LOW-THRUST DEORBITING AFTER DUAL REACTION WHEEL FAILURE: AUTONOMOUS ATTITUDE GUIDANCE AND HYBRID ACTUATOR CONTROL MODE

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ABSTRACT

For large constellations comprising several hundred satellites, the risk of double failure will no longer be extremely improbable. Today a configuration where only two reaction wheels out of four remain functional (in a 3-for-4 redundancy scheme), means a complete loss of attitude control, especially when the bus does not have a chemical reaction control system to fall back to. The resulting inability to perform deorbiting operations would leave a derelict satellite at the mission altitude of the constellation, becoming a permanent collision hazard. Avoiding this risk today for sustainable space development calls for adding a redundant wheel to the avionics design, or deorbiting the satellite after the first failure.

To address this, Airbus DS has developed a new de-orbiting mode called CONDOR (CONtingency DeORbiting) which recovers sufficient controllability to effectively de-orbit the spacecraft using only two functional RWs. The use case is here the ARROW platform, which serves as the generic avionics architecture for constellations and small sat programs, but the mode could be extended to any spacecraft with a magnetic control capability.

The main design principle consists in completing the partial loss of controllability (along the normal to the controllability plane of the two remaining RWs) with magnetic control. As magnetic control authority is generally far below that of reaction wheels, the direction of the magnetic field in the satellite's frame must be optimized. Along with the need to maintain thrust attitude while protecting the field of view of at least one star tracker against blinding or obstruction at all times, this leads to an over-constrained attitude guidance problem. Using an innovative autonomous potential-based guidance function with a very robust obstacle avoidance feature, we have extensively consolidated the feasibility of guaranteeing 3-axis attitude controllability during the thrust phase, allowing the spacecraft to perform de-orbiting even after a dual RW failure.

1 General CONDOR description

1.1 An introduction on USM

In the scope of constellation, and more largely to the New Space context, Airbus is faced with some unique and interesting challenges. One of the main ones is to ensure reliability, robustness and limit operation time loss through intelligent and groundbreaking concepts for safe mode. Therefore, new concepts such as Unified Safe Mode (based on ESA STEAM STR-based mode) and DFLECT has been implemented. This safe mode allows a complete modularity of the control and estimation modules based on the allowed or available sensors and actuators.

Furthermore, the capability of de-orbiting in very degraded configurations is key for constellations and New Space applications, where

- A great number of spacecraft are working on a same mission
- The COTS component utilized are less reliable

Currently, Safe Modes are not generally specified to resist a dual loss of inertial actuators, and if so the AOCS is not able to perform de-orbiting. Furthermore, a passage into Normal Mode is not feasible in this configuration. Thus, the deorbiting phase must be fully integrated into the Safe Mode architecture. This is the main concept of the CONDOR mode which can act as a sub-mode of the USM providing a supported deorbiting function in the Orbit management.

The main problematic when dealing with dual reaction wheel failure is of course the loss of three-axis controllability, making recovery challenging and de-orbiting at first glance impossible. In order to safely deorbiting spacecraft in such configuration, Airbus has developed CONDOR mode that utilizes magnetic capacity to compensate for the loss of controllability.

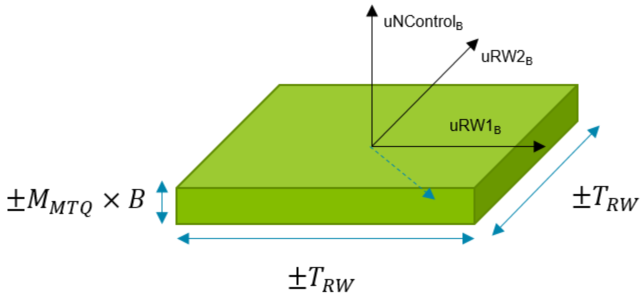


Figure 1 Illustration of the Torque capacity

The general purpose of CONDOR is to build a pseudo three-axis control by completing the full capacity plane left with the two remaining RW with the magnetic torque capacity. Of course, given that a reaction wheel has much more torque capacity than a magneto-torquer, this leads to highly anisotropic controllability. Furthermore, the magnetic torque capacity is dependent of the magnetic field direction. Hence the main challenge of CONDOR is to ensure the maximization of the magnetic control in the direction where RW torque capacity has been lost. This of course is antagonist to the keeping of a thrust direction (even if in that matter, one degree of freedom is available).

1.2 CONDOR guidance using DFLECT

To make de-orbiting possible, the guidance is key to respond:

- The thrust efficiency shall be maximized, meaning that the direction of thrust shall be as close as possible to the wanted direction (in our case, an anti-velocity direction for de-orbiting maneuvers)
- STR must be allowed in Safe-mode and at least one of them must not be blinded (otherwise, the loss of a three-axis knowledge makes the deorbiting impossible)
- The magneto-torquer capacity must be maximized toward the direction of lost capacity by the RW

This leads to an over-constrained problem that can be solved using Lyapunov-based algorithm, called DFLECT. The detailed description of the DFLECT algorithm is not the object of the present paper, however it is important to know the main principle: to define attraction and exclusion laws with associated weights in order to solve the problem in an optimal matter.

First, we need to identify hard constraint: in our case the protection of at least one optical head of the STR from Earth and Sun Blinding. This is defined through two “exclusion rules”: OH1 line of sight kept at an exclusion angle of both Earth and Sun. Then the “attraction rules” are defined as the desired attitude, with an associated weight. In our case we want to maintain thrust efficiency as high as possible, while maximizing the torque capacity. This translates into the following attraction rules: direction of thrust towards the spacecraft speed direction (X in our satellite reference frame), and direction of the normal of the RW capacity plane towards the perpendicular with the local magnetic field.

