ASSISTED REENTRY FOR ALL-ELECTRIC LEO SPACECRAFT

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ABSTRACT

There have historically been only two strategies to choose from when designing a deorbiting capability for future LEO missions: *controlled reentry* requires a very large deorbit boost and thus a powerful propulsion system, while *uncontrolled reentry* is only allowed for small satellites that (almost) entirely burn up when reentering.

Assisted Natural Reentry (or assisted reentry for short) intends to fill the continuum between these two extremes, thus offering more design flexibility to comply with the 10^{-4} rule in terms of ground risk.

The main rationale is that if the orbital plane can be phased correctly with the Earth and the distribution of potential reentry locations can be confined to at most a couple of revolutions, the vast majority of reentries will occur above oceans, reducing the casualty risk by an order of magnitude with respect to uncontrolled reentry.

We present the design and performances of an assisted reentry strategy tailored for an all-electrical LEO satellite, i.e. one with a very low thrust/mass ratio which only allows quasi-static perigee lowering and offers minimal control authority. We demonstrate with extensive simulation campaigns that despite high uncertainty in aerodynamic and atmospheric models and large density swings due to unpredicted changes in solar activity, the final dispersion can be kept within 2 orbits of the nominal reentry location, reducing casualty risk by a factor >5.

1 ASSISTED REENTRY: RATIONALE AND REQUIREMENTS

1.1 Rationale

Assisted reentry strategies are motivated by the need for future missions to comply with end-of-life disposal regulations to mitigate the accumulation of space debris. Over the years, these regulatory requirements have become stricter, with thorough enforcement first by ESA (see [1]), and now by sovereign states, with a formal introduction into French law in 2008 (see [2]). For missions in low-earth orbit, end-of-life disposal requires de-orbiting, by provoking or guaranteeing atmospheric reentry, with the two following approaches, roughly depending on spacecraft size:

- either a *natural reentry* (or uncontrolled reentry), which only requires lowering the perigee to ~500 km to take advantage of residual atmospheric drag, such that the spacecraft re-enters in less than 25 years. This option is acceptable only for smaller satellites, for which demise analyses prove that the spacecraft is entirely destroyed in the upper atmosphere or that the probability that the surviving fragments can cause a casualty is less than 10^{-4}.

- or a *controlled reentry*, when the fragments reaching the ground are too large or too numerous and the resulting casualty risk would exceed the 10^{-4} limit in case of an uncontrolled reentry. Controlled reentry requires a large propulsion capacity to lower the perigee down to 40 km in one big apogee manoeuvre.

Assisted reentry (short for 'Assisted Natural Reentry') is intended as an intermediate solution for medium-sized satellites, relevant when the uncontrolled casualty risk estimated from breakup analyses is moderately above the 10^{-4} limit. If the perigee can be lowered to ~150 km while still controlling the satellite, the uncontrolled part of the descent will be short enough to know where the reentry will occur with an uncertainty less than a few revolutions. By precisely timing the descent so that the last orbital arc is correctly phased with the Earth, we can then ensure that the vast majority of potential reentries occur over oceans or otherwise uninhabited regions, substantially reducing the global casualty risk compared to an uncontrolled reentry.

1.2 Dispersion performance requirements

Since the final portion of the descent is uncontrolled, errors in the initial conditions and uncertainties on the aerodynamics and the atmosphere will disperse the potential reentry locations. Approximating the distribution by a Gaussian distribution (with standard deviation σ_N) centered on a nominal reentry location, it is possible to estimate the reduction in casualty risk (compared to uncontrolled reentry) for every possible nominal location and for every possible value of σ_N. Selecting the optimum nominal location for each value of σ_N, it is then possible to derive the chart in Figure 2.

From the value of the uncontrolled casualty risk predicted by the demise analyses (using dedicated tools such as Debrisk and Electra [3]), we can determine the required risk reduction ratio. For instance, if the estimated risk for an uncontrolled reentry is 3×10^{-4}, we need to reduce the risk by at least a factor 3. The chart then tells us that such a reduction ratio is achievable if the dispersion σ_N can be kept below 1 revolution, i.e. the along-track 1-sigma uncertainty on the actual reentry location must be less than ±1 orbit.

Additionally, the phasing of the orbit (i.e. the timing of the passage at the last ascending node) must be sufficiently accurate to remain within the optimum longitude corridor where the ground track is mostly oceans: limiting the longitude error to ±2.5 deg requires a timing accuracy of ±10 minutes.

Developments conducted by ADS with CNES and ESA since 2015 have shown that this can be done with electrical propulsion: even with modest thrust-to-mass ratios, it is often possible to control the descent with sufficient accuracy and down to a sufficiently low altitude to keep the final dispersion within an acceptable range, thus complying with the 10^{-4} limit

1.3 Main principles and key objectives

The main descent strategy consists in performing electrical apogee manoeuvres to lower the perigee, down to progressively lower and lower altitudes. Below a certain perigee altitude, drag becomes too high and the process is irreversible: the operators then have to commit to a reentry date and time such that the orbital plane (and the argument of perigee) is optimally phased with the Earth's rotation (and the equator respectively).

Later on along the descent, drag torques near the perigee can become too large for the attitude control system if the spacecraft is left in its mission attitude: two slew manoeuvres are commanded before and after each perigee pass, to align with the equilibrium attitude and thus minimize transverse aerodynamic torque.

All the while the apogee manoeuvres are adjusted (based on orbit determination observations) in order to stay on track with the reference descent profile despite all the uncertainties and variability.

