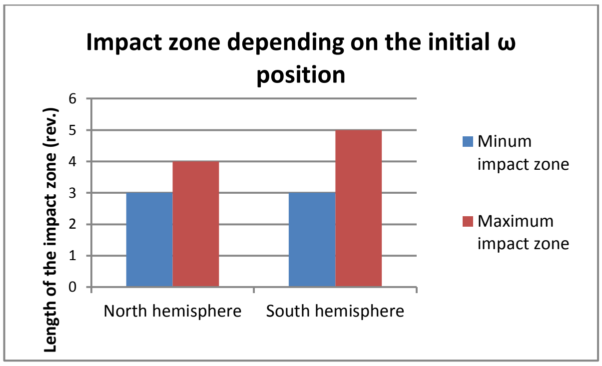

Figure 6. Length of the impact zone, in terms of orbital revolutions, depending on the re-entry perigee argument position.

2.2.4 Longitude phasing impact

The risk of the ground track impact zone varies depending on Earth's phasing. Indeed, the most populated zones present greater risks, than, for example, the oceans. Thus, the positioning/phasing with the Earth has a huge impact. The risk also depends on the number of fragments and their casualty area; the bigger the object, the higher the risk.

The risk has been computed with the method described in § 2.2.1 for several orbits with different perigee arguments and for two sets of survival debris: 15 or 6 debris, including or not structure panels. Nowadays, as there is a certain concern about the ground risk, satellites are more prone to be designed with materials that favour their burning during the re-entry. Then, the case of 15 debris represents the worst possible case where the structural panels are not burned, and the 6 debris case represents an improved design or more favourable scenario where these panels are burned.

For each case, the risk has, then, been re-estimated with longitude translations representing different Earth phasing possibilities with initial orbit: the best Earth phasing gives the minimum risk value for a given initial orbit. These minimum risk values are shown in Figure 7 for different perigee arguments.

In all cases, the minimum risk is below the random risk (uncontrolled re-entry). Thus, a reduction of the impact length and an adequate Earth phasing can reduce the risk provided by an uncontrolled re-entry.

It can be seen that for the 15 debris case, only perigee arguments in North hemisphere allow to find a longitude phasing with risks below the specified limit risk (10^{-4}), being compliant with the FSOA (French Space Operation Act).

For the 6 debris case, a favourable longitude phasing is easier to find whatever the perigee argument is. Indeed, the risk level is one order of magnitude lower than with 15 debris case: this shows the extreme importance of minimizing the number and surface of survival debris. It is also important to improve our knowledge and predictive capability regarding satellite fragmentation and surviving debris: for the same satellite, different tools (eg NASA DAS, ESA DRAMA or CNES DEBRISK) may give different sets of survival debris, which has a non-negligible impact on final risk estimation.

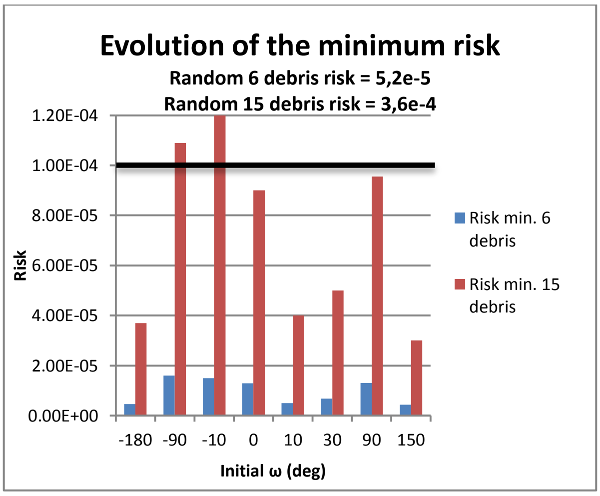

Figure 7. The minimum uniform distribution risk, taking into account the best longitudinal positions of the impact zone.

The maximum risk may also be evaluated (Figure 8): it corresponds to the worst possible Earth phasing in longitude. The risk value is, then, higher than the uncontrolled re-entry. One could tend to believe that in some cases the uncontrolled re-entry is better, but it is not really the case. In fact, the results are worse due to the definition of this risk itself. As in the fast re-entry deorbitation the ground track length is finite (length of several orbits), when it is badly placed and touches very populated areas, the risk is increased. The random re-entry evaluates all the zones of the Earth from a minimum to maximum latitude, smoothing the higher risks. A few days before an un-controlled re-entry, its trajectory would be known and could perfectly match one of the bad phasing possibilities, producing even a higher risk.

