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Space Debris Mitigation using dedicated Solid Rocket Motor

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Abstract

The growing problem of space debris forces to look for new mitigation methods for future space operations. Further generation of new debris can be significantly reduced by implementation of the end-of-life disposal techniques. Deorbitation and reorbitation of the obsolete spacecrafts, from most densely populated orbital regions, is in fact recommended by international standards like ISO 24113:2011. One of the most reliable propulsion systems for single orbital manoeuvres historically were the solid rocket motors. Therefore, it is considered as promising solution for an autonomous disposal systems. Although, modern retractions on acceleration levels and solid particles generation requires dedicated solid rocket motor and propellant. This article gives an overview of the development of such deorbitation motor, performed by the Institute of Aviation and its partners in Poland. A brief overview of a new solid propellant composition elaborated under ESA's project: "Pre-Qualification of Aluminium-free Solid Propellant" is given. Selection process and test results are presented, including burn rate, performance, solid particles generation, manufacturability, storability and radiation impact considerations. The main focus of the presented work is put on preliminary design of the deorbitation motor itself and current status of follow-up ESA's project: "Solid Propellant De-orbit Motor Engineering Model Development". Relatively high total ΔV demands for direct deorbitation and limited acceleration due to fragile satellites' appendages result in unique set of the requirements. The performance calculations influenced by thrust limitation and extreme burn time are given. The latter factor is also crucial for a thermal insulation design, which is outlined in this article. Potential solution for a nozzle throat erosion is presented, alongside with preliminary tests and other material trade-offs performed for design mass optimisation. The use of clusters of motors is envisaged, therefore, system-level integration is also highlighted. Since need for the TVC implementation was identified, a dedicated trade-off and outlook is also provided. The design description is summarised with recommendations and plans for further development.

1. Introduction

The number of space debris, induced by not only newly launched objects but also break-up events and collisions, pose a far-reaching threat to preservation of the space accessibility. This problem was addressed and mitigation measures were defined in several international guidelines by COPUOS, IADC, ESA and others. International cooperation in this field also resulted in developing a dedicated ISO standard (ISO 24113:2011). One of the most important method of debris generation reduction, covered by all mentioned documents, is end-of-life disposal of all new space objects. This includes reorbitation on a graveyard orbit or preferably deorbitation with a minimum on-ground casualties risk (below 10^{-4}) [1]. If the latter cannot be fulfilled with an uncontrolled deorbitation, control of on-ground footprint must be applied. This applies mainly to the massive or made out of non-dismissible materials objects. Controlled deorbitation requires active propulsion system and is achieved by lowering orbital perigee below 120 km altitude [2] ensuring atmospheric re-entry (predictable deorbitation). Exact value, defining path angle, depends on desired footprint size.

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1.1 Solid propulsion for deorbitation

Solid propulsion has a significant space heritage and solid retrorockets were already used for deorbitation of Mercury [3] and Gemini [4] capsules. Also Soviet reconnaissance missions utilized SRMs for deorbiting capsules and satellites [5]. Wide range of kick stages are provided for example by the Northrop Grumman Innovation Systems (former Orbital ATK) [6]. The STAR 3A motor is advertised as useful for deorbitation missions. Although, with peak thrust of 800 N and burn time of 0.5 s (total impulse 284 Ns), this motor would generate high acceleration for small objects and would not provide sufficient total impulse for bigger ones. Early concepts of the dedicated SRM for deorbitation were elaborated [7,8] and several systems for CubeSats are being developed [9,10,11]. However, no solution for large satellites was identified.
The main advantages of solid propulsion, which makes it an attractive candidate for controlled deorbitation, are the following:

- simple construction resulting in low dry mass, cost and high reliability,
- wide range of thrust levels, limited only by maximum acceleration, allowing direct deorbitation,
- density specific impulse is about 20% higher (for high performance solid propellants) than for hydrazine bi-propellant systems,
- low energy consumption for conditioning and operation allows development of autonomous modules.