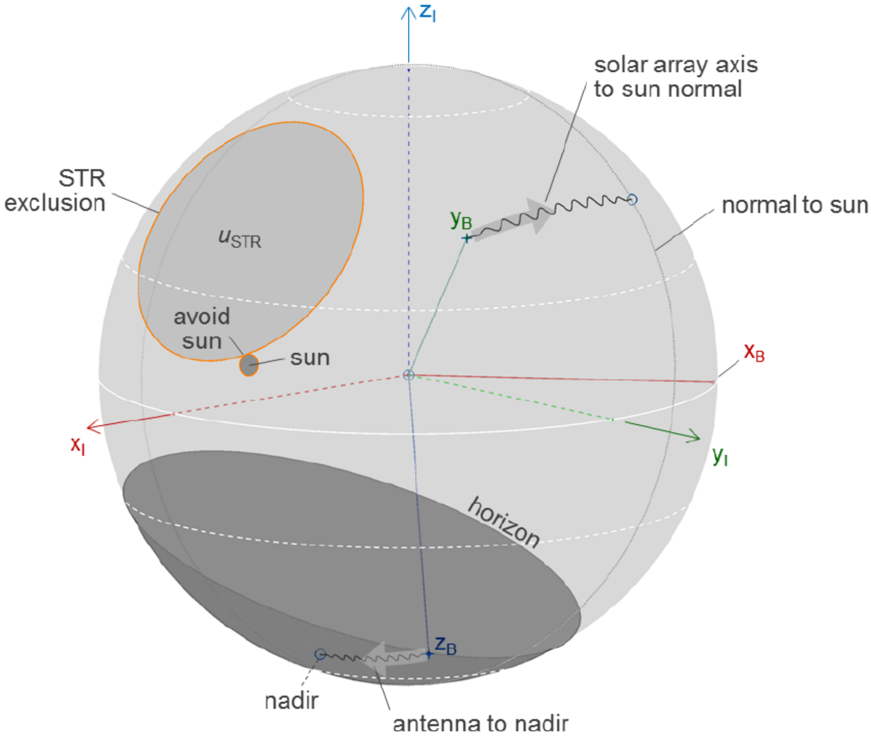


Figure 2 Illustration of DFLECT guidance applied for CONDOR

The antenna is attracted to the earth, while the rotation axis of the solar arrays is attracted to the plane normal to the sun's direction (to ensure full power). Additionally, hard constraints can be added, such as protecting the STR against blinding (by the sun or the earth)

• **Attraction laws**	• **Exclusion laws**
• Line of sight : $\overrightarrow{n_{RW}}\|_B = \dfrac{\overrightarrow{RW_i} \wedge \overrightarrow{RW_j}}{\left\|\overrightarrow{RW_i} \wedge \overrightarrow{RW_j}\right\|}$ • « World-attached » plane normal : $\vec{B}\|_I$	• Line of sight : $\overrightarrow{LOS_{OH_1}}\|_B$ • « World-attached » direction : $\overrightarrow{earthDir}\|_I$ • Exclusion angle : 25°
• Line of sight : $\overrightarrow{HETdir}\|_B = \overrightarrow{X_B}$ • « World-attached » direction : $\overrightarrow{v_I}$	• Line of sight : $\overrightarrow{LOS_{OH_1}}\|_B$ • « World-attached » direction : $\overrightarrow{sunDir}\|_I$ • Exclusion angle : 35°

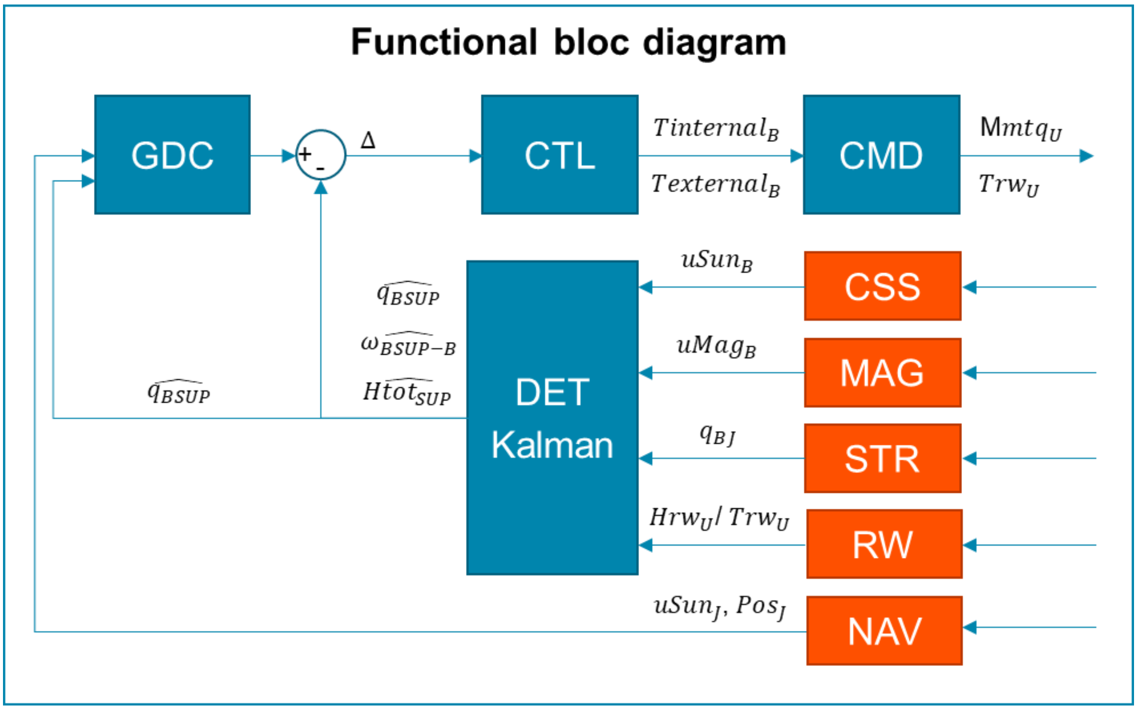
Table 1: Set of attraction and exclusion guidance laws (protection of one OH)

