Comparison of Chemical Propulsion Solutions for Large Space Debris Active Removal

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Abstract In recent years, the development of active removal missions, to face the growing risk of catastrophic collisions and new debris generation because of the high density of orbital debris in LEO, is widely discussed in the international space community. Besides legal and political issues, active removal measures are strongly hampered by high the costs involved. Chemical propulsion might represent the preferred way to perform the controlled reentry of large abandoned objects and, in the perspective of cost reduction, hybrid rocket technology is considered a valuable option, due to the potential lower fabrication and operational costs if compared with bi-propellant liquid systems. The possibility to use non-toxic propellants, besides

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their lower prices, reduces the complexity of handling, storability and loading operations, decreasing the connected costs and avoiding the need of a special staff. This study deals with the preliminary design and mass budget comparison between solid, liquid and hybrid propulsion modules used for active removal missions.

1 Introduction

Since the 1980s, the increasing relevance of the orbital debris problem, in order to guarantee the sustainable future use of circumterrestrial space, became progressively evident [1]. In addition to the voluntary adoption of specific mitigation measures by single space agencies, the need of addressing the issue on a global basis led to the creation of the Inter-Agency Space Debris Coordination Committee (IADC), which approved in 2002 the first version of a set of recommended space debris mitigation guidelines [2]. These inspired national laws or guidelines and international standards, intended to direct the actions of government and private entities in space. Finally, in 2007, recommendations derived from the IADC guidelines were endorsed by the United Nations [3].

During the past quarter of century, the efforts were concentrated on the adoption of mitigation measures, aiming at reducing or preventing the production of new orbital debris. These measures included the passivation of spacecraft and upper stages at the end of their operational life, in order to prevent accidental explosions, the choice of hardware and procedures to minimize the release of mission related objects, the end-of-life removal of spacecraft from relatively crowded space regions, as the Geostationary Orbit (GEO) ring, the limitation of the residual orbital lifetime of abandoned spacecraft and rocket bodies in Low Earth Orbit (LEO), below the altitude of 2000 km, and the prevention of accidental catastrophic collisions with conjunction assessments and, if needed, avoidance maneuvers [4]. Unfortunately, notwithstanding the progresses observed over the course of the years, the global level of compliance with the mitigation recommendations, in particular those dealing with the re-orbiting of spacecraft and upper stages after mission completion, is still relatively modest, being around 2/3 in GEO [5] and short of 60% in LEO [6].

Moreover, during the last decade, the results of some long-term simulations of the debris evolution suggested that mitigation measures alone, even if duly implemented in more than 90% of the cases, might not be sufficient to stabilize the number of orbital debris > 10 cm in the currently most crowded altitude ranges in LEO, being such objects the typical projectiles able to cause the catastrophic fragmentation of average spacecraft or rocket bodies at characteristic collision velocities in excess of 10 km/s [7, 8, 9]. Invoking the recourse to remediation, by managing the existing space debris population through Active Debris Removal (ADR), became, therefore, more and more popular in recent years [10, 11, 12], and various ranking schemes were developed to prioritize the potential target objects to be removed first, in order to favor a more benign evolution of the circumterrestrial environment on the long-term [10, 13, 14, 15, 16, 17].

Unfortunately, the ranking schemes developed so far are affected by some evident drawbacks, because very often provide indications either obvious or nonsensical, and try to find a deterministic order in a problem, the long-term evolution of orbital debris in LEO, which would be driven, after several decades at most, by a few tens (over 200 years) of catastrophic collisions, having in practice, from the occurrence point of view, the properties of discrete-time Markov chains [18]. In other words, the debris long-term evolution would be driven by a memoryless random process, i.e. the occurrence of catastrophic collisions, characterized by an extremely wide range of possible cumulative outcomes, in terms of debris number and distribution. In addition to that, the modeling of the long-term debris evolution is affected by large uncertainties in several critical areas [18, 19, 20, 21, 22], so the justification of expensive remediation actions on quite uncertain assumptions and results may still be premature.

Even though the evidence available so far is not reliable enough to support a specific remediation strategy, some basic facts cannot be disputed. Most of the cross-sectional area and mass in orbit (approximately 7000 metric tons) are concentrated in about 4750 abandoned intact objects, i.e. spacecraft and rocket bodies, plus a further 1250 operational spacecraft [23, 24]. In LEO, the abandoned objects and the associated mass are not evenly distributed, but quite often concentrated in relatively narrow altitude-inclination bands, where the probability of catastrophic collision is significantly above the average. This clustering pattern frequently involves a substantial number of identical objects, as upper stages of the same model. For instance, the former Soviet Union, Russia and Ukraine have left in orbit 291 Cosmos-3M second stages, with a mass of 1400 kg, 110 Tsiklon-3 upper stages, with a mass of 1410 kg, and 22 Zenit second stages, with a mass of 8900 kg [23]. The combined mass of these three types of objects alone is around 758 metric tons, representing more than 1/4 of the mass resident in LEO [14, 25].

It seems quite reasonable that any plausible remediation scheme, if and when deemed necessary, should start considering the active removal of abandoned mass (as much and rapidly as possible) from crowded LEO regions [26], and the targeting of very similar objects in general, and of upper stages in particular, would offer a lot of advantages already discussed elsewhere [27, 25]. A very attractive target for active removal would then be represented by the Russian Cosmos-3M second stages, with diameter of 2.4 m and length of 6.5 m, mainly concentrated in four critical altitude-inclination bands: 650-850 km, i = 74 deg; 850-1050 km, i = 83 deg; 900-1000 km, i = 74 deg; and 900-1050 km, i = 66 deg (Figure 1).

Considering the current presence of 291 stages of this kind in orbit, any developed approach, capture and removal techniques or procedures might be used many times over several decades. The moderate size and mass of these upper stages, coupled with their simple and compact shape, would be safer for initial demonstration missions and for routine operations as well. Moreover, it could be possible to operate in at least four separate altitude-inclination critical bands, the reentry risk assessment for de-orbiting (fragmentation analysis) should be needed only for one representative object, and the de-orbiting kits might be tailored for small series production. In addition, multiple rendezvous might be feasible within a single mission,

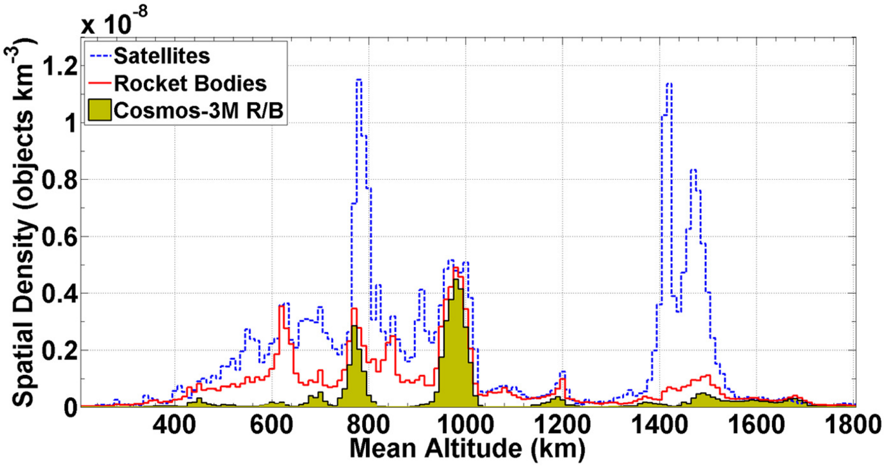


Fig. 1 Spatial density in LEO of intact satellites and rocket bodies. The spent Cosmos-3M second stages account for most of the abandoned rocket bodies in two of the most crowded altitude shells, between 750 and 1050 km.

because, for any given inclination, an average of about two stages would be present in each 5 deg bin of right ascension of the ascending node, with more favorable concentrations around specific orbit planes [14, 25]. Furthermore, the technology kit developed, once proven its feasibility, might be applied to larger abandoned rocket bodies, like Zenit second stages. Last, but not least, the choice of the Cosmos-3M second stages would offer the occasion for a broad and systematic cooperation with Russia, which would be of paramount importance for the knowledge of the rocket body itself, for the eventual availability of launchers at low cost, and to solve a number of legal issues affecting any future active removal mission.

