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Solid Rocket Motors for the De-Orbiting of Satellites

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Abstract
This paper describes the use of solid propellant rocket motors for systems that de-orbit or re-orbit satellites at the end of their service life or in case of a failure. The paper gives an introduction into the technology of solid propellant rocket motors. Emphasis is put on specific issues associated to the use for de-orbiting of satellites and the description of the first orbital solid propellant rocket motor demonstrator. The paper closes with conceptual considerations on de-orbiting systems using solid propellant rocket motors.

Nomenclature

Parameters

A	[m^2]	Area
a	[m/s^2]	Acceleration
a	[1]	Constant factor in Vieille´s law
I_{spec}	[m/s]	Specific impulse
L	[m]	Length of a body
m	[kg]	Mass
n	[1]	Exponent in Vieille´s law
P_{coll}	[1]	Probability of collision
p	[Pa]	Pressure
r_b	[mm/s]	Burn rate of propellant
T	[K]	Temperature
t	[s]	Time
v	[m/s]	Velocity
ρ	[kg/m^3]	Density

Subscipts

b	Burn operation
CC	Combustion chamber
c	Combustion
cs	Cross section
e	Condition at nozzle exit
f	Flight
op	operation
soak	Soak condition
V	Vehicle
vac	Vacuum conditions
∞	Ambient condition

Abbreviations

APC	Ammonium perchlorate
BC	Bayern-Chemie
CC	Combustion chamber
CG	Center of gravity
DPM	Double pulse solid rocket motor

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EPDM	Ethylene Propylene Diene Monomer rubber
LEO	Low earth orbit
MEOP	Maximum expected operating pressure
NVA	Nozzle-valve assembly
PSD	Pulse separation device
RESI	Reduced signature composite propellant
SRM	Solid rocket motor
STS	Space transport system (with Space Shuttle)
STANAG	NATO standardization agreement
TACS	Thrust and attitude control system
TRL	Technology readiness level
TVC	Thrust vector control

1. Introduction

The increasing number of uncontrolled space debris, failed satellites and used orbit insertion stages already put the usability of essential low earth orbits and the trespass at risk [1]. An aggravating element is the oncoming use of large constellations of communication satellites intended to enable the operation of the internet of things. The projected number is in total tens of thousands active satellites. Following the current rule that a satellite after end of service has allowance of 25 years to leave the orbit [2], we can expect another 10^4 to 10^5 major objects in or near the used orbits. A method to clear the orbit is to integrate into the satellite an independent system that de-orbits the satellite at the end of its life or in case of failure [3].
Another aspect is that more and more satellites use electric propulsion for orbital manoeuvring. Even if the satellite and its propulsion systems work properly, de-orbiting using the electrical main propulsion system has significant shortcomings:

- The thrust of electric propulsion thrusters is very low and allows only extremely gradual changes of the orbit. As the satellites approach the upper layers of the atmosphere, the re-entry trajectory is determined by the local and actual state of the atmospheric gas and not by the deceleration generated by the thrusters. Hence, the location of re-entry and impact on ground of the satellite is unpredictable. A Solid propellant Rocket Motor (SRM) with sufficient thrust allows to put a satellite with electric thrusters on a well-defined final dive with a subsequent impact in an uninhabited area
- The initial descend with electric main propulsion systems needs many orbits until the satellite reaches the altitude where the braking effect of the atmospheric drag becomes effective. During this phase, the risk of collision in a specific altitude band is proportional to the time needed for the satellite to cross this altitude band. A rocket motor that generates higher thrust produces a steeper and shorter dive, and therefore, reduces the collision risk significantly. Collision risk in this sense does not only mean the risk to create additional debris by collision. The orbits of descending satellites are well known and collision avoidance manoeuvres can prevent collisions in most cases. But, collision avoidance manoeuvres consume impulse of each participating vehicle. Each N··s of impulse that is used for collision avoidance is lost for station keeping manoeuvres and shortens the usable lifetime in orbit

The dichotomy of reliability and cost also supports the use of an independent on-board de-orbiting system: The satellites for the constellations have to be produced in comparatively large numbers and production cost is a key factor in business success. It is common knowledge that a slight relief on reliability requirements produces significant savings in production cost. Such a reduction in reliability of the satellite´s systems can be tolerated if it has an independently operating on-board de-orbiting system:

- In case that the satellite fails, the de-orbiting system carries it back to re-entry
- If the de-orbiting system fails, the satellite can de-orbit by its own propulsion means. In this case the de-orbiting process is more lengthy and the impact zone not predictable, but the orbit is cleared

Notice that the reduced system reliability requirement that entails cost savings applies to both the satellite and the de-orbiting system, reducing the production cost of both.

