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Impact of controlled and semi-controlled re-entry strategies on spacecraft design

Executive Summary (ES)

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Disposing of satellites safely for the people on Earth

There is an increased attention to safeguarding the Earth orbital environment, as reflected by the number of relevant regulations that are being set forward by national governments and international organisations. Among others, ESA has published a **Space Debris Mitigation Policy for Agency Projects** (ESA/ADMIN/IPOL(2014)2), supported by the corresponding **ESA Space Debris Mitigation Compliance Verification Guidelines** (ESSB-HB-U-002). The French government has set the Space Operations Act (**Loi relative aux Opérations Spatiales**), which is fully applicable to European launchers and satellites manufactured and/or operated from French territory.

Low-Earth Orbit (LEO) has been designated as a protected region, being a valuable region to operate satellites and consequently the most congested Earth orbit region. Therefore, the regulations dictate that satellites reaching their end of life must leave these protected regions, typically by re-entering the Earth atmosphere within a maximum of 25 years. In doing so, satellites will typically burn up in the atmosphere, apart from some components which may survive the heat re-entry. Safety is a major concern, thus, the various regulations also stipulate that satellites must clear the LEO region in a way that is also safe to people on the ground.

The casualty risk of a satellite drives the type of re-entry

There are three ways for satellites to leave the LEO protected region.

An **uncontrolled re-entry** is particular well suited for smaller satellites, where less than a dozen of parts may survive the re-entry. It is the most economical in terms of propellant as the perigee is reduced to an altitude of about 500 km, and the orbit decay within 25 years. The satellites debris may fall anywhere on the Earth, within a latitude range bound the inclination of the orbit, and the probability of a human casualty is extremely low, below the permissible risk set by the regulations.

Controlled re-entry is suited to larger satellites with several tens of surviving debris. The satellite is made to re-enter the Earth atmosphere so that its debris fall within a pre-defined uninhabited area at a precise time. The South Pacific Ocean Uninhabited Area (SPOUA) has been used for the disposal of the MIR station, the cargo vehicles re-supplying the International Space Station (ATV, Cygnus, Progress) and it is also the designated re-entry target for the design of the MetOp-SG satellites.

Assisted Natural Re-entry (previously known as semi-controlled re-entry) fills the gap between controlled re-entry and uncontrolled re-entry. As it is an intermediate solution, it is well adapted to medium-sized spacecraft which would marginally exceed the regulatory requirement for overall casualty risk of an uncontrolled re-entry. Assisted natural re-entry requires a last controlled perigee typically around 150km. The final decay lasts less than a day, so that the associated uncertainty can be less than one orbit. Contrary to controlled re-entry, perigee lowering can be done gradually, and is compatible with low-thrust propulsion.

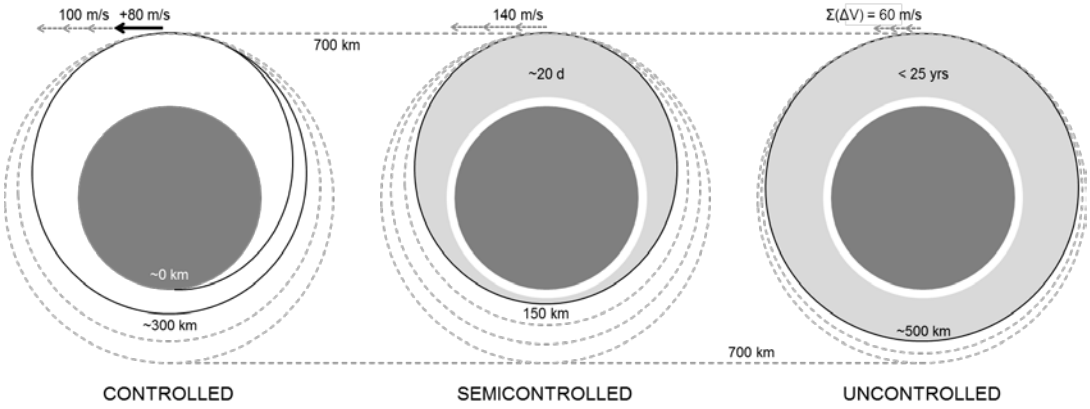


Figure 1. comparison between controlled, semi-controlled and uncontrolled re-entry

Assisted Natural Re-entry: managing the risk by controlling the dispersion

Depending on the amount of debris that might reach the ground, and on the expected longitudinal dispersion of impacts around the nominal point, a risk value can be computed for every location of the nominal impact point, resulting in a risk map for a given orbital plane. The map highlights how much the risk can be decreased with respect to uncontrolled re-entry when selecting an appropriate target impact point.

Some uninhabited areas such as the South Pacific Ocean are not particularly suitable because there are populated landmasses upstream and downstream. In comparison, the Atlantic Ocean is more suitable, as there are large tracts of empty Pacific Ocean upstream and downstream.

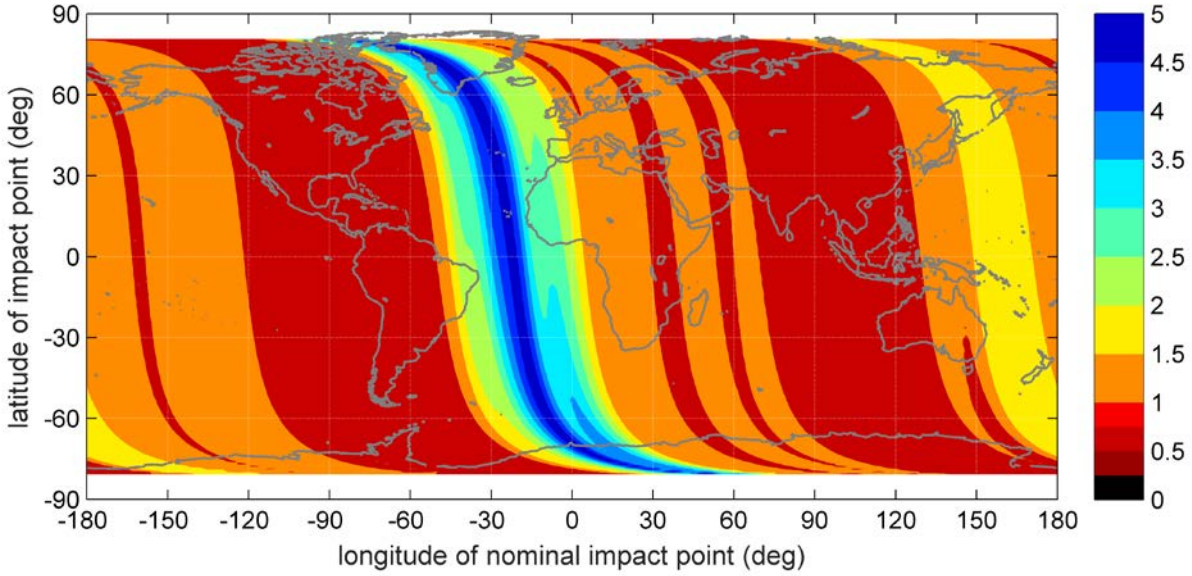


Figure 2: example risk map, with risk reduction ratio (for σ_N = 0.75 and i = 98 deg)

Intuitively, a high risk-reduction target requires a low dispersion, so that the ground track where the debris might be dispersed does not extend over populated landmasses.