The objective is control the descent down to the lowest feasible interface orbit (e.g. 150 × 300 km), after which the satellite is left uncontrolled until it breaks up (at around 80 km) and reenters. In order to limit the dispersion of the uncontrolled phase, the interface orbit must be reached accurately and at the right time.

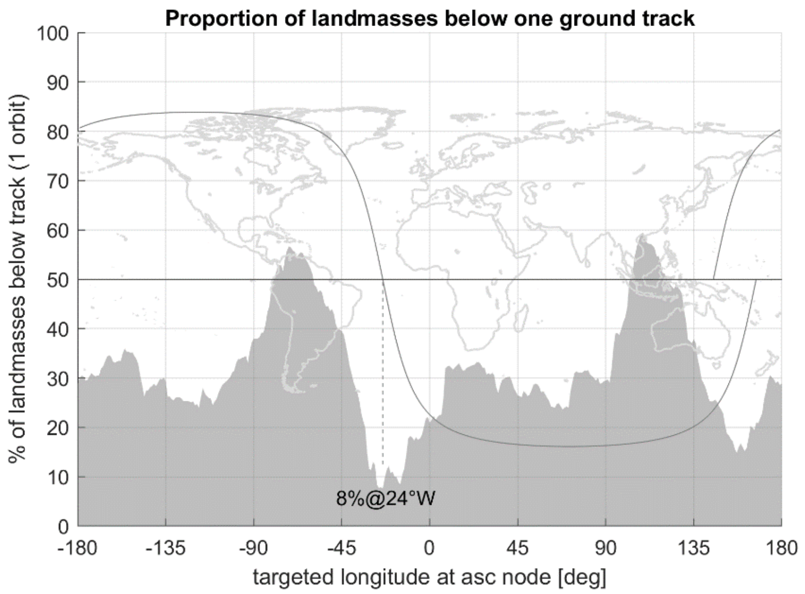

Figure 1 - Finding the optimum phasing with the Earth's oceans

For a 98° SSO orbit, the optimum longitude of the ascending node is 24W: the ground track is 92% oceans, and thus only 8% landmasses (4 times less than the 29% world average).

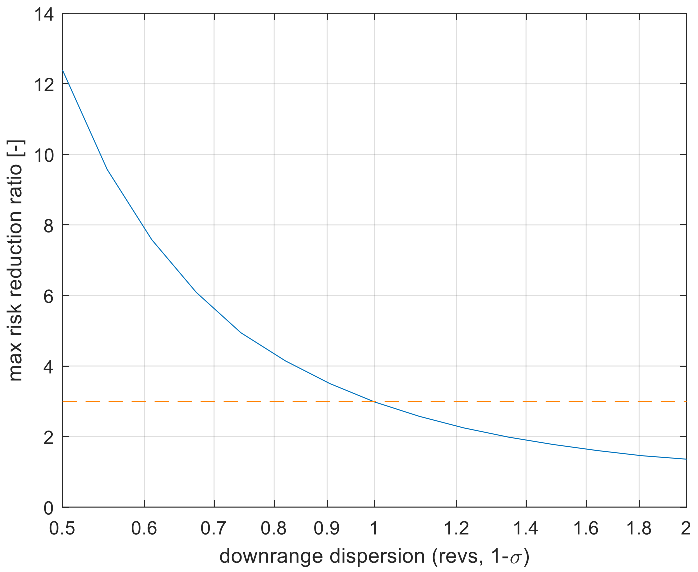

Figure 2 – Dispersion vs casualty risk reduction

If the predicted casualty risk for an uncontrolled reentry is 3×10^{-4}, then the dispersion resulting from the uncontrolled phase at the end of the assisted reentry scenario should be less than 1 revolution (1-sigma). This reduces the risk by a factor 3, down to 1×10^{-4}.

2 FEASIBILITY LIMITS AND TECHNICAL CHALLENGES

2.1 Feasibility domain

The main determinants of feasibility are related to the need to still be able to control the orbit (and thus also the attitude) down to the interface orbit, including its perigee where aerodynamic effects are several orders of magnitude higher than at mission altitude.

The *aerothermal limit* corresponds to the altitude where the aerothermal flux ($\approx 0.5 \times \rho V^3$) starts to cause noticeable heating on exposed surfaces. This typically occurs below 140 km: to avoid imposing additional constraints on the satellite's thermal design, the last controlled perigee should be above that limit. Note that the *mechanical limit*, where mechanical loads from air drag become significant, is much lower than the aerothermal limit and thus need not be considered.

The *power limit* sets a lower bound on the apogee altitude at which the orbit can still be controlled: the lower the apogee, the shorted the portion of the orbit where aerodynamic disturbances are low enough to allow sun pointing to recharge the batteries. When the batteries can no longer be recharged, the orbit can no longer be controlled.

The *attitude control limit* is determined by the need to withstand unpredicted transverse wind patterns around the perigee. Unless the AOCS is modified to weathervane[1] autonomously, it will try to fight the aerodynamic torque caused by transverse wind components. Given the momentum capacity of the reaction wheels and the longitudinal distance between the centre of mass and the centre of pressure, the cumulated drag that can be withstood over one perigee pass is determined by the magnitude of the transverse wind gusts. The resulting upper bound on the maximum cumulated drag delimits a boundary in terms of controllable apogee and perigee altitudes.

The *thrust limit* translates the fact that the descent is basically a tortoise-and-the-hare race between slow perigee lowering (with apogee manoeuvres) and accelerating apogee decay (from air drag around perigee). To each possible target altitude for the last controlled perigee, there corresponds an upper bound on the final controllable apogee, which depends on the thrust-to-mass ratio and the initial altitude. A low thrust-to-mass ratio will limit how much the perigee can be lowered before the apogee is too low for continued orbit control.