In order to secure safe de-orbiting of satellites, a number of different methods have been considered and discussed:

- Drag sails [4] are a means that is cheap and easy to deploy. The shortcoming is that a drag sail operates with reasonable efficiency only in very low orbits and that the sail increases the effective collision cross-section area. The contribution of a vehicle to the probability of collision P_{coll} can be expressed by the following formula:

$$P_{\mathrm{coll}} \sim A_{\mathrm{cs,V}} \cdot v_{\mathrm{V}} \cdot t_{\mathrm{f,V}} \quad (1)$$

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$A_{cs,V}$ is the cross-section of the vehicle perpendicular to the velocity vector, v_V is the velocity of the vehicle, in orbit rougly orbital velocity and $t_{f,V}$ is the time of flight of the vehicle. Hence, the reduced time for descend $t_{f,V}$ is balanced by the higher collision cross-section area $A_{cs,V}$ and the trade-off can end in a lower or higher overall probability of collision

- Electrically conducting tethers [5] use the force produced by the magnetic field of the earth in a conductive file if there is relative movement between the file and the earth´s magnetic field. This means is even easier to deploy than a drag sail and also cheap, but entails also the shortcoming of an increased $A_{cs,V}$. The situation is better than with a drag sail in so far as the tether is oriented along the velocity vector of the vehicle. But it still increases the cross-section in all other collision directions
- Saving sufficient propellant for a final descent thrust [6] is a method that relies on the promise that a satellite that is functional is deliberately de-commissioned, and that the residual amount of propellant can be estimated with sufficient precision

All these methods cannot produce a descend trajectory that targets precisely a specific area. And the last one relies on the correct function of the satellite and provides no redundancy for a secure de-orbiting.

2. Solid Propellant Rocket Motors for De-Orbiting of Satellites

2.1 Brief description of solid propellant rocket motors

The technology of SRM is well described in literature, see for example [7]. Hence, we can reduce the description to some elements that are of particular importance for the described purpose. Figure 1 shows the typical elements of a SRM.

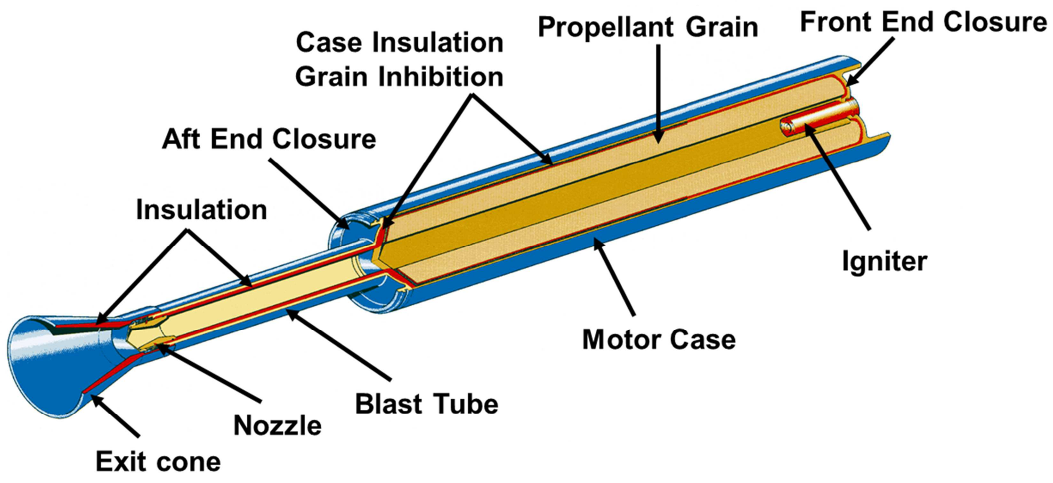

Figure 1: Typical elements of a solid rocket motor

The key feature of a SRM is the identity of tank and combustion chamber. The consequences are:

- A simple architecture that needs no means for propellant management
- No sloshing of the propellant, causing random acceleration
- No impact of gravity or non-gravity on the distribution or behaviour of the propellant
- No issue with compatibility of propellant and material of structures
- Design issues because all parts are subjected to combined mechanical and thermal loads
- The identity of propellant tank and combustion chamber allows high thrust density or short operation times with no inert mass penalty. For long operation times, the identity of tank and combustion chamber means that in the end the whole SRM structure is exposed to internal heating from the hot combustion products.
- The propellant grain has to maintain its ballistic, mechanical, chemical, physical and bonding characteristics within tight limits over the life time. Damage or dis-integration of the propellant grain during operation causes overpressure, bursting of the SRM case and destruction of the entire vehicle
- Immediate readiness for use at any time, no lead time and easy initiation by a current of 50 V and 1 A. More sophisticated initiation methods, i. e. exploding foil or exploding bridgewire initiators are also

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available and mature, if the space-proven through-bulkhead initiators should not provide a reasonable solution
- Easy handling during ground operations, motor- and satellite-integration

Over more than 60 years Bayern-Chemie (BC) and it´s predecessor companies have developed
- A wide variety of solid propellants, particularly composite propellants. This means an aluminium content from 0 % up to 20 %, burning rates up to 50 mm/s at standard conditions of operation (p_c = 70 bar, ambient soak temperature) and specific slow-burning and smoke-free composite propellants for gas generator applications. Recent development efforts lead to a composite propellant with a very low burning rate r_b = 2.0 – 2.5 mm/s at standard conditions of operation (p_c = 70 bar, ambient soak temperature). In combination with requirements asking for small accelerations, i. e. low thrust and long burn duration, this propellant is especially suited for the use with de-orbiting SRM
- Effective insulation materials for combustion chambers, aft closures and blast tubes, heat shields and surface covers of propellant grains
- Design methods for valves, pintles and nozzles that are able to endure intensive heating for long times of operation

The developments and concepts described in the following are based on this experience.