This can be confirmed by computing the risk map for all possible values of the longitudinal dispersion parameter σ_N and determining for each map the maximum risk reduction ratio that can be achieved. This results in a risk-vs-dispersion chart (Figure 3), which serves as a basis for defining the descent strategy.

Based on this chart, we can determine the maximum acceptable dispersion for the distribution of the final impact (for instance, to achieve a risk reduction of 5, the longitudinal dispersion on the distribution of the final impact point must be limited to 0.7 orbits 1-sigma).This in turn sets limits on the duration of the uncontrolled phase, and on the accuracy of the initial conditions of this uncontrolled phase.

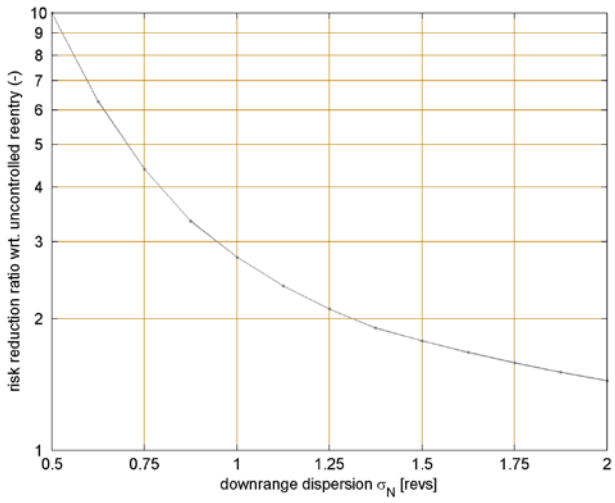


Figure 3. Maximum risk reduction ratio vs. downrange dispersion σ_N (for i = 98deg)

Limiting the dispersion of the assisted natural re-entry

To minimize dispersion on the prediction of the longitudinal location of the impact point, it is necessary to:

- control the descent for as long as possible to make the last uncontrolled phase as deterministic as possible (less initial uncertainty, less time for uncertainties to cause divergence)
- reach a last controlled perigee with the lowest possible altitude to make the uncontrolled phase as short as possible (faster orbital decay)
- maintain attitude control as long as possible despite during high disturbances low perigee, in order to minimize dispersion on drag

Although spiralling strategies might be possible, they require a specific aerodynamic configuration for the spacecraft in order to avoid the premature loss of attitude control. As satellites are not generally designed for specific aerodynamic characteristics, it is necessary to have an elliptical orbit for at least part of the descent scenario:

- On the lower portion of the orbit, the aerodynamic torque would be too high to maintain thrust attitude: instead, the satellite has to fly in a **specific aerodynamic attitude near the perigee**, to avoid saturating the reaction wheels.
- On the upper portion of the orbit, the atmospheric density is low enough to resume normal operations, alternating between sun pointing to recharge the batteries, and a thrust sequence around the apogee to continue lowering the perigee.

The objective is to keep controlling the descent down to **a perigee of typically 150 km**.

The sharp spikes in dynamic pressure near the perigee tend to precipitate orbital decay in the vicinity of the perigee. Essentially, if a given perigee pass does not cause immediate orbital decay, it is less likely that the residual drag on the rest of the orbit will be sufficient, and only the dynamic pressure spike when nearing the next perigee will terminate the descent.

By **targeting a perigee near the equator** for the final uncontrolled descent, this effect can be compounded with the oblateness of the Earth's atmosphere, further increasing atmospheric density around perigee and maximizing the concentration of the impact point (i.e. minimize downrange dispersion).

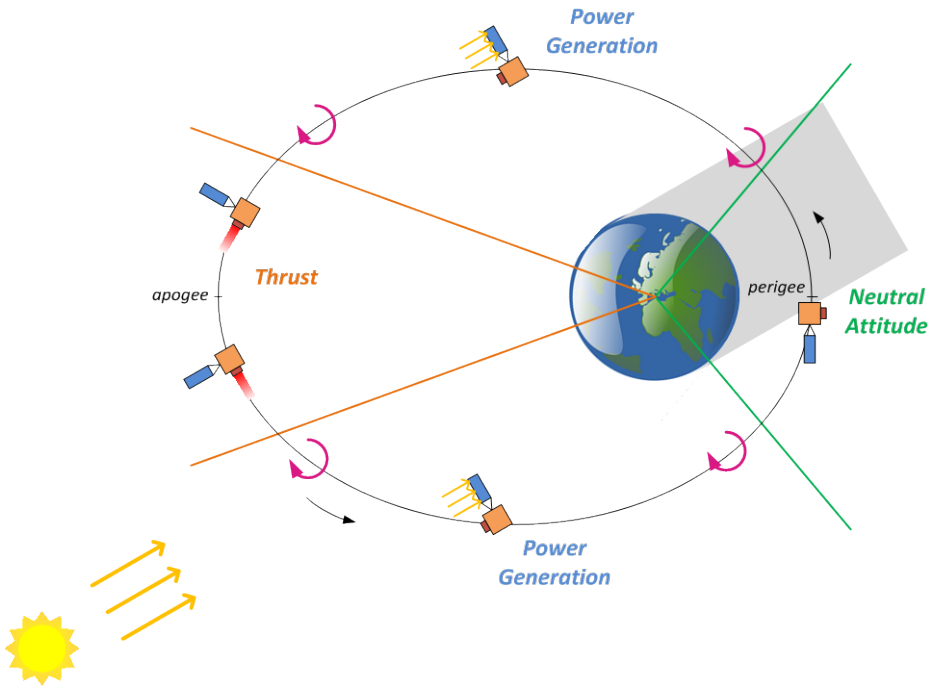


Figure 4. Illustration of the operations during the assisted natural re-entry descent.