The above arguments indicate that development of the SRM for deorbitation is an attractive solution for growing demand. Especially regarding emerging mega-constellations, which will require end-of-life disposal to maintain operative orbital slots.

2. Mission and requirements

The main target are LEO satellites from the most densely populated region of SSO (Figure 1). Considered system is required to enable both deorbitation and reorbitation of the spacecraft. Since the ΔV requirement for reorbitation and uncontrolled deorbitation are no higher than tens of m/s, and for controlled deorbitation ranges from tens to hundreds of m/s (depending on initial altitude), the latter should be studied extensively. Alongside, size-adjustable design and extended storability should be considered for GEO application.

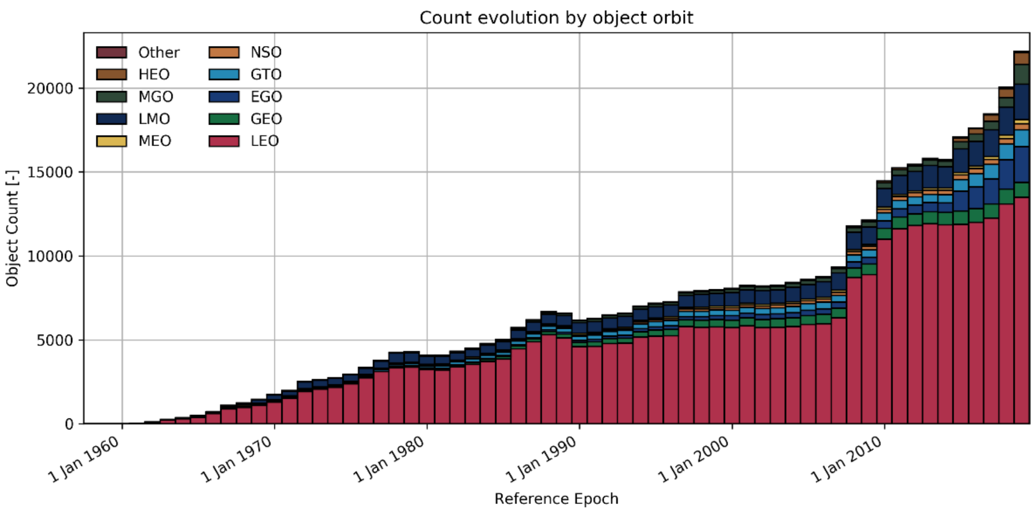

Figure 1: Spatial density vs altitude for LEO (ESA)

Controlled deorbitation manoeuvre (Figure 2) using a chemical propulsion, such as SRM, can be considered as the Hohmann transfer. A high thrust and relatively low burn time (with respect to an orbital period) allows to assume that the whole velocity change would be delivered at one point. This is beneficial for specification of orbital position for chosen re-entry area and therefore minimizing probability of casualties. To control atmospheric entry localisation, it is required to lower final perigee altitude to 80 km. The burn is planned to be delivered by one or multiple SRMs (organised in a cluster).

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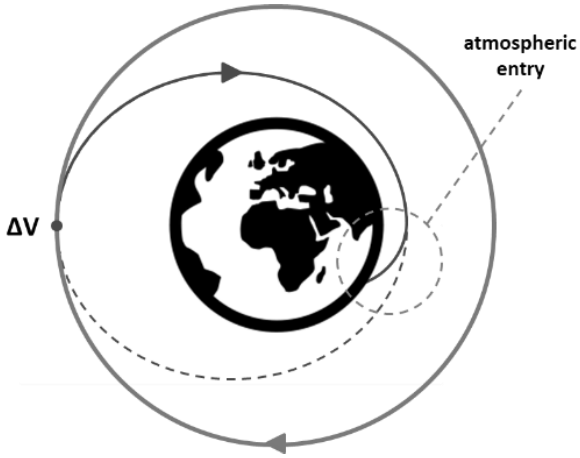