Finally, the *uncontrolled duration limit* translates the casualty risk requirement: compliance with the 10^{-4} rule imposes an upper bound on the dispersion of reentry locations. This dispersion strongly depends on how long the uncontrolled part of the descent lasts, once the orbit can no longer be controlled. Making assumptions (to be confirmed by detailed simulations) on this dependency sets a limit on the duration of the uncontrolled phase, which can in turn be converted into an upper bound for the last controlled apogee and perigee altitudes.

Those limits can be derived semi-analytically from a few macroscopic parameters of the mission (e.g. thrust-to-mass ratio, ballistic coefficient, power requirements, casualty area). They can then be plotted as forbidden regions in a chart where the horizontal and vertical axes represent the altitudes for the perigee and apogee on the last controlled orbit (see Figure 3). A non-empty central region predicts that an assisted reentry scenario might be feasible, and suggests eligible values for the altitudes of the apogee and perigee that should be targeted at the end of the controlled descent.

[1] Align with the true incoming wind

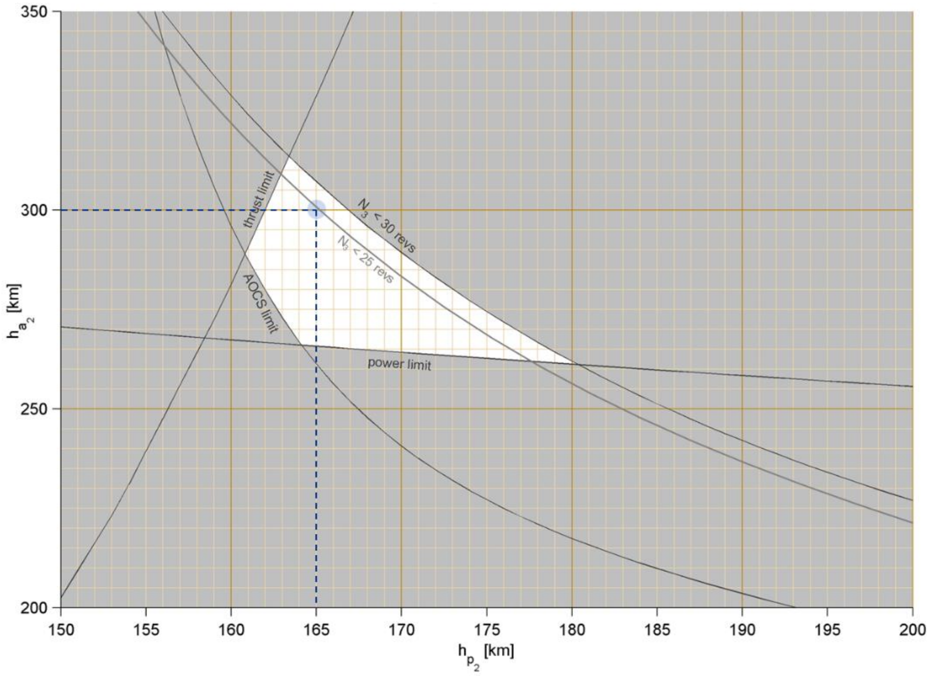

Figure 3 – Feasibility chart for the reference mission
Each point in the chart represents a possible combination of perigee and apogee altitudes for the interface orbit (i.e. the last controlled orbit before the final uncontrolled phase). Greyed-out regions correspond to orbits that cannot be reached with the satellite's characteristics.

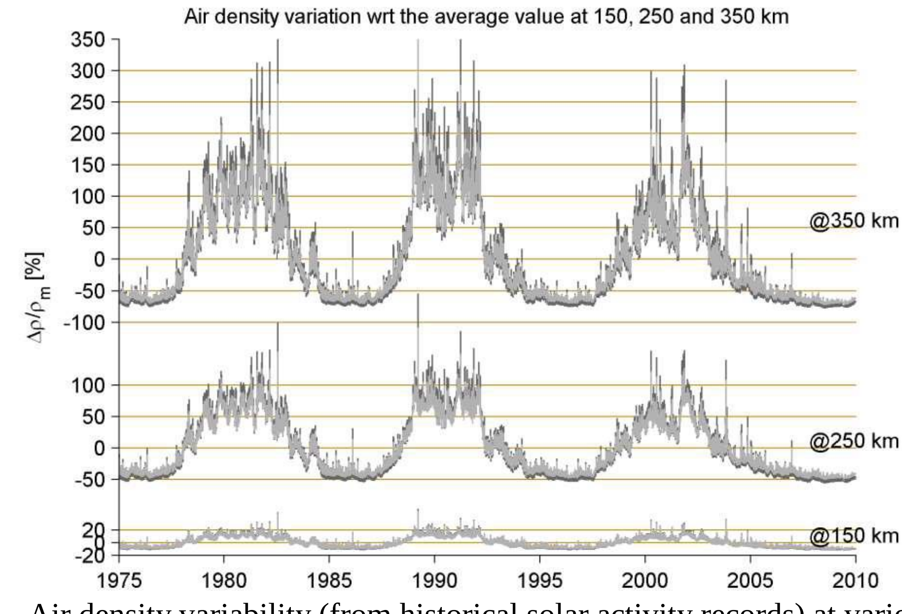

Figure 4 – Air density variability (from historical solar activity records) at various altitudes
Using the historical values for solar and geomagnetic activity proxies (F10.7 and Ap), the NRLMSISE-00 model predicts the above fluctuations for atmospheric density, at 150, 250 and 350 km respectively.