2 Algorithm description and tuning

2.1 Algorithm description

CONDOR algorithm is included in the Unified Safe Mode (USM) in order to address the deorbiting application. The generic functional architecture of the mode is composed by:

- Estimation: provide best attitude estimation of the satellite using available equipment.
- Guidance: generate a target attitude profile in function of the set of attractions and avoidance rules (based on DFLECT). For this study, the rules are described in Table 1: Set of attraction and exclusion guidance laws.
- Control: deduce a physical torque to be generated in order to follow the target guidance (internal) and perform angular momentum offloading (external).
- Command: distribute the actuation request torques to the available actuators. In this application, the control torque in the uncontrollable direction is sent to the magnetorquers (MTQ).



The main actuators used in constellations are reaction wheels, a quite complex equipment integrating software, electronic and hardware components. Furthermore, wheels hardware is composed by mechanisms and mobile parts sensible to different phenomena as aging, temperature and lubrication. At least three reaction wheels are required to control the satellite attitude (one per axis), compensating disturbance torques and providing agility. However, a fourth reaction wheel pyramidal configuration is often used to improve satellite reliability and allowing a less demanding operating profile for the wheels.

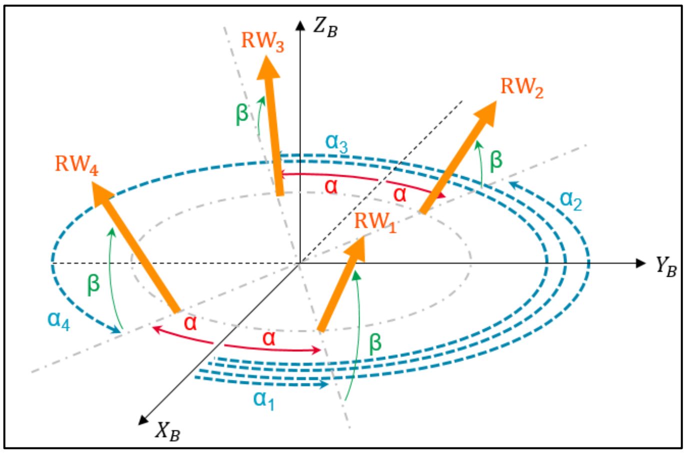


Figure 3: Typical pyramidal reaction wheel configuration

The unit vector of each reaction wheel can be expressed in the satellite frame (B) in function of azimuth (α=) and elevation (β) of the wheel rotor axis:

$$\overrightarrow{RW_i} = \begin{pmatrix} \cos(\alpha_i)\cdot\cos(\beta) \\ \sin(\alpha_i)\cdot\cos(\beta) \\ \sin(\beta) \end{pmatrix}_B \quad \text{with}: \alpha_i = \left\lfloor \frac{i}{2} \right\rfloor \cdot \pi - (-1)^i \cdot \alpha \tag{1}$$

In case of a double reaction wheel failure, the satellite controllability domain is included in the plane composed by the rotation axes of the remaining two wheels. The normal plane of the operating wheels can be expressed as:

$$\overrightarrow{n_{ij}} = \frac{\overrightarrow{RW_i} \wedge \overrightarrow{RW_j}}{\left\|\overrightarrow{RW_i} \wedge \overrightarrow{RW_j}\right\|} = \frac{1}{\sqrt{1 - cos^2(\beta)\cdot\sin^2\left(\frac{\alpha_i - \alpha_j}{2}\right)}} \cdot \begin{pmatrix} \cos\left(\frac{\alpha_i + \alpha_j}{2}\right)\cdot\sin(\beta) \\ \sin\left(\frac{\alpha_i + \alpha_j}{2}\right)\cdot\sin(\beta) \\ \cos\left(\frac{\alpha_i - \alpha_j}{2}\right)\cdot\cos(\beta) \end{pmatrix}_B \tag{2}$$

Magnetorquers (MTQ) are usually used for angular momentum offloading. In this study, they are also used to generate the attitude control torque in the uncontrollable direction of the satellite. The actuation principle is to use the interaction between the Earth magnetic field and magnetic field generated by a coil set in the satellite. Even with three orthogonal magnetorquers, the control principle is nonlinear because control torques can only be generated perpendicular to the local Earth magnetic field. That means that magnetic control has also a non-controllable axis which is collinear to the local Earth magnetic field. The magnetic field evolves (in direction and amplitude) in function of the orbit and the satellite attitude. Resulting torque is proportional to the magnetic moment induced by the coils:

$$\overrightarrow{T_{MTQ}} = \overrightarrow{M_{MTQ}} \wedge \overrightarrow{B_{earth}} \tag{3}$$

However, the CONDOR concept combines the support of the magnetic control with an intelligent target guidance optimizing the remaining wheel configuration capacity. The goal is to maximize the magnetic control capacity building a pseudo 3-axis control. The desired result is to be able to guarantee a thrust direction good enough to perform the satellite deorbiting after a double failure case.

2.2 Tuning strategy

A trade-off have been carried during the tuning phase of the algorithms in order to obtain, on one hand, acceptable performances and reactivity and, on the other hand, feasible command torques by the considered actuators.