The three phases of the Assisted Natural Re-entry

The descent can be divided into three phases (Figure 5), 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 the aerodynamic disturbances preclude manoeuvres in the lower portion of the orbit
- phase 3: when aerodynamic disturbances are too high for any orbit control capability

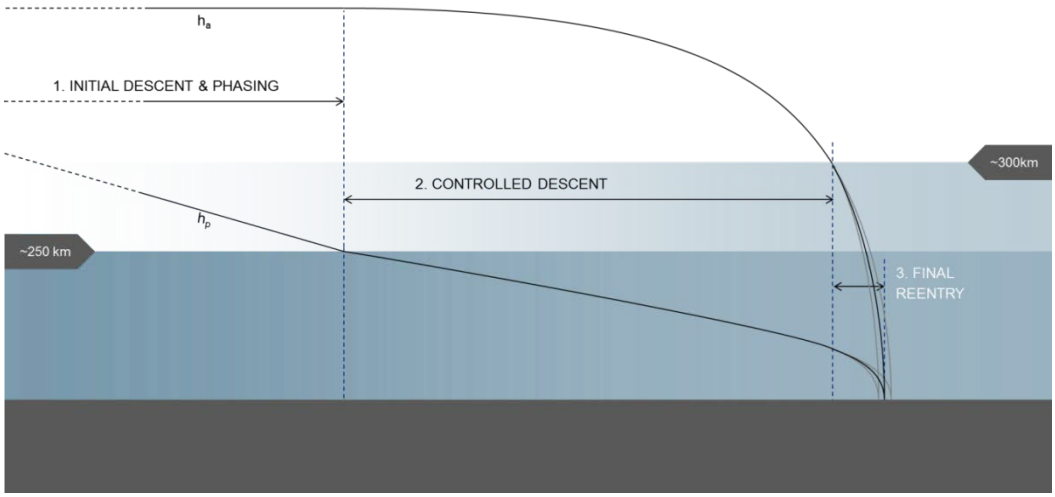


Figure 5: Schematic representation of the 3 phases in the descent

Phase 1 – initial descent and phasing

The objectives of the first phase of the descent are:

- pre-positioning the argument of perigee (by anticipating its natural precession over the predicted duration of the whole descent scenario) so that the last controlled perigee is located optimally;
- lowering the perigee (main objective);
- maintaining the apogee to avoid premature circularization while lowering the perigee;
- preparing the orbital phasing, i.e. ensuring the next phase starts at a deterministic moment in time.

Phase 2 – controlled descent

The second phase starts when it becomes impossible to maintain the apogee altitude:

- 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 reach the desired last controlled apogee altitude;
- to reach the desired last controlled perigee altitude;
- at the right time, to ensure the desired phasing 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 optimised so as to avoid saturation of AOCS actuators. As the perigee and the apogee altitudes decrease, this represents an increasing proportion of the orbit.

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 parts (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 circularisation, final decay and breakup.

The critical aspect of the assisted natural re-entry is to predict the duration of the uncontrolled phase (expressed in number of orbital revolutions, N_3) and its uncertainty expressed as standard deviation, σ_{N3}.

To avoid losing orbit control prematurely:

- the attitude control system should be robust to aerodynamic disturbances, down to the target perigee altitude
- the propulsion system should be able to lower the perigee fast enough to avoid premature circularization

Feasibility Assessment of Assisted Natural Re-entry

Airbus has developed **a set of semi-analytical expressions that are used to generate feasibility chart**, illustrated by Figure 6, to be used in early design phase.

It is important to capture the major contributors to dispersion, so that a preliminary feasibility status can be established, with possible system-level recommendations to adapt the overall mission design. This is why we choose to concentrate the effort on a reduced set of contributors.

Major contributors

- uncertainty on solar activity dominates all terms of error
- uncertainty on initial conditions of phase 3 dominates the dispersion at the end of the uncontrolled phase

variability of ballistic coefficient in phase 3 (from random attitude motion) is a major term of phase 3 dispersion (in addition to initial conditions)

The description of the methodology is discussed in the Final Report of the study.

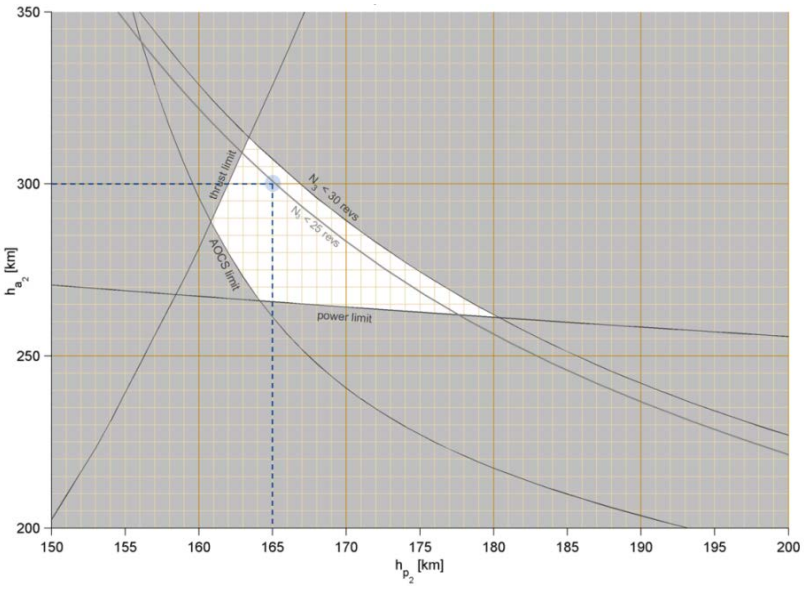


Figure 6: Combined system limits and feasibility domain (on an example mission)

For any given mission, a statistical simulation campaign (several thousands of simulations) is performed, spanning Phases 2 and 3. Each of those simulations starts at a random date within the 33 years (latest 3 solar cycles). For the duration of the simulation, the solar activity considered for the atmosphere model corresponds to the historical records (with daily variations of F10.7 and 3-hourly variations of Ap). That way, each simulation represents what would have happened if the re-entry had started at that date.

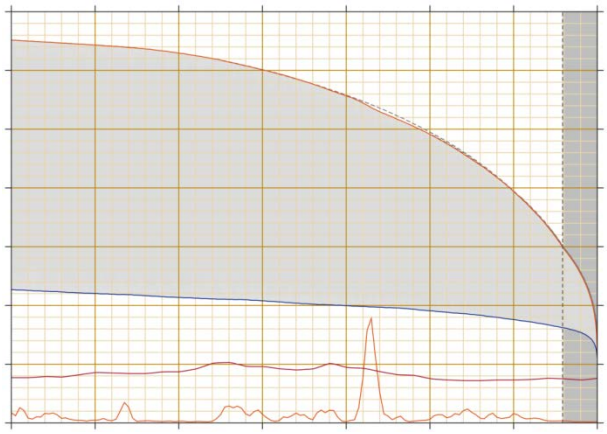


Figure 7. Illustration of a single simulation case

Altitude and solar activity (F10.7 and Ap) vs. time. The large spike in Ap 13 days before impact results in an unexpected increase in perigee drag, leading to a dent in the apogee altitude (compared to reference descent profile). The descent control loop then updates the apogee manoeuvres to adjust the perigee altitude, so that the drag level and the apogee profile both return to their nominal values.

