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JUST-IN-TIME COLLISION AVOIDANCE MISSION: REACTIVE SYSTEM FOR BRAKING SPACE DEBRIS

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Abstract

The context of increasing proliferation of space debris makes space agencies study various means for reducing risks. An initiative of CNES focuses on the issue of avoiding collision between non-maneuvering large debris (such as upper stages, end-of-life or damaged satellites) by braking one of the debris. In the frame of preliminary concept analysis, an assessment of this idea is built upon technological survey, physical analysis leading to orders of magnitude, then first modeling aiming at measuring credibility/readiness of the concept and at identifying technological barriers.

In order to avoid a most probable collision between 2 orbiting objects (JCA: just in time collision avoidance), a suborbital rocket (sub-orbital trajectory) could be launched from Earth in order to decrease the speed of one object therefore modifying its trajectory, thanks to a cloud of gas and particles. This Earth-based option would be less expensive than an orbital solution and would be deployed only when required.

The system is supposed to sequentially operate as follows:

- a sub-orbital rocket (either launched from ground or air-launched) places a generator of gas and particles onto a target altitude close to the one of the debris to deflect,
- once onto target altitude, the generator is powered up ensuring that a cloud of gas and particles is created,
- the debris passing through the cloud thus generated, experiences a drag force inducing a slight slowdown,
- the trajectory of the debris is therefore being modified and, after several drifting orbital revolutions, thus eventually avoids the other debris

Feasibility of braking effect had first been set with the use of CFD computations, and success criteria on the shape of the cloud pointed out, such as positioning parameters, size and density of particles (results presented during EUCASS 2017). Preliminary sizing has highlighted ways of generating a cloud with improved efficiency, regarding the example of the plume out of a Solid Rocket Motor.

This paper will complete the previous study and will be focused on the system and mission aspects:

- On mission side, type and number of launch bases needed to fulfill both reactivity and “rendezvous” objectives, and comparison of alternatives
- Time and space margins required by environmental constraints, and their allocation to sub systems
- Pre-sizing of launch vehicle, including propulsion, aero-dynamics, payload space and mass, carrier if any

Resulting architecture, principles and orders of magnitude will be presented in this paper. These different studies used innovation and design techniques (creativity, multidisciplinary working group, MBSE/ARCADIA) and modelling tools (CT Paris’ home-made CFD software CPS_C™, home-made HADES software platform, commercial optimization environment modeFRONTIER, open source MBSE tool Capella).

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1. Introduction

1.1 Context

Collision between large non-maneuverable objects rarely occurs but results in generating a large number of new debris of various sizes at each occurrence, thus rapidly increasing the uncontrolled orbital population and consequently collision risks. To counter these collisions among large objects, two approaches can be considered, as part of the Large Debris Traffic Management:

- The "strategic" one consists in retrieving a certain number of large debris each year, thus reducing the probabilities of major collisions. This Active Debris Removal (ADR) strategy [1] has been intensively studied throughout the world for more than 10 years. It appears to be technically feasible, but hard to finance, and raises numerous non-technical problems such as legal or political ones.
- The "tactical" one [2] is aimed at avoiding an "announced" collision by acting on one of the two debris some time prior to the predicted collision date. This strategy is called Just-in-time Collision Avoidance (JCA). The first ideas were presented some 10 years ago, and among the solutions which have been proposed, slightly slowing one of the debris via an impulsive drag force appears promising [3]. For the tactical approach, flexibility and reactivity are, with the perennial cost criterion, the main challenges.

In the following chapters of this paper, we describe a non-orbital system aiming at braking a debris for JCA. The feasibility of the braking effect through a dense cloud generation has been shown in [4]. The objective of this paper is to assess the operational feasibility of the overall system, including the launching system for the envisioned cloud generator.

1.2 Mission

The typical mission sequence is the following: two large debris A and B are identified, having a probability of collision higher than a given threshold. Their orbits are determined, by the Joint Space Operations Center (JSpOC) for instance, with a good precision several days before the potential collision, this information being refined over time thanks to dedicated radar measurements (on-ground and/or orbital observations). The decision to perform a JCA operation on one of the two concerned debris is decided by an ad-hoc international body, to be defined.

A sounding rocket is launched from one of the dedicated bases in the world. On top of the upper stage is mounted a gas and particles generator, several orbits prior to the expected collision. While the sounding rocket is at apogee, this generator fires a cloud towards the trajectory of the debris right before its transit time. The debris passes through the cloud and hits the particles which, by energy exchange, slightly reduce the debris orbital velocity, thus changing its trajectory. This slight change of trajectory is propagated until, at the expected collision date, the debris misses the collision point. As the generator is on a sub-orbital trajectory, it falls back to Earth immediately after the operation, potentially recovered and reused.

2. The system

2.1 Modelling of the system

It has been decided to deploy a Model-Based System Engineering (MBSE) approach on the project using the Arcadia method and the associated Capella tool. This approach offers several advantages, such as the detection of sensitive modeling properties, the gathering of all the data in the model, and the traceability of the requirements that are given to the system.