Nevertheless, in this example all the maximum risks are higher than the 10^{-4} FSOA limit, even in the favourable case of 6 debris. It means that the phasing with the Earth is mandatory in order to obtain an acceptable risk.

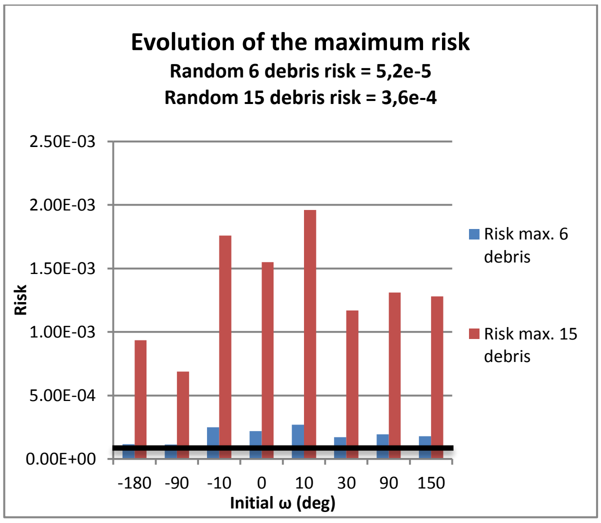

Figure 8. The maximum uniform distribution risk, taking into account the worst longitudinal positions of the impact zone.

Thus, the safest re-entry orbit should come from the good perigee argument and good date / longitude phasing to target a chosen Earth zone, in order to minimize the risk and be compliant with the FSOA.

The previous risks were determined through a uniform risk distribution. However, it is also possible to use a Gaussian distribution which reduces the risks. The Figure 9 shows how, when using a Gaussian distribution, the risk is acceptable regardless of the perigee argument value. The choice of the distribution type depends on the confidence in the ballistic coefficient dispersions. A Gaussian distribution is acceptable if there is a good confidence in the mean ballistic coefficient. On the other hand, if this is not the case, a uniform distribution would be more appropriate.

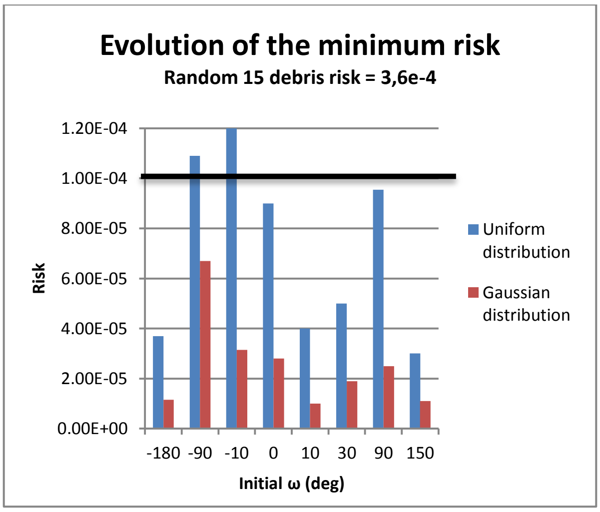

Figure 9. Risk comparison between a uniform distribution and a Gaussian distribution.

3 DEORBITATION STRATEGY

This section addresses the manoeuvring phase which brings the satellite to the re-entry orbit.

These studies are done considering a typical observation satellite with Pleiades-like characteristics: see Table 1.

Orbit altitude	700 km
Eccentricity and ω	Frozen, North Pole
Local hour	22.5 h
Mass	1000 kg
Surface	10 m^2

Table 1. Reference satellite characteristics : ~ Pleiades

3.1 Chemical propulsion

The following studies present some results reproducing a case with chemical propulsion.

Table 2 shows the propulsion characteristics of Pleiades satellite.

DeltaV / man	2.5 m/s
Thrust	1N · 4 engines
Manoeuvre duration	10 min
Isp	210 s

Table 2. Pleiades propulsion system characteristics. "Isp" is the specific impulse.

