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Research and test activities on sustainable space propulsion systems at TU Darmstadt

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Abstract

Sustainable space propulsion has gained increasing interest as more payloads are launched. Growing numbers of objects in the orbit has led to increased collision risk and space debris, endangering satellites and future missions. Compact propulsion systems are needed to maintain space cleanliness. They enable manoeuvrability and safe de-orbiting. Ongoing research and innovation are crucial to meet the growing demand for effective, efficient, and environmentally friendly propulsion systems. These developments aim to ensure sustainability and safety in space operations. This paper presents an overview of the research activities on sustainable space propulsion systems at the Technical University Darmstadt, including green propellants and miniaturized thrusters for CubeSats and micro-satellites.

1. Introduction

Over the past decade, sustainable space propulsion systems have experienced increasing interest and larger numbers of payloads are being launched into orbits [1]. However, their growing number leads to an increased collision risk and the generation of space debris, endangering operational satellites and future missions in the same orbit. As such, there is an urgent need to maintain the cleanliness of space and tackle the challenge of rising space debris by developing compact propulsion systems for these satellites [2]. These propulsion systems are necessary to enable manoeuvrability during a mission and to facilitate de-orbiting either at the end of their useful lifetime or in the event of an unexpected failure.

Currently, space propulsion systems predominantly rely on hypergolic and storable fuels like hydrazine, which are easy to store and do not require external ignition to initiate combustion. However, these fuels are highly toxic and harmful to the environment, making their handling complex and challenging. Furthermore, there is a possible ban for the use of hydrazine and similar fuels in the future due to the EU REACH regulation, which creates an urgent need for sustainable and green alternatives such as hydrogen or methane [3], [4]. The Technical University of Darmstadt's Institute of Gas Turbines and Aerospace Propulsion (GLR) is taking action to develop innovative solutions for the challenges facing CubeSats and other micro-satellites. Amongst others GLR has currently ongoing projects in the following research areas, which are further explained in chapter 3:

- Clean Water Electrolysis Propulsion System, a technology using water as a propellant to create a small thruster system for de-orbiting. Water electrolysis propulsion systems have emerged as a promising and environmentally friendly concept for powering satellites in space[5], [6]. Unlike traditional propulsion systems that use hypergolic and storable fuels such as hydrazine, water electrolysis propulsion systems produce propellant in space by decomposition of water into hydrogen and oxygen through the process of electrolysis. This approach allows production of propellant in orbit only when needed. One concept focuses on a water electrolysis thruster for CubeSats, where very small dimensions are required. However, there are challenges in minimising complex geometries, such as the injector, where 3D printing can be employed as a manufacturing method. Further, the thermal management of the thruster needs to be assessed, as the combustion of the hydrogen/oxygen propellant combination leads to very high combustion temperatures of up to 3000 K.

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- Small-scale Rotating Detonation Engine (RDE): Rotating Detonation Engines have received significant attention in recent years as a compact propulsion system that could provide high thrust and efficiency, resulting in a significantly higher specific impulse, Isp, for space propulsion [7]–[9]. RDEs use detonation, which is a pressure-gain combustion process, resulting in large pressure increases and higher temperatures compared with deflagration in conventional engines. Several physical processes in RDEs are still to be understood. Current research is mostly focusing on larger dimensions, however using small-scale RDEs this technology and its benefits can also be applied for in-orbit space propulsion. However small-scale RDEs remain a challenge due to high heat loss and an increased frequency of the rotating detonation wave. The physical process and scaling effects need to be comprehensively assessed to expedite this technology and its application in the space propulsion sector.

- Miniaturized ignition concepts: As the demand for compact sustainable propulsions systems continues to grow, the development of miniaturized ignition concepts will play an important role in the future of space exploration and satellite technology. The ignition concept/process is a crucial part in the operation of a propulsion systems and can directly affect the reliability of the propulsion system, as a delay or failure of the ignition can damage the thruster and thus directly affect the mission success. Challenges of ignition systems include the time-controlled ignition to ensure efficient use the propellants, and the miniaturisation of ignition concept, especially for the application in CubeSats with their small size limitations.