Below are the first steps of this modeling method, which are required to describe the stakes of the project. First, the actors involved in the system are defined:

- **Space debris**: The space debris that is considered cannot be maneuvered from ground. It can be an outworn satellite, an empty stage of a rocket or any large space debris.
- **Owner of satellites**: Satellites orbiting in LEO have three main purposes: scientific, commercial and military. As their owners are the main beneficiaries of the space security, a tax on the activity of operational satellites to clean up LEO could be considered. But states also have a major interest in contributing to the funding of the mission, to guaranty access to space for their industry and/or economy.

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- **Space surveillance organism**: The Joint Space Operation Center (JSpOC) is constantly monitoring the objects orbiting in LEO. When a probable collision is detected, it provides Conjunction data Messages (CDM) to satellites owners tenth of hours before the incident. They can then maneuver their satellite in order to avoid the collision. The new observation system Space Fence will raise the accuracy of the measures, and avoid a large amount of false positive collision detections.
- **Legal environment**: Current legislations are not adapted to the current situation. There is no real legal framework for organizing an emergency system to avoid a collision between two debris. Indeed, the legislation of both the country where the debris came from (owner of the object, country from where it had been launched, etc…) and the legislation of the diverting system (to which we have to add the legislation of the launching country) have to be considered. In addition, as it is impossible to know in advance which country will be involved, it seems that a worldwide moratorium on that topic has to be organized, in order to create an international organization with the authority to launch a mission in case of emergency.

These actors and their functions are described in Figure 1.

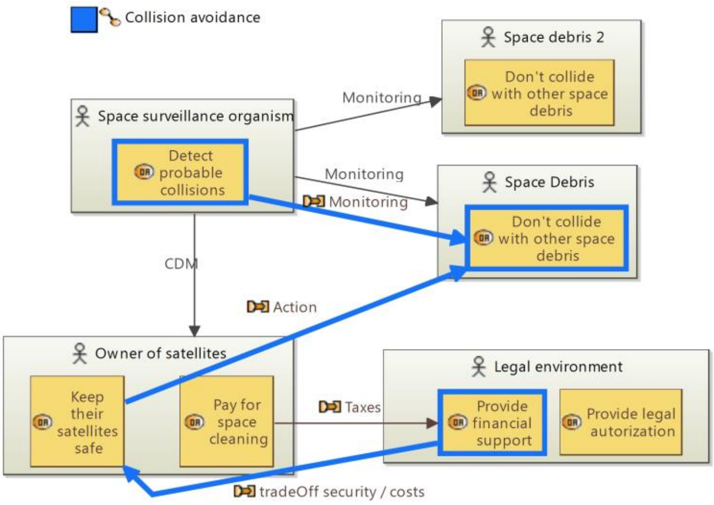

Figure 1: Operational Architecture of the System

The diagram in Figure 2 summarizes the main mission of the system and the capabilities it has to fulfill.

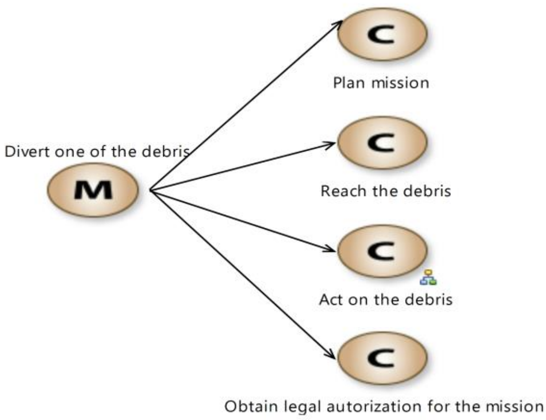

Figure 2: Missions and Capabilities of the system

From these capabilities, the main operational functions of our system (in green), and of the external actors (in blue) are shown in Figure 3.

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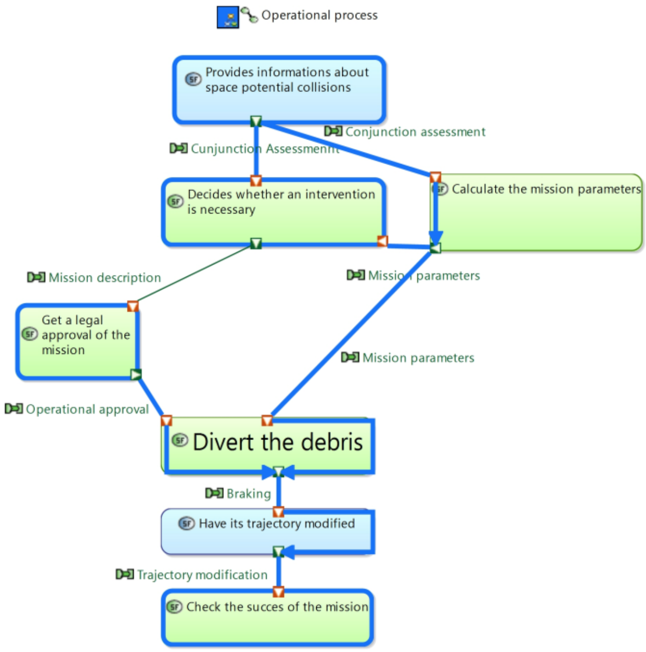

Figure 3: System main Data Flow Blank

Figure 4 shows a simplified diagram representing the main components of the system with the main functions that are assigned to them: it is at this point that the idea of the particles cloud appears.