The latter aspect cannot be ignored, because the Liability Convention [28] states that launching states are liable for damages procured in space or on the ground by a space object, like those targeted for active removal. Additional legal issues arise from the Outer Space Treaty [28] and the Registration Convention [28]. Moreover, specific national directives or laws often address the ground safety issues, creating a quite complex web of national and international regulations to be met. Any significant and effective remediation initiative will therefore need a substantial amount of international cooperation.

2 Active Multi-Removal Mission Concept

An active debris removal mission requires a very complex functional spacecraft, capable of performing a number of challenging operations, from rendezvous to debris capture and disposal. The ADR system complexity grows if a multi-removal capability is required, since the above mentioned operations have to be repeated a certain number of times [29]. In this case, indeed, a space system must be capable of reaching, capturing and removing several targets, typically orbiting at different altitudes and inclinations. Nowadays, it does not exist a standard mission outline, because of the presence of different target typologies located in correspondence of several orbital altitudes and inclinations. The most critical aspect of an ADR mission is the target capture, because of the important challenge represented by the docking with a large non-cooperative object, whose attitude might be unknown, requiring a safe relative rotation reduction [30, 31, 32, 25]. In this respect, different technological solutions are under investigation. However, assuming the capability of target capturing, its consecutive disposal plays the main role. In regards to the body size of the considered objects, instead of their transferring to a 25-year residual lifetime orbit or to a higher graveyard orbit outside LEO, a controlled atmospheric reentry toward a non-inhabited region might become a fundamental requirement in next future [30, 32, 33, 14, 25]. In fact, the latter solution allows for the control of the impact footprint and location of the object's fragments, providing a fast disposal and significantly reducing the collision probability that affects long-term disposal approaches.

To reduce the high cost and risk related to the design and development of a such complex system, as well as its operation control, a possible solution regards the use of the launcher's upper stage to place in orbit a certain number of de-orbiting kits (DeoKits), each one removing a single large abandoned object [34]. The DeoKit shall be able to perform the rendezvous with the selected target and, being equipped with an automated device, the safe debris capture and mating, as well as the target safe de-orbiting and reentry into the Earth atmosphere, by means of a suitable primary propulsion module [25, 14, 34]. Among the space launchers owned by the European Space Agency (ESA), concerning the ADR mission concept here analyzed, the upper stage of Soyuz (Fregat) might represent the best candidate for DeoKits carrying and distribution, once verified the ADR economical feasibility. Specifically, Fregat can be restarted up to 20 times and has a total ΔV capability of about 4.7 km/s [35]. This provides the possibility of performing multiple removals with a single launch, using Fregat propulsion to deploy each DeoKit close to the debris to be removed. Of course, once the candidate targets have been identified, the removal sequence has to be optimized based on Fregat capability in terms of total velocity increment, as well as number and mass of the de-orbiting kits embarked on board. The multi-removal mission under analysis involves several steps and critical aspects. Apart from setting the best rendezvous sequence, an effective mid-range rendezvous maneuver is required for each DeoKit, exploiting its primary propulsion module, to reach the selected debris, after separation from Fregat. Once in the vicinity of the target, a secondary propulsion system made by a Reaction Control System (RCS) must allow for close-proximity operations [30, 14], as well as object capturing. In addition, in order to perform the debris disposal, the DeoKit must be rigidly and reliably connected to the target external structure with a mechanism such as, for example, the one proposed in [29, 27, 14]. With reference to Figure 2, the different steps of a single removal phase can be summarized as follows:

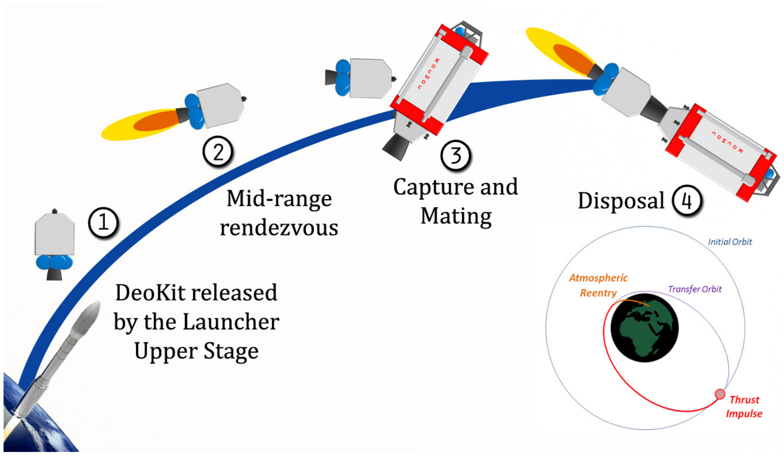


Fig. 2 ADR mission conceptual scheme: removal of a Cosmos-3M second stage.

1. the DeoKit is released by the upper stage at a prefixed distance from the first target in the same orbit plane;
2. the DeoKit uses its own propulsion unit to perform terminal rendezvous;
3. once reached the object, the DeoKit proceed with capture and mating to rigidly connect itself with the debris, by means of specific mechanisms [14, 25];
4. the propulsion module of the DeoKit is remotely ignited to perform the target disposal: target and de-orbiting kit are de-orbited in the mated configuration, after a phasing time needed to aim at the selected zone of the Pacific Ocean.

Once the first DeoKit has been released, the upper stage moves towards the next target on a different orbit and operations from 1 to 4 are carried out by a second DeoKit. After the release of the last de-orbiting kit, Fregat is de-orbited as well. In order to perform preliminary analyses, a demonstrative two-removal mission is investigated considering the removal of two Cosmos-3M second stages, namely Cosmos-3M 11112 and Cosmos-3M 22676, located at an average altitude of 767.62 km and 777.97 km, respectively, and 74 deg inclination. To this end, it is assumed that the removal system is injected in a 700 km circular orbit in the same plane of the first target. Considering the rendezvous strategy and the assumptions introduced in [25], the two rendezvous maneuvers would require a total velocity increment of about 100 m/s. The additional ΔV needed to nullify the right ascension of the ascending node (RAAN) differences can be estimated in about 600 m/s, considering that, typically, an average of about two stages would be present in each 5-deg bin of RAAN. Thus, the total velocity increment required for each removal would be lower than 700 m/s. Considering Fregat total ΔV capacity, this would allow the removal of more than 5 objects within a single mission. On the contrary, if we limit the multi-removal mission to 5 objects, as suggested by the debris population evolution scenarios, the ΔV which could be used for each rendezvous maneuver would be slightly lower than 1000 m/s, thus allowing to nullify RAAN differences up to about 7°-8°. Nevertheless, it is worth outlining that the actual required velocity increment for each removal depends on the strategy being used for the multi-removal mission, which is not described in details, being it beyond the aim of this preliminary analysis.