2.2 Solid propellant rocket motors in space

For the use with a satellite de-orbiting system, some specific requirements apply:
- If the satellite has deployed sun paddles, antennas or other devices, the fixation of these devices is designed to withstand accelerations that are typical for orbital manoeuvres. A typical value for tolerable acceleration is 0.4 m/s^2 [3]. This means that for operation in the original orbit, a de-orbiting motor has to operate at very low thrust and for long operation times. For SRM, the consequence is to use a cigarette burner type propellant grain and, depending on grain diameter and thrust level, a slow burning propellant.
- If the SRM is used for the final precision dive of a satellite that has reached the edge of the atmosphere, dis-integration of delicately attached parts is no real concern because at that altitude very few vehicles carry out regular missions, and the lightweight disintegrated parts are slowed down soon by the atmospheric drag and the low ballistic coefficient m/A_{cs}. In this case, the deceleration may be higher than 0.4 m/s^2 and the burning duration of the SRM shorter.
- A general requirement for SRM that operate in orbit is that they must not exhaust particles because even small particles can cause tangible damage to structures of other space vehicles in case of a collision. Hence, solid propellants to be used in orbit have to be aluminium-free. Bayern-Chemie has such propellants in its portfolio, already fully qualified and used in series production of tactical SRM, the Deep Stall Recovery System (DSRS) SRM that was integrated in the tail cone of one A 400 M test airplane and a couple of technology programs .
- The SRM have to circle in orbit for many years without degradation. For the long-time behaviour of SRM and solid propellant grains under orbital condition no experience is available. The SRM of Bayern-Chemie orbited for about 3 months with the 3U-Cubesat D-SAT [8] and operated safely after this time of exposition to space conditions.
- Within the duration of a typical orbit of a LEO satellite of about two hours the satellite travels through bright sunlight and deep shade, imposing severe and short temperature cycles upon the satellite structures. In space, the heating scenario is dominated by radiation and heat conduction whereas convective heat transfer can be neglected (with the exception of heat pipes). Hence, thermal management is a key issue of the design of any orbital SRM.
- Space radiation is another factor that has the potential to change properties of solid propellant grains. For the D-SAT SRM, samples of similar inert composite propellant have been exposed to Beta radiation with a maximum dosis of 300 krad. Subsequent testing on propellant sample level by tensile test with mini dog-bones, on propellant density and on cross-link density showed no change of the properties of the tested composite propellant. In order to reduce the impact of space radiation on the propellant grain, a combustion chamber made of aluminium or steel is preferred over combustion chambers made of fibre reinforced resin.

2.3 The D-SAT mission

The D-SAT mission was carried out by the company D-Orbit SRL, Italy, which developed and operated the D-SAT satellite, a 3U-cubesat [9]. Fig. 2 shows pictures of the satellite. For the D-SAT satellite BC developed, built and

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qualified a solid propellant rocket motor. The primary goal of the D-SAT mission was to orbit the Earth in a Low Earth Orbit (LEO) for a period of approximately two months and subsequently demonstrate a de-orbit manoeuvre where the D-SAT motor decelerates the satellite such that it will re-enter the earth's atmosphere in a controlled manner. Effectively the D-SAT was launched on June 23, 2017 as a secondary payload on the Indian PSLV-XL rocket. It stayed in orbit for about 3 before the ignition of the SRM on October 2, 2017. In orbit, D-SAT also carried out different experiments on earth observation [8].

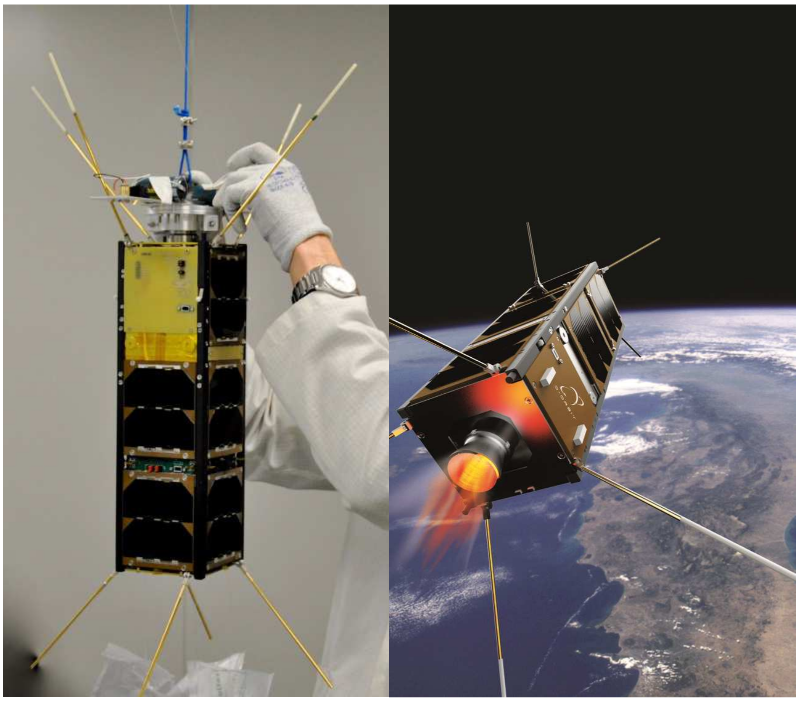

Figure 2: Picture of the D-SAT satellite and artist´s impression of the de-orbiting manoeuvre