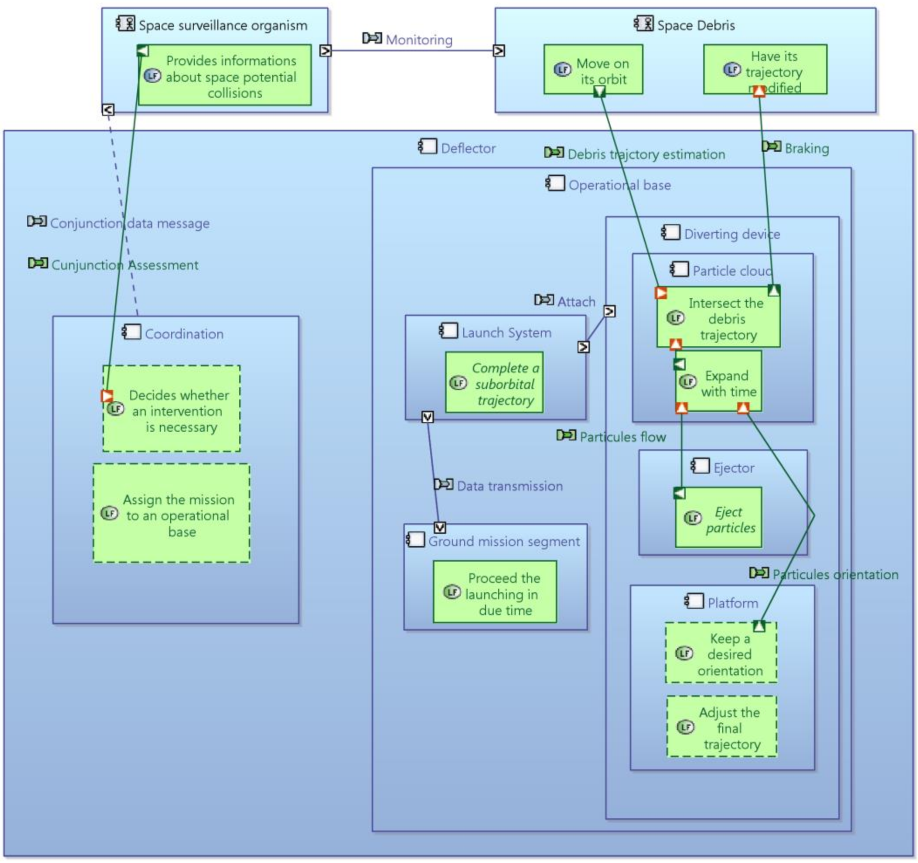

Figure 4: System Logical Architecture

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The whole model developed for the project is much more complete and detailed. The glimpse of this model presented here only aims at giving a global outlook on the whole system. The MBSE approach proves to be interesting for the management of the requirements of the system, and the management of the interface with potential subcontractors.

2.2 Main requirements

Several requirements apply to the mission and to the system. First, the system‘s trajectory shall stay sub-orbital, in order to avoid inserting any new object on orbit. Also, in order to reduce the risk of a collision between the debris and the generator, it has been stated not to pass over the debris trajectory, and to ensure a safety distance between the two objects. This safety distance has been set to 500m but needs an appropriate analysis to be confirmed. These two requirements constrain the positioning of the system. Indeed, the cloud generation has to be performed around the apogee of the suborbital trajectory, in order to be as close as possible under the debris path while respecting the safety distance.

Moreover, it has been shown in [4] that the generator’s trajectory shall be in the orbital plane of the debris during the generation. Indeed, if the generator’s apogee velocity is transverse to the orbital plane, the cloud generated will also have a transverse velocity, all these would require an even more precise timing for the particles ejection to ensure the cloud is on the trajectory when the debris passes.

It has been decided that the system shall be able to perform the JCA for debris in orbit up to 1200km and in any inclinations. This has been considered as the maximum altitude in which the collision risk between large debris is substantial, especially with regard to the future mega-constellations planning on using LEO orbit up to this altitude.

Eventually, timing constraints are also very restrictive. Indeed, the observations of the space object’s position suffer uncertainties which are propagated to the ephemerids of the two debris future trajectories. The longer the time between the observation date and the expected collision date, the higher the uncertainty on the debris trajectories. The requirement has been set to 24 hours between the date the JCA is triggered and the collision expected date. It is expected that future improvements of the debris observations shall contribute to lower the uncertainty on the trajectories, enough in order not to trigger the JCA too often. The effect of this constraint on the mission timeline is presented in section 3.1.

3. Mission analysis

The mission analysis aims at studying the effect of the system on the target debris and at finding a flight scenario for placing the generator in position for the braking. Indeed, inserting a generator into a sub-orbital flight for rendezvous with an orbiting object requires flight mechanics, launcher performance and rendezvous analysis, which are presented in this chapter.