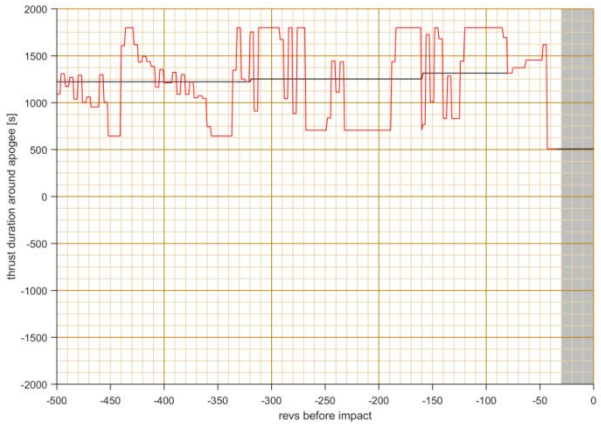


Figure 8. Illustration of a single simulation case: apogee thrust manoeuvre duration

The descent control loop adjusts the duration of apogee manoeuvres to compensate for variations in the atmospheric density, so that the nominal descent profile is followed correctly. This ensures that the actual impact point will remain close to the nominal impact point.

For each simulation, the geographic location of the impact point is recorded. The casualty area is then distributed over the fragmentation footprint around/downstream of the impact point, and the corresponding risk is computed by combining this distribution with the population density.

The statistical risk for the campaign is the average of the risk values over the whole set of simulations.

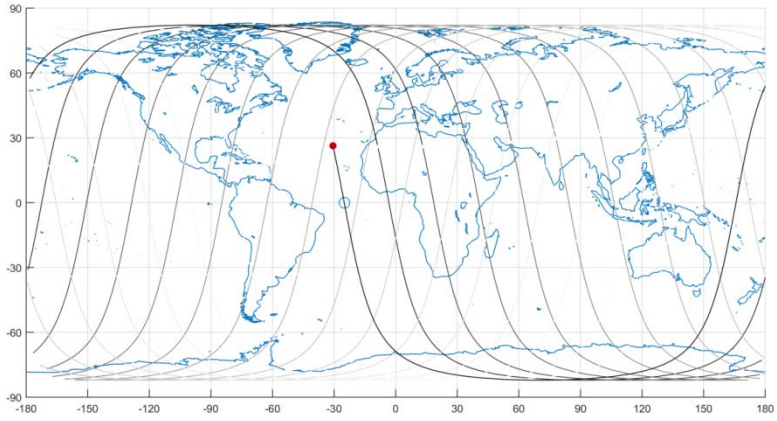


Figure 9. Illustration of a single simulation case: last few ground tracks, and impact point

This demonstrate that despite a large spike in geomagnetic activity a few days before the end of the descent, the system was capable of maintaining optimal phasing with the earth's rotation and control the rate of descent, even with the very low control authority (electric propulsion).

Case Study of the Assisted Natural Re-entry

As part of the study, three mission cases have been studied. The missions (satellite configuration, propulsion system and initial conditions) were selected in concertation with ESA.

For each of the three reference cases, the simulation campaign demonstrates that for each of the reference cases the casualty risk is well below the 10^{-4} regulatory value. This is explained by the fact that:

- the design point (h_a,h_p at the end of the controlled phase) for each reference case was not chosen *at* the dispersion limit (upper-right boundary) but well within the feasibility domain, therefore resulting in reduced dispersion.
- the risk map used for assessing maximum allowed dispersion uses a conservative Gaussian distribution for along-track dispersion of impacts. In reality, impacts tend to concentrate nearer the equator, i.e. in the middle of the ocean.
- the real dispersion from control errors was less than the expected dispersion (from which the design point was chosen).

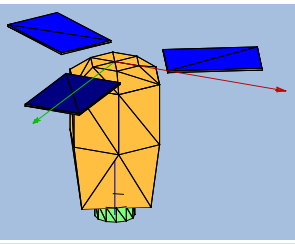
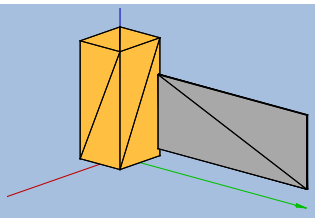
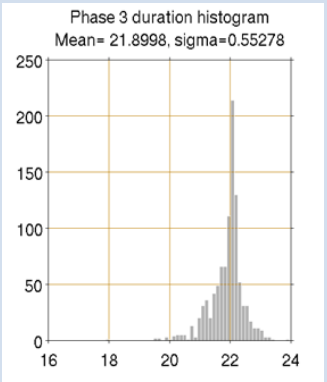
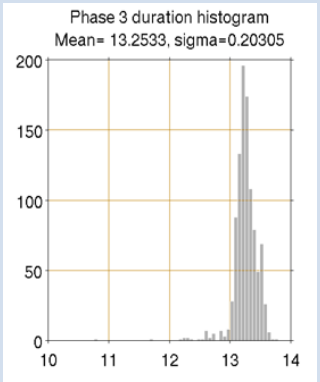
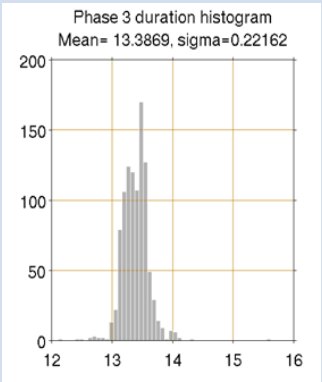
	Case 1	Case 2	Case 3
Geometry			
Mass	750 kg	1500 kg	
Propulsion	HET	Arcjet	HET
Thrust	80 mN	240 mN	120 mN
Ballistic Coefficients **Thrust-pointing** **Equilibrium attitude** **Random tumbling**	 68 kg/m^2 45 kg/m^2 52 kg/m^2	 170 kg/m^2 110 kg/m^2 60 kg/m^2	
Casualty risk area	20 m^2	25 m^2	
Reaction wheels momentum authority	20 N.m.s	40 N.m.s	
Results			
N3 standard deviation spec	1.1	0.9	0.9
Phase 3 duration histograms (showing average number of orbits, N3, and its standard deviation)			
Casualty risk	**5.0 x 10^{-5}**	**1.0 x 10^{-5}**	**1.6 x 10^{-5}**

Table 1. Summary of the Assisted Natural re-entry study cases.

Assisted Natural Re-entry is ready for adoption by future missions

The semi-controlled re-entry strategy enables the utilisation of high-specific-impulse, low-thrust propulsion system, such as arcjets and Hall-Effect Thrusters (HETs). However, these have high power demands (1-2 kW), which require dedicated power processing units and an adapted satellite power subsystem design, where a dual-power-bus PCDU may be best suited. This is likely to come with a hardware mass penalty, as LEO satellites seldom need such PCDUs. This mass penalty (possibly including solar array increase too) is to be more than compensated by the higher specific impulse and improved propellant mass usage.