2.4 The SRM for the D-SAT mission

According to the specification, well-known materials, components and design features have been used. The nominal overall dimensions of the motor are 121.55 mm long with a maximum diameter of 97.0 mm. Hence, it occupies a little more space than what is available in one cube element.

The design of the D-Sat rocket motor is based on a particle-free propellant family (RESI, REduced SIgnature) which has been developed and fully qualified according to military standards by Bayern-Chemie for a series production SRM and also gas dynamic devices. RESI propellants have also been used for the big SRM for the Deep Stall Recovery System [DSRS] that was integrated in the tail cone of one A-400 test airplane and for several technology programs, including a major program on insensitive munitions. The propellants have excellent ageing characteristics: 6 month of artificial ageing @ 60 °C yielded no significant impact on

- Chemical stability
- Sensitivity and ignitability
- Mechanical properties and glass transition point
- Ballistic properties

In addition to that, the RESI propellant, artificially aged for 10.5 years at +62 °C (lifetime >100 years at room temperature !!!) had similar properties as the new propellant, only the strain-capability at low temperatures of -54°C was slightly but not critically reduced.

The key properties of the Bayern-Chemie RESI propellants are:

- Burn-rate r_b: 12 – 17 mm/s (p_c = 100 bar at T_{soak} = 20 °C)
- Burn-rate exponent n: 0.25 – 0.50
- Density ρ: 1700 kg/m^3
- Specific impulse I_{spec}: 2280 - 2420 m/s (expansion ratio = 70:1)

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- Specific impulse $I_{spec,vac}$: 2700 m/s (vacuum condition)
- Soak temperature range: -54°C to +71°C
- Shelf life: minimum 12,5 years, up to 16 years

The properties of the RESI propellant that was used for the D-SAT SRM are:

- r_b = 12.5 mm/s at p_c = 100 bar and at T_{soak} = 20 °C and represents the lower end of the burn-rate range at the time of development
- At all conditions, i. e. 40 bar < p_c < 140 bar, -30°C < T_{soak} < +71°C, the pressure exponent n (from Vieille´s law: $r_b = a \cdot p_c^n$) is n < 0,43 which is a value that stands for stable combustion.

Radiation tests were performed at small sample levels using an inert propellant that is representative of the RESI propellant family in terms of the binder material. Irradiation levels with a maximum dosis of 300 krad indicated that the mechanical properties, the density and the cross-link density did not change in comparison to the not irradiated samples. The mechanical properties were determined by using tensile tests in combination with mini dog bones. The cross link density gives information on the "connectivity" of the binder material polymer chains and therefore indicated that the applied maximum radiation dose of 300 krad did not cause any molecular structural change or, in other words, any break-up of the polymer chains or their binding to each other.

The propellant grain is a tube inner burner with a burning face at the nozzle side. In order to ease the handling of the explosive component at transport, integration and launch in different countries and under varying jurisdiction, a cartridged propellant grain was chosen over a case-bonded propellant grain despite of the significantly higher inert mass. This allowed to conduct the final assembly of the SRM at the launch complex, together with the satellite structure.

Due to the comparatively small size of the rocket motor, the caseing is made out of Al-alloy which kept the design and its structural tolerances in realistic and also affordable manufacturing limits. The internal thermal insulation is conventionally made out of EPDM rubber for the case surface and Silica-Phenolic for the aft closure and the nozzle exit cone.

The nominal key parameters of the D-SAT SRM are:

- Total motor mass: 0.9 kg
- Propellant mass: 0.3 kg
- Maximum pressure: 70 bar
- Nominal pressure (20 °C): 50 bar
- Nominal thrust: 375 N
- Nominal burn-time: 3,2 s
- Impulse (vac.): 836 N·s
- ***Nominal*** $I_{spec,vac,eff}$: 2613 m/s
- Throat diameter: 7.5 mm
- Nozzle expansion ratio: 20
- Operational temperature range: -30°C to +71°C (qualified at -34 °C)

The D-SAT Motor Test Firing and Qualification Program was designed in order to qualify the rocket motor step by step, building confidence and gaining experience in terms of motor ignition behavior, motor burn characteristics and motor performance. Hence, the qualification of the D-SAT rocket motor was conducted in three consecutive phases, i.e. the Functional, Verification and Qualification phase.