3.1 Post braking effect

The trajectory of the debris after being braked has been simulated. To do so, the Clohessy Wiltshire Equations [5] are used, their results shown in Figure 5. This simulation shows the result of a braking of $\delta V = 7.7mm/s$ propagated during 12h on a 1200km circular orbit debris. The figure shows the new debris trajectory in a reference frame fixed to the initial debris trajectory. The reference frame is the LVLH (Local Vertical Local Horizontal) relative to the initial debris trajectory, i. e. without braking, with the axis shown as Vbar for the velocity direction (i.e. local horizontal) and Rbar for the direction from earth to the debris (i.e. local vertical). It allows seeing the effect of the braking on the debris, compared to a fictive debris which has not been decelerated.

This figure illustrates that despite being braked, the debris is actually moving forward from its initial trajectory. The sequence is the following: at the initial (0, 0) position, the braking is applied, one can see that the debris slightly moves backward, in –Vbar. This shows that the debris velocity is lower than its initial velocity V, as expected due to the braking δV. Due to this lower orbital velocity, the debris perigee is lowered by 30m. On the other hand, the apogee is kept unchanged, which introduces an orbital period difference δT compared to the initial orbit period T. After one revolution, the debris comes back to the initial apogee, $\delta L = 140m$ ahead of the initial trajectory.

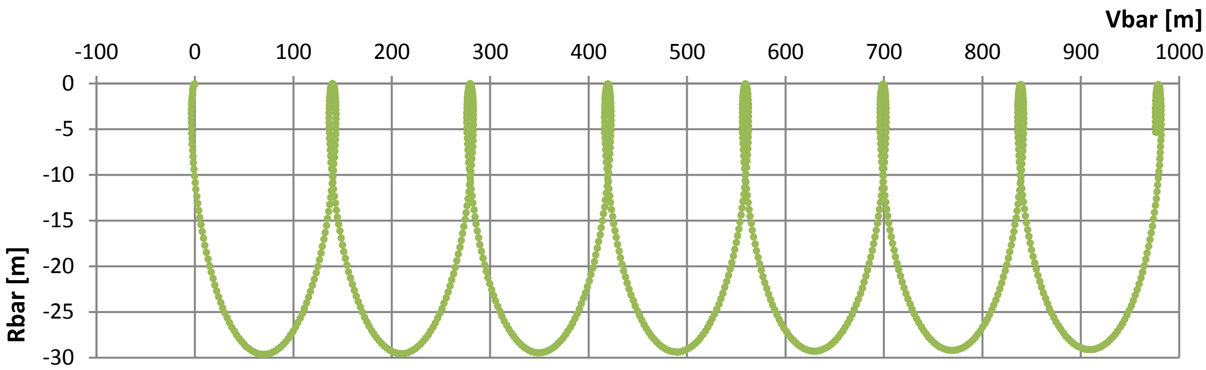

Figure 5: Debris trajectory after braking in the LVLH frame fixed to the trajectory before braking
Braking of 7,7mm/s, simulation duration 12h

After $n = 7$ revolutions, i.e. almost 12h, the new debris trajectory is almost 1000m away from its initial trajectory. The avoidance distance thus depends on n, the number of revolutions the braking effect can be propagated on (see section 3.1 for timings) but also on δV, the deceleration transmitted to the debris. The Clohessy Wiltshire Equations give the following formula for period difference δT induced by a deceleration of δV:

$$\delta T = -3.T.\frac{\delta V}{V}$$

Leading to the avoidance distance δL:

$$\delta L = -3.n.T.\delta V$$

This calculation can be coupled with the debris mass and friction coefficient (included in δV calculation) to compute the required mass of particles hitting the debris, called effective mass. This is shown in Figure 6 for a 1500kg, 800km altitude debris with a friction coefficient of 1.7. The graph shows the required effective mass depending on the avoidance distance, for 6 and 12 hours of propagation of the braking effect.

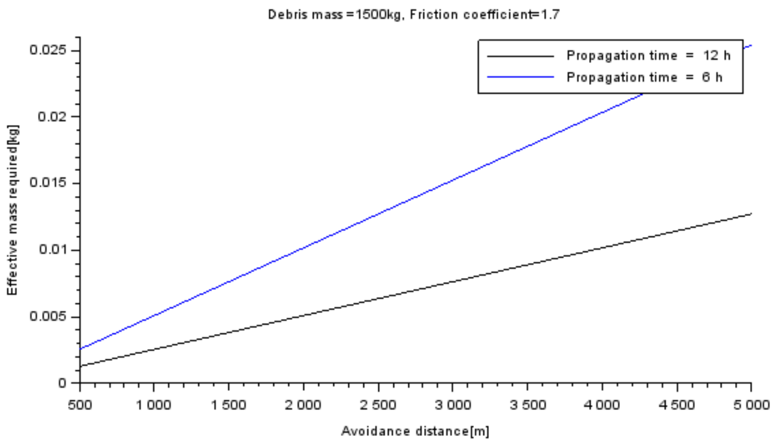

Figure 6: Effective mass required for avoidance distance and propagation duration

This figure illustrates the very small effective mass requirement. Indeed, only few grams are required to brake sufficiently the debris. The main difficulty is to set up this mass to the correct position and on the correct timing via a particles ejector. The ejector itself is not discussed here but one solution is discussed in [4]. The solution to bring this ejector system in position and time is discussed in the following paragraphs.