Attitude **estimation** function mainly depends on the STR measures availability. The protection of one optical head is deemed sufficient to provide enough accuracy for CONDOR feasibility. In this study we consider the protection of the same optical head during the whole simulation (a more intelligent choice could be analyzed in a second time). In the next chapter, we present the dedicated campaign carried out with the goal to analyze the impact in the results and the sensibility of the concept in function of the choice of the protected OH.

Moreover, the default values of the generic estimation filter of the Unified Safe Mode (USM) and **controller** has been retuned for this new application case, especially in terms of :

- Measurement noise covariance matrix (R)
- Process noise covariance matrix (Q)
- Controller response time (τ_{CTL})

Here below the main justifications of the proposed **guidance** tuning (based on the generic DFLECT algorithm):

- **Attraction laws** are defined in function of the satellite accommodation and local Earth magnetic field (see §1.2). Each law is tuned with a specific weight factor. They are tuned in order to ensure satisfactory ratios of magnetic and thrust controllability. After a preliminary analysis, both attraction rules ("thrust direction pointing" and "magnetic field pointing") have been tuned with the **same weight** (comparable to control stiffness).
- **Avoidance laws** are tuned with typical Sun Exclusion Angle (SEA) and Earth Exclusion Angle (EEA) values of the Star tracker sensors. Avoidance laws have been tuned with the highest priority weight in order to always ensure an available estimation (at least one STR OH not blinded). This choice induces a guidance degradation in some cases.
- **Response time** (τ_{LYAP}) of the Lyapunov guidance algorithm is set to 60 sec. It is equivalent to the past period information processed in the algorithm to build the current target attitude.

3 Mission application

Considering that the risk of double failure is a question of probability intimately related to the number of satellites and equipment units in orbit, it seems particularly interesting to focus the preliminary analysis of the de-orbiting capacity with a very degraded configuration for a large constellation scenario. That is why the use case selected to model and prove the CONDOR concept is the Arrow platform which is the basis of the Airbus DS generic avionics architecture for constellations in Low Earth Orbit (LEO).

For this specific use case, the interests of this concept are numerous and very important: increase the satellite lifetime by avoiding a satellite deorbiting after the first reaction wheel failure, reduce the risk of space debris generation and collision avoidance operations and contribute to a more sustainable space development. All these advantages without adding redundant units to the current hardware and avionics design. Thus, an innovative solution for the future missions but also compatible with the production chain and in-orbit platforms.

3.1 Satellite

The central body is a roughly a 1 meter-sided rhomboid with two mobile solar arrays, for a mass of 150 kg. The two solar wings are deployed on +Y and –Y faces and have two degrees of rotation in order to optimize the power capacity. The 2 GWA are deployed on +X wall. The Ku antennas are mounted on +Z wall.

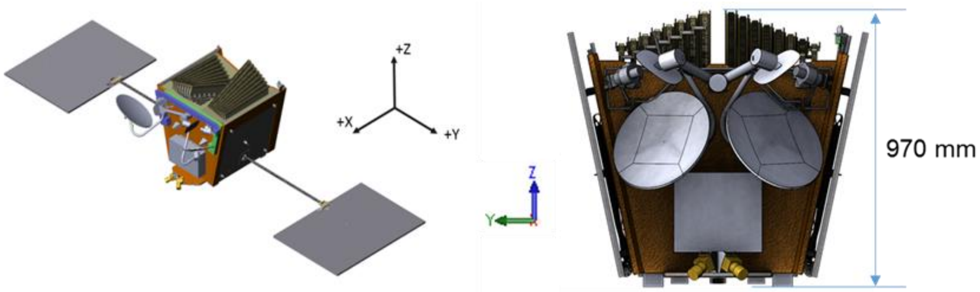


Figure 4: Arrow satellite external overview (deployed and stowed)

3.2 AOCS avionics

Here below the generic Arrow avionics used in the AOCS loop:

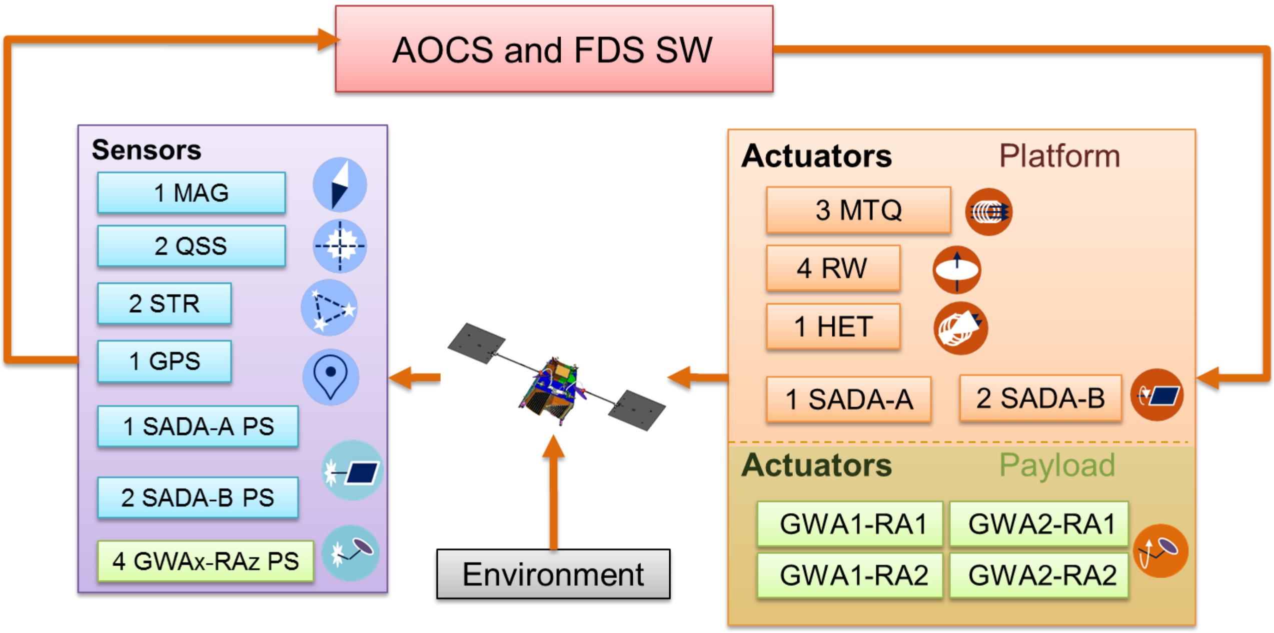


Figure 5: Arrow GNC/AOCS equipment

Three magnetorquers offer an external torque capacity to the platform. A cluster of four reaction wheels offers fine attitude control capability and internal angular momentum storage capacity. Hall Effect Thruster (HET) is used to ensure the orbit control, the orbit raising and the de-orbiting. As we can imagine, the goal and results of the study are mainly related with actuators specifications and constraints (torque capacity, reliability, operation limitations). The approach followed is to apply the concept to an already existing COTS equipment configuration but a specific sizing integrating this mission phase could be also envisaged.

The coarse angular rate measurement in Safe Mode is realized with Magnetometer (MAG), and the Sun detection is ensured by a pair of Quadrant Sun Sensors (QSS). The inertial attitude is computed thanks to two Star Trackers (STR). The position and timing is given by the GPS.

3.3 Thrust sequence

During EOR and station keeping phases of the OneWeb mission, the maneuvers sequence considers discontinuous thrusts of the electrical propulsion. The strategy takes into account all the constraints and needs at system level (energy, propulsion hardware and flight dynamics) and represents a compromise between the different subsystems.

For deorbiting purposes, a propulsion duty cycle lower than 100% is also acceptable. A symmetrical sequence is preferable in order to minimize the perturbation of the orbit eccentricity. However, the main goal of the deorbiting is to reduce the orbit apogee until the atmosphere reentry. The orbit knowledge can be determine autonomously using the GNSS measures or by the ground system managing also the collision risk avoidance. One of the main goals of this study is to prove that there are sufficient slots in the orbit with an acceptable thrust vector error to accomplish a deorbiting with two wheels.

4 Preliminary tuning and performances

4.1 Flight Dynamics feasibility

A preliminary Flight Dynamics study has been conducted in order to show the feasibility of the de-orbiting. A satellite trajectory with an inertial guidance profile is considered, built geometrically so that the HET axis can be aligned in anti-velocity twice an orbit. Only the RW 3 and 4 are working.

Two missions cases are analysed, and compared with the reference mission (described above) :

- **Case 1** : At each orbit, 1x20min maneuver placed at the Argument of Latitude (AoL) that minimizes the angle between the thrust direction and anti-velocity direction at centroid and the closest to the apogee
- **Case 2** : At each orbit, 1x20min maneuver placed at the AoL that maximizes the efficiency of the burn in terms of $\frac{dHp}{dt}$ (integrated over the maneuver, Hp being the perigee altitude)

	Reference Mission	**Case 1**	**Case 2**
Ha/Hp	1080km / 300km	935km / 300km	970km / 300km
ΔV	214.6m/s	267.2m/s (+25%)	262.5m/s (+22%)
Final mass	140.76kg	140.13kg (-0.62kg)	140.19kg (-0.56kg)
Transfer duration	114.9d	153.6d (+35%)	151.1d (+33%)
HET ON/OFF	1609s	2162s (+34%)	2124 (+32%)

Table 2: Cases assumptions

This guidance provides with 2 thrust opportunities per orbit (only the most effective is retained), but with a quite poor overall efficiency:

- The proper alignment of HET with anti-velocity is achieved over a very brief period of time. The mean “thrust error” (the angle between the HET axis and the actual velocity direction) is typically 18° (which leads to 5% loss of the HET thrust efficiency)
- The thrust opportunity can be quite far from the apogee, ideal thrust location to decrease perigee.

Those constraints being combined, the thrust efficiency ranges between 30% and 95% as seen in the Figure 6 :