2.1 De-orbiting Phase Design

Once the DeoKit has been rigidly connected to the selected debris, it is ignited to start the controlled de-orbiting and atmospheric reentry. For the disposal, both single apogee burn and multi-burn strategies could be envisaged, even though the large size of the reentering objects suggests to limit the number of burns to allow an immediate reentry, so to relax the attitude control requirements of the mated configuration. Indeed, below 300 km the atmospheric torque can significantly affect the controlled reentry maneuver [36]. On the other hand, a multi-burn approach might be considered when the abandoned object orbits at high altitudes in order to achieve a better control of the conditions for the final atmospheric reentry and impact footprint [32]. Due to the large size and mass of the reentering objects, the destruction process in the atmosphere could be incomplete, with a high residual risk of ground impact. Hence, the reentry shall be controlled and directed to a specific location on Earth (South Pacific or Atlantic Ocean). With reference to previous studies on LEO de-orbiting strategies [36], a disposal strategy is pursued in which the mated configuration (DeoKit-debris) is steered to an elliptical transfer orbit with a perigee enough low to allow for an immediate atmospheric capture. In addition, to limit the ground impact area of fragments surviving the atmospheric phase, a sufficiently steep Flight Path Angle (FPA) is required. In particular, a preliminary, non-optimized, reentry trajectory analysis is performed in which an elliptical disposal orbit with a perigee below 60 km and FPA < -1.5 deg at 120 km is considered.

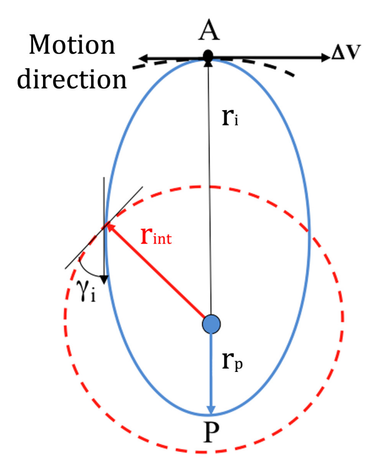


Fig. 3 De-orbiting maneuver schematic

Figure 3 illustrates the de-orbiting strategy, where A is the apogee of the disposal ellipse and P is its perigee, whereas the red circle represents the reentry interface located at 120 km altitude. Under keplerian orbit assumption, the Eq. 1 relating the perigee radius r_p to the flight path angle at the reentry interface, γ_i, is found:

$$r_p = \frac{(r_i - r_{int}) r_{int} cos^2 \gamma_i}{r_i - r_{int} cos^2 \gamma_i} \tag{1}$$

where r_{int} is the radius corresponding to an interface of 120 km altitude, while r_i is the radius of target's orbit. From Eq. 1, the perigee radius corresponding to a given FPA (or vice versa) is computed, and then, the ΔV needed for de-orbiting can be obtained. This study is focused on the comparison of different chemical propulsion technologies, as well as their respective motor performance and preliminary mass budget. From the point of view of multi-removal missions, small and compact propulsion modules are preferred, in order to maximize the number of DeoKits (i.e. of removed objects) carried on orbit by a single launch. Once released by the launcher upper stage, each DeoKit is an autonomous spacecraft composed by a primary propulsion module and a service platform, here called ADR platform.

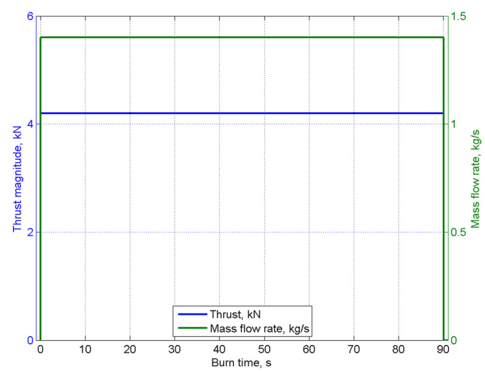


Fig. 4 Average thrust (green) and propellant mass flow rate (blue) needed to remove 1400 kg by means of a DeoKit of 566 kg, satisfying the constraints on FPA and perigee altitude

The latter includes the needed avionics, instrumentation and sensors for mission operations, the thermal system, batteries for power supply, the capture and mating mechanisms, as well as a secondary propulsion system (RCS) for the spacecraft attitude control. With the purpose of propulsion modules comparison, the same target and disposal profile are considered for all engine types. More specifically, the object to be removed is a Cosmos-3M second stage at an altitude of 770 km, requiring a ΔV of about 200 m/s. The reentry phase is simulated using a 3-deegres-of-freedom (DOF) standard dynamics model taking into account longitudinal motion only [37], spherical gravity and US76 Standard Atmosphere. A simplified hypersonic aerodynamic model of the Cosmos-3M mated to the DeoKit is used, based on assuming purely ballistic reentry and a Mach-independent drag coefficient, yielding a ballistic coefficient of about 100 kg/m^2.

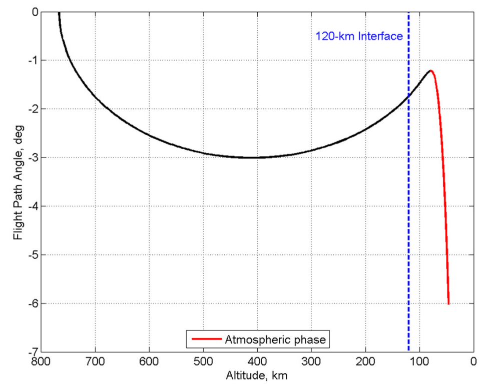


Fig. 5 Flight path angle change during the disposal maneuver

The disposal simulation is performed by a single apogee burn, assuming a DeoKit mass of 566 kg estimated in previous work [34]; in this manner, a FPA of -1.74 deg at 120 km and a perigee altitude of about 54.7 km are achieved with a motor thrust of 4.2 kN for 90 s of combustion time (keeping the average acceleration level within 0.4 g). Figures 4 and 5 show, respectively, the average thrust and propellant mass flow rate required to satisfy the mission constraints and the FPA variation during the disposal flight of the mated configuration (DeoKit-debris).

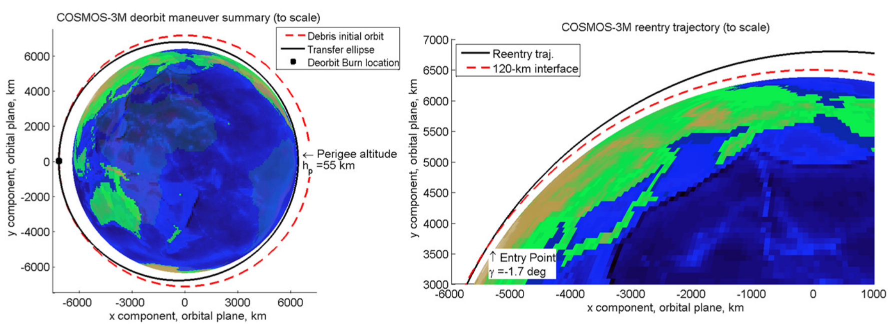


Fig. 6 Disposal trajectory summary: elliptical transfer orbit with FPA of -1.74 deg at 120 km and perigee altitude of 54.7 km

In Figure 6, one can see the disposal trajectory. The propulsion module mass budget resulting from the preliminary design will be expressed as a percentage of total DeoKit mass (assumed as fixed value), thus identifying the amount of load, assigned to the ADR platform, which the engine can carry in addition to the target mass (1400 kg). According to this preliminary estimation, the Fregat would be able to carry 5 DeoKits (566 kg each one), for a total mass of 2830 kg, preserving about 2000 kg on the total payload capacity to LEO, confirming the Soyuz as possible candidate for a multi-removal demonstrative mission. However, the actual number of DeoKits carried in orbit shall be checked against the volume available within the

Soyuz fairing, the target orbits and a more detailed mass budget including also the dispenser, as well as the possible presence of a satellite.