Figure 2: Direct deorbitation from circular orbit

Thrust of the SRM would be limited by the maximum acceleration for the whole S/C and cumulated total impulse should be sufficient for the required ΔV. A typical satellite, in deployed configuration, is mechanically resistant only to very low accelerations (fragile appendages). Higher values may disintegrate fragile parts generating dangerous debris. High total impulse alongside with limited thrust level requirements result in long SRM burn time. This can be achieved only by low burn rate of the propellant in combination with end-burning grain configuration and constant thrust profile. This affects thermal insulation and nozzle throat design significantly.
Deorbitation cannot be followed by generation of new debris, which exceed in size objects from the natural orbital environment. Thus SRM shall not generate solid particles over 1 mm and propellant over 30 μm (propellant generates majority of ejected mass) in diameter during firing. Using aluminium in the propellant composition results in generation of slags and solid aluminium oxide agglomerates in the exhaust, therefore aluminium and other metalized compounds are excluded from the propellant composition.

2.1 Main challenges

Analysis of the functional requirements led to formulation of the main problem areas. Brief description with potential solutions is provided below.

- Propellant burn rate and performance

Performance augmentation of the AP-HTPB propellant without aluminium can be achieved by increasing AP content. This is although limited mostly by decrease of the mechanical properties and casting technology. On the other hand, high AP content causes burn rate increase. Stable combustion, especially at low chamber pressures (essential for low regression rate), may pose a difficulty due to limits concerning adding metal powders to the propellant.
Significant reduction of the burn rate has to be achieved with the use of suppressing additives. Most of them however, decrease propellant performance by their retarding effect. Many of burn rate suppressants increase the temperature at which AP decomposition occurs, thus leading to possible difficulties in SRM ignition.
A summary of the propellant development challenges and solutions are presented in the table below.

Table 1: Propellant development challenges and solutions summary

Challenges	Solutions	Implementations
High total impulse	State of the art propellant High Isp	AP/HTPB system Optimized oxidizer-fuel ratio
Limited thrust (long burn time)	Low burn rate	End-burning grain Low chamber pressure Burn rate suppressant Multimodal AP
Solid particles generation	No metalized compounds	Aluminium-free propellant
Storability	Storability analysis and testing	Vacuum, accelerated aging, radiation testing

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- Low erosion nozzle throat

Presently used throat materials in European SRMs for space applications have erosion rates of approximately 0.1 mm/s (depending on the mass flux). Considering 10 mm nozzle throat diameter and the burn duration of the motor, the throat area would increase 9 times which is not acceptable. However, relatively low chamber pressure as well as no metal content in the solid propellant will influence the final erosion rate. Additionally, refractory metal inserts are considered for the throat, such as tungsten or molybdenum.

- Low ablation rate of thermal insulation

For long burn durations a low ablation rate of the internal thermal insulation is needed in order to limit the SRM mass. This requirement is also consistent with the one for low density of thermal insulation. The challenge will be to select (on the basis of existing solutions) and test ablative materials in real conditions.
Combustion of ablative materials can cause generation of the solid particles, and since larger than 1 mm in diameter are unacceptable the following must be taken into consideration: combustible fibres and low solid content. On the other hand, combustible fibres may increase the erosion rate of the ablative insulation.

3. Propellant selection

Low regression rate, no metallic additives and stable combustion in low pressure combined with high performance required elaboration of a new propellant composition. The following methods of lowering burn rate were considered:

- Lowering propellant pre-ignition temperature
- Lowering chamber pressure
- Using multimodal AP (different fractions)
- Using burn rate suppressants

Only first method was excluded from the final solution, not to add unnecessary complexity to the system. Implementation of the other three solutions was done in phases of the propellant composition selection, basing on laboratory scale tests. General propellant development logic is presented on the Figure 3 below.