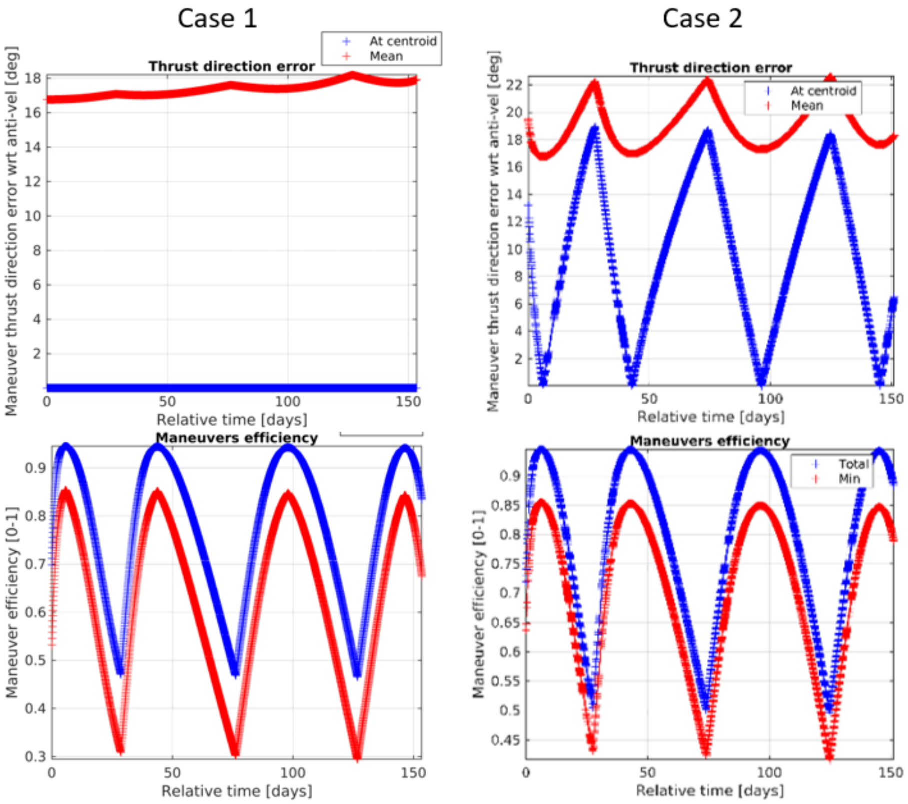


Figure 6: Thrust direction error and HET efficiencies evolution over time, depending on the case

However, despite those contraints, this study has shown that de-orbiting can be achieved with an acceptable overcost of ~35% with reference to nominal de-orbiting (in terms of duration, ΔV, HET ON/OFF cycles), and still well below available budget (30 m/s remaining).

For the followings sections, we will consider an arbitrary thrust direction error threshold equal to 45°.

4.2 Assumptions summary

Table 3: Use case assumptions

	Value
Initial altitude and inclination	1200 km / 87.9 deg
Initial eccentricity	~0
Initial RAAN and true anom.	Randomly selected between [0; 360] deg
Satellite mass	150 kg
Propulsion system	1×15mN , Isp = 1200 sec
Simulation duration	~4 orbits
RW torque	30 mNm
RW available	2
MTQ torque	~1 mNm
Target ang momentum	0 Nms
Disturbing torques	HET residual, magnetic, solar

The HET is ignited after the first orbit. When switched on, the HET thrusts until the end of the simulation. The changing of altitude is considered negligible and is not computed, which is relevant with the duration of the simulation.

The uncontrollable axis directly depends on the available reaction wheels:

Available RW	3+4	2+4	1+4	2+3	1+3	1+2
Uncontrollable axis	$\begin{pmatrix} 0 \\ sin(\beta) \\ sin(\alpha) \cdot cos(\beta) \end{pmatrix}_B$	$\begin{pmatrix} sin(\alpha) \\ cos(\alpha) \\ 0 \end{pmatrix}_B$	$\begin{pmatrix} sin(\beta) \\ 0 \\ sin(\alpha) \cdot cos(\beta) \end{pmatrix}_B$	$\begin{pmatrix} -sin(\beta) \\ 0 \\ sin(\alpha) \cdot cos(\beta) \end{pmatrix}_B$	$\begin{pmatrix} -sin(\alpha) \\ cos(\alpha) \\ 0 \end{pmatrix}_B$	$\begin{pmatrix} 0 \\ -sin(\beta) \\ sin(\alpha) \cdot cos(\beta) \end{pmatrix}_B$

Table 4: Uncontrollable axis, depending of the available wheels

4.3 Simulation campaigns

Several simulation campaigns have been performed:

- First campaign (C1): this is the reference campaign considering the protection of OH1. It is the comparison basis for the next campaigns.
- Second campaign (C2): simulations considering the protection of OH2. This analysis aims to show the influence in the results of the OH protection strategy.
- Third campaign (C3): strategy improved with the protection of OH1 from the beginning of the simulation and addition of a Nadir-pointing attraction of the Z_B axis.

	Simulations
C1	1000
C2	100
C3	100

Table 5: Summary of the campaigns

The selected key performance indicators (KPI) :

- HET efficiency: ratio of collinearity (dot product) between the target thrust vector and the true state.
- MTQ efficiency: ratio of collinearity (dot product) between the target magnetic vector and the true state.

4.4 AOCS performances

Campaign 1

Here below the main conclusions of the reference campaign:

- All the simulations ensure a mean HET efficiency better than 67% (in other words, an alignment between the HET axis and the velocity direction of 48°)
- 50% of the simulations guarantee a mean HET efficiency better than 95% (in other words, an alignment between the HET axis and the velocity direction of 18°).
- 85% of the simulations ensure an HET efficiency >75% during the 85% of the time (in other words, an alignment between the HET axis and the velocity direction of 41.4°).

	OH1 protected		
Statistics (mean, after HET ignition)	**1σ**	**2σ**	**3σ**
HET efficiency (%)	87	71	50
Total time during which HET efficiency > 0,75	86	58	26
MTQ efficiency (%)	72,3	64	51
Angle between nadir and Z_B (deg)	105,4	140,0	141,3

Table 6: Summary of the campaign 1 results

The angle between nadir and Z_B is also only given for all purposes, but it has to be kept in mind that at this time, no alignment law is given.