3.1.1 Direct re-entry

With chemical propulsion, it is possible to perform a controlled direct re-entry, which is the best way to ensure security and comply with international and national regulations. However direct re-entry is very demanding: 180 m/s are needed to re-enter from the mission orbit.

On the other hand, it is also possible to deorbit the satellite through a lot of small manoeuvres executed to decrease its altitude down to 250 km for example, and then perform one final large thrust to re-enter. This final thrust would require around 90 m/s, which means 6 hours thrust with 4N propulsion capacity. This is of course not realistic. Direct re-entry needs a powerful and dedicated propulsion subsystem with a high thrust capacity (> 100 N) in order to be able to do this final re-entry manoeuvre rapidly enough (10 to 15 minutes).

3.1.2 Fast re-entry deorbitation

Fast re-entry deorbitation allows to perform operations with a standard propulsion subsystem, and to save the consequent hydrazine mass needed for the final re-entry thrusts.

The deorbitation can be performed through two different strategies: one that penalizes the duration and the other one that penalizes the consumed mass.

3.1.3 Elliptical strategy

This strategy performs only decelerating apogee manoeuvers that lower the perigee. On the other hand, the apogee decreases naturally due to a higher drag force near the perigee. The chemical engine is powerful, so the frequency of manoeuvres becomes an important parameter. Indeed, if the manoeuvers are too frequent, the perigee will reach the target altitude too early, with an apogee still too high. A valid deorbitation strategy should find the perfect frequency of apogee manoeuvers to reach the target re-entry orbit: right apogee and perigee altitude at right date. For Pleiades satellite case, the right frequency is one manoeuver every two days, as shown in Figure 10.

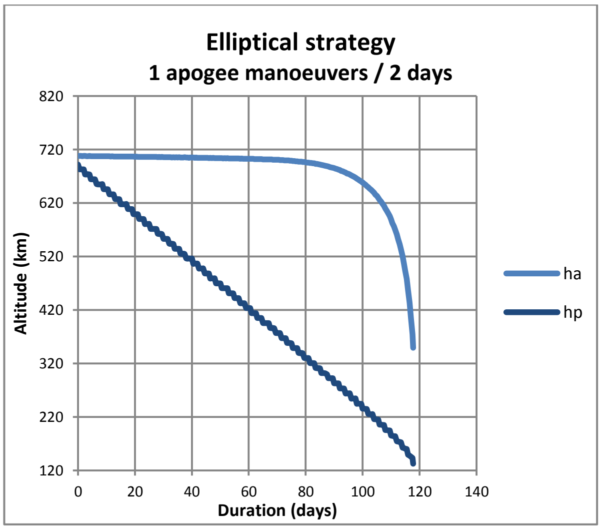

Figure 10. Elliptic deorbitation strategy with chemical propulsion. Ha and hp are apogee and perigee altitude.

This strategy allows saving fuel because natural drag force at perigee is used to decrease apogee instead of the propulsion system, but it requires a lot of time: four months.

3.1.4 Circular strategy

It is possible to add, to the previous strategy, perigee manoeuvers which lower the apogee.

A theoretical example is shown in Figure 11 with apogee and perigee manoeuvres performed at each orbit (in a real case the frequency would certainly be lower, or there would be regular interruption which would enable a correct orbit determination). When the altitude is too low, perigee manoeuvre are stopped and apogee manoeuvre continue until target perigee altitude is reached.

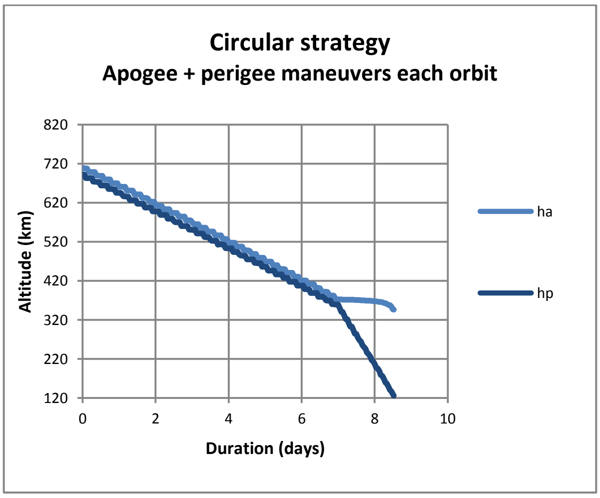

Figure 11. Circular deorbitation strategy with chemical propulsion.

