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Satellite Water Propulsion:
Satellite Water Propulsion: 3D Printed Ceramic Thruster & Space Debris Mitigation Concept

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Abstract

With the impending ban on hydrazine in the space sector, a great interest in green propulsion systems has developed. Also, the desire for cost reduction of propulsion systems, because of the upcoming disposal regulations, drives the interest even further. Therefore, the Institute of Space Systems of the University of Stuttgart is developing a water electrolysis propulsion system (WEP) that is scheduled to fly on the institute's ROMEO mission at the end of 2025. This work focuses on the ceramic thruster development [7] showing first experimental results, and space debris mitigation aspects for WEP in general and the ROMEO mission in specific.

1. Introduction and Motivation

The Institute of Space Systems (IRS) at the University of Stuttgart is developing a green electrolysis-based water propulsion system for satellites. One primary goal is to replace the commonly used hydrazine for higher thrust propulsion. Due to its extreme toxicity, using hydrazine is a risk for operators and the environment in case of an accident and might be banned in the EU soon. Using water instead, which is decomposed in orbit, increases fuel efficiency, and drastically decreases the handling complexity and costs, making it accessible also to smaller institutions. The water propulsion system in development at the IRS is one of the technology demonstrations on the institute's ROMEO mission, an 80 kg small satellite with a planned launch date in 2025, where the water propulsion system shall be used to reach an elliptical medium earth orbit from a low earth orbit [1]. An overall system prototype with flight-like electrolysis unit and thruster has previously been successfully built and tested at IRS [2, 3]. In the first part this paper shows first experimental test results of the previously presented development of a novel cost-effective 3D printed ceramic thruster with tri-coaxial injector [7]. The second part of the paper deals with different orbit-raising strategies and systems design aspects exclusive for impulsive water electrolysis propulsion systems, which are assessed with respect to ROMEO's special space debris mitigation concept using the institute's WESPAT tool [5]. Since the water propulsion system technology demonstrator is going to raise the satellite to an elliptical medium earth orbit, where, in case of a system failure, it would stay for decades before reentering earth's atmosphere, these strategies target to minimize natural orbital lifetime at all times before further reducing it at the end of the nominal mission and the beginning of the extended operation.

1.1 Motivation

Hydrazine-based chemical thrusters are the standard for satellite propulsion systems [4]. Even though the number of active satellites with electrical propulsion systems is steadily increasing [5], they are not generally a suitable replacement for chemical propulsion systems for every satellite mission, as they come with a high-power demand, which usually drives the satellite's electrical power system dimensions and consequently increases their dry mass. Furthermore, they have a much lower thrust level, which prolongs transfer maneuvers. Most LEO or MEO missions do not have a high Δv-demand, so their increased propellant efficiency is often not worth the drawbacks.

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Today, most chemical thrusters are based on hydrazine, as this propellant offers simple and light systems with relatively high efficiency, either as monopropellant or bipropellant with NTO. Its imminent ban however, further encourages the development of green propulsion systems. Many of them are less toxic than hydrazine and could act as an alternative if the ban took effect. Nonetheless, they would not solve the other downsides, such as the immense costs involved due to the necessary safety measures, making it impossible to use for smaller institutions.

The strength of water electrolysis propulsion (WEP) is that it runs on inert water. Only in orbit the water is decomposed through electrolysis and gradually stored in small separate gas tanks for oxygen and hydrogen. After reaching a certain pressure level, a thrust maneuver can be performed by injecting both gases in the combustion chamber, where ignition is initiated by a catalyst. The fluid schematic for a WEP system is illustrated in Fig. 1.

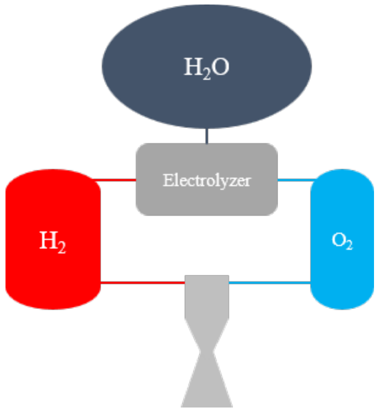

Figure 1: Fluid schematic of the water propulsion system

NASA has investigated water electrolysis propulsion since the 1960s [6] but the first and, until today, only water electrolysis propulsion system in orbit was launched in 2021 on NASA's Pathfinder Technology Demonstrator and is called HYDROS® [8]. It delivers a nominal thrust of 1.2 N with 310 s specific impulse and is operated in the 5-25 W range according to Tethers Unlimited Inc. specifications [9]. The research and development on the thruster at IRS aims to achieve both an increase in efficiency and cost reduction at the same time.

2. Ceramic Thruster Prototype

Building on Harmansa's first flight-like prototype [3] made from Inconel®, illustrated in Fig. 2, the development of the next generation thruster started in 2021. Within a re-analysis phase in combination with the experimental characterization results it was decided to keep the catalytic ignition concept, which performed reliably during Harmansa's test campaign. Also, the hydrogen cooling film concept was kept for the next generation. However, the achieved specific impulse (I_{SP}) of 230-250 s was the main point to be improved. According to a simulation, that was verified by the experimental results, only about 44 % of the gases reacted until the end of the nozzle.