- In the Functional phase two motor firings were test fired at the extreme operating soak-temperature levels of -30°C and +71°C in order to show the functionality and the performance level. Both tests were successful and met all specifications
- In the Verification Phase three rocket motors were environmentally pre-loaded with random vibration tests and temperature shock cycles and subsequently fired at -30°C, +20°C and +71°C soak temperature. All three tests were successful and within the specified requirements
- In the qualification phase four artificially aged rocket motors were exhibited to the full environmental qualification program. Two motors each were fired at a slightly extended lower soak temperature level of -34°C and at +71°C. All four qualification tests were successful and within the specified performance limits

The firing tests were carried out in vertical nozzle-up orientation, as shown in Fig. 3. Fig. 4 shows the firings at low soak temperature -30 °C and high soak temperature +71 °C. Notice that the exhaust plumes are over-expanded

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because they are designed for expansion into vacuum and not into ambient pressure at the test site. Geometrical limitations drove the selection of the comparatively low nozzle opening ratio of 20 which is far from optimal for expansion into vacuum.

Fig. 3: Vertical test stand for all D-SAT the motor firings

2.5 The result of the orbital SRM firing of the D-SAT mission

On October 2, 2017 the SRM of the D-SAT was fired with the goal to initiate the re-entry of the D-SAT. The SRM ignited and produced the required impulse. Unfortunately, the impulse was oriented into the wrong direction and the orbit of D-SAT was lifted instead of lowered. But the satellite was operational after the burn of the SRM what indicates that the loads generated by the SRM were within the specified and tolerable limits. The root cause for the malfunction of the de-orbit manoeuvre is not known to the authors.

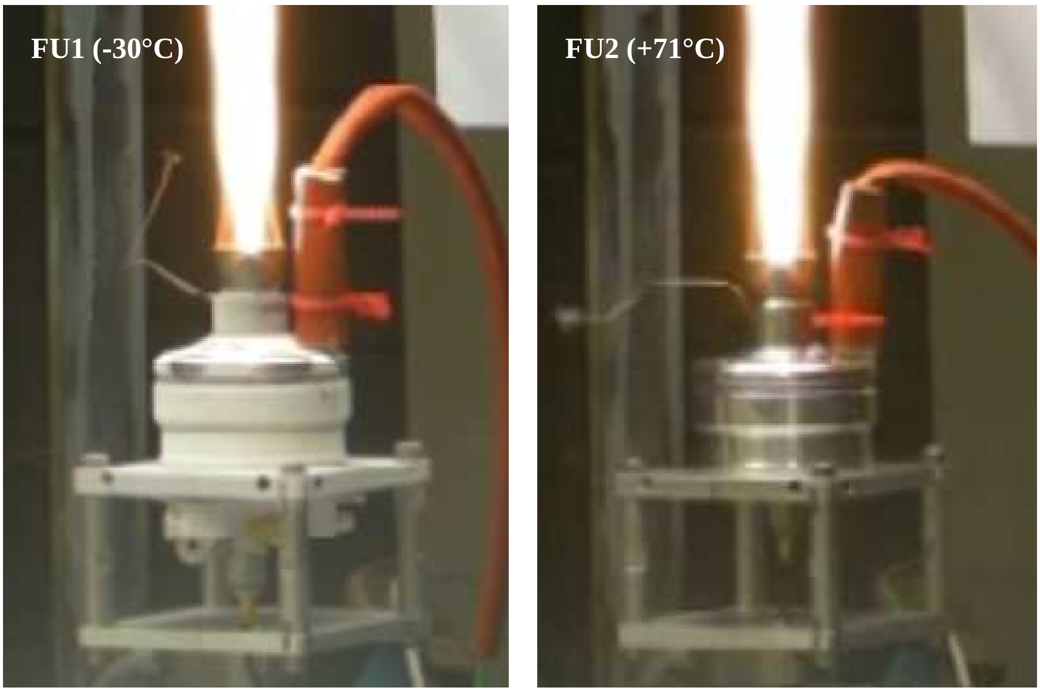

Fig. 4: Separation of the exhaust jet in the nozzle (Functional Tests FU1 and FU2)

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3. Concepts for De-Orbiting Systems with SRM

From the general considerations in chapter 2.1 and the experience gained from the D-SAT mission, specific attention should be paid to the following points of a SRM for de-orbiting satellites:

- SRM have minimal need and possibility for testing during integration and before the launch. Checkout tests may be done for the ignition chain and the mechanical assembly. With respect to the ignition chain, by nature of an SRM, only indirect testing is reasonable
- When travelling in orbit, similar indirect functional tests as before launch are possible.
- The SRM must produce as few as possible ejecta if it is set into operation. Ejecta may be the fragments of environmental seals or particles shed by the ablative internal heat shield of the SRM, or particles produced by the propellant itself.
- Because mono-pulse SRM by nature have burnt and exhausted their explosives completely at the end of operation, passivation activities are not necessary.
- Thrust vector control (TVC) is very likely needed because the movement of the vehicle during operation of the SRM is very sensitive against small deviations between the line of action of the force vector and the satellite's Center of Gravity (CG). The situation is aggravated by the fact that the exact position of the CG is not precisely known along the duration of the mission. Residual liquids in tanks may also move the CG. For SRM, various methods for TVC have been used with success in series motors, i. e. with very high Technology Readiness Level (TRL):
    - Movable nozzles with flex-seal bearing are well known from big SRM boosters of STS, ARIANE, VEGA and others. This method is very elegant and does not reduce the axial thrust level beyond the unavoidable 1-cosine effect. But it is expensive and heating management of the flex seal assembly becomes more difficult with decreasing dimensions. The thrust vector variation is limited to some degrees. This should be well beyond the requirement for the TVC capability of a de-orbiting SRM
    - Jet vanes can generate very high lateral thrust component, but the penalty is a significantly reduced efficiency of the SRM because the jet vanes block a significant part of the nozzle cross-section. And the leading edges of the jet vanes will blunt within some seconds of operation. For long times of operation, the loss of total impulse due to the drag effect of jet vanes is immense compared to the low requirement on thrust vector deflection capability. Another problem is the heat conduction along the support structures of the jet vanes. For long times of operation, say in the order of a minute or more, the transient heating of the actuation train requires specific design solutions. And sealing against the hot gas is another story. Hence, even if the heat conduction problem can be mastered, jet vanes are not the preferred solution for a TVC of an SRM with long operation times
    - The injection of gas laterally to the wall of the nozzle exit cone has also been used with success, but it is complicated in so far as not only a control mechanism for the gas flow is needed. The gas itself has to be provided. Gas taken from reservoirs can be taken into account if only a small amount of gas is needed. Gas tapped from the Combustion Chamber (CC) of the main SRM is available, but very hot if the SRM contains a high performance propellant. Even without Aluminum, the RESI propellants have a T_c of about 3000 K, too high for mechanical control systems if the time of operation exceeds several seconds. The injection of liquids is much easier, however, the volume source generated by the evaporation of the liquid is accompanied by an energy sink generated by the latent energy that is needed for the phase change from liquid to gas. Source and sink may compensate each other. An additional energy source can be introduced if the injected evaporated liquid reacts with the SRM gas. Critical in this respect is that the gas produced by SRM is always fuel rich and does not contain oxidizer that could be used for some kind of "afterburning" process. And the lateral injection of gas from a hot gas generator also adds a lot of complexity to the system
    - Flaps and spoilers are a simple method if just a limited lateral thrust component is needed. A technical advantage is the good spatial separation between the hot surface at the jet flow side and a cold surface at the actuator side, i. e. a clean separation of functions. If no lateral force component is needed, the flaps or spoilers are retracted from the nozzle cross-section. Hence, flaps and spoiler are well suited for mild thrust vectoring as this is required to point the thrust vector towards the position of the CG

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- Another idea is to move the complete SRM. Whereas conventional gimbal or two-axis bearings are mechanically complex, particularly if they have to operate after long years of no or occasional movement, a solution using a bearing without relatively moving contact surfaces could provide a good solution for small TVC angles.
- Derived from the control of missiles, a combined Thrust-and-Attitude Control System (TACS) could also be a solution. Fig. 5 shows the architecture. The gas generated by the SRM is fed into a Nozzle-Valve-Assembly (NVA) which distributes the gas according to the required thrust vector. In all cases the main part of the gas will flow down the thrust nozzle; depending on the required lateral trust component the lateral nozzle/valves open or close. If the operation times are longer than a couple of seconds, the T_c has to be reduced and adapted to the level that the mechanical parts can tolerate along the foreseen operation cycle. This of course reduces the effective I_{spec} of the overall SRM.

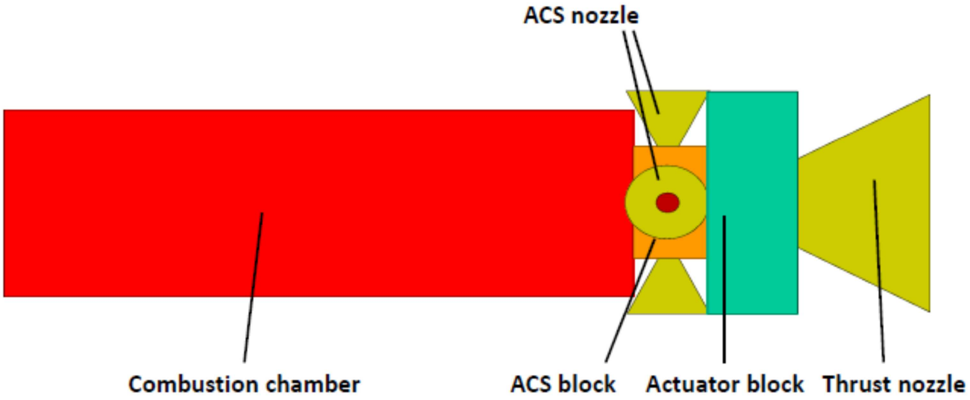

Fig. 5: Architecture of an SRM with TACS

- The preferred method of use of SRM is to control the final descent of the space vehicle in that way that it will impact in a pre-defined unpopulated area, e. g. in the South Pacific. If two thrust pulses are needed, Double Pulse Motors (DPM) have been tested with excellent results, see [10] and the literature cited therein. The key characteristic of a DPM is that the CC holds two solid propellant grains which are separated by a Pulse Separation Devices (PSD) [11]. Fig. 6 shows the longitudinal cut of the flight demonstrator motor. The second pulse propellant grain can be ignited at any time after the first pulse grain has burnt out. Advantages of a DPM over a two-motor-array are
  - The gas of both pulses exhausts through the same nozzle, producing the same thrust vector orientation
  - In the typical axially aligned configuration both solid propellant grains are located on the thrust vector axis and the mass change at operation does not induce disturbing moments
  - Triple- and multi-pulse SRM are feasible, but the technical complexity and the expenses for verification increase over-proportional with the number of pulses