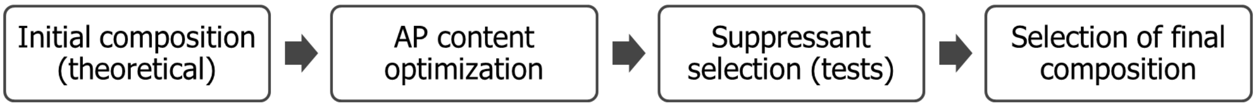

Figure 3: Propellant development logic

Selection of the most effective burn rate suppressant and propellant composition was the main purpose of this project. After extensive literature research, the following candidates were selected for further testing: melamine, urea and oxamide. The latter was found most suitable in terms of performance, suppressing effect and manufacturing. Final propellant composition and basic performance is presented in Table 2 below.

Table 2: Final propellant composition and basic properties

Final composition	
Ammonium Perchlorate	83%
HTPB system	12%
Oxamide	5%
Basic properties	
Burn rate (@ 10 bar)	2.85 mm/s
Density	1.71 g/cm^3
Density specific impulse	472 s
Theoretical I_{sp} (vacuum, 92% efficiency)	276.0 s
Demonstrated I_{sp} (static test, sea-level nozzle)	174.3 s
Propellant TRL 6	

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Final propellant verification was done in semi-full scale (equivalent thrust, lower burn time) dedicated Test Motor. Results of the reliable tests were gathered in the table below (Table 3) and confronted with simulated parameters. Differences between test results are mainly related to non-ideal propellant inhabitation and significant difference (up to 15°C) of temperature conditions between test days. Thrust level is higher than expected because model did not include additional mass flow from ablative thermal insulation, which is significant in this case. This affects also total and specific impulses results. Characteristic velocity efficiency is lower than assumed, although, still high as for test motor.

Table 3: Test Motor results summary

Parameter	Test 3	Test 6	Test 11	Mean	Calculations	Deviation
Mean pressure [MPa]	0.94	0.92	0.84	0.90	0.97	-3.5%
Max pressure [MPa]	1.02	1.19	0.88	1.03	1.01	0.7%
Mean thrust [N]	165.87	154.82	148.95	156.54	150.78	10.0%
Max thrust [N]	184.52	202.04	158.32	181.63	157.16	17.4%
Burning time [s]	30.75	30.81	32.67	31.41	31.88	-3.5%
Total impulse [Ns]	5 098.95	4 785.25	4 857.45	4 913.89	4 803.07	6.2%
Specific impulse [s]	181.42	170.26	172.83	174.83	170.91	6.1%
Characteristic velocity [m/s]	1 341.92	1 323.00	1 267.38	1 310.77	1 467.10	6.1%
Characteristic velocity eff. [-]	91.5%	90.2%	86.4%	89.3%	96.0%	-6.9%