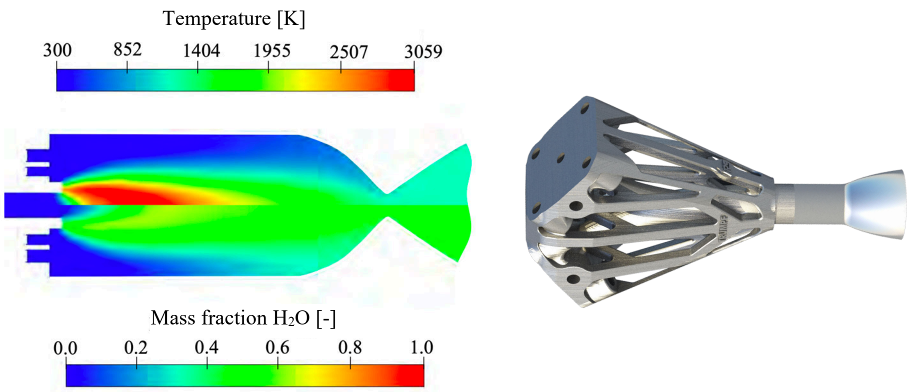

Figure 2: Left: Combustion simulation of former prototype by Harmansa; central oxygen injection, hydrogen injection on the outside; top half shows the fluid temperature distribution, bottom half the water mass fraction; adapted from [2].
Right: Render of former prototype by Harmansa; adapted from [3]

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To increase the I_{SP}, it was tried to increase the combustion efficiency. Therefore, multiple combustion chamber variations were analyzed in simulations, and it could be shown that longer combustion chambers hardly contributed to a more complete combustion while drastically increasing wall temperatures and increasing thermal losses. According to the simulations, the produced water acts like a film barrier and mostly prevents the gases from further mixing and reacting. To increase gas intermixture, a swirl injector could have been an option, but was not further analyzed, as it was assumed that the swirling gas would disturb the hydrogen wall cooling film and therefore lead to too high wall temperatures. Simulations with impinging injectors did not show as good intermixing potential as for liquid injectors. The impinging gas stream is almost directly deflected and develops a laminar-like flow field. Also, injector variations with higher differences in flow velocity were not leading to a promising design. As an alternative, the amount of alternating gas layers was doubled to increase the reactive area between oxygen and hydrogen. Here, a promising design was found, that increased the combustion efficiency to 75 % at the end of nozzle resulting in a theoretical I_{SP} of > 350 s with an expansion ratio of 200. The according illustration, prototype photo and simulation are shown in Fig. 3.

Figure 3. Left: 3D combustion simulation of new thruster generation: H_2O mass fraction; top half with section cut in hydrogen inlet plane, bottom half with section cut in oxygen inlet plane, flow direction from left to right [8].
Centre: Photo of thruster prototype with lab interface.
Right: Section cut of 3D model with focus in injector and combustion chamber (same planes as in left picture).

However, while reducing the cooling film mass flow to further increase the efficiency, the wall temperature increased as well, exceeding the safe operation temperature of Inconel®. Most metals are prone to oxygen at high temperatures and to protect them appropriately, especially close to the throat, a relatively big share of the hydrogen mass flow would have to be invested into cooling. In combination with transpiration cooling in particular, further investigations could potentially extend the material choice. However, platinum could be a very well performing but costly material choice.

For the new thruster generation, it was decided to experiment with unconventional manufacturing methods and materials: 3D printed ceramic. At first, Alumina was analysed due to its high operating temperature of over 1650 °C according to the manufacturer Lithoz. But analysis showed that the thruster would probably break from thermal shock just after ignition. Therefore, the less thermal shock sensitive Silicon nitride, SiAlON, was chosen, that compromises a bit on the maximum operating temperature with 1200-1500 °C. Although more viscous than Alumina slurry, the green part is also LCM printable, a stereo lithographic process, that is highly precise and allows for a very low surfaces roughness R_a of 0.7 µm. More information on the development and the simulations can be found in the corresponding previous publication [7]. To enable the clean printing of injector channel with 0.5 mm diameter, the process needed further adaptation, but was finally archivable by Alumina Systems in cooperation with Lithoz. In Fig 4. on the left a CT scan of the thruster is shown, demonstrating the manufacturing precision. The diameters of the injector channels matche almost exactly the CAD part for manufacturing. The graphic compares to the original part from the simulation with 0.4 mm in diameter instead of 0.5 mm for the small channels.

2.1 Experimental characterization

In 2 Pa vacuum chamber experiments, an I_{SP} of up to 303 s could be measured in steady state firing with a stochiometric mass flow of 315 mg/s (design point) after 30 s, which is the expected maximum burn duration of the ROMEO mission. The center picture of Fig. 4 shows an operating thruster. The laboratory interface setup is illustrated in Fig. 5 on the left while the planned flight-like interface is illustrated on the right. The first prototype already performed > 500 ignition cycles with the same 0.17 g catalyst charge and counting. Furthermore, it passed a sine and random vibration shaker test at ESA standards for both relevant axes. Also extended burn durations of up to 60 s were performed multiple times, reaching an outer wall temperature of > 1 000 °C, where according to the combustion simulations in Ansys Fluent only ~ 800 °C as maximum inner wall temperature were estimated. Besides, this simulation assumed an almost perfect thermal barrier with only 5 W heat conduction just behind the attachment, leading to further deviation between

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simulation and experimental results. In combination with the lab thruster interface and gas supply part the heat conduction can be roughly estimated between 150 and 200 W. On the one hand, this is one potential explanation of the lower I_{SP} compared to the simulations. On the other hand, improving or implementing a thermal barrier would lead to even higher wall temperatures.