The semi-controlled re-entry strategy devised by Airbus Defence and Space has been designed to fit existing, “classical” AOCS architectures. In essence, no additional hardware is required (although slightly larger reaction wheels may be needed, subject to the satellite configuration and ballistic characteristics).

The control loop to follow semi-controlled re-entry strategy is largely autonomous, and the role of the ground segment is almost limited to monitoring the evolution of the descent, as well as commanding the transition between the three phases. During the descent, a Fail Operational must be followed, since an excessive interruption of the nominal descent would result in an unrecoverable descent strategy. At worse, this would lead to an uncontrolled re-entry, which represents the worst case possible in terms of casualty risk.

There has been continuous investment at Airbus since 2015 on all topics that need to come together for offering a mature semi-controlled re-entry function for the projects that might need it:

- Flight dynamics strategy
- Attitude control at very low altitudes
- Sizing approach
- Simulation tools
- Operational concept
- End-to-end statistical performance campaigns

This investment also benefited from significant support from space agencies:

- CNES R&D 2015-16: Contrôle d’attitude des satellites à très faibles altitudes
- ESA R&D 2016-18: Impact of Controlled and Semi-Controlled Re-entry on Spacecraft Design (the current study)

Thus, the present assessment, based on robust analyses, clearly shows that **Assisted Natural Re-entry is technically realistic with current technologies**. However, it is important to stress that assisted natural re-entry strategies with low-thrust systems are not a universal solution for all missions. In particular, it only applies to medium-sized platforms, considering that with electric propulsion, the casualty risk can typically be reduced by a factor 4-5 at most. In order to achieve higher risk reduction ratios (for satellites with a higher casualty area than ~30 m²), chemical propulsion would be needed, and the mass savings would be much less significant when compared to controlled re-entry.

Moreover, there are costs associated with the strategy. These costs need to be taken into account in the trade-off activities, before deciding whether semi-controlled re-entry is the preferred solution.

The first element of additional costs is the need for verifying/qualifying an extended flight domain, with much lower perigee altitudes and much higher aerodynamic drag:

- Mission- and platform-dependent analyses to establish feasibility
- Additional analyses (AOCS, FDS) to verify system robustness
- FDIR adaptations to take into account new flight domain (e.g. higher disturbance torque)
- materials qualification / adaptations (environmental conditions at low altitude)

The second element of additional cost is the need for longer end-of-life operations:

- Controlling and monitoring the descent
- Ground workload + ground station usage

This second cost contribution can however be largely reduced by increasing the level of automation (on-ground) or even autonomy (on-board) for semi-controlled re-entry functions would significantly reduce operations costs. This is an important next step in future R&D on the topic.

Controlled Re-Entry still provides the safest disposal

While satellite recovery remains a distant option, controlled re-entry is the only option to safely bring a satellite and its debris down in an uninhabited area. This is particularly relevant for satellites with large casualty risk areas and/or where Assisted Natural re-entry is not an option.

The key parameters of the controlled re-entry are:

- The length (ellipse major axis) of the debris cloud, L_{dc};
- The error / variation of the length of the debris cloud, δL_{dc};
- The error / variation of the position of the debris cloud and thus its centroid, $\delta\lambda_{dc}$;
- The altitude of the targeted perigee, h_{tp}.

The length of the debris cloud is driven by the two debris with the lowest and highest ballistic coefficient, resulting in a length of typically 1000 km or so.

The altitude of the targeted perigee, h_{tp}, and its argument of perigee are key design parameters to ensure that the debris fall down within the Safety Re-entry Area (SRA).

For near-polar orbits and orbits with sufficiently large inclinations (above 45°), the SPOUA is the safest area to target.