3 Propulsion Module

The target size, the disposal strategy and the propulsion technology are important aspects with a strong impact on mass budget, system volume, and cost of the propulsion unit. Considering a large object, the capability of throttling and reignition may represent a stringent requirement for the adequate control of the rendezvous and disposal maneuvers, whereas compact design is important for easier docking to the target and for dynamic stability of the final assembly. A compact volume may request a higher average propellant density but may collide with the ΔV requirements for a controlled atmospheric reentry, needed for the LEO objects in the highest altitude. Thrust level should stem from a trade-off choice about the risk of debris fragmentation, especially for large spacecraft, and long mission duration (correlated to propellant storability and collision risk during the maneuver). In the frame of chemical propulsion, solid propellants represent a simple, reliable, and proven technology with good specific impulses, but feature limited flexibility and not suitability for multi-burn missions, whereas liquid bi-propellants fill the gaps left by the solid systems, but larger volumes and higher degree of complexity are requested. In fact, the walls of combustion chamber and nozzle require to be cooled to sustain the aggressive combustion environment with which they are in direct contact, so high resistance and more expensive materials are needed; also the injection and feeding systems require a complex design, thus involving a quite high level of costs for liquid propulsion technology. Furthermore, the propellants handling must be carefully considered, due to the high toxicity of typical liquid substances used for space applications (NTO, MMH, UDMH, etc.). In this respect, a great interest is oriented to hybrid rocket technology, due to the high specific impulse achievable, intrinsic safety, possibility of green propellant use, low cost technology and, especially, reignition and thrust throttleability. Among the latter, also owned by liquid propulsion technology, the first allows for multi-burn disposal, which is highly recommended when the target orbits at high altitudes, in order to favor a better reentry trajectory selection; while the second might represent a key aspect to avoid the risk of fragmentation for the most fragile components of a large abandoned satellite during the de-orbiting maneuver.

3.1 Hybrid Propulsion Module

A hybrid rocket motor typically features the oxidizer in the liquid or gaseous state, while the fuel is in the solid state. Its safety is guaranteed by no-contact between fuel and oxidizer, except during the combustion phase. A hybrid rocket can also be built with particular geometries, by means of tangentially oxidizer injection, resulting very compact in size and highly efficient [38, 39]. Such characteristics could be the right solution for space debris mitigation, by supplementing new satellites with this engine type in the next future. Clearly, considering long time space missions, the solid fuel selection must account for materials characterized by negligible properties shifting because of aging. This technology seems very promising even in the field of space debris remediation, making possible the active removal in LEO of large intact objects (several metric tons), both spacecraft and rocket bodies, by using a chaser vehicle equipped with a Hybrid Propulsion Module (HPM) for the controlled reentry maneuver and several micro thrusters, for the attitude control, spilling directly the HPM liquid oxidizer to be used as a monopropellant (dual-mode use) [40, 27].

Table 1 Conceptual comparison between different features of chemical propulsion systems

	Solid Propulsion	Liquid Propulsion	Hybrid Propulsion
Toxicity	Reduced	High	No
Performance I_s [s]	250-310	300-500	250-340
Safety	Low	Intermediate[a]	High[b]
Complexity-Cost	Low	High	Reduced
Throttleability	No	Limited	Yes[c]
Reignition	No	Yes	Yes
TRL	9	9	7

[a] explosion hazard due to leak of both fuel and oxidizer at gaseous phase
[b] only the oxidizer presents hazard of leaks at gaseous phase, fuel is solid and inert
[c] the effective throttleability range of hybrid rocket is still under investigation among researchers

Overall, a hybrid motor represents a solution that mediates benefits and drawbacks from both liquid and solid rocket technology. In Table 1, a conceptual comparison between the three different propulsion systems is resumed. On one side, it is bestowed the throttleability and reignition capability typical of liquids, specific impulse levels that fall in between the performance of solid and liquid propulsion, and a higher mean propellant density due to the use of a solid fuel. Nevertheless, a technological gap exists due to its late development (i.e low technology readiness levels) and lack of in-orbit demonstration. In the simplest possible configuration, a hybrid rocket is made by a central-perforated solid fuel placed in the combustion chamber where an injector blows liquid/gaseous oxidizer. This grain configuration shows a quite high volumetric efficiency ($\sim$80%) for fuel amounts below 800 kg, with length-to-diameter L/D smaller than 20 [41], that would be the case of ADR missions. The main drawbacks of hybrid technology are, besides low fuel regression rate, poor combustion efficiency, hence unburned products, and oxidizer-to-fuel ratio (O/F) shifting with consecutive specific impulses losses. However, different means are considered, especially for the enhancement of mass burning rate, spanning from the use of advanced additives to different injection approaches (swirling oxidizer and vortex combustion) [38, 39, 42, 43]. In this respect, innovative designs of the combustion chamber, such as vortex pancake, provide higher combustion

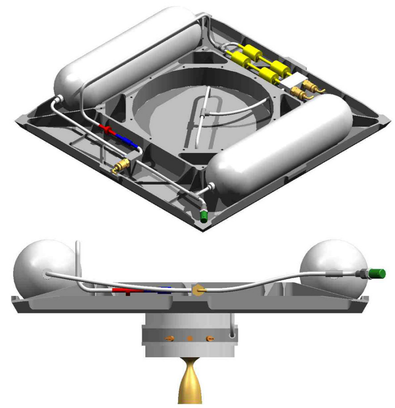


Fig. 7 hybrid motor alternative geometry configuration suggested by Gibbon et al. for satellites orbital transfer [46]

efficiency, low performance variation during combustion, and, in the case of solid metal additives, reduced emission of condensed combustion products (CCPs) thanks to the vortex combustion [38]. The vortex pancake configuration might represent a very interesting solution for Post-Mission Disposal (PMD) maneuvers [44, 45, 46]: this small and compact hybrid rocket could be easily integrated in the design of new satellites (see Figure 7), providing both the maneuver capability and the final disposal to a 25-year residual lifetime orbit, or a direct atmospheric reentry.

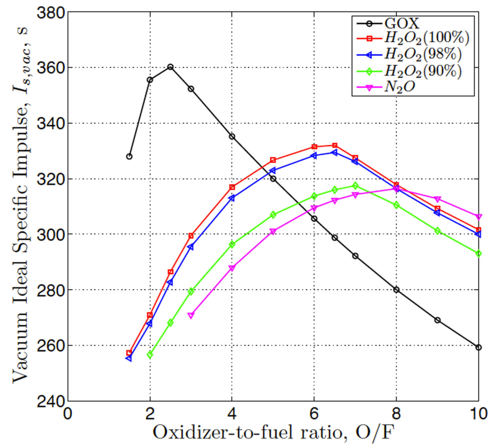
Therefore, cheap technological solutions might allow for a more easy approach to ADR by the international space community. Hybrid propulsion could help in this direction; in fact, once reached its complete maturity, the costs of hybrid motors might become very low, due to cheap propellants and construction materials, as well as technological solutions required. For example, aluminum can be used for the combustion chamber, which does not need a cooling system for walls, since they are protected by the solid fuel, which is itself an insulating material. The exhaust nozzle can be realized without a cooling system too, if the burn times are small ($90 \div 100$ s) [41]. Moreover, the injection system does not need a complex design, as well as the tanks and feed system, depending on the oxidizer selected. The ignition, if allowed by the oxidizer, can be performed by catalyst reaction, avoiding the need of a pyrotechnic ignition system (instead required for solid motors). Overall, this so reduced complexity could provide an important decrease in design and manufacturing costs, making hybrid motors more suitable for controlled reentry than liquid motors, in the perspective of ADR missions.