2.2 Main challenge and design driver: drag uncertainty

Assisted reentry design and operations raise a number of challenges, but the high level of uncertainty on drag is probably the biggest of those. It results from many contributors:

- Inaccuracy of the *atmospheric model*: the semi-empirical models for predicting air density in the thermosphere do not claim to be more accurate than a few 10%. Moreover, the data points for calibrating those models are very scarce below 200 km.
- Variability due to unpredictable *solar activity*: even if the atmospheric model was perfect, changes in solar activity (which are not predictable) can cause very large fluctuations. For instance, the atmospheric density around 165 km can change by up to 10% in a single day.
- Uncertainty on the *aerodynamic model*: the interaction coefficients between the various satellite surfaces and the atmosphere are not known accurately, and the detailed physical phenomena (adsorption, bounces, shocks) are hard to model and predict.
- Uncertainty on the *attitude during the uncontrolled phase*: when the satellite is left uncontrolled, the randomness in attitude introduces randomness in drag[2], causing additional dispersion in the descent profile.

Our solution addresses these challenges by controlling the descent in closed-loop: the descent itself becomes a means of calibrating drag and ensuring that its value is (nearly) deterministic at the start of the uncontrolled phase, despite all the sources of uncertainty.

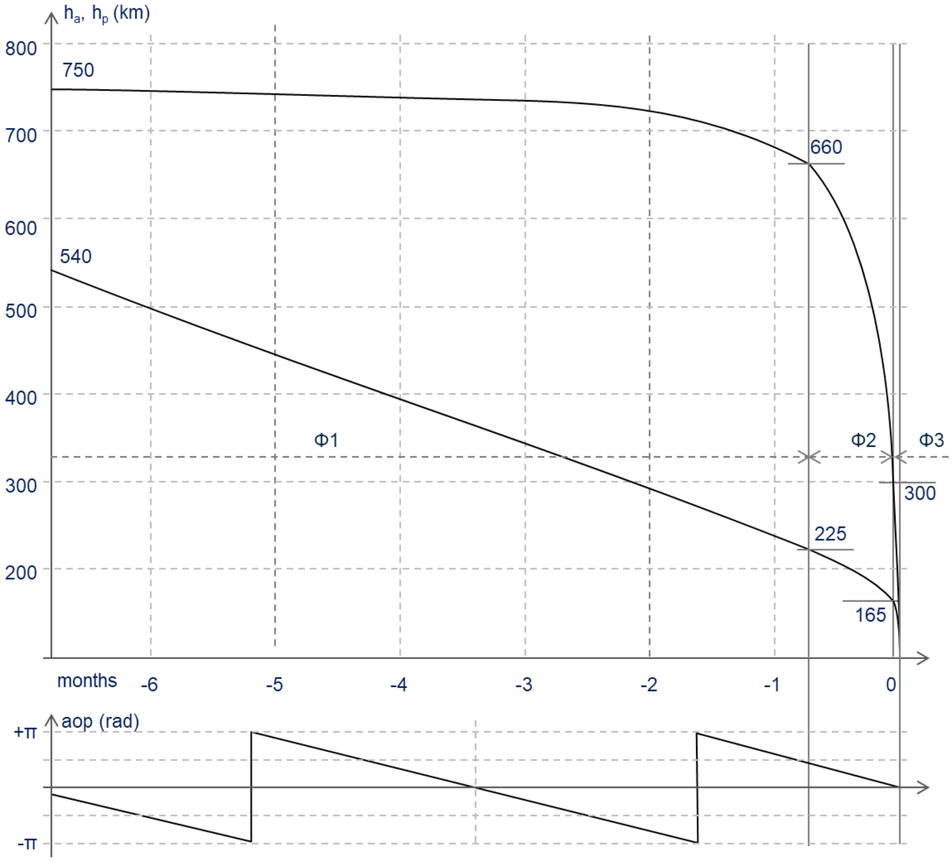

Figure 5 – Descent profile, showing the three phases of the descent

With the low thrust-to-mass ratio of the reference mission, the descent lasts a full year.

[2] Unless the geometry exhibits a very stable equilibrium attitude, which is fortunately the case for the reference mission

3 DESCENT SCENARIO AND CONTROL STRATEGY

3.1 Descent phases

The descent can be divided into three phases, delimited by specific breakpoints in terms of aerodynamic disturbances and the resulting constraints on the capability to control the orbit:

- phase 1: when the attitude is unconstrained, and thus all orbit control manoeuvres are possible
- phase 2: when aerodynamic disturbances preclude manoeuvres in the lower part of the orbit
- phase 3: when aerodynamic disturbances are too high for any orbit control capability

Phase 1: initial descent and phasing. The initial apogee and perigee are those of the nominal mission orbit. During this phase, aerodynamic effects are mild enough and there are no limitations on attitude, allowing normal operations. The objectives of this phase are:

- pre-positioning the argument of perigee (anticipating its natural precession over the predicted duration of the descent scenario) so that the last controlled perigee is located optimally
- lowering the perigee (main objective during most of the descent)
- preparing the orbital phasing, i.e. arranging for the next phase to start at a specific time

We consider that the above objectives correspond to a relatively classic problem of low-thrust orbital transfer with a time-longitude rendezvous constraint, for which orbit control issues are essentially decoupled from AOCS aspects; they are therefore not described in detail here.