Since the small channels were enlarged for easier manufacturing of this first prototype, a simulation with the dimensions from the CT scan were performed and show an I_{SP} loss of ~ 20 s at same wall temperatures, compared to the original simulation. Also it is estimated, that the sealing concept of the laboratory model is prone to leakage, especially after extended operation times.

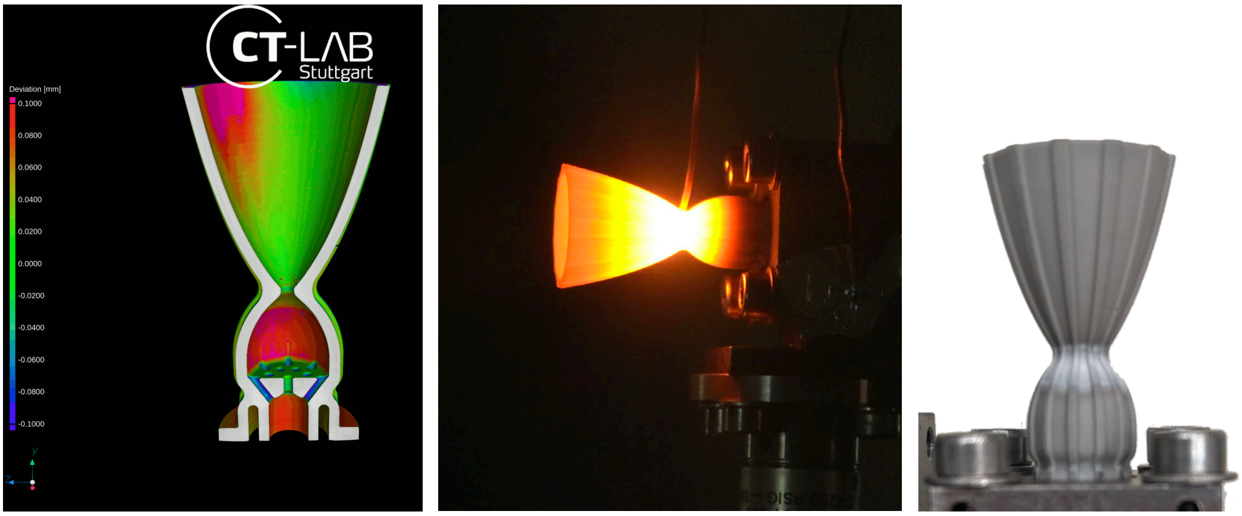

Figure 4: Left: CT scan of 1st ceramic thruster prototype after multiple extended thrust cycles.
Centre: Operation with 315 mg/s stochiometric mass flow in vacuum chamber.
Right: Change in colour at the hot spots after multiple extended operation times.

In order to characterize the gas tank blow down, multiple stoichiometric 40 s burns were conducted with decreasing total mass flow. No clear trend could be identified, as the specific impulse varies between 261 and 303 s down to 63 mg/s of total mass flow. However, when further decreasing down to 36 mg/s, a clear ISP drop below the 200 s mark could be witnessed.