3.2 Ground bases location

The optimum ground bases locations obviously depends on the debris orbital parameters. In order to reach any orbit, a certain number of ground bases are required. This number depends on:

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- The capability of lateral deport catch-up by the system. Indeed, due to timing constraint, it is unlikely that the debris trajectory will pass right above the ground base. It is thus necessary to catch-up the lateral deport to join the debris orbital plane.
- The delay between the decision to engage the procedure and the expected collision. Figure 7 shows the timeline of the process: the acting time T_A is the time between the decision to engage D_E and the moment when it is too late to act on the debris D_F. One of the requirements is that the acting time shall be small enough to cope with the precision of the debris positions prediction (T_P=24h) and the braking propagation time required (T_B=12h).

The time T_A is used to wait for a suitable debris pass and to launch, catch-up with the lateral deport and reach the apogee where the systems acts on the debris. Its maximum value is 12 hours but the baseline for the study is 6h

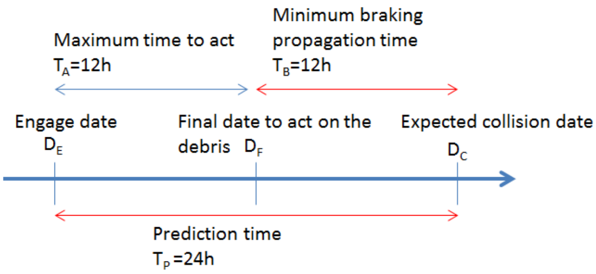

Figure 7: Timeline

Another requirement is that the bases are located near the equator in order to be able to reach any orbit inclination. Moreover, one of the drivers of this analysis is the orbital drift, and due to the equatorial constraint, the equatorial drift. Figure 8 shows the worst case in the range of orbit studied for this analysis (maximal altitude 1200km and polar inclination 90°): up to 3060km can separate two orbits at the equator. This number will be used as requirement.

Figure 8: maximum equatorial drift

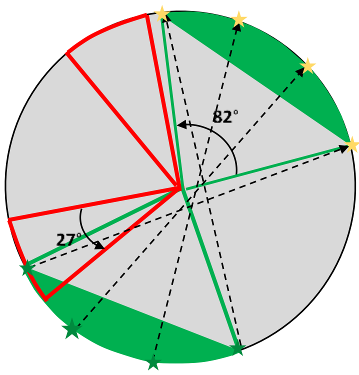

Figure 9: ground bases location

A geometrical approach can be used to compute the number and locations of the ground bases. Figure 9 shows in dashed line the orbit of the debris during 6 hours: the ascending and descending nodes will be shifted by 82° in longitude in 3 periods (5h30), the equivalent of 3x3060km around the equator. The green zone shows the equatorial footprint for both nodes. One possible solution of coverage by the ground station is shown by the two red triangles. They represent the necessary coverage zone (at the equator): indeed if both coverage zones are 27.3° large in longitude and 90° one from the other, at least one dashed line will always fall into one of the triangle. This shows that at least one opportunity for reaching the orbit occurs in 6h. If the triangles are smaller or with a different angular separation, this is not ensured anymore.

The number of bases required in this coverage zones can be computed using the apogee range of the system. Indeed, the 27.3° longitude represents 3060km on the equator. The number of bases per triangle is 3060km divided by the apogee range of the launcher. The Table **1** shows the number of coverage zones and the number of bases required for a given acting time and a given launcher range.

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Table 1: number of bases required

Acting time (TA)	Launcher apogee range (km)	Total number of bases required	Number of coverage zones required
0h	3060	6	6
6h	3060	2	2
12h	3060	1	1
0h	1530	12	12
6h	1530	2	2
12h	1530	1	1
0h	765	25	25
6h	765	4	2
12h	765	2	1
0h	383	51	51
6h	383	8	2
12h	383	4	1
0h	191	104	208
6h	191	16	2
12h	191	8	1

This table shows a worst case of 104 bases, which would simply be too expensive and complex to manage and keep operational 24/7. This explains the need for a large range or a large acting time in order to reduce the number of bases. The air launched system presented in the following section is a solution to improve the range. In that case, the range will be done by the aeronautic carrier. Improvement of the Space Situation Awareness and the precision of the observations and ephemerides of the space debris are required to increase the acting time.

On the other hand, it is important to mention the redundancy of the bases. Indeed, if for a meteorological reason, one of the bases is unable to perform the required launch, another base shall be able to replace it and perform the launch. This particular problem is very global and has not been addressed in this study but will impose extra bases to the system.

3.3 Insertion in the debris orbital plane

As seen before, the launch system is used to join the orbital plane of the debris, i.e. to catch the lateral deport between the launch site and the orbital plane. Thus, the first part of the trajectory crosses this orbital plane. However, it is mandatory to inject the system in the orbital plane of the debris so, a second maneuver to change the velocity vector and align it with the orbital plane is necessary. This is shown in the Figure 10.