Phase 2: controlled descent. This phase starts when the apogee altitude can no longer be maintained:

- either because the drag around perigee exceeds the thrust capacity
- or because maintaining a thrust attitude around the perigee would result in excessive disturbance torques and saturate the AOCS actuators

From then on, the apogee can only go down, and the descent is inexorable. However, it is still possible to control the rate of descent by adjusting the thrust duration around the apogee. The objectives of this controlled descent phase are:

- to follow a predetermined profile of apogee altitudes (and apogee decay rates) for which the predicted reentry occurs after an predetermined number of revolutions
- to control the time along this descent profile, so that the longitude of the ascending node at the time of predicted reentry will be phased correctly with the Earth's rotation.

As the aerodynamic torque around perigee can exceed the attitude control capacity, a specific guidance scheme is required during this phase: in the lower part of the orbit, the commanded attitude profile is modified to avoid saturation of AOCS actuators.

Phase 3: final uncontrolled descent. This last phase starts when the orbit control capability is lost:

- either because there is not enough time in the upper part of the orbit to recharge the batteries and achieve a sufficient ΔV.
- or because the aerodynamic disturbances near the perigee exceed the capability of the attitude control system (momentum saturation).
- or when aerothermal effects near the perigee exceed the qualification temperatures for exposed organs (such as the solar arrays)

From then on, as orbit control is lost, the final descent is passive. The orbital energy is dissipated through aerodynamic drag at each perigee pass, until circularization, final decay, breakup and reentry.

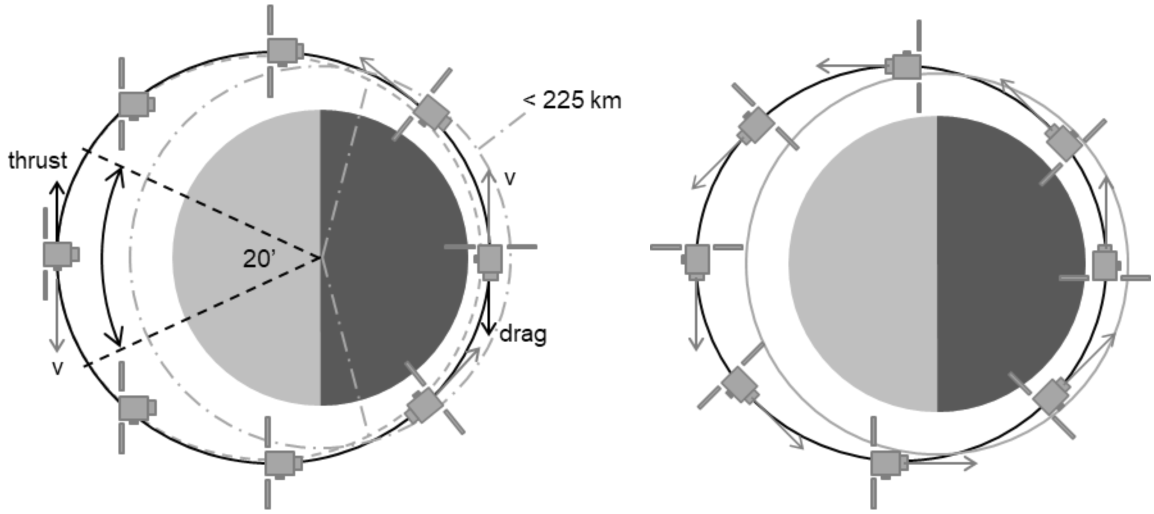

Figure 6 – Attitude profile, during phase 2 (left) and during phase 3 (right)
In phase 2, the commanded attitude alternates between sun-pointing, thrust-pointing (apogee) and wind-pointing (perigee). In phase 3, the spacecraft is captured in its stable equilibrium attitude

3.2 Descent control strategy in phase 2

A *reference descent profile* is computed beforehand, based on the best current knowledge of the aerodynamics and the atmosphere. Taking into account the attitude and power constraints along the descent, it covers all three phases and is designed such that the predicted reentry location matches the target location exactly.
In addition to the nominal duration of all the apogee burns, this guidance profile thus determines a sequence of apogee and perigee altitudes, along with the associated timing, that the system must follow as accurately as possible until the end of phase 2 (after which the satellite can no longer be controlled).

In the real world, the aerodynamic coefficients and the atmospheric density will differ from the assumptions considered when computing the reference scenario. Priority is then given to following the sequence of apogee altitudes: by adjusting the duration of apogee burns and thus the evolution of perigee altitudes, we can make sure that the drag encountered at perigee actually follows the drag predictions on the reference profile.

This *closed-loop control* strategy takes advantage of the fact that the atmospheric density depends exponentially on altitude: correcting a prediction error of e.g. 20% on air drag only requires nudging the perigee by a few kilometers up or down. This process can be formulated exactly like a control law: the current apogee altitude and apogee decay rate errors are estimated from navigation data (onboard GNSS or ground-based OD) and a controller[3] computes the adjustment in apogee burn time. The response-time of this closed loop is low enough (a few days) that it does not need to be implemented onboard: this can all be performed from the ground by uploading the appropriate attitude profiles and maneuver programs, avoiding any significant modifications of the satellite's AOCS.

A *longitude phasing control* function operates in parallel to the main control loop, adjusting the evolution of the semi-major axis so that the final phasing of the orbit corresponds to the intended optimum longitude upon reentry. Consequently, the successive orbital periods actually corresponds to the reference profile, making the total duration of the decent fully deterministic.

[3] with appropriate gain-scheduling to account for the change in vertical density gradient as the perigee altitude decreases

Periodically, *guidance updates* are performed: because the control authority is very limited, the large variations in atmospheric density due to changing solar activity could cause the control loop to saturate (large apogee altitude errors leading to command excessive or negative burn times). In these instances, rather than trying to return to the initial reference scenario, it's preferable to revise the reference profile based on an updated knowledge of the environment (current solar activity, calibrated aerodynamic model).