This strategy enables a higher frequency of manoeuvres and an important reduction of the duration but the fuel consumption increases.

3.2 Electrical propulsion

The electrical engines provide smaller thrusts at higher specific impulse. Thus, the strategy is completely different from the one using chemical engines. In this study, the Pleiades satellite is also used as reference but with electrical propulsion characteristics.

Thrust	82 mN
Manoeuvre duration	20 minutes max. 10 minutes min.
Isp	1400 s

Table 3. Typical electrical propulsion system characteristics.

3.2.1 Basic strategy - main influence parameters

The simpler re-entry strategy is applied: one maximum apogee manoeuvre at each revolution to lower the perigee. An example of result can be seen on Figure 12: indeed this leads to long operations (around 100 days).

It is important to evaluate the effect of different parameters.

3.2.1.1 Mass and surface

Mass and surface have a different influence.

The satellite mass impacts on the manoeuvre efficiency during all deorbitation. A small mass is favourable because the manoeuvres are more effective, the perigee decreases faster, and the total duration is reduced.

The satellite surface has an impact only when the drag force is more important, so, in the lowest altitudes near the end of deorbitation. The satellites with a higher surface cause more drag and, thus, the apogee decreases faster: altitude limit for apogee manoeuvre may be reached before perigee has reached a low enough altitude. On the other hand, the satellites with smaller surface cause less drag, the apogee decreases slower, incrementing the available duration for lowering the perigee. Then, the surface changes principally the final perigee altitude, being possible to obtain it lower when the surface is smaller.

As a conclusion, the deorbitation becomes more difficult when the satellites are huge in terms of mass or with large surfaces.

3.2.1.2 Batteries charging

The electrical propulsion needs a lot of energy and depends on the batteries charging. Thus, the duration of the manoeuvers cannot always be maximum: it depends on the battery charging time available (outside eclipses).

Simple hypotheses were taken into account to show the impact, supposing that the batteries charging is possible during the manoeuvres when they are illuminated:

- Manoeuvers done at each apogee pass.
- Manoeuvre duration is maximum when it takes place outside the eclipse.
- Manoeuvre duration is reduced when it takes place totally or partly inside the eclipse.

The Figure 12 shows the difference between using constant duration manoeuvres and eclipse dependant durations. Logically, the strategy with constant manoeuvres duration is more efficient, reducing the whole deorbitation duration. Moreover, the perigee reduction rate is constant, as each manoeuver produces the exactly same perigee reduction.

On the other hand, in the case of variable manoeuvre duration, the perigee reduction rate is not constant. The duration depends on the apogee position in relation to eclipses, with an apsis axis rotation of ~100 days period; it becomes more difficult to predict the total duration and the final perigee altitude.

In the previous simulations the starting date, local hour or perigee argument did not have any important impact. Nevertheless, the position of the eclipses changes depending on these input parameters and so does the deorbitation simulation, the final result becoming very dependent from the input characteristics. The orbits with mean local hours at 6h or 18h are exempt of eclipses, thus, they can use the maximum duration of the manoeuvers all the time.

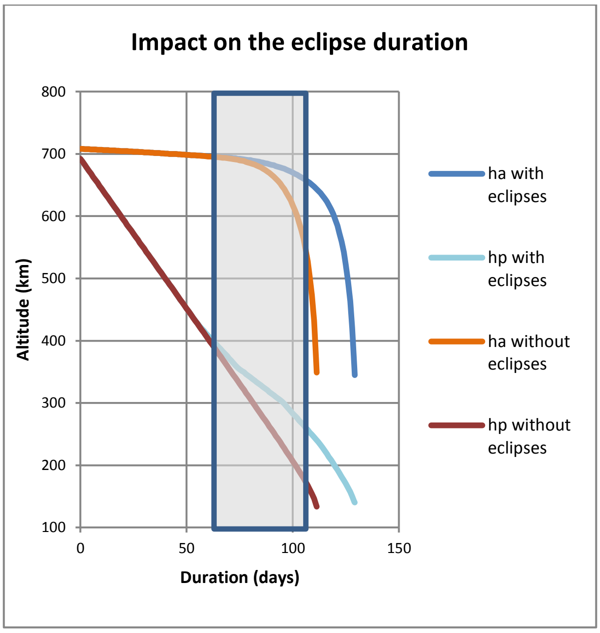

Figure 12. Simulation with and without duration reduction due to eclipses. The shadowed part shows the points where the apogee is in eclipse.