3.1.1 Propellant Selection

A great advantage of hybrid propulsion is the use of non-toxic propellants, that are also significantly cheaper than common substances used in solid and liquid propulsion. Typical hybrid propellants almost do not exhibit explosion hazards since the oxidizer and the fuel remain separated during all manufacture, storage and transport operations. This level of safety, together with their non-toxicity, reduces the overall costs of all ground operations before the mission. Different solid materials are considered as solid fuel, especially within the family of hydrocarbons [39]. During the years, the attention of research studies was mainly focused on carbon-based polymers and paraffin wax materials, depending on their costs, mechanical properties and combustion performances. In the polymers group, typical fuels are, Polyethylene (PE), Polymethylmethacrylate (PMM) and Polybutadiene (PB) with hydroxyl or carboxyl ion as chain terminators. These materials are quite cheap if compared with typically employed liquid propellants [41]. Moreover, cracks and voids in solid fuel are not so critical as in solid propellants, thus reducing the need of industrial-level high quality control and assurance, in term of both manufacture and final product inspection. This aspect also contributes to reduce costs. The Hydroxyl-Terminated Polybutadiene (HTPB) is the most popular and well-known solid fuel for hybrid propulsion, also due to its large use in solid propulsion as binder. It is a rubber compound very safe to handle, allowing to be easily powered with metal additives, maintaining very good mechanical properties. HTPB shows an isotropic behavior during the combustion [47], resulting very suitable for multiple burns missions. Although the regression rates provided are not so high with respect to paraffin-based fuels, HTPB is a quite energetic and very safe material, since if soaked in liquid oxygen it is not explosive [41]. Metal additives allow to reach greater regression rate levels, also providing a little increase in fuel density, which, for pure cured rubber, is about 915 kg/m^3 [41].

Compared to HTPB, PE and PMM are cheaper, but the latter, for large grains, can be subjected to cracks due to heat loads during the combustion process [41], introducing the hazard of nozzle throat obstruction. Because of this, PMM might results expensive, owing to the need of X-ray computed tomography and ultra-sonic inspections for large size grains. Returning to HTPB, for example, not cured polymer can cost about 10 \$/kg, depending on the supplier and purchased amount. Considering that for an ADR mission a relatively small quantity of fuel is required, due to the low ΔV involved, a solid grain charge would not have a strong impact on overall mission costs. On the other hand, paraffin wax materials are recently acquiring large interest in the research community, due to the higher regression rate provided, up to two or three times that of HTPB [47]. Nevertheless, paraffin-based fuels display anisotropic combustion and poor mechanical properties, the latter requiring the addition of polymeric additives in order to increase the elastic modulus and avoid the risk of cracks due to strong thermal and fluid dynamic stresses [47, 39]. In the frame of a multi-burn mission, the anisotropic behavior might represent a significant drawback, since the shape and conditions of the fuel perforation after the firing would not be easily predictable (as regards the oxidizer mass flux estimation for the following burn). In the light of these considerations, HTPB seems to be the most suitable choice for ADR applications, being a well known and safe material, tested in different operating conditions, able to provide quite high performance, with ideal vacuum specific impulses $I_{s,vac}$ close to that of paraffin wax [27]. However, paraffin materials, if fitted out with additives for mechanical properties improvement, in the

Fig. 8 Ideal vacuum specific impulse for HTPB with different oxidizers (chamber pressure 3.0 MPa and nozzle expansion area-ratio 50) [34]



case of single-burn mission might represent an interesting solution, especially for small PMD on-board engines.

The most interesting oxidizers are gaseous or liquid oxygen (GOX or LOX), nitrous oxide (N_2O) and hydrogen peroxide (H_2O_2) at high concentration, also known as High Tested Peroxide (HTP) [41, 39, 48, 44, 38]. Their performance, in term of $I_{s,vac}$, are compared in Figure 8, estimated using the Chemical Equilibrium for Applications (CEA) software [49, 50], assuming 3.0 MPa of chamber pressure, an area-ratio of 50 and shifting nozzle expansion. HTPB is selected as solid fuel and introduced in the software by means of an empirical chemical formula evaluated in SPLab [34]. All these oxidizers provide ideal $I_{s,vac}$ above 300 s in correspondence of their optimal O/F, representing a valid option for rocket applications. Of course, liquid oxidizers with higher density are preferred for system volume constraints. However, before entering in contact with the fuel port, the oxidizer must be properly vaporized, in order to limit combustion inefficiencies, not regular usage at the head section of the solid grain and the annihilation of the tangential velocity component in case of swirling injection [41, 51]. Therefore, LOX requires a separated gas generator to provide hot gases in the pre-combustion chamber, and, being a cryogenic substance, a more complex and expensive system is necessary for its storage. These features also entail a significant addition of mass on the overall propulsion system.

On the contrary, the hydrogen peroxide can be easily vaporized by means of a catalytic decomposition, so injecting in the combustion chamber gaseous O_2 and H_2O at temperatures up to 1000 K, depending on HTP concentration and reaction efficiency. In this case costs and complexity can be strongly reduced, just requiring a catalyst system for the oxidizer injection. The temperature of the produced gaseous mixture is enough for the HTPB ignition ($\approx$ 800 K) [27], avoiding the need of a complex ignition system. Hydrogen peroxide is a well-known substance used for different applications in commercial, aerospace and defense industries during the last 100 years. In the 1930s, its decomposition reaction was exploited to develop the first monopropellant system [52, 40, 48]. A very low vapor pressure characterizes the hydrogen peroxide, making it easier to handle with respect to other liquid oxidizers or monopropellants, such as LOX and the toxic group made by nitrogen tetroxide (N_2O_4) or NTO, Monomethyl-hydrazine (MMH) and Unsymmetrical dimethyl-hydrazine (UDMH).