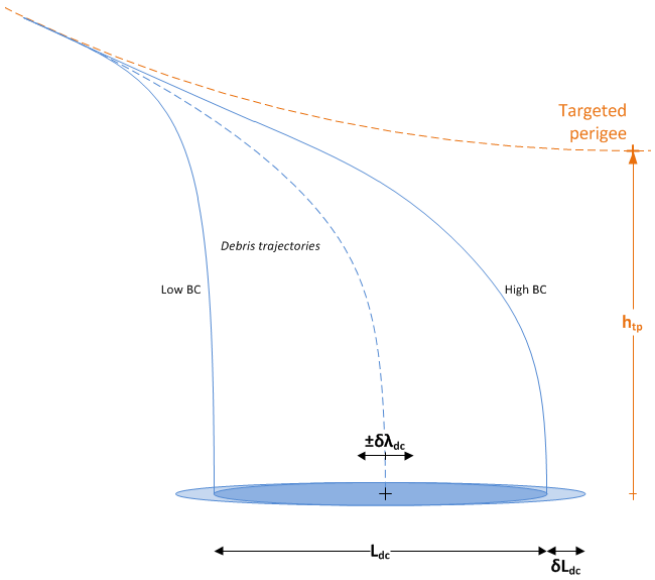


Figure 10. Illustration of the controlled re-entry key debris cloud parameters

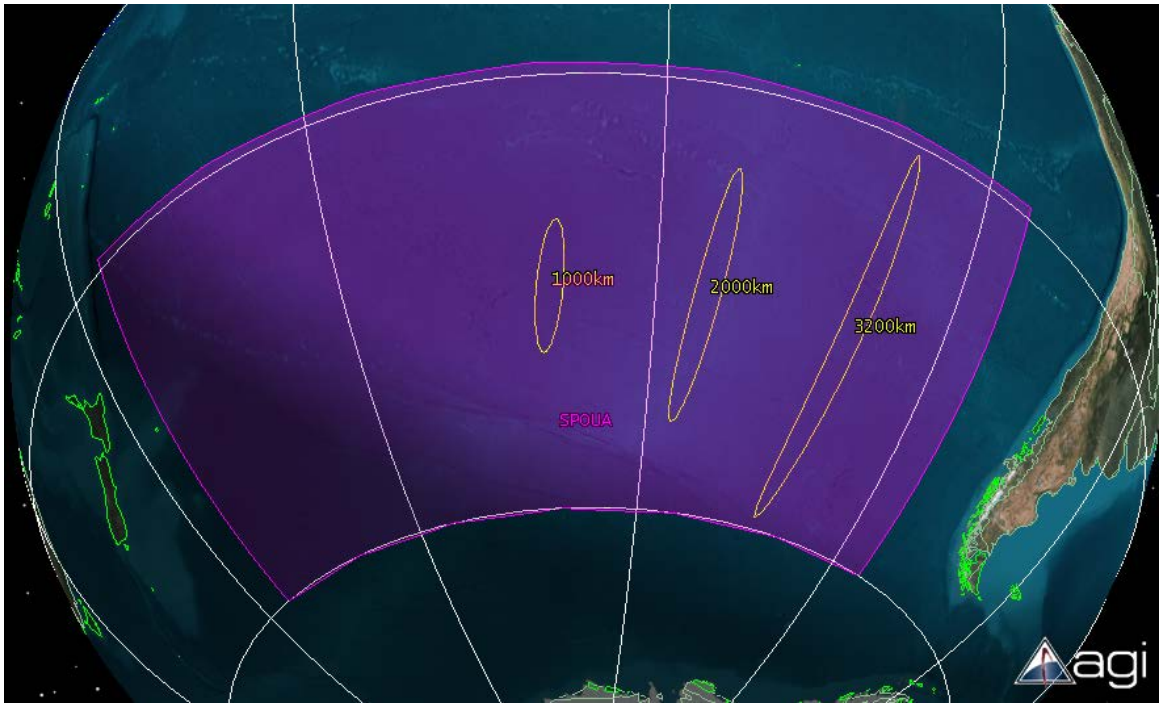


Figure 11. Illustration of the SPOUA represented in STK, with scales of imaginary landing ellipses

The controlled re-entry disposal strategy will most often be based on a **multi-step impulsive manoeuvre sequence**, which is particularly well suited for initial orbits above 500 km. This strategy would be split in three phases (Figure 12):

- Phase 1: Initial Hohmann transfer to lower circular orbit to free operational orbit (e.g. 20-50 km lower altitude), allows still nominal spacecraft operation during preparation for controlled re-entry.
- Phase 2 (.1 to .n): One to several perigee lowering manoeuvers to reach pen-ultimate perigee, number of manoeuvres depends on spacecraft capabilities, possibility for validation of correct disposal system function and correction in case of manoeuvre errors
- Phase 3: Final re-entry boost leading to re-entry and break up in the atmosphere, one thruster left open to initiate tumbling motion of the spacecraft

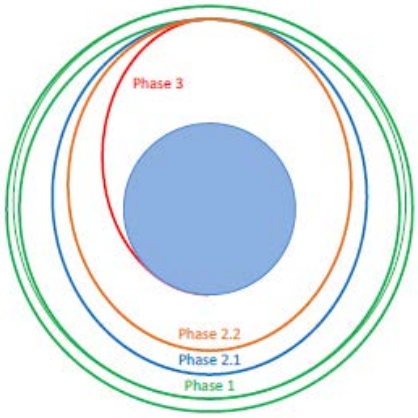


Figure 12. Orbit evolution for a multi-step controlled re-entry strategy

This strategy makes the magnitude of each manoeuvre more easily manageable. It is also interesting for ensuring that the duration of the overall disposal phase remains short but with the possibility of monitoring the orbit evolution and apply corrections where necessary. However, the increased duration of the disposal phase increases the operational effort required.

Simple guidelines for Controlled Re-Entry

Targeted perigee

For most satellites, those with a ballistic coefficient roughly in the range 50-150 kg/m^2, a targeted perigee altitude in the range 40-60 km is sufficient for an orbit at 800 km. More generally, the recommended re-entry flight-path angle (at 120-km altitude) is between -1.5° and -2.5°.

The choice of Argument of Perigee – and through it, the direction of flight (North-South or South-North), see Figure 13 – leads to noticeable variations in the maximum length of the debris cloud. However, it has little impact on the feasibility of the controlled re-entry, as the cloud length is significantly smaller than the length (along the latitudes) of the SPOUA. The AoP can be approximately defined as a function of the targeted altitude (for apogee altitude of 800 km) or re-entry flight-path angle as:

- Ascending pass: $\omega_{tp} \approx 315 + 0.7(60 - h_{tp}) \approx 315 + 60(-1.7 - \varphi_{120})$
- Descending pass: $\omega_{tp} \approx 225 + 0.7(60 - h_{tp}) \approx 225 + 60(-1.7 - \varphi_{120})$

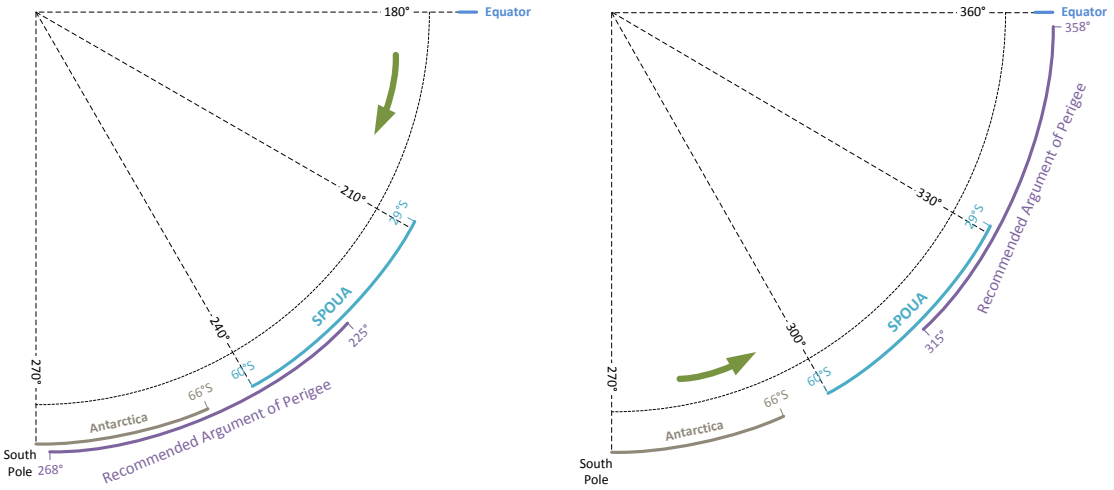


Figure 13. Illustration of the range of argument of perigee to be targeted for a fall-out in the SPOUA.

Thrust authority

The following guideline for the design of the satellite propulsion system for a control re-entry:

- Threshold > 30 N/ton (for low penultimate perigee AND long burns)
- Objective > 50 N/ton (for low penultimate perigee OR long burns)
- Goal >75 N/ton (for design with comfortable margins, for large satellites)

It is possible to write the change in perigee altitude (penultimate perigee altitude to targeted perigee) as a function of the impulse of the manoeuvre over the spacecraft mass.

$$\Delta h_{peri}^{[km]} \cong 200 \times \frac{F_T^{[N]} \Delta t^{[min]}}{m_{sat}^{[kg]}}$$

There is a negligible effect from the apogee altitude and the propulsion system specific impulse.

For a given satellite mass, the trade-off is between:

- The thrust delivered by the propulsion system (dependent on the propulsion architecture and hardware);
- The duration of the manoeuvre;
- The change in altitude of perigee resulting from the manoeuvre.