This controlled descent strategy essentially makes the drag profile *deterministic*: by controlling the apogee decay rate, the system guarantees that the drag is as prescribed by the reference scenario. Therefore the interface conditions at the end of the controlled phase are deterministic: regardless of the uncertainty in aerodynamics and atmospheric density, the perigee drag is precisely as intended. Since the perigee attitude and altitude are unchanged when switching to phase 3, this means that the starting conditions of the uncontrolled phase are known with much more accuracy than the underlying atmosphere and aerodynamics. This is decisive for narrowing the final dispersion despite all the model uncertainties.

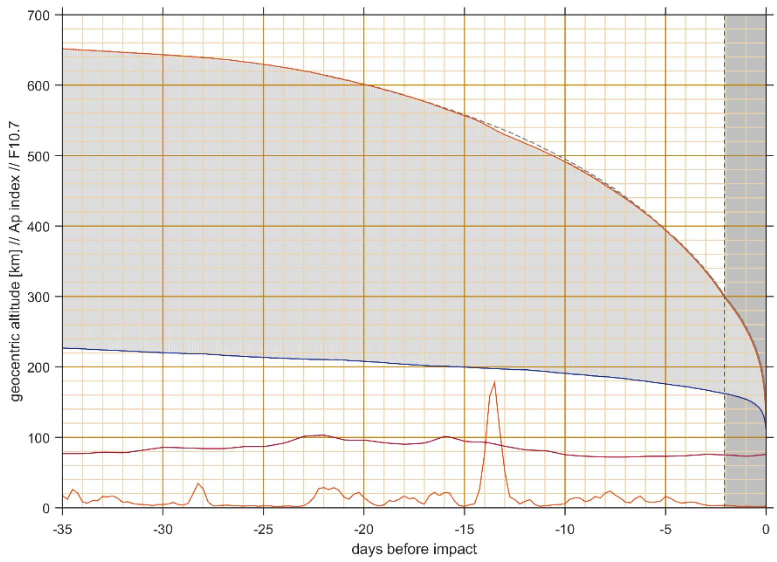

Figure 7 – Illustrating the robustness of the controlled descent strategy
The top part of the chart represents the evolution of the apogee and perigee altitudes, while the bottom 2 curves represent solar activity (F10.7 and Ap, units not represented). Two weeks before reentry, a solar flare causes a spike in drag and a drop in apogee altitude (the dotted line represents the guidance profile). By reducing the pace at which the perigee is lowered, the control law manages to compensate the excess drag and bring the apogee altitude back on track.

3.3 Summary of operational concept

The operational concept of the assisted re-entry has been implemented within the operations tools, so it can be applied on upcoming Airbus satellites. Most of the functions required to perform such a re-entry are fully automated and only the following critical functions are performed manually:

- At the beginning: the evaluation of the initial strategy to phase the argument of perigee
- At the end of phase 1: modification of the reference profile to target the final longitude and the final downrange
- During phase 2: guidance updates to adjust to revised solar activity forecast

The remaining flight dynamics functions required to perform the ANR are fully automated in the operational concept:

- Orbit estimation
- Drag and thrust efficiency calibration
- Control of the real trajectory (duration and apogee decrease) wrt. the reference profile
- Trajectory prediction
- Collision assessment and collision avoidance

With this operational concept, the reentry is mainly automated, and manual operations are limited to a few interventions. Additional checks and automatic monitoring of the real trajectory are also implemented to ensure a good health of the system during re-entry.
The control laws explained in §3.2 have also been designed to handle degraded configurations that could cause the satellite to drift from the reference profile, for instance:

- Large drag prediction errors
- Collision avoidance manoeuvre
- Safe mode triggered on-board
- Loss of TC links (up to a few days)

If such a degraded situation occurred, the control law would automatically readjust the manoeuvre profile to cancel the difference between the real trajectory and the reference profile.

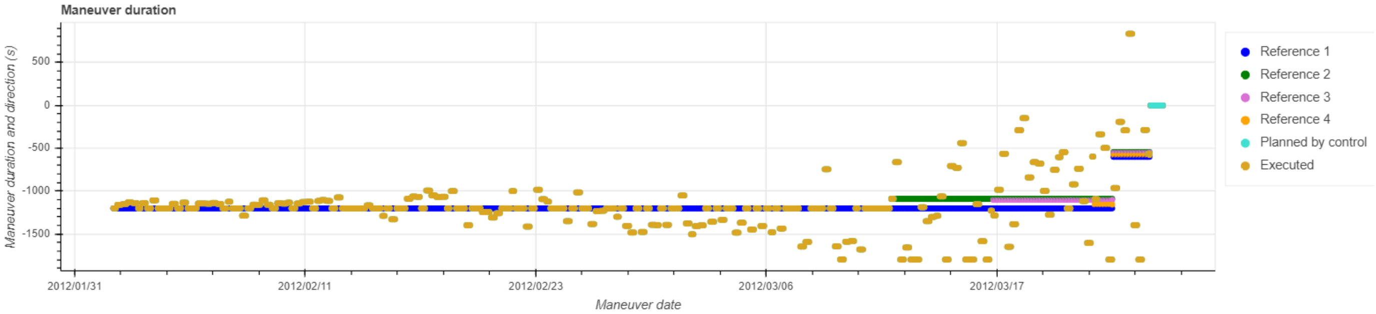

Figure 8 – Operational implementation of the control strategy

The first 4 profiles correspond to the manoeuvre durations commanded by the successive updates of the guidance profile. The more noisy values correspond to the activity of the closed-loop control law, continuously adjusting to compensate fluctuations in air drag due to solar activity

4 PERFORMANCE VALIDATION ON A REPRESENTATIVE USE-CASE

The whole strategy has been extensively validated, for implementation on an Airbus LEO mission with electrical propulsion only. This section describes the setup and the results of the validation campaign.