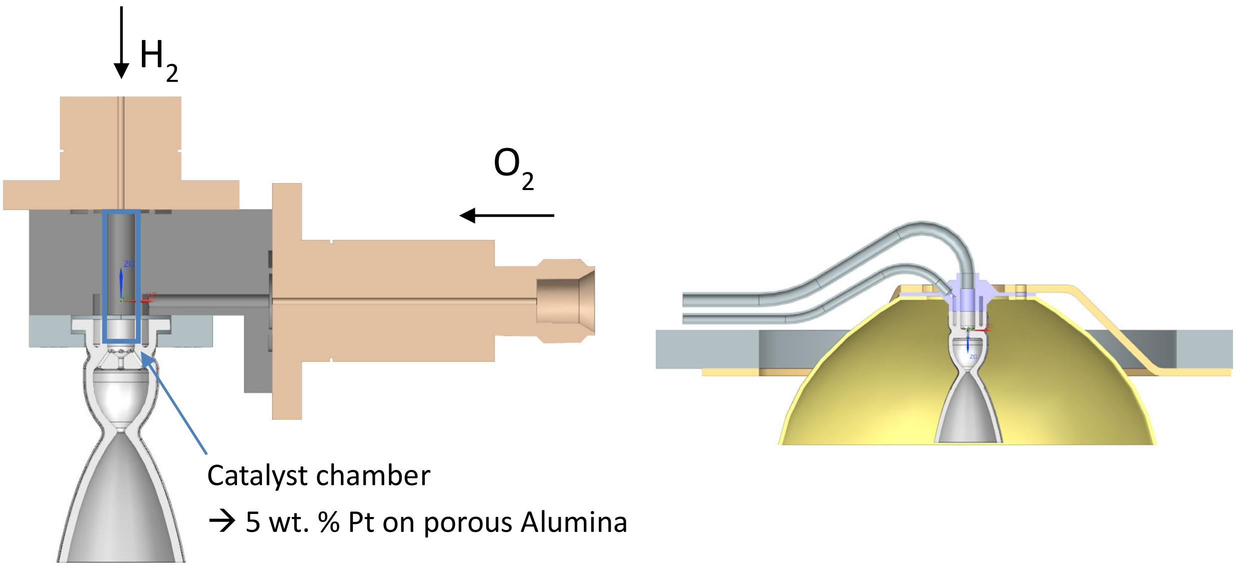

Figure 5: Left: Laboratory interface part with extended catalyst chamber and Sigraflex® flat seals as modular test setup with easy accessibility; high heat conduction and thermal losses.

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Right: Potential setup on ROMEO with a thermal barrier and a miniaturized Invar interface part, the thruster is glass soldered to; in total below 100 g incl. IEP valves from Lee and excl. screw.

2.2 Ignition concept

As stated above, a catalytic ignition concept is used to achieve the planed ~1 000 ignitions during the ROMEO mission. Currently K-0176 from Heraeus is used as catalyst. This is granular highly porous Alumina with 5 wt. % of Platinum particles. This results in a surface area of 90 m^2/g and therefore high reactivity. Harmansa's original, reliable ignition concept with short oxygen pre flow in the order of 15 ms could not be reproduced 1.5 years after his test campaign. This is probably due to slow molecular oxidization of the platinum that reduces the reactivity. However, the catalyst should be reactivated at temperatures above 450 °C [10].

In separate tests with pre-mixed gas in a vacuum chamber, also pure platinum mesh with 0.2 mm diameter and 0.5 g was tested at 10 bar inlet pressure but no ignition could be achieved, in contrast to the K-0176. The current thruster ignites reliably with an adapted ignition strategy: The oxygen pre-flow is routed through the hydrogen feedline with the catalyst chamber by an additional valve for 10-100 ms, depending on the exposure duration to vacuum, to increase the amount of oxygen present in the catalyst chamber. When the 3rd valve is shut, the main hydrogen valve is opened at the same time as the main oxygen valve. The gases ignite and are pushed into the combustion chamber by the hydrogen and reach steady combustion within 7 ms according to a transient combustion simulation.

In the first firing tests, sparking occurred: In the moment of ignition a spark burst could be observed followed by single sparks every few seconds. However, this effect strongly reduced over time. Generally, it is assumed that loosely bound porous particles of the catalyst base material, which is prone to thermal shock, are flaking, but more analysis is still pending.

In separated tests, a leakage path from the O_2 line to the H_2 line was induced at the combustion chamber causing a fuel rich steady pre-combustion in the catalyst chamber. This concept allows a direct ignition without the cold-gas loss of the oxygen pre-flow. Since this ignition concept was promising, the next manufacturing batch will include a lab model where the leakage flow can be measured and regulated for closer analysis and optimization.

2.3 Current development steps

Currently, the 2nd generation of the ceramic thruster is in development, where multiple different thruster variations are going to be tested. Further combustion simulations showed that if the combustion chamber wall falls off steeper to the throat (60° instead of 30°), the wall temperature can be reduced about 300 °C while losing only a few seconds of I_{SP}.

Another current development is the quad-coaxial injector, illustrated in Fig. 5, further increasing the contact area between the two gases to increase combustion efficiency. In this case, the hydrogen supply and catalyst chamber would be on the outside, and oxygen on the inside of the thruster interface. This allows for twice as many channels for the cooling film to achieve a more consistent film. A drawback of the tri-coaxial injector was the alternating pattern of H_2/O_2 injectors which leads to a more uneven film and therefore temperature distribution. Currently, we are iterating mass flow proportions, since we are facing an undesired disturbance of the cooling film caused by the impinging O_2 injector, driven by the motivation to design straight injector channels to improve the manufacturing process. If successful, this design could, in theory, reach up to 370 s according to an adiabatic (!) simulation.

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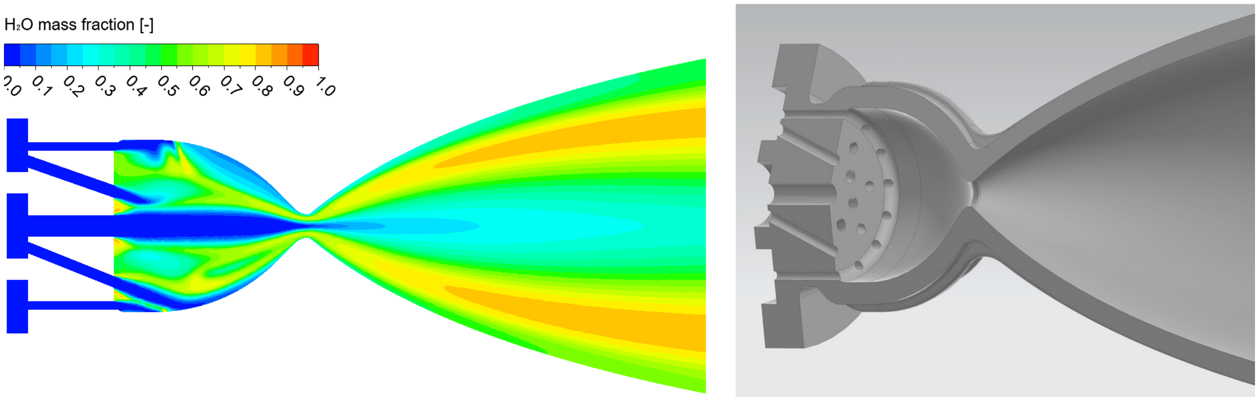

Figure 5. Left: 3D combustion simulation of quad-coaxial injector: H_2O mass fraction; top half with section cut in hydrogen inlet plane, bottom half with section cut in oxygen inlet plane, flow direction from left to right.
Right: Section cut of 3D model (same planes as in left picture).