There is another problem associated to the batteries charging. When the perigee is lower than a certain altitude (typically 250 km), the satellite loses its attitude performance and may not be able to orient solar panels to charge the battery. Then, the manoeuvers duration is even more reduced because of this lack of charging time. This is another constraint which adds complexity to the deorbitation and degrades the final perigee altitude.

3.2.1.3 Solar activity

The solar activity has an impact in the final characteristics on the re-entry orbit.

- A higher solar activity implies a faster apogee reduction. Then, the available duration to reduce the perigee is smaller and the final perigee altitude remains higher.
- A lower solar activity decreases slower the

apogee and, thus, there is more time to apply apogee manoeuvers that lower the perigee. Moreover, the apogee manoeuvres are more effective, improving the perigee descending rate. So, surprisingly, in this phase a low solar activity will be favourable, allowing reaching a lower perigee altitude.

During the manoeuvring phases, a low solar activity improves the results. However, once the satellite is in the re-entry orbit, the high solar activity is the one that improves them.

3.2.2 Strategies depending on solar activity

As it was explained in the previous section, there are lots of parameters that influence the deorbitation. In this section, three solutions are presented for three different cases: low, medium and high solar activity.

The duration of the manoeuvres is variable, as it is shown in § 3.2.1.2. Moreover, the duration is even more reduced when the perigee attitude is no longer controllable.

The requirements of the final re-entry orbit are:

- Orbit 350x130 km.
- The argument of the perigee must be set in the North hemisphere.
- In order to maximize the manoeuvre duration in the final phase which is the most critical, the apogee cannot end in eclipse. As it is an orbit with local hour 22.5h, the argument of the perigee should finish close to 0º.

3.2.2.1 Final perigee argument constraint

With the last two constraints the position of the perigee argument is very limited. It should be set close to the eclipsed Equator but only in the North hemisphere part. For an orbit with local hour 22.5h, ending during the Summer solstice gives the worst condition because the eclipse is much positioned in the South hemisphere, as it is shown in Figure 13. In order to prove the validity of these strategies during the whole year, simulations are performed in this worst case: ending close to Summer solstice, the feasibility for any other date being automatically demonstrated.

It is possible that using different satellites or different threshold altitudes, the strategy could only be performed ending at the winter solstice, imposing a huge constraint in the deorbitation strategy.

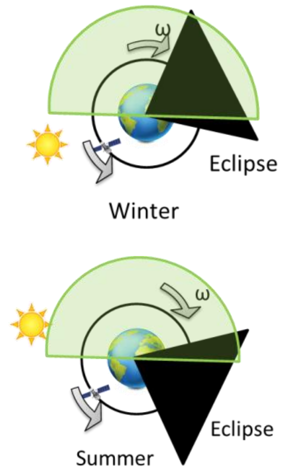

Figure 13. Shadowing effect difference between summer and winter (in black). Target ending zone (in green).

3.2.2.2 Setting the initial perigee argument

LEO orbits are nearly circular. The frozen eccentricity often used for stability reason is around 10^{-3}, perigee oriented towards North direction. Apogee manoeuvres will increase eccentricity and the apsis axis will rotate with a period close to 100 days.

To reach the desired perigee argument at the end of deorbitation, it is necessary to initialise the perigee argument to the appropriate value at the beginning of operations, taking into account the expected rotation of the apsis axis during the operations.

For cost reason, it will be possible neither to maintain the perigee argument in a constant position during operations nor to move the apsis axis to a desired value at the end of operations when eccentricity is higher. It would neither be appropriate to wait until achieving naturally the target perigee argument because the waiting orbit at low altitudes is very expensive due to high drag force and should only be used in order to phase the satellite with the Earth.

Thus, the positioning of the perigee argument should be set as soon as possible: dedicated manoeuvre will ensure the appropriate setting of apsis axis. This will induce an extra cost, except in the ideal case where the desired initial perigee argument is the same as station-keeping one.