4.1 Characteristics of the reference mission

Initial orbit. The orbit is a noon-midnight SSO orbit with an inclination of 98° and an ascending node at 22:30. The simulation campaign concentrates on the altitudes where aerodynamic effects are significant, starting with an apogee at 550 km and a perigee at 250 km. We assume that the argument of perigee is already positioned such that the natural precession due to J2 effects will cause it to be located above the night-side equator (arg. of p. ~0) at the end of the descent.

Low thrust-to-mass ratio. The electrical propulsion system offers a very low thrust-to-mass ratio of 0.03 mm/s². This is equivalent to a 15 milliNewton thruster on a 500-kilogram satellite. The resulting Delta-V of 4 cm/s for each 20-minute apogee burn allows to lower the perigee by ~2 km per day. The whole descent scenario thus takes nearly a full year. On the other hand, with such a low thrust capacity, the power constraints are very relaxed, with almost always enough time to recharge the batteries between burns, even when the burns occur during eclipse.

The ballistic coefficient takes two very different values depending on whether the satellite is in its nominal flight attitude with its solar arrays edge-on (B = 80 kg/m²), or in its equilibrium attitude with its solar arrays face-on (B = 30 kg/m²). This equilibrium attitude is passively stable, guarding against tumbling during the uncontrolled phase – this makes the uncontrolled phase more predictable.

4.2 Simulation setup and assumptions

As the requirement for assisted reentry is statistical in nature (global probability), the most relevant method for assessing the system's performance is through Monte-Carlo simulation campaigns.

The simulator comprises three main modules:

- the *space segment*, corresponding to the simulation of the 'real world', with the satellite's propulsion system, the orbital dynamics, the atmosphere and the aerodynamic model.
- the *ground segment*, simulating the operations. It computes the reference guidance profile, predicts the reentry location, implements the descent control loop, updates the guidance profiles and uploads the manoeuvre plans.
- The *postprocess* module computing various performance indicators. In particular it computes the casualty risk for each reentry simulation, and its global average over the whole campaign.

The key models in the *space segment* are listed below:

- The *propulsion model* simulates a retrograde thrust (antiparallel to the velocity vector) over an extended portion of the orbit around the apogee, with a duration dictated by the manoeuvre profile commanded by the ground segment.
- The *attitude control system* is considered perfect: the attitude follows the reference profile commanded by the ground segment. This avoids simulating a 10-Hz control loop within a 6-month-long simulation.
- The *aerodynamics model* predicts the drag force and torque for all flight attitudes. The drag force feeds into the propagation of the orbit, while the accumulation of drag torque is compared to the saturation limit of the AOCS actuators.
- The *orbital dynamics* is simulated with a model of the Earth's gravity, including at least the effect of J2 (for sun-synchonicity drift and precession of apsides)

- The *atmospheric model* uses the JB2008 atmosphere model.
- *Solar activity*. In order to capture the fluctuations of the atmosphere in a statistically representative way, we use historical records of solar and geomagnetic activity: for each descent simulation, we pick a random date in the past and use the historical solar activity data from that date onwards as inputs to the atmospheric model.

The key functions in the ground segment are:
- The *propagation simulator* which is almost a clone of the space segment, for computing and iterating on guidance profiles. But in order to represent the fact that the real world is not known precisely, a number of discrepancies are intentionally introduced:
    - The aerodynamic model is perturbed with a multiplicative bias (up to 10%)
    - The atmospheric model is different (NRLMSISE-00 instead of JB2008). This introduces very large differences in predicted density, dominating all other errors
    - The solar activity data are not the actual observations but the historically recorded *predictions* (made at the time the guidance profile is supposed to be computed).
- The *control function* (called at each visibility), which updates the duration of the next apogee manoeuvres based on the observed apogee altitude and apogee decay rate errors (and on the current longitude error).
- The *guidance function* (called infrequently), which iterates until a suitable reference profile is found, starting from the current estimated orbital state and ending at the target reentry location and at the right date and time.
- The *navigation function*, which simulates typical orbit determination errors.
- The *ground station* model, which simulates the average delay (6 hours) between downlinks and uploads.

The main *postprocess* is about the computation of the casualty risk. For each individual simulation, the probability density of debris is represented as a Gaussian fragmentation footprint with an elliptical shape located and oriented according to the simulated reentry location, and for which the surface integral corresponds to the casualty area[4] of the spacecraft. The casualty risk for that simulation is the surface integral of the product of this probability density of debris (non-dimensional) with the population density[5] (units = m^{-2}). The result is the probability of causing a casualty *for this specific reentry*. Considering that all possible reentries have equal probability, the casualty risk for the whole campaign is then the average of the risk values of the individual simulations.