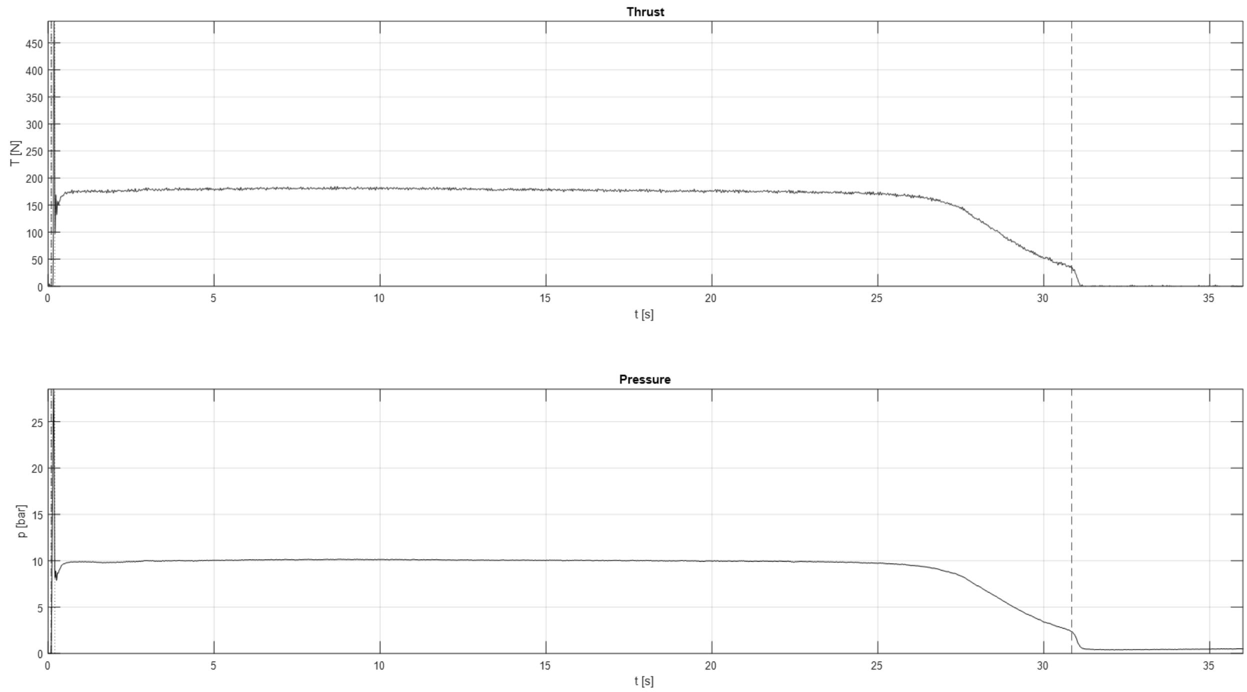

Figure 4: Example of the test results

4. Preliminary deorbitation motor design

Design process started with sizing activity, which allowed clarification of requirements for single motor in cluster. This process is presented in Figure 5.

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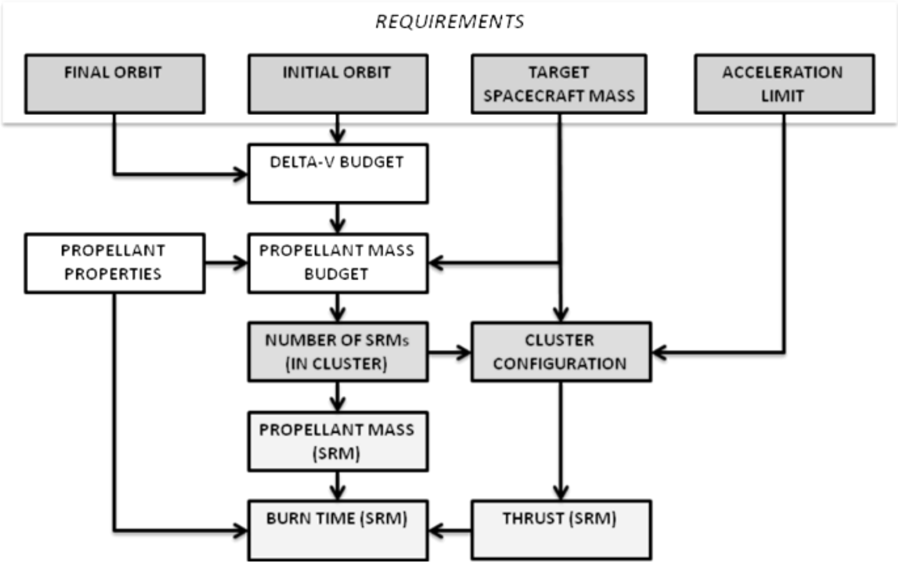

Figure 5: The process of basic SRM parameters definition

Calculations assumed lowering 1500 kg class satellite's orbit perigee to 80 km, starting from circular 800 km orbit (typical for SSO). Upper acceleration limit of 0.04g was a requirement for this project. Results of the sizing activity are gathered in Table 4.