A toxic propellant involves higher costs, especially for handling and ground operations: specialized staff and plants are required for the safe management of toxic substances and their price is subjected to environmental laws, that will tend to become more strict in the next future (with consequent costs growth). In fact, by way of example, the average price of MMH and UDMH during the 1990s was respectively about 17 \$/kg and 24 \$/kg, but later, due to the upgrade of environmental regulation, the price of MMH jumped to 170 \$/kg [53, 54]. The non-toxicity of hydrogen peroxide together with its easy handling, as well as its large diffusion, reduced the average price of HTP (with concentrations of 90-95%) down to about 1 \$/kg, during the 1990s. Although the propellant cost represents a very small portion of the economical effort required by an ADR mission, a so large difference in costs and handling, together with a less complex propellants legislation, makes the hydrogen peroxide, as well as hybrid propulsion, a very suitable option for new commercial companies that decide to deal with the development of propulsion systems for small satellites orbital transfer and post-mission disposal. For rocket applications, the high concentration of H_2O_2 is blended with water; typically at 90%, the most common grade, but even up to 98%. The latter provides higher mixture gas temperatures and better performance, but its price becomes greater and the significant change in the adiabatic decomposition temperature involves a more complex design of the catalyst bed and more particular materials, with respect to typical systems used for HTP(90%) [40]. Because of the relatively low temperatures, the catalyst system (chamber, pipes, etc.) can be implemented with stainless steel, thus keeping low the costs, while the catalyst bed is generally made by silver. The oxidizer density is about 1390 kg/m^3 for HTP(90%). The change in concentration entails the change of other properties, such as the freezing point which, for a percentage of 90%, is about 261.77 K (-11.5°C) [52]. Therefore, speaking of missions in space, a thermal system for the control of the oxidizer temperature must be considered. Despite this, hydrogen peroxide would represent a key choice for hybrid propulsion units designed for ADR, also because of its dual-use capability. The latter consists in the possibility to use HTP as oxidizer for the primary HPM and as monopropellant for the secondary propulsion system, made by several micro-thrusters for attitude control, spilling the oxidizer directly from the main tank [25, 27, 40]. Historically, the hydrogen peroxide became famous as a hazardous substance due to some incidents mainly happened between the 1930s and the 1960s, when the industrial practices for handling and rocket development were still immature and characterized by incidents even with other substances. In more recent years, two notable incidents, described and discussed in [55], enhanced the negative opinion about hydrogen peroxide, but they were provoked by the use of incompatible materials and system-design failures. Despite this, H_2O_2 with concentrations below 90%, under the correct precautions, is enough stable both for what concern typical industrial uses and space missions, as demonstrated by the COMSAT spacecraft with a stored time of 17 years in vented tanks [56]. Modern industrial techniques are able to produce high hydrogen peroxide quality, with very low presence of impurities, making it safer both for handling and for storing. Research studies have been carried out with the aim to improve the propellant properties especially for space applications; for example, the natural decomposition rate of hydrogen peroxide can be significantly reduced if stored in tanks made by specific materials, thus improving its long-term stability [55]. In conclusion, with regards to active debris removal, hydrogen peroxide with concentration up to 90% might represent a good candidate oxidizer together with HTPB solid fuel, despite the lower specific impulse achievable with respect to LOX. The latter, besides greater system complexity and costs, can unlikely be stored for long times, limiting the possibility of its employment for ADR missions, that can last several months.

3.2 Preliminary Design and Mass Budget

The preliminary mass budget of a DeoKit for the active removal of a Cosmos-3M second stage is carried out. Besides the ADR platform, made by avionics, communication, measuring and support systems, as well as the soft and hard docking mechanisms [27, 34], the principal component is the propulsion module. Different configurations can be designed depending on the rocket technology selected: solid (SPM), liquid (LPM) or HPM. In Figure 9, one can see possible conceptual DeoKit designs with respect to various motor solutions. In case of a HPM powered by HTPB+HTP(90%), the propellant for RCS is spilled from oxidizer tanks. Moreover, the high L/D ratio of HPM solid fuel requires arranging the engine within the ADR platform volume. On the contrary, the LPM, due to its compact combustion chamber, can be attached on the aft section of the service platform. For the SPM, a specific integration should be performed, since its size strongly depends on the solid grain geometry. Furthermore, an additional propulsion unit for the initial mid-range rendezvous is required, due to the no-reignition capability. In this case, depending on the required thrust, a small solid rocket or a more powerful RCS system might be exploited.

The mass budgets of liquid and solid propulsion modules, regarding the same mission requirements and ADR platform, are finally compared with the hybrid solution. Of course, reduced motor masses (i.e. lighter DeoKits) would favor multi-removal mission strategies, performed with a single launch. The mass and size criteria of the different components assembled in the propulsion modules, here considered, are based on reference design concepts for rocket motors [41].

For the hybrid propulsion module, a more complex calculation method was carried out in order to estimate the fuel regression rate for the couple HTPB+H_2O_2, of which poor data are available in the literature survey. A computational tool was implemented for internal ballistics analysis and engine sizing based on the approach suggested by Funami and Shimada [57]. The calculation domain consists of the

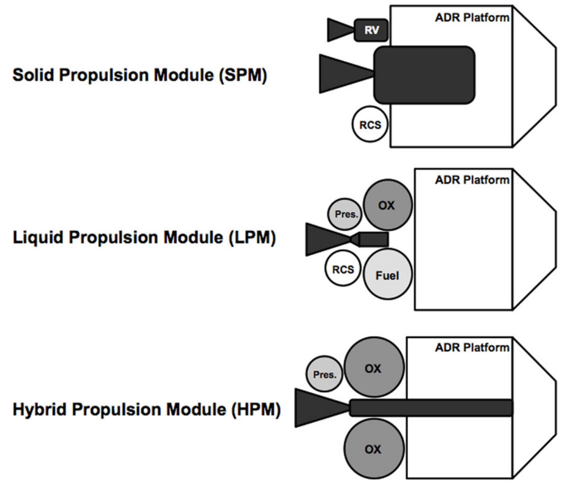


Fig. 9 conceptual DeoKit configuration with different chemical propulsion modules

solid fuel cylindrical perforation and a de Laval nozzle. The flowfield is described by quasi-one-dimensional non-viscous compressible Euler equations with the addition of mixture fraction conservation. The mass release from the solid fuel surface is considered as a source term, in which the fuel mass flux is evaluated with Eq. 2, the Marxman model, assuming pure convective regime [58, 59]

$$\rho_f r_f = 0.03G\left(\frac{G\cdot x}{\mu}\right)^{-0.2}\left(\frac{St}{St_o}\right)\left(\frac{u_e}{u_{fl}}\right)\frac{\Delta h}{\Delta H_{v,eff}} \qquad (2)$$

where G is the local specific mass flux, μ is the gas-phase viscosity, x is the axial location, St/St_o is the ratio between the Stanton number with and without fuel suction, u_e is the main stream velocity, u_{fl} is the velocity at the flame and Δh is the difference between the enthalpy at the flame and at the wall of the gas phase. $\Delta H_{v,eff}$ is the HTPB vaporization enthalpy. Oxidizer and fuel are considered at gaseous phase, with instant mixing and reacting inside each finite volume [57]. The vaporized HTPB is represented by gaseous butadiene monomers (C_4H_6), whereas the hydrogen peroxide enters in the combustion chamber already decomposed in oxygen and steam. For the combustion, chemical equilibrium is applied to a reduced number of selected chemical species [49, 50, 57] in order to obtain thermochemical and transport properties of the gaseous mixture, required in Eq. 2. A detailed description of the computational method is available in [57, 34]. The calculated time- and space-averaged regression rate was compared with the experimental results of Shanks and Hudson [60] for the couple HTPB+GOX [34]. With this computational tool, the Marxman et al. model underestimates the regression rate of about 16% at high mass flux and up to 50% at low mass flux. This lack between numerical solution and experimental data can lay at both the use of a pure convective energy flux balance at the fuel surface and the effects of injector geometry on the flow path of the experimental motor [39, 61]. The second can significantly affect the flame development within the turbulent boundary layer, producing regression rate trends quite different from that predicted by the classic theory. However, despite its limitations, this numerical approach provides regression rates of realistic order of magnitude, resulting useful for propellant consumption estimation, preliminary rocket investigations and design choices. Moreover, the underestimation of the regression rate plays a conservative role on the preliminary sizing. Motor performance is evaluated assuming Bray expansion [62], two-dimensional nozzle losses and throat erosion rate; the latter based on numerical results of Bianchi and Nasuti for the couple HTPB+H_2O_2 with a graphite nozzle [63]. The code assumed not diluted hydrogen peroxide, so the final performance are corrected with respect to the effective H_2O_2 concentration, by means of correction factors estimated with NASA CEA software [49, 50]. For the combustion chamber an aluminum alloy is assumed, whereas phenolic material, due to its higher resistance than graphite-based, is considered for the nozzle. The latter is not cooled, so short burning times ($\sim$100 s) must be considered. The liquid oxidizer is stored in 4 spherical aluminum tanks, pressurized, by means of a pressure-regulated system, with gaseous nitrogen loaded in a spherical titanium tank. The catalyst system for motor ignition is designed assuming the validity ranges tested for HTP(90%) by Ventura [40], with silver-based catalyst beds.