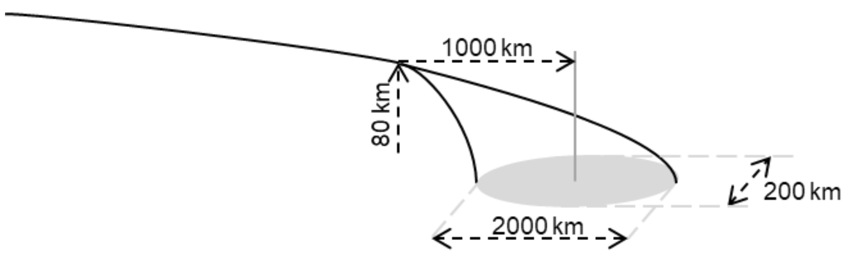

Figure 9 – Representation of the fragmentation footprint

The centre of the distribution of debris (for each individual reentry) is considered to be 1000 km downrange of the breakup point. The Gaussian distribution is such that σ_x = 1000 km, σ_y = 100 km, and its integral is equal to the casualty area predicted by demise analyses.

[4] The *casualty area* is a well-established mathematical construct capturing the number and size of the hazardous fragments surviving reentry, combined with the typical cross-section of a person.

[5] GPWv4 map, with its 2020 projection, extrapolated to 2033 (+15% scaling)

4.3 Results of the performance campaign

The simulation campaign with the most pessimistic assumptions (including dissimilar atmospheric models between space-segment and ground-segment) was intended as a demonstration of robustness of the whole strategy. It comprised 1500 detailed simulations.

For achieving a 3-fold reduction in casualty risk (to decrease the risk below10^{-4}), the required dispersion is 1 revolution 1-sigma. However, in order to cover a nonzero probability of an uncontrolled reentry (e.g. from a hardware failure interrupting the operations), the target value for a successful reentry was lowered to 0.75×10^{-4}, tightening the target dispersion to 0.9 revolutions.

Compared to these targets, the casualty risk was estimated at 0.3×10^{-4}. This corresponds to a dispersion of 0.5 revolutions 1-sigma, with hardly any reentries occurring further than +/- 2 orbits uprange or downrange of the nominal location (in the South Atlantic Ocean).

The accuracy of longitude control also contributing to this low risk. The final across-track dispersion (longitude drift due to the accumulation of errors in the orbital period) is less than 3 deg.

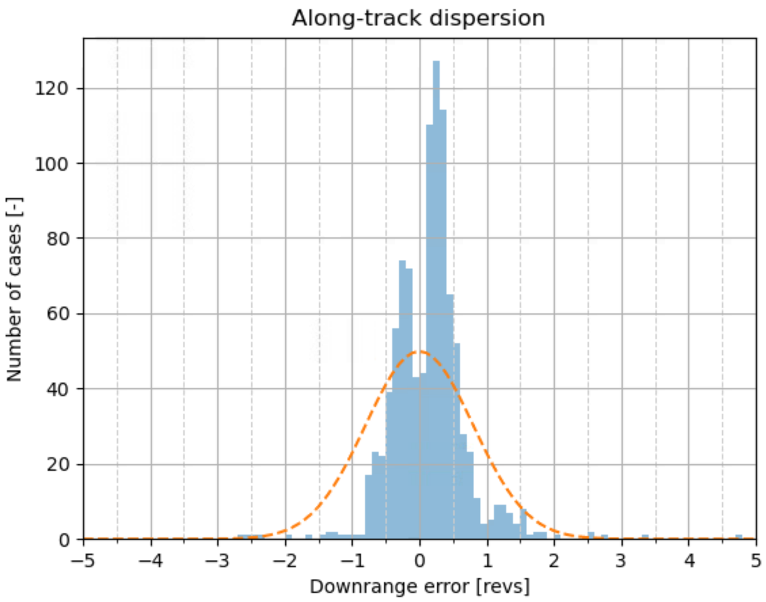

Figure 10 – Results of the simulation campaign

Thanks to the control strategy (and the deterministic stable attitude during the uncontrolled phase), the distribution of reentry locations is even more compact than the requirement (dotted bell curve). This allows to reduce the casualty risk by almost an order of magnitude compared to uncontrolled reentry, despite the very low thrust.

5 CONCLUSIONS AND PERSPECTIVES

Although challenging on many aspects, assisted reentry with a low-thrust system has now been proved feasible, with performances exceeding the requirements. This makes it a noteworthy candidate solution for small-to-medium all-electric LEO spacecraft which would otherwise be marginally non-compliant with the 10^{-4} limit. The solution developed and validated by Airbus is ground-based, requiring no hardware changes and minimal adaptations in the onboard software (mostly FDIR thresholds for the low-altitude flight phases).

Thanks to uninterrupted R&D effort at Airbus since early 2015, with significant support from CNES (see [4] & [5]) and ESA (see [6] & [7]), assisted reentry as a whole can be considered a mature solution. The design and validation process is now fully stabilized, and the tools for supporting operations are being finalized. The low-thrust version has already been adopted by Airbus on an all-electric mission, while the chemical-propulsion version will soon[6] be implemented by ESA for Aeolus. Assisted reentry solutions can thus fill the gap between controlled and uncontrolled reentry, offering more design flexibility to comply with safety regulations.

At least as an interim solution until improvements in demisability make all but the largest satellites naturally compliant, assisted reentry can prove very useful in the following contexts:

- For medium-sized LEO satellites, when the estimated casualty risk for an uncontrolled reentry is marginally above the regulatory limit. This is especially useful for all-electric platforms, as it avoids having to add a complete chemical propulsion system just for reentry.

- For extending mission lifetime, for example of a valuable scientific satellite in LEO without a ready replacement: for a mission initially designed to perform a controlled reentry, switching to an assisted reentry can save upwards of 40 m/s.

- For retrofitting currently flying satellites which were not meant to be deorbited, on a best-effort basis. This is typically the case of Aeolus

6 REFERENCES

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