A further improvement in all variations is the direct accessibility of all injector channels, allowing better inspection and cleaning and therefore further reduction of minimum channel diameters.

Also using a nozzle designed by the Method of Characteristics (MOC) instead of the current parabolic approximation of a thrust optimized contour (TOC) shall be characterized. For the other variations, the nozzle will be cut off to save iteration costs.

In addition, a test campaign is currently prepared to gradually test with wet gases to analyze the impact on performance, temperature, and ignition behavior. However, results from Ariane Group show only little influence during the thruster ramp up phase and almost none later on [11].

3. Space Debris Mitigation Considerations

The satellite population in low earth orbit (LEO) raises drastically, especially due to upcoming mega constellations and the new space approach, supported by the lowering launching costs. The satellite mission ROMEO has the goal to research and demonstrate technologies to promote missions to the medium earth orbit by lowering the costs of the radiation tolerant hardware components. Many of the LEO satellites could also use a medium earth orbit (MEO) but are held back by the high costs of the necessary radiation tolerant hardware.

Even without large constellations, NASA predictions show a huge increase in the LEO population until 2210, but also show the influence of post mission disposal (PMD), as illustrated in Fig. 6. To prevent the Kessler syndrome, agencies and space companies currently discuss and pass regulations, to reduce future space debris. These upcoming requirements will increase the need for propulsion systems also for smaller satellites. Therefore, especially the demand for low-cost alternatives will rise in the near future, boosting the development of green propulsion systems.

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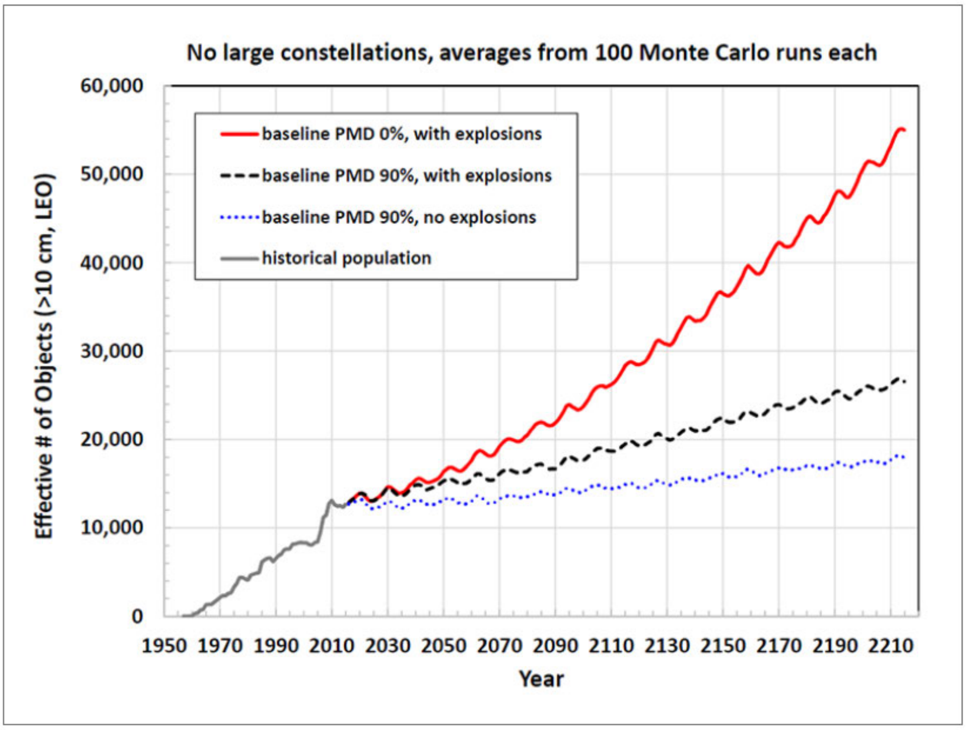

LEGEND-simulated historical LEO environment and results from three different future projection scenarios. Each projection curve is the average of 100 MC runs. The effective number is defined as the fractional time, per orbital period, an object spends between 200 km and 2000 km altitudes. Credit: NASA ODPO.