Table 4: Sizing activity results

Name	Value
System	
Total number of motors in cluster	4
Number of motors fried simultaneously	2
Maximum thrust generated by the system	500 N
Total required ΔV	200 m/s
Total required propellant mass	116 kg
Motor	
Maximum thrust	250 N
Minimum propellant mass	29 kg
Minimum total impulse	78.5 kNs
Nozzle expansion ratio	220

The grain geometry must be suited for the requirements. Beside delivering required total impulse, the thrust level must be within desired range and the burn time shall be minimized to reduce thermal insulation mass. In this case, with a high total impulse required (78.5 kNs) and relatively low thrust level (250 N), the shortest burn time will be obtained for constant thrust (314 s). There are no additional requirements on thrust profile due to operating in vacuum conditions. Therefore, near-to-constant burn area with easy to control web thickness (defining burn time) is needed. The end-burning grain geometry is, therefore, an obvious choice.

This type of geometry has, however, one disadvantage. The burn area proceeding down the chamber expose the wall to hot gases. Combining that with significant burn time (exceeding 300 s) will require heavy ablative thermal insulation. Although, exposure time (assuming near-to-constant burn rate) will be a linear function of the chamber length, with 0 at the aft end and equal to total burn time at the nozzle entrance. Minimizing construction mass leads to use of regressive (in direction from nozzle to the aft end) thickness of the chamber insulation, with only minimum thickness at the aft end. The grain shape that include this feature is outlined in Figure 6.

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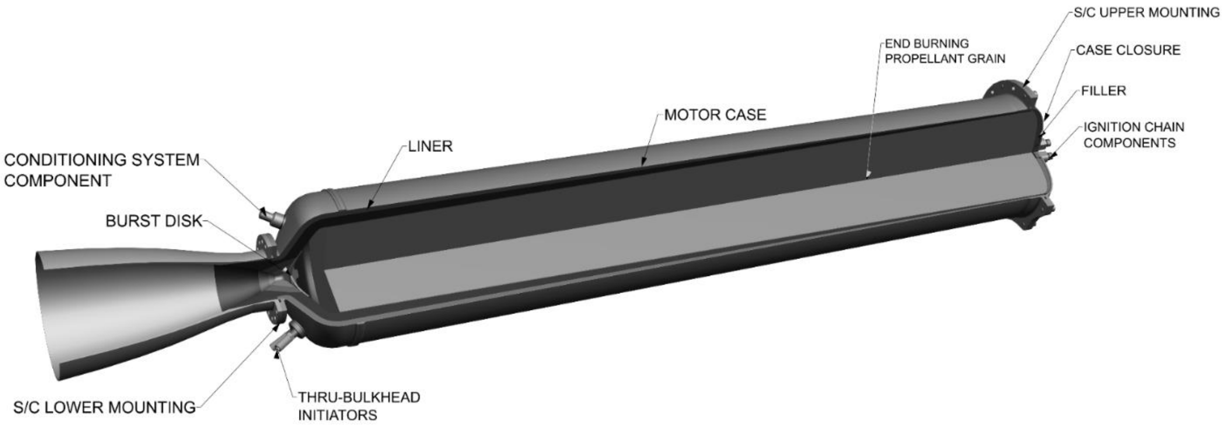

Figure 6: SRM outline

Optimal thermal insulation thickness induces conical shape of the propellant grain. This results in progressive burn area, which will affect a thrust profile. Although, regression of the nozzle throat will compensate this effect in certain extent. This shows that thermal insulation and nozzle throat regression rates are two main design drivers. An overall calculated and predicted performance is listed in Table 5 below.

Table 5: Deorbitation SRM performance summary

Performance	
Mean thrust	245 N
Max thrust	250 N
Action time	324 s
Total impulse	79 kNs
Specific impulse	278 s
Propellant mass	29 kg
Total mass	44 kg

4.1 Ignition chain

The system is responsible of initiating the burn of the first motor in the chain (cluster) and transferring the flame front to the next one. It is designed to be fully autonomous once the ignition information to the Safe and Arm device was transferred. Redundant lines are proposed to ensure the reliability of the system, as shown in Figure 7.