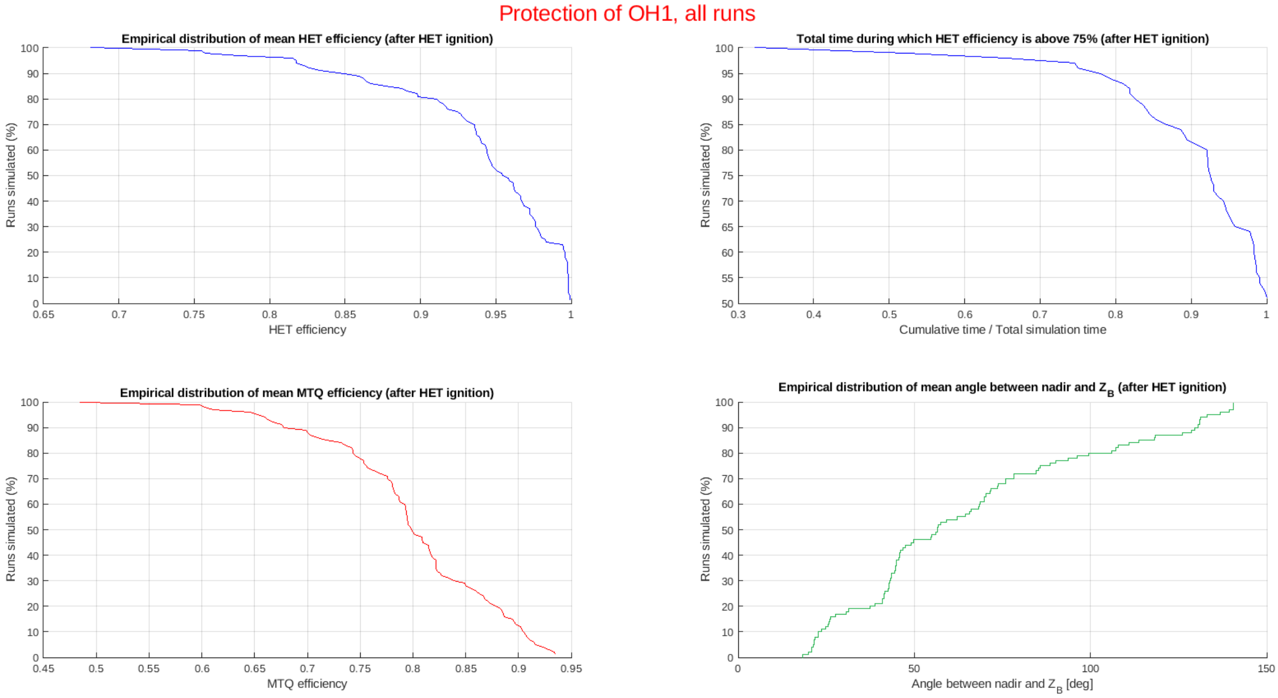


Figure 7: Indicators of the campaign C1

Campaign 2

When protecting the OH2 instead of the OH1, the results are similar, and so are the conclusions. We can neglect at this stage the sensibility of the results to the choice of the OH protected.

Campaign 3

In order to reduce the number of the STR tracking loss observed during the first campaigns, a new strategy have been analyzed. The protection of the OH is performed from the beginning of the simulation. Also, a new attraction law is integrated in the guidance algorithm (DFLECT). With the previous attitude laws, there is a degree of freedom around the thrust vector (anti-velocity) inducing a non-controlled rotation of the OHs. In order to minimize the OH blinding, especially by the Earth, a Nadir-pointing attraction of the Z_B axis is added. Even if the satellite has not a 3 axis control capacity, this geocentric attitude induces a passive protection of the OH by design and geometry.

	OH1 protected (with nadir attraction)		
Statistics (mean, after HET ignition)	**1σ**	**2σ**	**3σ**
HET efficiency (%)	84	53	48
Total time during which HET efficiency > 0,75	82	47	35
MTQ efficiency (%)	55	52	52
Angle between nadir and Z_B (deg)	23,2	45,6	64,3

Table 7: Summary of the campaign 3 results

As seen in Table 7, the results improved the STR availability but degraded a bit the magnetic control and HET efficiencies.

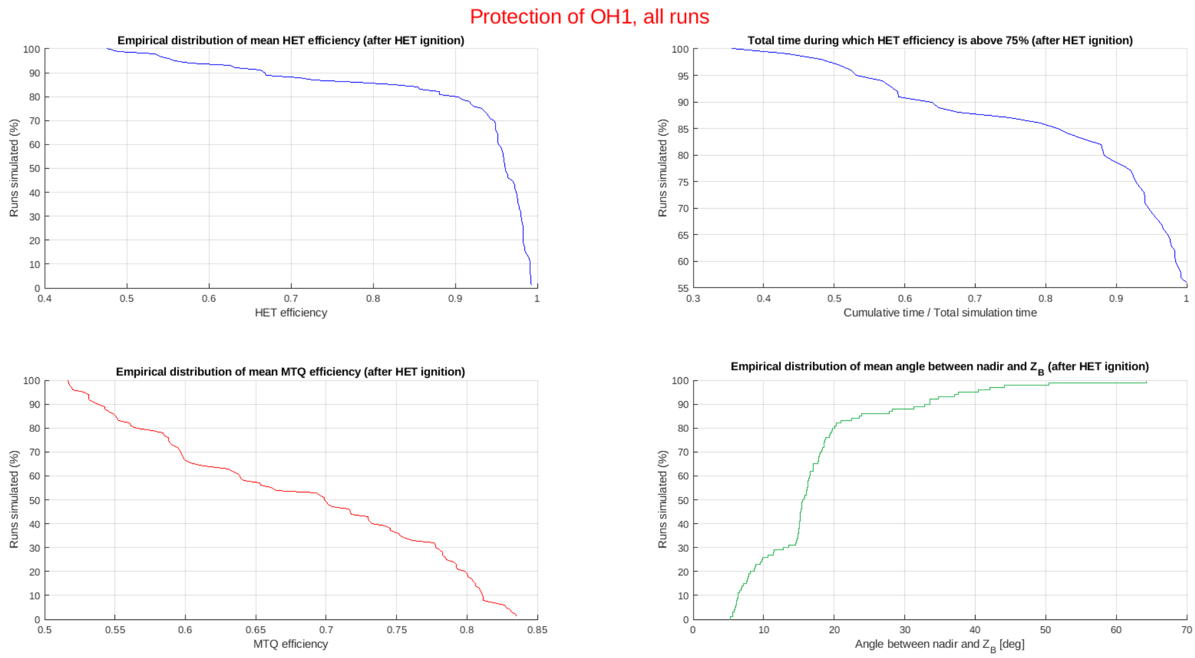


Figure 8: Indicators of the campaign C3

4.5 Offloading performances

In order to prove feasibility of the mode strategy, the offloading performances during thrust have been checked. Indeed, the wheel cluster must be able to compensate for external disturbing torques (caused by both environment and thrust dispersion). Some particular cases are displayed on figure below. For a continuous thrust, the offloading is able to keep wheel saturation for at least three orbits.