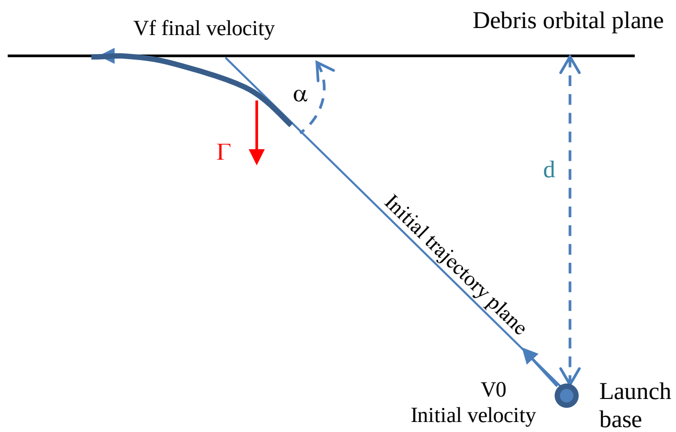

Figure 10: Trajectories, projected on a horizontal plane

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Angle α represents the angle between the orbital plane and the initial trajectory (after the take-off boost). Discussion on the angle α can be found in [7]. Γ is the constant acceleration given by the rocket to inject the system in the debris orbital plane. The Γ boost being finite, the trajectory draws a curve until it reaches the orbital plane.

An analysis has been performed to study the effect of α on the launch system mass. The hypothesis of a three stages rocket is taken: the 2 first stages used to give the apogee altitude and the range and the 3rd stage to inject into the debris orbital plane. The following parameters are used: d=500km, debris altitude H=800km, acceleration $\Gamma=40 m/s^2$, payload mass 250kg and constructive index (stage dry mass / propellant mass) of 13%, 15% and 25% for the 1st, 2nd and 3rd stage respectively. This analysis uses a trajectory optimizer to optimize the 1st and 2nd stages boosts to bring the system to the so-called initial trajectory.

	Losses (m/s)		**α=5°**, ΔV_deport_catch= 562,7 m/s, ΔV_initial = 7459 m/s					
	Aero	Incidence	K (%)	Isp	Dry mass	Propellant mass	Complementary mass	ΔV
Stage 1	70	50	13	270	2160	16619	0	3750
Stage 2	0	50	15	290	353	2355	0	3879
Stage 3	0	0	25	280	21	84,4	100	562,7
Launcher total mass: 21 942 kg								

	Losses (m/s)		**α=30°,** ΔV_deport_catch = 1113 m/s, ΔV_initial = 5125 m/s					
	Aero	Incidence	K (%)	Isp	Dry mass	Propellant mass	Complementary mass	ΔV
Stage 1	70	50	13	270	551	4242	0	2519
Stage 2	0	50	15	290	198	1319	0	2776
Stage 3	0	0	25	280	50	200	100	1113
Launcher total mass: 6 910 kg								

	Losses (m/s)		**α=90°**, ΔV_deport_catch = 1153 m/s, ΔV_initial = 3908 m/s					
	Aero	Incidence	K (%)	Isp	Dry mass	Propellant mass	Complementary mass	ΔV
Stage 1	70	50	13	270	247	1903	0	1854
Stage 2	0	50	15	290	132	883	0	2224
Stage 3	0	0	25	280	52	210	100	1153
Launcher total mass: 3 778 kg								

The conclusion of this analysis is that the case with α=90°, i.e. aiming straight at the debris orbital plane and then braking to stop the system horizontal velocity, is the most mass efficient method. Indeed, with α<90° the gain obtained on the ΔV_deport_catch do not compensate the necessity of a larger range which give a larger ΔV_initial. In such a case, the velocity of the system when arrived at the apogee is brought to zero and the trajectory after rendezvous is a vertical fall.

This shows that a roughly 4 tons three stages rocket is able to bring a 250kg payload into a rendezvous with a debris at 800km from a base 500km away from the orbital plane of the debris.

3.4 Air-launched solution

Another possible option for this launch system is to use an air-launched rocket. Indeed, the air-launched solution gives the opportunity to increase the apogee range of the rocket with the plane range in order to catch-up with the debris orbital plane. Moreover, using the manoeuvrability of the aircraft, the release of the rocket is done in the debris orbital plane, which means that the rocket is already injected in the orbital plane. In fact, due to the earth rotation there is always a longitudinal component in the rocket's velocity which might introduces a small ΔV to be compensated in order to be completely injected in the orbital plane.

Note that as mentioned, the trajectory time (between the launch and the apogee) is included in the acting time. When thinking about conventional or sub-orbital rocket, this trajectory time is around few minutes so it does not have a large impact on the operational timing. On the contrary, for an air-launched rocket, the time for flying towards the orbital plane before the release of the rocket can be up to several hours so it has to be taken into account. On the other hand, for an air-launched rocket, the apogee range includes the flying range of the aircraft, which can be much larger than the rocket apogee range itself.

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Thus, applying these results to an air-launched solution, would give a fair amount of ground bases:

- For a 6h acting time and 600km range for the aircraft and rocket combined, a total of 4 bases are needed
- For a 12h acting time and 600km range for the aircraft and rocket combined, only 2 bases are needed

A cruise speed of 900km/h would give a flying time of 40min, which has to be included in the acting time.