3.2.2.3 Initial apogee raising

Once the perigee argument is well positioned, the deorbitation manoeuvers start. The simpler strategy is to perform tangential braking manoeuvers at apogee to lower the perigee.

Unfortunately, this solution is not always sufficient: it may happen that when the satellite reaches the minimum apogee altitude, the perigee altitude is still too high. This problem occurs when there is not enough time to decrease the perigee because the apogee has decreased too fast (the higher the solar activity, the more common this problem is).

Different solutions can be envisaged: maintain the apogee during operations, increase the initial apogee and leave it evolve naturally afterwards, or increase the initial apogee and maintain it afterwards. All these solutions slow down the reduction of the apogee altitude and give more time to achieve the desired perigee lowering.

Maintaining the apogee altitude requires perigee manoeuvres during deorbitation. The available batteries charging time is shared between perigee and apogee manoeuvres, so, the apogee manoeuvre frequency is reduced when both manoeuvres are performed simultaneously, and the cost and duration of operations increases significantly. These are the reasons of why this method has been discarded.

Then, the best solution is to perform only perigee manoeuvers at the beginning of the strategy to reach the target apogee altitude and then continue with only apogee manoeuvers to lower the perigee. Of course the drawback here is that an initial apogee altitude, which depends on predicted drag force and solar activity level, should be assumed. The effective drag force may be different during deorbitation and it will have to be compensated during operations.

3.2.2.4 Results for several solar activity levels

The results of these simulations are shown in Figure 14 and Table 4. In order to end with the same conditions, the cases with higher solar activity need to increase the initial apogee causing a longer duration and a higher consumption. The perigee argument change is also different in each case because it depends on the deorbitation duration.

	Low solar activity (F10.7=65, Ap=0)	Medium solar activity (F10.7=140, Ap=15)	High solar activity (F10.7=200, Ap=30)
Duration (days)	148	159	177
Consumption (kg)	12	14	15
ha final (km)	350	350	350
hp final (kg)	130	130	130
ω initial (deg)	90→138	90→160	90→177
ω final (deg)	0	0	0
deltaV (m/s)	164	186	208
ha max (km)	710	800	888

Table 4. Main characteristics of the deorbitation strategies when re-entering during Summer solstice.

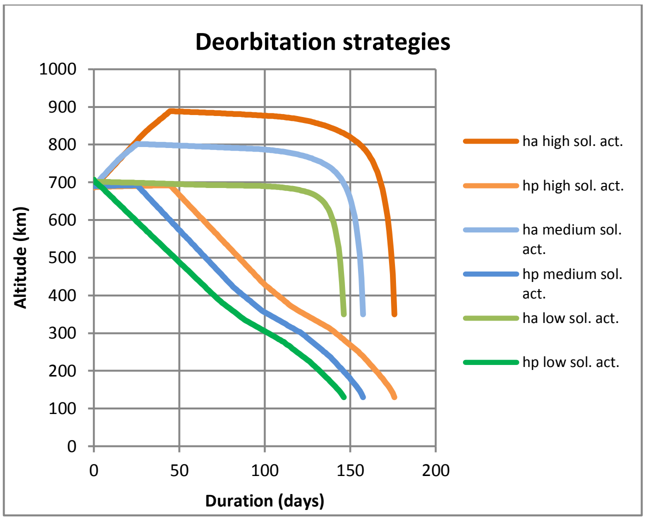

Figure 14. Perigee and apogee altitudes during deorbitation depending on the solar activity.

3.3 Comparison between the strategies

The Figure 15 shows a summary between the different strategies. The chemical propulsion implies the highest consumption; however, it can lead to a reduction of the duration (9 days instead of 159!). Moreover, the chemical propulsion has another very important advantage: the duration of the manoeuvres is independent of the eclipses. Thus, the chemical propulsion is easier to simulate and to foresee. Its velocity increment is also higher, so, in case of manoeuvre avoidance it would also offer better performance. Nevertheless, it has a main disadvantage of needing a huge mass and the necessary volume to contain it. The Figure 15 also shows how the elliptical chemical strategy does not offer a significant advantage with respect to the electrical strategies: its duration remains long and its consumption is much bigger. The circular chemical strategy is more interesting with a huge improvement in duration.