The duration of the burn is constrained by the amount of gravity losses that can be tolerated, and which are typically compensated for by an even longer burn duration. This can be expressed as a percentage of an impulsive manoeuvre Delta-V, and is a function of the burn duration (Δt), expressed in minutes:

$$\Delta V_{GL} \approx 0.85\% \times e^{(\Delta t/14)}$$

The change in the altitude of perigee is driven by the altitude of the targeted perigee (discussed on the previous page) and the altitude of the penultimate perigee, which is itself primarily dependent on the satellite AOCS capability.

The trade-off equation works well for thrust per spacecraft mass above 50 N/ton approximately, as the required duration is moderate and the gravity losses are very small. It is also possible to express the required thrust authority as a function of Δh_p. A typical figure of 0.2 N/ton/km ±25% appears to be a good and simple first approximation.

The justifications for the above guidelines are detailed in the Final Report.

Existing technology remains the best solution for Controlled Re-entry

Monopropellant hydrazine is the most widely used option for LEO satellites, although the low specific impulse makes it difficult for large Delta-V budgets. For the de-orbit phase, a sufficient level of thrust is required, which can be achieved either by a set of small thrusters or by a main engine. In addition, re-pressurisation or pressure-regulated systems (rather than blow-down) can ensure an optimal thrust level is achieved for a given thruster. In the event of a very large Delta-V budget, it may be possible to incorporate arcjets (with specific impulse in the order of 500 s) into monopropellant hydrazine architectures to cover the satellite needs at different stages of its mission.

Bipropellant offer a 50% improvement on specific impulse but the mass saving is much less due to the heavier hardware. Bipropellant systems are more complex, with more components and thus less reliable. The modular design approach is not possible and therefore bipropellant systems are more time-consuming for AIT activities. Bipropellant systems are thus only interesting for very large vehicles (above approximately 5 tons, such as the ATV) or may be an option for very specific cases of missions with very large Delta-V budgets AND where launch vehicle mass capability is critical.

Solid rocket propulsion has been investigated for de-orbit systems, independent and autonomous from the satellite they would be embarked on, in case of the failure of the latter (e.g. Envisat). They offer high thrust over a short period of time. However, there are many issues with using solid rocket propulsion, notably with

AOCS and reliability for long-duration missions. Solid rocket propulsion might be considered for all-electric satellites, with a large casualty area where Assisted Natural re-entry would not be satisfactory.

Thus **monopropellant hydrazine systems currently remain the best option for the controlled re-entry** of satellites in low-Earth orbit.

The disposal of low-Earth orbit satellites can rely on existing technology

Various architectures have been considered, but it is possible to cover the wide range of satellite mass with two architectures of 4 x 20-N, with or without a 400-N main engine. In case a main engine is used, the 20-N thrusters are used for a thruster-based attitude control and as a back-up (with its own disposal scenario) in case of failure of the main engine.

These systems can operate either in blow-down or re-pressurised, depending on the specific needs of the mission.

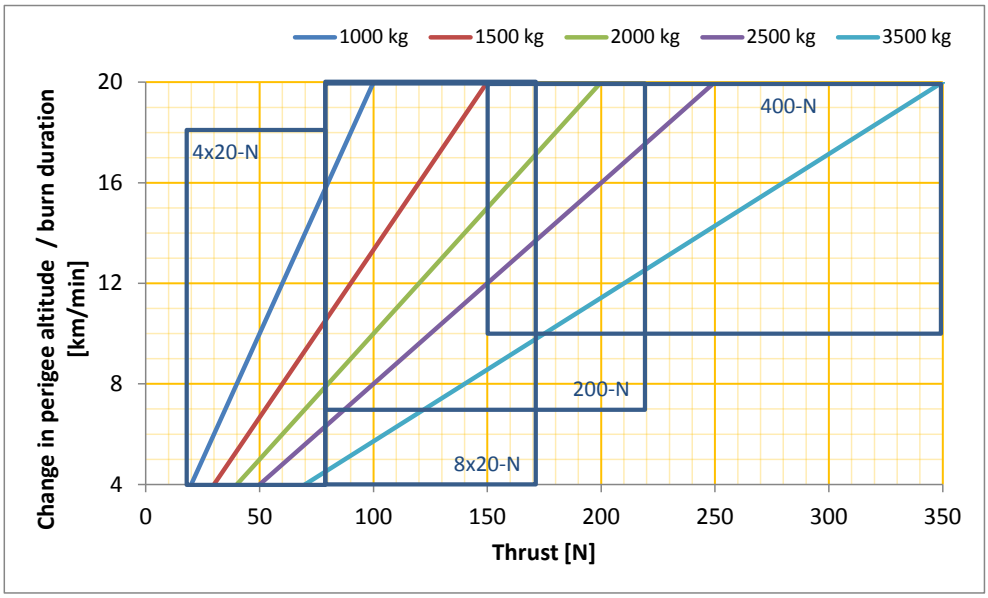


Figure 14. Approximate applicability of candidate monopropellant hydrazine systems.

This has been confirmed through the case study performed for satellites with a mass of 1000, 1500 and 2500 kg. The cut-off point for a 400-N main engine lies around a satellite mass of 2000 kg, and is subject to the specifics of the mission. For satellite mass of 2500 kg and above, the back-up scenario would be to use a single or two (nominal plus redundant) branches of 4x 20N thrusters.

In summary, monopropellant is the preferable choice for controlled re-entry, primarily because it is a good compromise across the range of trade-off criteria:

- It delivers a satisfactory thrust level, such that the final manoeuvre duration is reasonable, without inducing excessive gravity losses
- It offers the simplest attitude control during the big propulsive manoeuvres
- It is based on high TRL, reliable hardware, without any major development required (some re-qualification may be necessary)
- It is versatile, well suited for nominal mission and disposal, and provides flexibility in the propulsion architecture
- It is the least fuel-efficient technology, but the mass penalty is largely compensated by the total hardware mass, for satellites in orbits below 1000 km or so.

		Case 1 : 1000 kg	Case 2: 1500 kg	Case 3: 2500 kg
Option 1	Thrusters	4x 20 N	4x 20N	1x 400N (+4x 20-N)
	Pressure Type	Blow-down	Re-pressurised	Re-pressurised
	Average Thrust	36.7 N	55.8 N	250.6
	Burn Duration	27.6 min	28 min	19.9 min
	Perigee Drop	200 km	200 km	400 km
Option 2	Thrusters	4x 20 N	4x 20N	2x 4x 20N
	Pressure Type	Re-pressurised	Blow-down	Re-pressurised
	Average Thrust	61.2 N	48 N	107.2 N
	Burn Duration	24.6 min	34 min	30.5 min
	Perigee Drop	300 km	200 km	250 km

Table 2. Summary of candidate propulsion systems for the controlled re-entry study cases.