Concerning the preliminary design of LPM, the motor performance, for the typically employed propellant couple NTO+UDMH, are evaluated with NASA CEA software (considering Bray expansion and two-dimensional nozzle losses) assuming a constant O/F of 2.2 during combustion. The mass budget of the overall system is estimated using semi-empirical design relations, based on historical motors database, as suggested in [41]. With regard to an ADR mission and the strong need of cost reduction, the combustion chamber and the nozzle are assumed to be protected by an estimated amount of ablative material. The liquid propellants are stored in 4 spherical aluminum tanks, pressurized, by means of a pressure-regulated system, with gaseous nitrogen loaded in a spherical titanium tank. The combustion chamber and nozzle are made by columbium, typically used for these applications.

For SPM, the propellant consumption during combustion is estimated by means of an experimental burning rate law (Eq. 3), evaluated at SPLab [64] for the solid formulation AP(68%)+HTPB(14%)+ μAl(18%)[1]

$$r_b = 1.08 \pm 0.03 \cdot p_c^{0.46\pm 0.01} \qquad R^2 = 0.996 \tag{3}$$

where p_c is the combustion pressure in bar and r_b is expressed in mm/s. This solid propellant has a theoretical density of 1.761 kg/m^3. The motor size and mass are estimated assuming a graphite cylindrical chamber (L/D $\sim$ 1.5) with a spherical dome and De Laval nozzle. The mass of the latter and of the other components, such as igniter and internal insulation, are evaluated with empirical relations and mass ratios based on historical solid motors database [41, 65]. Just as for LPM, motor performance evaluation is carried out with NASA CEA software.

For all designed motors, a nozzle area-expansion ratio of 80 is imposed. A chamber pressure of 8.0 MPa is assumed for SPM, while a pressure of 1.7 MPa is con-

[1] Aluminum powder with average diameter of 30 microns.

sidered for HPM and LPM. To the latter, an increase of 40% on the inert mass is applied, to account for structure connection elements, bosses, valves, cables, etc. Concerning the SPM, being characterized by a simpler structure without tanks and feed system, an increase of 10% is considered [41].

Each propulsion unit is able to provide about 4.2 kN of thrust for 90 s of combustion, necessary to perform the disposal of a Cosmos-3M second stage from an altitude of 770 km, by means of a DeoKit with a total mass of 566 kg (see Section 2.1). Such maneuver requires a ΔV of 200 m/s, to which a 10% is added as safety margin to account for performance losses. Further 15 m/s are assumed, for HPM and LPM, to account of the propellant mass useful for a mid-range rendezvous with the target. The total velocity increment of 235 m/s (220 m/s for SPM) is provided by each motor with a single burn. In Table 2, the preliminary mass budget results

Table 2 Preliminary design results for different chemical propulsion modules

	Solid Propulsion	Liquid Propulsion	Hybrid Propulsion
M_{PM} [kg]	153.6	188.0	216.5
$\%^{PM}_{deokit}$	**27.1%**	**33.2%**	**38.3%**
M_{ADRpt} [kg]	412.4	378.0	349.5
$\%^{ADRpt}_{deokit}$	**72.9%**	**66.8%**	**61.7%**
M_{prop} [kg]	141.2[a]	141.1	150.2
average $I_{s,vac}$ [s]	301.4	328.1	300.2

M_{PM} propulsion module mass, M_{ADRpt} ADR platform mass, M_{prop} propellant mass
[a] the propellant amount of SPM is based on the disposal maneuver only

are summarized. The mass percentages of propulsion module and ADR platform are evaluated with respect to the DeoKit mass (566 kg), for each motor type, and compared in Figure 10.

In the case of HPM, the disposal flight simulation was recalculated with motor's effective performance, since hybrids are characterized by O/F shifting (increase) and consequently not constant thrust (decrease) during the combustion. A maximum theoretical I_s of 312 s is achieved with a mixture ratio of 6.5, which, during the 90 s of combustion, grows from 5 to 9 yielding, respectively, 308 s and 294 s of specific impulse. The produced thrust almost linearly decreases from 4.5 kN, at the beginning of combustion, to 4.0 kN after 90 s. The chamber pressure decreases from 1.7 MPa to 1.3 MPa, while the gaseous mixture temperature also drops from an initial value of 2750 K. Low pressure and temperature limit the throat erosion rate, in this instance estimated as 5 mm of diameter enhancement at the end of combustion. Hydrogen peroxide allows for lower combustion temperatures with respect to gaseous or liquid oxygen, producing a lesser nozzle throat erosion [63] and related performance losses. At this chamber conditions, the inlet oxidizer mass flow rate is imposed to achieve an initial mass flux (G_{ox}) of about 600 kg/(m$^2 \cdot$ s), thus estimating a space-averaged regression rate of about 1 mm/s, which, accordingly to Marxman model, decreases with the G_{ox} drop, due to the growth of fuel port diameter during the combustion. As previously asserted, the calculated regression

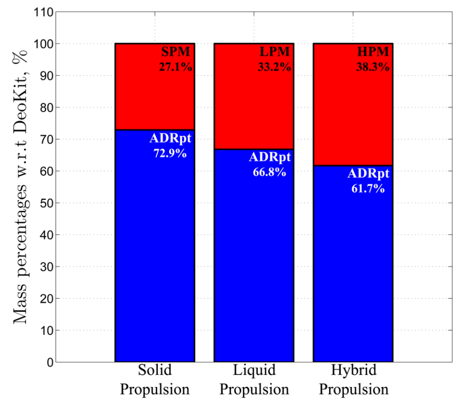


Fig. 10 propulsion modules mass percentage with respect to DeoKit total mass

rate shows a certain level of underestimation, but this plays a conservative role in the HPM preliminary design. The HPM allows to achieve a FPA of -1.77 deg at an interface of 120 km and a perigee altitude of 52.3 km, satisfying the mission requirements. With regards to solid propulsion mass budget, it is important to highlight that the SPM is designed only for the disposal maneuver, hence an additional propulsion unit (monopropellant, hybrid or solid) must be accounted for the mid-range rendezvous between the DeoKit and the target.

Despite this, solid rocket solutions can provide very compact motors, of which 92% of their total mass is the propellant. These results suggest that solid propulsion might be exploited with a different mission approach: a chaser spacecraft could carry few SPMs to be attached (if the feasibility will be demonstrated) to different targets. Each solid motor should be equipped with the instrumentation for the disposal control and the mating system for the joint with the target. The chaser should be a complex spacecraft able to perform multiple precise rendezvous, so loaded with a great amount of propellant, also required for its final disposal (once released all SPMs). The chaser might be powered by a liquid or hybrid rocket, or an existing launcher upper stage could be directly modified and equipped with the ADR platform, as well as a certain number of SPMs. This alternative solution should be explored, in contrast with the approach of multiple autonomous DeoKits described in Section 2. Besides the ADR possibility, solid motors represent a valuable option for the post-mission disposal of new satellites (boarded engine).