This shows that the air-launched solution gives the opportunity to greatly reduce the number of ground bases. On the other hand, it is necessary to develop an air-launched system, both plane and rocket, with the required apogee and range to cope with that solution.

3.5 Rendezvous with the debris

In the rendezvous concept, three synchronization situations exist:

1) the timing is nominal: the debris and the system are synchronized, as planned
2) the system is in advance: the debris will pass the rendezvous point after the system
3) the system is late: the debris will pass the rendezvous point before the system

These dispersions are expected to happen in the operational phase, due to different factors such as errors in the debris position and trajectory assessment or launch system dispersions. One requirement is that the rendezvous is made at the apogee of the system trajectory, to avoid any vertical velocity other than the ejection velocity. One option for synchronizing the system with the debris is to reduce (or increase) the flight duration (i.e. the time to the apogee) of the system in order to meet the debris before (or after) the nominal rendezvous date. This is done with two boosts presented in Figure 11. It shows the example of the system being late with respect to the debris, the initial trajectory in red. The objective is to accelerate the system with a first boost so that it reaches the debris orbit more quickly. This acceleration has two main effects. First, it move the rendezvous point backward on the debris trajectory, has shown in the figure form R_n to R_a, but the debris velocity being so large that this has only a minor impact on the rendezvous date. Moreover, due to this acceleration, the apogee is not at the debris altitude anymore, but higher. As it is required to have the apogee at the debris altitude (minus the safety distance), a second boost is necessary to slow down the system so that the apogee altitude is brought back to nominal.

The larger the time between the two boost the less ΔV is required. Obviously there are some constraints for the timing of the two boosts, for instance 30s might be required between the last boost and the rendezvous for controllability of the system.

The characteristics of the orbit and the maneuvers for an example case with a debris at 800km and a positive delay of 30s are given in Table 2.

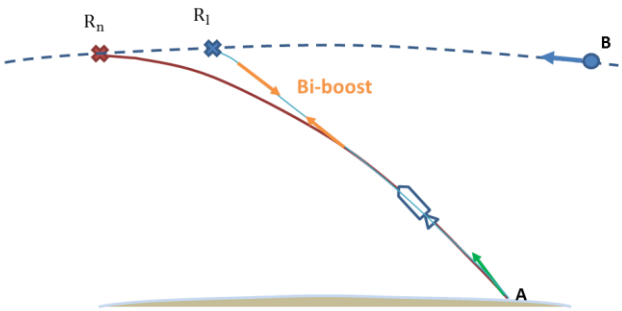

Figure 11: rendezvous with the debris, maneuver for a system being late

The total ΔV is 292.4m/s. Figure 12 shows the ΔV required depending on the synchronization offset. This ΔV has to be taken into account in the propellant budget of the terminal stage and its amount will depend on the precision of the nominal trajectory and of the debris observation. For instance, a 400m/s ΔV margin allows for a synchronization offset of -120s to +100s, thus given a launch window of 220s.

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Table 2: rendezvous sequence example

Maneuver	Altitude (km)	Velocity (m/s)	Time before reaching apogee (s)	Apogee altitude (km)	Delay with respect to the debris (s)
	100	3465.7	418.2	800	+30
Boost1 ΔV1 27.7 m/s	100	3493.5	422.55	812.5	0
	796,5	497	64.33	812.5	0
Boost2 ΔV2 -264.7 m/s	796,5	232	30	800	0

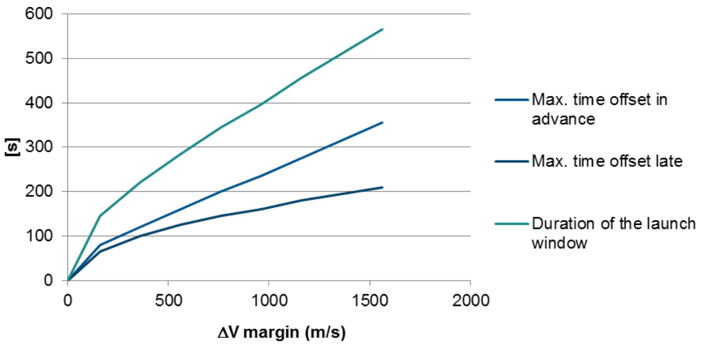

Figure 12: deltaV margin versus synchronization offset for an altitude apogee of 800km

This also shows that depending on the launch and observation conditions, the total propellant mass might not be required, thus the necessity for the capacity of shutting down the engines of the terminal stage.

Other maneuvers might be necessary to correct potential out-of-plane errors in the trajectory, i.e. if the trajectory plane has a horizontal offset with the debris orbital plane. With usual observation means, the debris orbital plane can be known with a good precision (typically 15-20m cross track error, see [6]). The out-of-plane maneuver is mainly to correct launcher trajectory error. An analysis has been done, not detailed here, on the bi-boost maneuvers required for out-of-plane dispersions correction and shows that the ΔV requirement is lower than the ΔV presented for the synchronization offset. Typically, around 100m/s of ΔV margin shall be allocated in the ΔV budget of the terminal stage for up to 10km out-of-plane error after separation with the second stage.