Figure 6: Prediction of objects in orbit until 2210, where PMD stands for Post Mission Disposal [12]

3.1 ROMEO

The ROMEO mission is taken as reference mission to analyze the effect of different space debris mitigation scenarios. This is a small satellite from IRS with 80 kg mass including ~ 12.5 l of water and scheduled to be launched by the end of 2025. It shall be launched piggy-back in a LEO polar orbit and then by means of the water electrolysis propulsion system, raise its apogee to 2500 km into the MEO. Since the satellite is incorporation many technology demonstrations special precautions are set in place. So, the launching orbit is limited to an altitude, where the satellite would deorbit by its natural decay within 25 years even though the solar panels did not deploy. Also, while increasing the apogee, the perigee is lowered gradually at the same time, to never exceed a natural remaining orbital lifetime of 25 years with 5 years margin. This orbit raising strategy is illustrated in Fig. 7 and the preliminary ROMEO satellite layout itself in Fig. 8.

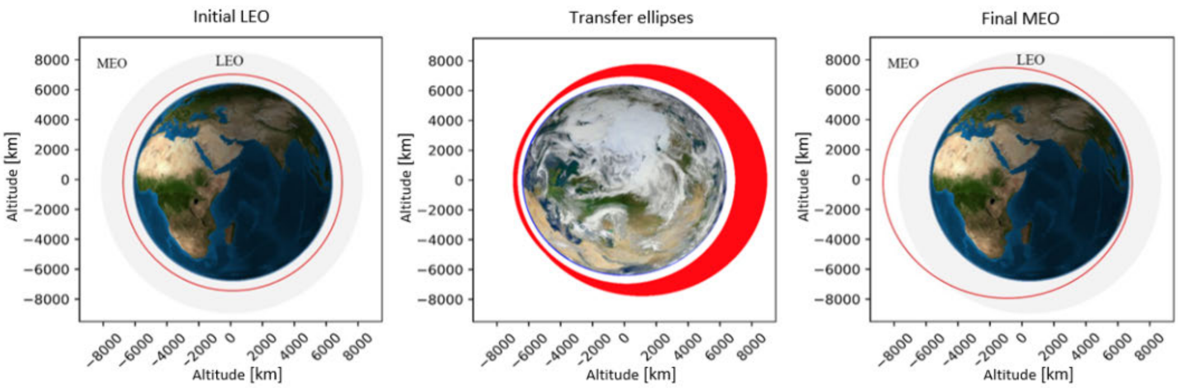

Figure 7: Different phases of the ROMEO orbit during its mission.

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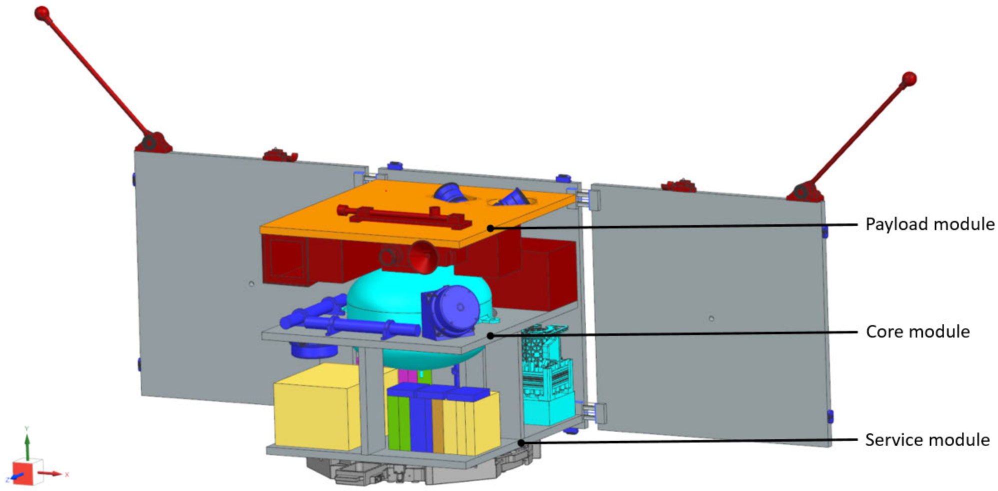

Figure 8: Overview of the ROMEO Satellite System

3.2 Collision probability

For the collision probability calculation, the WESPAT tool [5] is used for the simulation of the transfer with the water electrolysis propulsion system, incorporating DRAMA simulations for deorbit and ballistic phases, and ARES simulations for collision probability calculations.

First simulations showed that a mission with a launch to a 25-years LEO orbit and no maneuvers until reentry the collision probability for objects > 10 cm is 0.00067, as illustrated in Fig. 9, and therefore barely below the requirement of the national space agency of 0.001. The simulation is aborted about 30 days prior to reentry. The diagram shows that the collision probability is highest between orbit altitudes of 560 and 600 km.

If a transfer manoeuvre to an elliptic 2500 km MEO, limited to 25-years lifetime, is conducted without any Lifetime Reduction Manoeuvre (LRM), the total collision probability with objects > 10 cm until reentry is increased to 0.00075, still below the required threshold. However, if a LRM to 10 years is performed one year into the mission, the collision probability is drastically decreased to 0.00032, as illustrated in Fig. 10 and 11 respectively for objects between 1-10 cm. It can be seen, that the collision probability is highest when the orbit apogee is below 1 500 km. Even if the satellite is always crossing the highly populated orbits on its elliptic MEO orbit, the collision probability decreases with further rising apogee, since the time spent in lower orbits decreases.

A LRM to 10 years remaining orbital lifetime, demands for about 185 g of water, while a LRM to 5 years would require about 400 g of water.