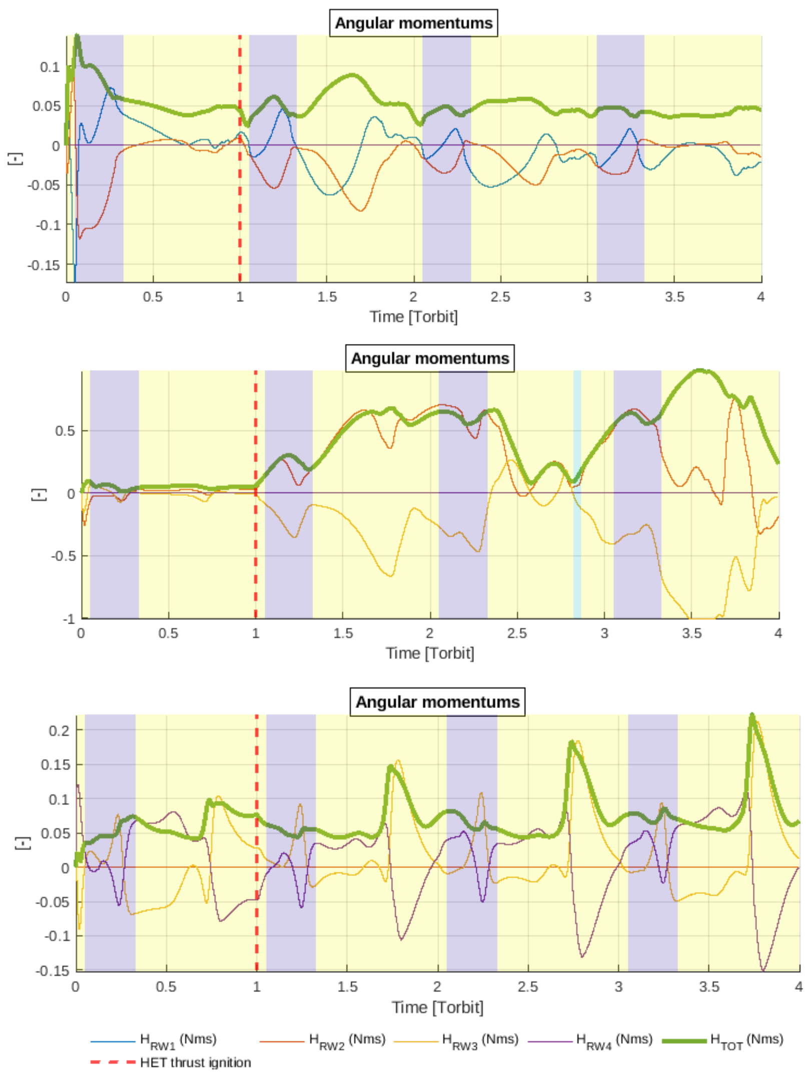


Figure 9 Evolution of the total angular momentum during thrust

4.6 Synthesis

The simulation campaigns on the Arrow use case have confirmed the feasibility of the CONDOR strategy to deorbit a satellite with only two reaction wheels available.

Table 8: Synthesis of the campaigns

	C1 OH1 protected			**C2 OH2 protected**			**C3 OH1 protected + nadir attraction**		
Statistics	1σ	2σ	3σ	1σ	2σ	3σ[1]	1σ	2σ	3σ[1]
HET efficiency (%)	87.1	71.3	50.2	89.1	75.8	65.9	83.6	52.5	47.6
Total time ratio with HET efficiency >75%	86.0	58.2	25.9	88	71	52	82	47	35
MTQ efficiency (%)	72.3	64.2	51.1	73.3	65.0	61.0	55.1	51.8	51.7
Angle between nadir and Z_B (deg)	105.4	140.0	141.3	105.4	140.0	140.6	23.2	45.6	64.3

[1] The number of runs is not enough to get relevant 3σ statistics. Their presence in the table is only indicative.

5 CONCLUSIONS AND WAY FORWARD

In the scope of Arrow platforms, CONDOR mode proves the feasibility of performing de-orbiting with a dual loss of reaction wheels. This application can secure missions functioning with degraded RW configuration, allowing the re-entry if a second wheel is lost. This capability is particularly important in large constellations where the probability of dual loss during mission lifetime becomes high due to large numbers of spacecraft and use of COTS components. Although studied in Arrow platform for constellation-type missions, this concept could easily be extended to all types of missions with magnetic control capability. In the context where clean space is an major topic, the ability to maintain a safe three-wheels configuration while ensuring a clean re-entry in case of further wheel failure is key.

The closed loop impact of the thrust vector error on the trajectory has not been considered in this study and can be a next step. Also, there is an identified opportunity to extend the feasibility of the concept to other flight domains and HW platforms impacting the current satellite reliability and the associated lifetime.

One important feature to be added is the possibility to optimize the protection of a set of STR, instead of protecting one Optical Head at all time causing the general guidance to be very constrained. Preliminary concepts show improvement of the overall guidance error, and thus thrust efficiency and/or availability.