For rendezvous precision reasons, it is recommended to have an onboard autonomous observation mean. Radar or optical sensors are possible options. The range has been evaluated to 1200km, which is 160s before the rendezvous, in order to assess the trajectory of the debris with an enhance precision and ensure a good synchronization both in position and timing.

3.6 Flight scenario

Possible launcher design and flight scenario have been drawn based on the air-launched solution proposed in section 3.4. The air carrier maximum payload capacity has been set to 2770kg, typical to a jet plane. The accommodation of a rocket on the jet and the cruise phase until drop have not been studied yet. The rocket design, trajectory and performances have been obtained using in-house tools.

The configuration chosen is a two stages rocket plus a terminal stage that includes the generator. The two propulsion stages are using H2O2/kerosene (baseline to have storable propellants), resulting in an ISP of 300s. For stage 1 and 2 respectively, the construction index are 13% and 15%. With the jet plane maximum payload capacity mentioned above, the stages designs are shown in Table 3.

Table 3: air-launched rocket design

Stage	Propellant mass (kg)	Inert mass (kg)	Combustion duration (s)
1st stage	1615	240	27
2nd stage	538	80	13
Fairing	0	50	0
Terminal stage including generator	Total mass 247		Dispersions dependent

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The total mass is 2770kg. Using this design, a flight scenario example for a rendezvous with a debris at 1200km is has been developed and is shown in Table 4 and Figure 13. In this scenario, the velocity at the apogee is 224m/s, in the debris orbital plane.

Table 4: flight scenario sequence

Time (s)	Event	Altitude (km)	Flight Path angle (°)
0	Rocket drop	12	20
5	1st stage ignition	12.181	4.1
32	1st stage end of burn	33.2	76.6
	28.4s coasting phase , 1st stage separation		
60.4	2nd stage ignition	86.9	74.9
71.1	Fairing jettisoning	113.8	84.1
73.4	2nd stage end of burn	122.4	85.0
	540.6s nominal ballistic phase to apogee, terminal stage guidance		
614.1	Nominal apogee	1199.5	0

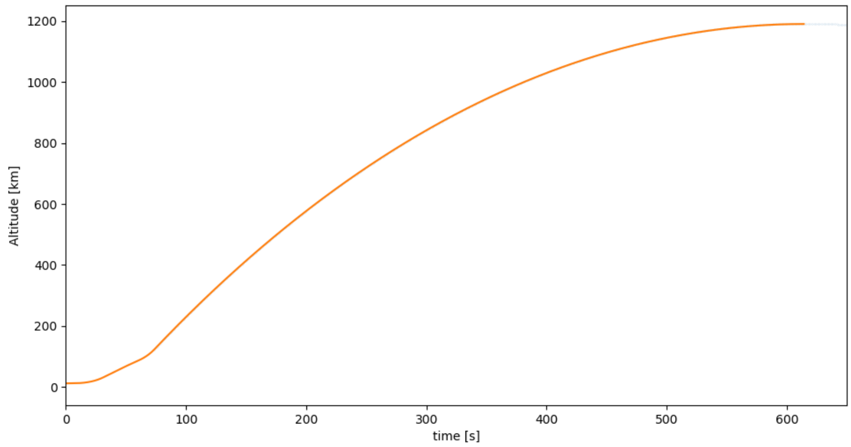

Figure 13: altitude vs. time

This analysis shows that rocket design that meets the requirements of the mission exists, hence showing the feasibility of the concept. The next step is a deeper study on the complete launcher system, including the air carrier, the ground bases locations and their concept of operation.

4. Conclusions and recommendations

The feasibility of the operation of a JCA has been studied in this analysis. First the overall system, the actors and their functions have been modeled using the MBSE/Arcadia method. Then, the mission itself has been studied to draw first design parameters.

A certain number of ground bases are required for the launch, depending mainly on the range of the launch system and the timing constraints inherent to the observations means. It showed that an air-launched system can be the solution to avoid a large number of bases all over the globe in order to catch the lateral deport implied by the timing. This also proved the necessity of improvement of the debris observation, in order to gain accuracy on the ephemerids of the concerned objects, which would give more acting time to the operation.

The rendezvous phase has also been studied, mainly the in-plane and out-of-plane errors correction. For the in-plane dispersions, a strategy to cope with synchronization offsets and apogee altitude error has been proposed, based on a bi-boost strategy that reduces or increases the flight time to synchronize the rendezvous with the debris. The out-of-plane maneuvers have also been evaluated, to draw a ΔV budget of the necessary margin necessary to the final guidance for rendezvous.

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An example of a possible rocket design and flight scenario has been drawn and shows that the air-launched solution is feasible using liquid propulsion for the first two stages. Because of the required maneuvers (flight time and apogee correction, transverse maneuver, collision avoidance maneuvers, stabilization and attitude control of the generator), general design considerations for the terminal stage can be drawn from this study. The propulsion architecture shall most certainly allow for a 3 axis attitude control, as well as to be able to implement a thrust in any direction.

Eventually, in order to estimate the precision of the trajectory system with respect to the different dispersions and perturbation, as well as the impact of the observation means, it will be necessary to model the on-board automatic guidance and to test it in simulations taking.

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