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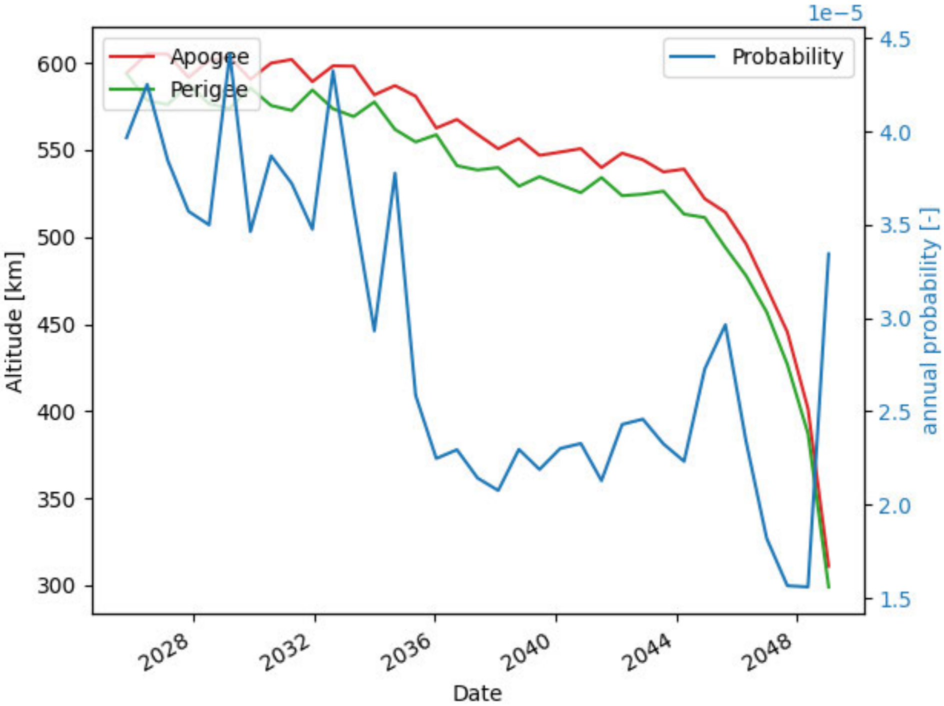

Figure 9: ***Collision probability for objects > 10 cm*** *and orbit trajectory for launch to a ~25-year lifetime LEO with natural decay until deorbit*

Assumptions: 80 kg wet-mass, Cross sectional area on the tumbling satellite of ~ 0.59 m^2.

<u>*Results: P(> 10 cm)*</u> = 0.00067 during orbital lifetime of 23.5 years → 0.001/0.00067= 1.49 (MoS = 0.49)

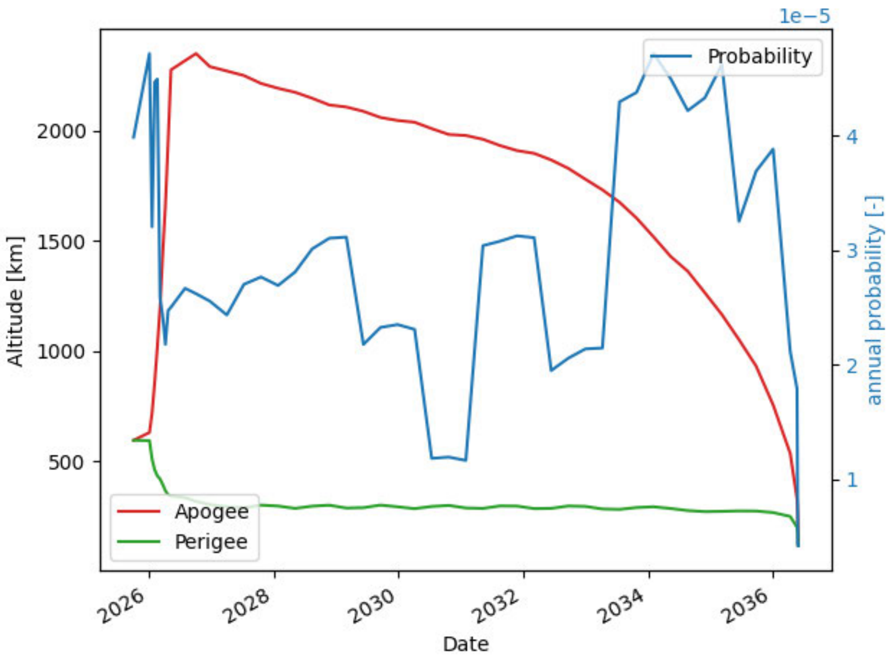

Figure 10: ***Collision probability for objects > 10 cm*** *and orbit trajectory for launch to a 25-year lifetime LEO, Transfer to 2500 km elliptic MEO, LRM to 10 years remaining lifetime.*

Assumptions: 80 kg wet-mass, 130-day manoeuvre phase, LRM by the end of 1st year, de-orbit mass of 68.615 kg, Cross sectional area on the tumbling satellite of ~ 0.59 m^2.

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<u>Results: P(> 10 cm)</u> = 0.00032 during orbital lifetime of 10.5 years → 0.001/0.00032= 3.13 (MoS = 2.13).