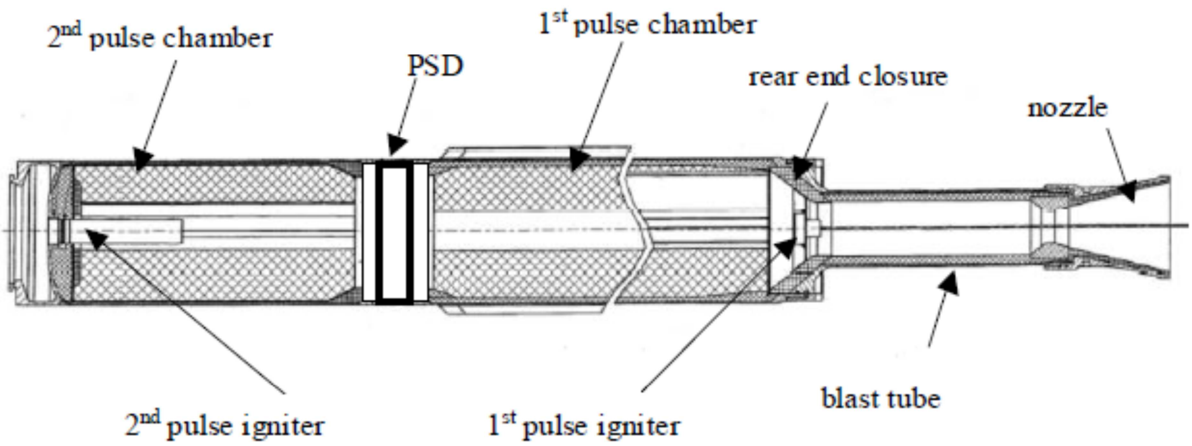

Fig. 6: Longitudinal cut of the flight demonstrator DPM SRM

- If the deceleration produced by the de-orbiting SRM has to be small, the time of operation needed to create the necessary velocity decrement is long and the thrust is low. With SRM, long operation times can be realized in combination with a cigarette burner propellant grain. Because the identity of propellant tank and combustion chamber is a key characteristic of SRM, some specific considerations have to be taken:
    - The thrust of a cigarette burner type motor is proportional to the burning area = cross-section area of the propellant grain and the burning rate r_b. Depending on r_b and duration of operation t_{op} the length of the cigarette burner propellant grain is $L_{Grain} = r_b \cdot t_{op}$. For a velocity decrement of Δv = 200 m/s, a propellant with a burn rate r_b = 10 mm/s and a deceleration of a = 2 m/s^2 we arrive at $L_{Grain} = r_b \cdot \Delta v / a$ = 1000 mm. From the viewpoint of SRM technology, grains of such dimensions have been used in series motors for military applications. The problem is rather the integration of such a long SRM in the satellite body in that way that its longitudinal axis points precisely to the CG. A shorter cigarette burner using a slower burning propellant is easier to integrate into the satellite. Unfortunately the I_{spec} of very slow burning propellants is significantly reduced which means higher propellant mass. Hence, a careful trade-off between burning rate and length of the solid propellant grain is needed to create an optimal solution
    - As the solid propellant slug burns down over time, an increasingly wide inner surface of the combustion chamber is exposed to the hot gas. Long exposure of the surface to the hot gas means an appropriately thick internal thermal insulation layer which adds inert mass to the SRM design. At the end, the trade-off between T_c and mass of internal thermal insulation is likely to end up with a compromise propellant with somewhat reduced I_{spec} and tolerable mass of the internal thermal insulation of the combustion chamber
    - In principle, the dimensional (length) and heating problem can be eased by using an array of smaller SRM that are fired consecutively. Practically, the shortcomings dominate: The SRM have to be distributed alongside the resultant thrust vector and thrust level over time as well as t_{op} have to match very precisely under any circumstance, e. g. different soak temperatures of the respective motors due to different exposure to sun radiation. One method to overcome the problem of un-synchronous operation of the individual SRM is to feed the gas into a plenum and to exhaust it through a central TACS. In any case, the complexity of the propulsion system increases significantly. Hence, a single-motor design is strongly recommended
- Like all monopropellants, solid propellant can produce explosions or, more precisely, deflagration and detonation. Stimuli are impact, friction, heat and electric discharge. In military environments, a variety of potential aggressions are possible, occur and are classified by standards, see for example STANAG 4439 [12] and related standards, e. g. for testing procedures. Because de-orbiting SRM are handled in a well-defined and controlled environment by educated personnel who execute precise procedures, the risk on ground reduces to accidents at handling or integration of the SRM. In orbit, the dominating risk is the impact of meteorites or debris particles on the SRM. If the de-orbiting SRM is exposed to potential impacting particles, impact shields have to be used as this is the case for other sensitive structures. Notice that the case structure of a SRM is more massive than for example the wall of a liquid propellant or liquid oxidizer tank. In addition, a layer of thermal insulation material is placed between the combustion case wall and the solid propellant grain, providing additional protection against the impact of micro-particles. Tests are planned to establish knowledge about the reaction of SRM propellant grains on the impact of small particles
- The experience gained up till now does not indicate that solid propellant grains are specifically sensitive against cosmic radiation. Nevertheless, testing for an orbital life time of many years has to be done. Static tests in ground testing facilities have the advantage that accelerated radiation ageing cuts the time needed to get the information
- The effect of particles like protons or molecular oxygen has to be studied as well. The key question is whether the enclosure of the grain can be designed in a way that the grain is shielded against the contact with those particles. Relief can be expected if the SRM can be sealed in that way that internal overpressure lasts over the entire lifetime of the de-orbiting motor. But the design of the seal has to respect the no ejecta requirement.