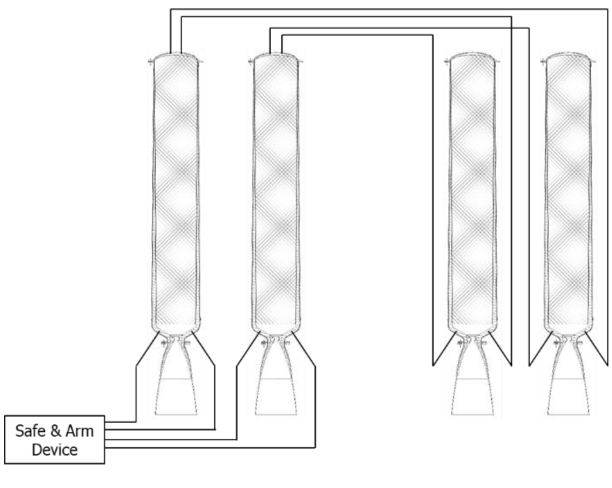

Figure 7: Ignition chain for the cluster of SRMs

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Although for current phase of the project the regular Exploding Bridgewire Initiator was used, the use of a TBI (Figure 8) is envisaged to be used in the Engineering Model development stage.

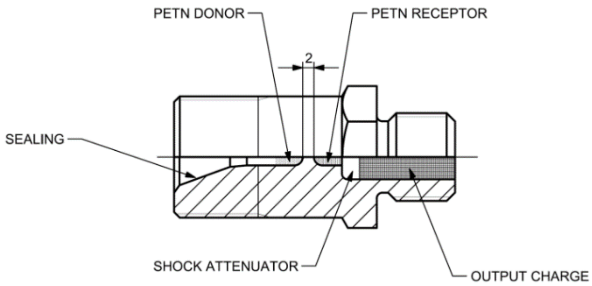

Figure 8: TBI design concept

Conducted research showed that the use of Shielded Mild Detonating Cord as a main part of the transfer line would be inefficient and the Shock Tube was selected. It provides the biggest flexibility from considered solutions and is one of the most reliable. Moreover, it is the lightest of the systems. It lacks in terms of speed but in respect to envisaged distances the maximum delay would be around 0.001 s which is more than acceptable.
During current phase of the project a selection of pyrotechnic charges were tested for ignition of the propellant. Finally, the most reliable proved to be BKN (Boron and Potassium nitrate mixture) and will be considered for further analysis in next development stages.

5. System-level analysis

SRM positioning on or in the satellite needs careful consideration for a number of reasons. Firstly, motor performance is dependent on a distance between thrust vector and spacecraft's CoM as well as temperature difference between two motors firing simultaneously. While the positioning affects the distance between thrust vector and CoM directly, the temperature difference is more complex matter which can be affected by positioning i.e. by mounting the SRM inside the satellite would allow much easier temperature control of the motor. Also, motors may not cause the spacecraft to exceed allowable volume during its launch. Clearly, motors should be positioned in a way that will not affect correct work of any other subsystem. SRM positioning will also influence on a change of spacecraft's moments of inertia which may or may not affect design of the AOCS system. General considered solutions are:

- Motors mounted inside the satellite to the side wall.
- Motors mounted inside the satellite to the bottom wall.
- Cluster of SMRs mounted to the launch vehicle adapter.

Of course, in case of a mission with large number of the SRMs a mixed approach is also possible.