Focusing on liquid and hybrid systems, the HPM results heavier than LPM, of about 28.5 kg, corresponding to a difference of 5% with respect to the DeoKit mass. At this preliminary design phase, the two technologies provide modules of comparable mass and also nearly equal width (~ 90 cm). They mainly differ in combustion chamber length, which for HPM results significantly longer, with a solid fuel grain of 1.6 m. In the light of these results, the selection of the best propulsion option for ADR missions deals with the most favorable features provided by each technology. Liquid propulsion might be preferred for disposal strategies based on low thrust with one long burn times, that also provide a further mass saving, according to the preliminary design approach described in this chapter: a thrust of 1.0 kN provided for 445 s could satisfy the disposal requirements with a LPM of about 177 kg (31.2% of the total DeoKit mass). Nevertheless, this mass reduction could be counterbalanced by the addition of a more efficient and complex cooling system, required for long combustion times, as well as its correlated greater design and realization costs. Despite this, a long burn strategy with reduced thrust levels entails low accelerations imparted to the target structure, decreasing the breaking risk for the target's deployed appendages and structures [25]. On the contrary, abandoned rocket bodies, due to the aim of their use, are able to sustain strong axial accelerations and structural stresses produced by larger thrust values. At present, the orbital debris characterized by the highest removal priority are all massive rocket bodies ($>$ 4000 kg) and only one is a spacecraft (Envisat[2]) [14, 66]. Therefore, a LPM might be employed in the frame of an ad hoc mission for Envisat recovery and disposal, whereas HPMs for a more continuing removal strategy of large abandoned rocket bodies. However, lower thrust levels can be provided also with a hybrid motor by implementing multi-burns disposal strategies, limiting each combustion phase below a fixed time.

Hybrid propulsion seems to be a valid alternative to liquid propulsion, in the light of its intrinsic safety and possibility of cost reduction for engine manufacturing, but also simpler ground operations such as components assembling, propellants handling and storage, due to the use of cheap and non-toxic substances. In addiction, the ADR platform mass must account of RCS for close-proximity operations (low-thrust) with the target and attitude control (high-thrust) during the mid-range rendezvous and disposal. By selecting HTP(90%) as monopropellant, a propellant amount of about 20 kg can be estimated assuming 2000 low-thrust impulses ($\sim$ 3 N) and 500 high-thrust impulses ($\sim$ 50 N). With respect to RCS, the use of hybrid motor represents an advantage due to the possibility of spilling the hydrogen peroxide directly from the main oxidizer tank, avoiding the need of an additional tank, with its own pressurization system, as required for SPM and LPM. Hybrid rocket technology has not been tested in space yet, but its possible implementation in the frame of active debris removal might promote its full development, as well as an effective cost reduction, strongly required to ensure that hybrid becomes a valid alternative to liquid propulsion for orbital transfers in the next future.

4 Conclusions

In recent years, the active debris removal concept is widely discussed in the international space community and, besides the technological challenge, significant obstacles are put by its economic impact and, in a more complex manner, by political and legal issues related to the implementation of specific technologies. In this work, the way to use chemical propulsion modules for the controlled reentry of large abandoned objects was discussed. This approach seems to be one of the most

[2] Earth-observing satellite lost by ESA in April 2012, after 10 years of service.

effective and safe, allowing for a fast removal of the selected targets from the most crowded orbital bands. The Cosmos-3M second stage represents a very interesting candidate for mission demonstration and study of multi-removal strategies, promoting the development of technologies directly applicable also to larger abandoned rocket bodies (e.g. Zenit second stages), characterized by masses above 4000 kg.

The removal mission might be performed by a de-orbiting kit, here called DeoKit, made by a primary chemical propulsion module and by an ADR platform, which includes avionics, sensors, capture mechanisms and a secondary propulsion system (RCS) for the attitude control. With the aim of multi-removal missions, several DeoKits might be carried in orbit by a launcher upper stage able to release each de-orbiting kit in the proximity of the selected targets. The Soyuz launcher, with Fregat upper stage, which exhibits the capacity to carry several DeoKits, might represent a valid candidate for a multi-removal demonstrative mission. Nevertheless, the number of carried de-orbiting kits must be evaluated taking into account the target orbits (i.e. ΔV to be provided by Fregat), the available volume of the payload fairing and a detailed mass budget for the overall effective payload.

From the propulsion systems comparison performed assuming the same disposal mission requirements, the solid motor results in a very compact system with a lower wet mass than that of liquid and hybrid motors. However, solid propulsion technology lacks in throttleability and reignition capability, but the advantage provided by its compactness might be exploited by a different ADR mission approach: a chaser spacecraft able to perform multiple rendezvous with different targets and to attach a SPM to each them. In this instance, the chaser could be conceived as powered by a LPM or HPM or, as alternative, made up by an existing upper stage modified and equipped with an ADR platform. On the other hand, a smaller mass difference is achieved between liquid and hybrid motors, the latter resulting 28.5 kg heavier. At this preliminary design phase, the relatively small mass difference between hybrid and liquid propulsion technology allows to mainly base the choice on their features comparison. In addition, the underestimation which affects the regression rate estimation allows for a certain margin for the improvement of HPM performance and mass budget. Hybrid motors provide several advantages with respect to liquid rockets, in particular an enhanced system safety, use of non-toxic propellants, dual-use of the oxidizer to supply also the RCS for the DeoKit attitude control and possibility of significant overall costs reduction. The latter is one of the basic aspects, since the development of ADR entails a large economical effort and would not provide any profit to the executor agency, making impossible eventual long-term amortization. The lower cost promised by hybrid propulsion might be helpful in this direction, making available relatively cheap motor systems for ADR missions.

Future steps of this research should improve the quality of propulsion system comparison, by considering a more complete analysis of performance losses and combustion efficiency for all propulsion systems, an enhanced regression rate prediction precision for hybrid motor, as well as a more detailed mass evaluation for rocket subsystems. Furthermore, the two described ADR mission approaches should be evaluated and compared in details, focusing the attention to the mass budget of the overall ADR system and the total velocity increment required for both rendezvous and removal of multiple targets.

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Acronyms

ADR Active Debris Removal
CCPs Condensed Combustion Products
CEA Chemical Equilibrium for Applications
DeoKit De-orbiting Kit
DOF Deegres-Of-Freedom
ESA European Space Agency
FPA Flight Path Angle
GEO Geostationary Orbit
GOX Gaseous Oxygen
HPM Hybrid Propulsion Module
HTP High Tested Peroxide
HTPB Hydroxyl-Terminated Polybutadiene
IADC Inter-Agency Space Debris Coordination Committee
LEO Low Earth Orbit
LOX Liquid Oxygen
LPM Liquid Propulsion Module
MMH Monomethyl-hydrazine
NTO Nitrogen Tetroxide
PB Polybutadiene
PE Polyethylene
PMD Post-Mission Disposal
PMM Polymethylmethacrylate
RAAN Right Ascension of the Ascending Node
RCS Reaction Control System
SPM Solid Propulsion Module
UDMH Unsymmetrical Dimethyl-hydrazine