Whereas the knowledge base about the behaviour of SRM in space over long periods has to be built up, the extensive technology base gathered from military applications over the last 80 years provides design solutions for those problems that are not genuine space-related.

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4. Summary and Outlook

Solid propellant rocket motor technology is well matured over decennies in the course of military applications. The principal usability as propulsion sub-system for de-orbiting of satellites has been demonstrated by the D-SAT mission. For the use as propulsion unit for satellite de-orbiting devices that have to operate after many years in orbit some verifications have to be done:

- The life time under orbital conditions, particularly the effect or non-effect of cosmic radiation have to be investigated. Artificial radiation ageing using ground testing facilities are an indispensable first step. In parallel, representative solid propellant samples should be brought into orbit for long duration tests at original load conditions. In an ideal way, instrumented solid rocket motors could deliver diagnostic data before being fired under orbital conditions. Propellant samples could be brought back to earth for laboratory investigations.
- High-performance solid propellants with very low burning rate. The novel BC propellant is an important step, but further activities intend to reduce the r_b below the current limit
- Propellants without Aluminum do not produce Aluminum Oxide particles or slag. The gas produced by the RESI propellants contains just sub-micron-size particles which are too small to be relevant. Attention has to be paid to particles that can be generated by the burning or erosion of the internal thermal insulation material. Experimental investigations are ongoing to measure the size and amount of the particles contained in the exhaust gas.
- Investigations on the effect of particle impacts on SRM are also helpful to check the hazard potential in order to design protective devices or to avoid over-quality due to excessive and unrealistic worst-case assumptions

In summary, the use of SRM for satellite de-orbiting devices makes use of valuable spin-ins from military technology that needs just limited additional verification to be used with space vehicles.

References

[1] Ijaiya, H., Space Debris : Legal and Political Inplications, Environmental Pollution and Protection, Vol. 2, No. 1, 2017
[2] European Code of Conduct for Debris Mitigation, Issue 1.0, 2004. Available at hpp://www.stimon.org/ wos/pdf/eurocode.pdf.
[3] Janovsky, R., Kassebom, M., Lübberstedt, H., Romber, O., Burkhardt, H., Sippel, M., Krülle, G., Fritsche, B., End-of-Life De-Orbiting Strategies for Satellites, Deutscher Luft- und Raumfahrtkongress, Stuttgfart, D, Sept. 23-26, 2002
[4] http://blogs.esa.int/cleanspace/2016/11/08/de-orbit-drag-sail-put-to-test/, June 19,2019
[5] Kawamoto, S., Okhawa, Y., Nishida, S-I., Kitamura, S., Development Status of Electrodynamic Thethers, Proc. 5th European Conference on Space Debris, Darmstadt, Germany 30 March – 2 April 2009, (ESA SP-672, July 2009)
[6] ESTEC CDF-Team, SPADES -system, External final presentation, ESTEC, NL, June 25, 2013
[7] Davenas, A., Solid Propulsion Technology, Pergamon Press, 1993
[8] https://space.skyrocket.de/doc_sdat/d-sat.htm, June 19, 2019
[9] Tussiwand G., Antonetti S., Rossettini L. and Di Tolle F., Benefit of Adopting a Spacecraft Decommissioning Device to Implement Orbital Access Sustainability, 7th IAASS Conference, Friedrichshafen. 20-22 October 2014
[10] Naumann, K. W., Stadler, L., Double-Pulse Solid Rocket Motor Technology – Applications and Technical Solutions, 46th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Nashville, 2010, AIAA-2010-6754
(11] Stadler, L., Trouillot, P., Rienäcker, C., Niedermaier, H., Audri, D., Ruiz, S., Hacker, A., Naumann, K. W., et al., The Double-Pulse Motor for LFK-NG, 42nd AIAA/ASME/SAE/ ASEE Joint Propulsion Conference & Exhibit, Sacramento, CA, USA, 2006, AIAA-2006-4762
[12] NATO - STANAG 4439 Policy for Introduction and Assessment of Insensitive Munitions (IM), Ed. 3, NATO, Nov. 20, 2018, https://standards.globalspec.com/std/13107202/STANAG%204439