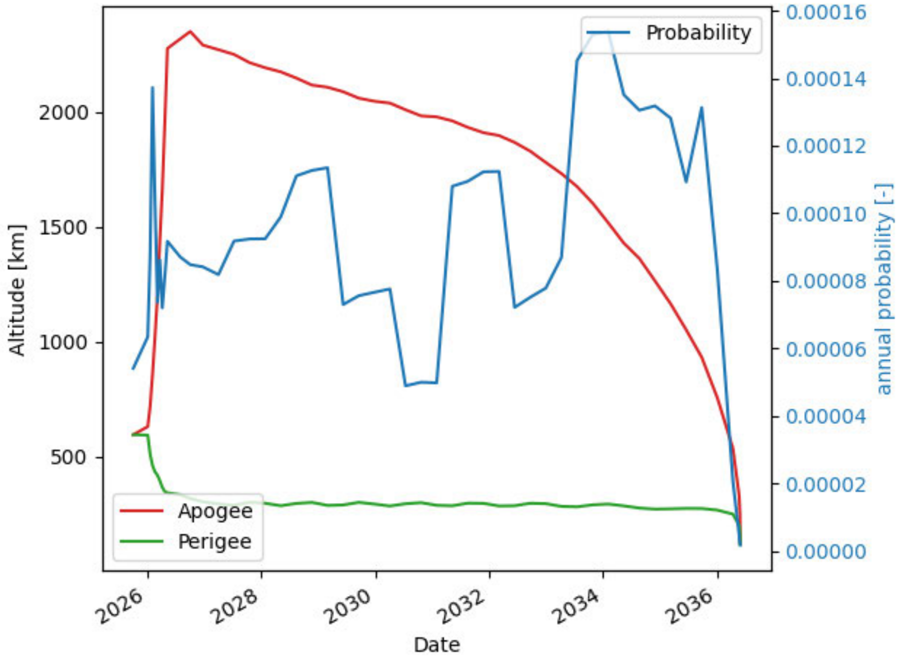

*Figure 11: **Collision probability for objects > 1 cm and < 10 cm** and orbit trajectory for launch to a 25-year lifetime LEO, Transfer to 2500 km elliptic MEO, LRM to 10 years remaining lifetime, same assumptions as in Fig. 9.*

<u>Results: P(< 10 cm)</u> = 0.00102 during orbital lifetime of 10.5 years → 0.01/0.00102 = 9.80 (MoS = 8.80)

3.3 Collision avoidance

ROMEO's water propulsion system will be able to provide collision avoidance (CA). Usually, recommended propulsion maneuvers are in the order of 10 cm/s Δv. Fully filled gas tanks on ROMEO will be able to provide about 65 cm/s Δv in case the thruster could not be further improved. A full charge takes below two hours of electrolysis time at 30 W power and faster if more power is available. It could also be decided to never fully deplete the gas tanks and always keep a reserve, to be able to react instantaneously.

Simulations including safety margins showed that a dedicated fuel budget of 0.24 l covering 8 CA maneuvers per year over 5 years should be sufficient for the planned trajectory.

3.4 Reentry

Controlled reentry maneuvers require for high thrust maneuvers. The respective gas tank size would not be feasible for common satellites. Therefore, an uncontrolled reentry is inevitable for water electrolysis propulsion systems just as for other electrical propulsion systems. This requires minimizing the risk for damage caused by reentering parts. Wherever possible, design for demise is chosen and this is the main reason why IRS is currently developing an aluminum tank instead of using a state-of-the-art titanium tank, since they normally fully survive the reentry due to their shape and the high temperature resistant material. Unfortunately, for all PEM electrolyzer units titanium is the state-of-the-art material choice due to the demanding acidic environment.

Simulations with ESA SARA showed that even if a design was found that allows for immediate disintegration of the electrolysis unit, also the thinnest titanium cell plates only decompose during reentry if they were not covered from the very beginning of the reentry phase. Therefore, a design for demise has to be neglected and design for containment is selected. Even though the risk that debris from the electrolyzer damages humans or human made structures is well

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below the required thresholds, further research to replace titanium should be conducted. This need increases with the amount of such propulsion systems in orbit. At IRS there currently is an ongoing investigation of an alternative electrolysis unit concept allowing to replace the titanium [13].

4. Summary and Outlook

With water electrolysis propulsion, an environmentally friendly alternative to hydrazine is available, which is also suitable for smaller institutions. The previously presented efforts on the ceramic thruster development could finally be experimentally tested and is proven to be fully functional. It was also demonstrated that the thruster withstands the mechanical stress during launch. First test campaigns towards life-cycle simulation look very promising, not only concerning the thruster itself, but also the catalytic ignition concept. The experimental characterization of the laboratory prototype does not yet deliver the performance estimated by the combustion simulations. This might be explained by a combination of small manufacturing adaptations of the injector, thermal losses of the laboratory interface without thermal barrier, and others. The lessons learned are currently transferred in the development of the next ceramic thruster generation, which will also include glass soldered models towards the planned flight interface.

Furthermore, general space debris mitigation analysis, collision avoidance, and reentry considerations of the water electrolysis propulsion system are presented with the ROMEO mission outlining the capabilities and limits of the current developments.

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