Table 6: Possible applications and their impact

	Launch Vehicle upper stage	**LEO satellite**	**Mega-constellation satellite**	**GEO satellite**
Lifetime	Few hours/days	Few years	Few years	>20 years
Radiation	N/A (very short mission)	Low	Low	High
Thermal cycles	N/A (very short mission)	Many	Many	Low
ΔV required	Low	Medium	High	Low
Market size	Tens per year	Tens per year	Hundreds per year	Tens per year

In the particular application of this project, TVC performance is crucial, as wrongly oriented satellite might not set the correct trajectory, and instead of going to the graveyard orbit or towards lower orbit, it can remain on common operational orbit, increasing the probability of colliding with an operational spacecraft and generating space debris.
Based on a trade-off results, outside jet vanes concept (Figure 9) was selected for further investigation. This configuration shall allow to be compliant with the requirements. Moreover, the variety of materials nowadays, that are available on the market and have good thermal properties, low erosion rates and proper selection process could eliminate potential disadvantages. Another advantage, when comparing to gimbal solution, is a potential mass

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reduction of the overall system (moving only the flaps, not entire motor). This concept assumes using three actuators, each that controls one deflector, that geometry is of extended curvature of the exit nozzle, in order to avoid shock wave generation.

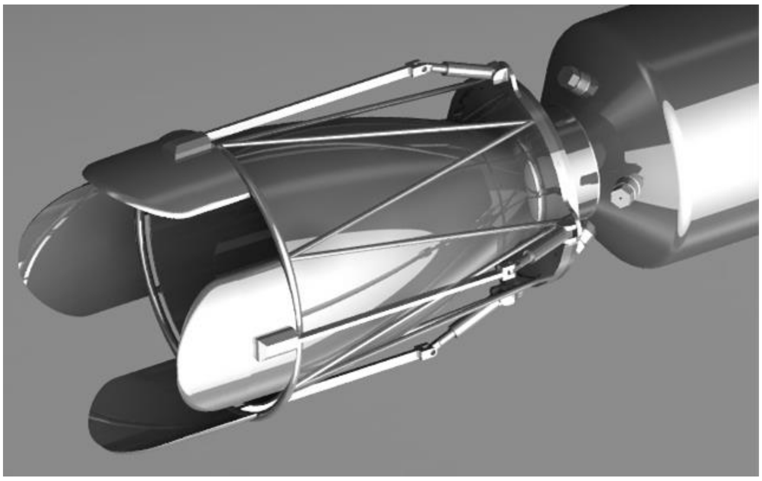

Figure 9: Outside jet vanes TVC concept

6. Conclusions

The following roadmap (Figure 10) is proposed on basis of the project outcomes. Further development should be divided in 3 parallel paths with an emphasis on SRM development. Another branch should be dedicated to TVC, which is a vital part for reliable deorbitation manoeuvre. Optional path of autonomous system development is also envisaged. All the activities shall converge in IOD.

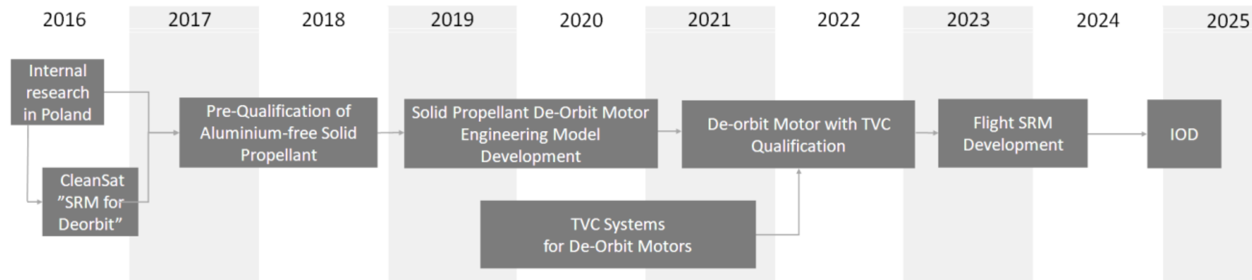

Figure 10: Proposed development roadmap

Development of the SRM for deorbitation is undoubtedly a difficult task. Consolidated requirements indicate the need to develop a dedicated motor with unique characteristic. Multiple material and construction choices must be made with respect to overall mass and performance. However, the most important step was already made with elaboration of the devoted propellant composition. Confirmation of its performance proves the attractiveness of using solid propulsion for deorbitation.

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