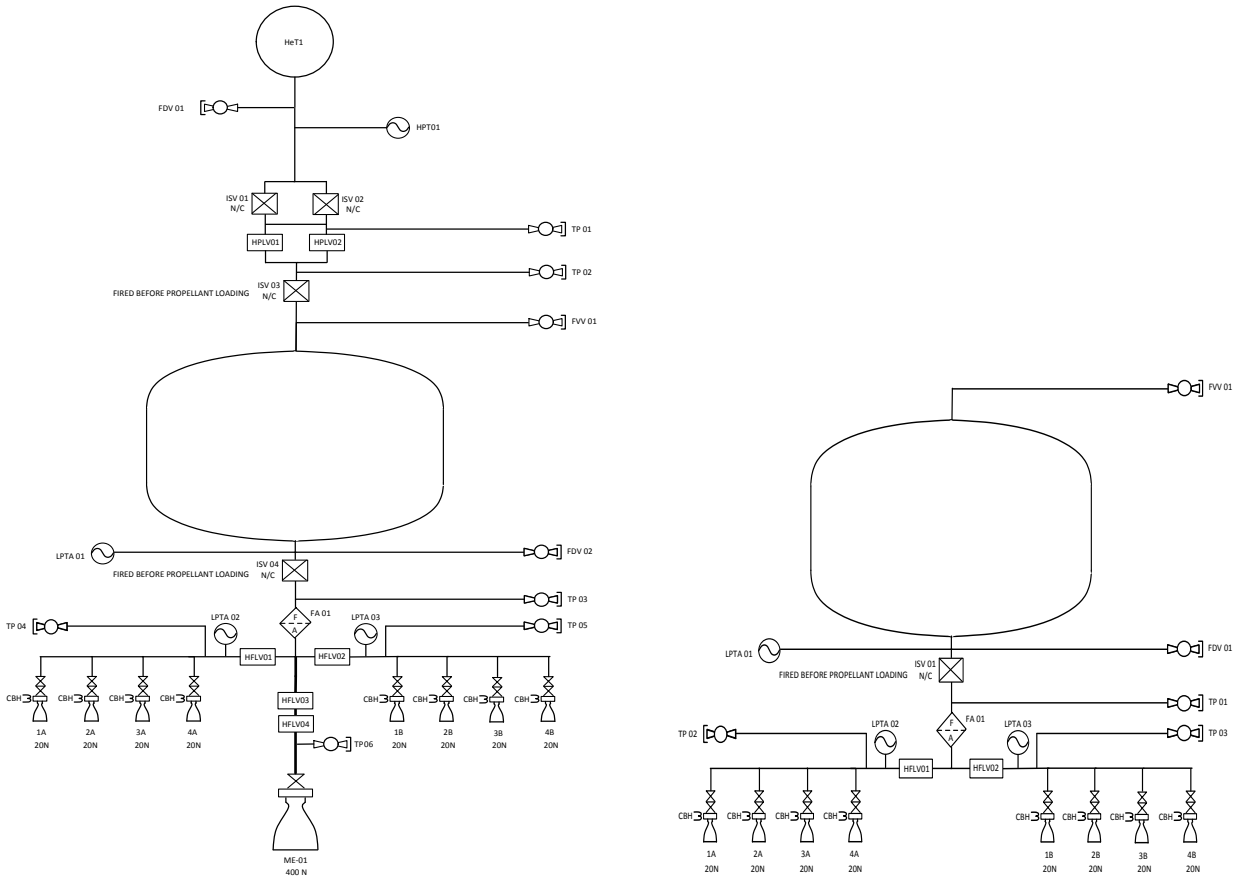


Figure 15. Illustrations of possible propulsion architectures for controlled re-entry

Left: re-pressurised system with 400-N main engine, and right: 4x 20N in blow-down.

Conclusions and Way Forward

Assisted Natural Re-entry / Semi-Controlled Re-entry

Assisted natural re-entry is a technically valid solution for ensuring compliance of low-thrust systems with end-of-life de-commissioning regulations. This is especially useful for all-electrical LEO missions, for which it can offer large mass saving and propulsion architecture simplification over a chemical or hybrid solution. However, assisted natural re-entry strategies with low-thrust systems are not a universal solution for all missions. In particular, it only applies to medium-sized platforms, considering that with electric propulsion, the casualty risk can typically be reduced by a factor 4-5 at most.

The current solution for running the descent control loop is compatible with a ground-based implementation, requiring no modifications of S/C hardware or control laws:

- the orbit control manoeuvres are computed on ground (from ground-based orbit determination or GNSS telemetry)
- the attitude guidance profile (for minimizing aerodynamic torque disturbance around perigee) is adjusted from AOCS telemetry and aerodynamic models

Therefore assisted natural re-entry does not require complex developments of new on-board functions, and is therefore ready for adoption by future missions.

Moreover, there are costs associated to the strategy, in terms of verification and validation (extensive additional analyses), as well as operations during the disposal phase. These costs need to be taken into account in the trade-off activities, before deciding whether semi-controlled re-entry is the preferred solution.

Controlled Re-entry

Controlled re-entry is best suited for satellites with larger casualty risk area, or for satellites where low-thrust propulsion is not an option (due to e.g. nominal mission constraints) or offer little advantage

Monopropellant hydrazine is currently the preferable technology for low-Earth orbit missions, with architectures based on off-the-shelf thrusters of 20 N and 400 N providing sufficient versatility to cover a satellite mass range. The usage of multiple 20-N thrusters enables a thruster-based attitude control during the disposal phase, and they can serve as a back-up to a non-redundant 400-N thruster.

The improved thrust and specific impulse of bi-propellant systems is only worth the additional complexity for large satellites (above 5 tons approximately).

Introducing hydrazine arcjets to complement classical monopropellant thrusters is a good option for satellites in higher orbits (above 1200 km or so) to save propellant mass and still be able to perform the controlled re-entry.

Solid propellant meet the thrust level required, and can be slightly more mass efficient, but this is all lost to the complexity of attitude control during the short burns. Spin stabilisation during the burn may be a solution, but the applicability is quite limited.

Way Forward

Semi-controlled re-entry is still theoretical, and while extensive simulation work is validating the concept, it is still not approved for implementation. Further analyses, in particular on the uncertainties of critical parameters, may help to alleviate the reticence. An important next step in future R&D on the topic is to increase the level of automation (on-ground) or even autonomy (on-board), as these functions would significantly reduce operations costs.

Controlled re-entry is already in use, and can rely on off-the-shelf technologies. The exception is for the special case of missions with large Delta-V budgets, which would typically be satellites in higher orbits (above 1200 km or so), where arcjets could become useful. There is currently no mature European arcjet available. However, the market is fairly limited: while large constellations are being implemented at higher altitudes, these satellites are rather small and uncontrolled re-entry is sufficient. Also, hydrazine is subject to the REACH regulation, and could become banned in a not-too-distant future.

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