On the other hand, electrical propulsion is less effective in term of duration: it lasts around 150 days, but very efficient in term of consumption with only 15 kg compared to more than 100 kg with chemical propulsion. The results are better when the solar activity is lower.

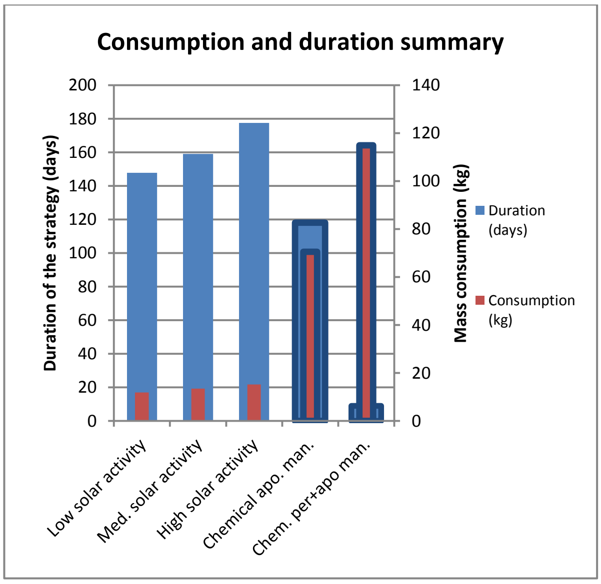

Figure 15. Summary of the durations and consumption between the different electrical and chemical strategies. The two chemical options are outlined in black.

4 CONCLUSION

Fast re-entry deorbitation seems feasible and represents a good alternative to the uncontrolled re-entries when a totally controlled re-entry is not feasible. With a good phasing of final perigee argument and longitude phasing, it is possible to reduce the casualty risk to an acceptable value.

The deorbitation manoeuvres could be performed using chemical or electrical propulsion. Chemical propulsion allows short operations but needs a lot of fuel. Electrical propulsion allows saving mass but means long operations, even with very frequent manoeuvres. It also requires high battery capacity and charging capacity.

The results presented in this study where based on a particular LEO satellite with its mass and geometry; on a Sun-Synchronous orbit with an ascending node in eclipse: all quantitative results are of course dependant of the satellite and orbit characteristics. However, the influence of several parameters has been studied, leading to some useful elements:

- Final perigee argument is better in North hemisphere and should be close to the Equator: it is necessary to anticipate perigee argument rotation at the beginning of operations.
- The perigee of the re-entry orbit should be low enough to assure an acceptable value of casualty risk. The perigee is lowered through apogee manoeuvers, thus, the altitude of the apogee should be high enough to perform these manoeuvres until achieving the targeted perigee. In case of high or medium level of solar activity, it requires an initial raising of the apogee.
- Longitude phasing with Earth is important: a good control of the operations duration is needed to preserve perigee and apogee altitude decrease rate.

Several elements have not been studied, among which the sensitivity of satellite, its orbit characteristics and the manoeuvring phase real time control to achieve the four important rendez-vous: perigee orientation, perigee altitude, apogee altitude and longitude phasing, taking into account variation of the solar activity, manoeuvre performance... Management of degraded cases such as collision avoidance or safe mode recovery also needs to be studied.

5 REFERENCES

1. S A. Gaudel, C.Hourtolle, J.F. Goester, M. Ottavian (2015). De-orbit strategies with low-thrust propulsion. *Springer International.*
2. C. Hourtolle, S. Delattre, J.F. Goester, E. De Pasquale (2015). Studies about a shallow re-entry for ATV-5. *25th International Symposium on Space Flight Dynamics ISSFD October 19 – 23, 2015, Munich, Germany*.
3. M. Ottaviani, J.F. Goester, A. Gaudel-Vacaresse (2013). Stratégies de désorbitation à poussées faibles. *CNES internship report*.
4. B. Hassan, C. Hourtolle (2015). Maneuvering strategies for semi-controlled reentries applied to low Earth orbits. *CNES internship report*.
5. G. Colas, A. Gaudel-Vacaresse (2014). Computation of risks during uncontrolled atmospheric re-entries on final orbit. *CNES internship report.*