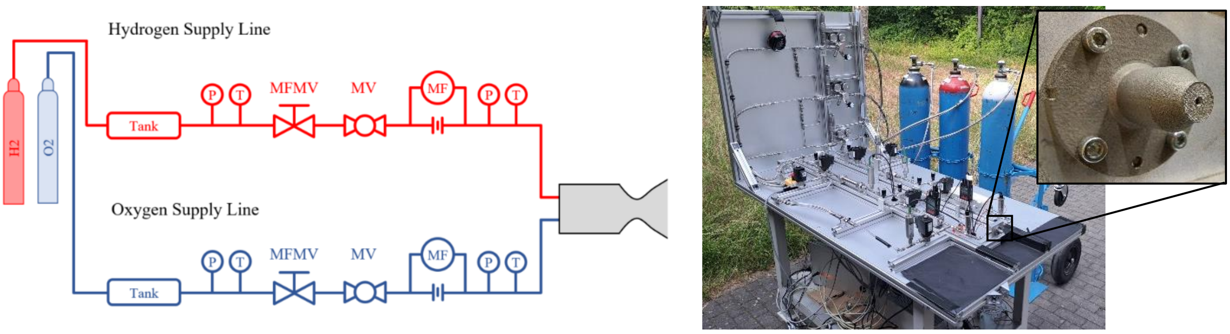
For the purpose of experimental testing of the above-described research activities, the GLR has developed and commissioned a test infrastructure for sustainable space propulsion systems. Two test benches offer test capabilities using green and environmentally friendly propellants over a combined thrust range from 500 mN to 40 N, thus ranging from thrusters for CubeSat applications to those for the AOCS of larger-sized satellites or launch vehicle applications. The GLR sustainable space propulsion test infrastructure is explained further in Chapter 2.

2. Infrastructure

At the Institute of Gas Turbines and Aerospace Propulsion, new test infrastructure for space propulsion systems has been built. These test benches, named PICO and LOTUS, are used for investigating combined cold and hot gas propulsion systems for satellites, with a focus on the use of sustainable and green propellants. Further these can be used for testing of individual components, such as the igniter systems. Additionally, a vacuum chamber (EVA) is available for testing propulsion systems and components under space conditions in vacuum. The individual test benches and experimental setups will be explained in more detail below.

2.1 PICO

The PICO (Propulsion Infrastructure for CubeSat Operations) facility is a test bench designed for the investigation of small propulsion systems for CubeSats. Figure 1 (a) shows the feed system architectures and Figure 1 (b) shows a picture of the PICO test bench. It is a mobile test bench, which is supplied from nitrogen, hydrogen and oxygen bottles. The PICO consists of two main supply lines which are capable to store propellant inside of build in tanks and are controlled through a main valve (MV). The mass flow of each supply line is set through metering valves (MFMV) implemented in front of the mass flow sensors (MF). Furthermore, pressure sensors (P) and Thermocouples (T) are implemented in the tanks and behind the main valve of each line to monitor the test bench during operation. The propellants are ignited using a spark plug.



(a) (b)

Figure 1 (a) Schematic of feed system architecture and (b) Picture of PICO test bench

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PICO allows the analysis of thermal behaviour and the evaluation of performance parameters of small propulsion nozzles with a thrust class of up to 1 N. The test bed is designed as a pressurized system and can therefore be operated in two modes, “steady-state” and “blowdown mode”. The “steady-state” mode enables operation in various operating ranges for extended test durations, allowing for detailed examination of individual components of a propulsion system. The “blowdown” mode simulates the process behaviour of a real satellite propulsion system. For this purpose, the dropping pressure during a thrust event is realized using the built-in tanks in the supply lines. Additionally, the test bench is used for more fundamental analyses of ignition processes. The nominal mass flow rate of the test bench is 0.77 g/s using a gaseous H2/O2 mixture, with a maximum pressure of 20 bar. Table 1 gives an overview of all key specifications of the PICO test bench.

2.2 LOTUS

The LOTUS (Liquid Oxygen Test Unit for Space) test bench extends the test capabilities at GLR into a higher mass flow range. The LOTUS test bench is supplied from gas bottles and enables testing with a larger total mass flow rate of up to 30 g/s. It can be operated with gaseous hydrogen and oxygen and also has a cryogenic cooling unit for operation with liquid oxygen (LOX). It can run with gaseous oxygen, and addition has a cryogenic cooling unit to operate with liquid oxygen (LOX). In addition to gaseous hydrogen, gaseous hydrocarbons can also be used as a propellant, as summarized in Table 1. Like the PICO test bench, the LOTUS test bench is a pressurized system that can be operated in either steady-state or blowdown mode.

Figure 2 (a) shows the feed system architectures and Figure 2 (b) shows a picture of the LOTUS test bench. The feed system consists of four feed line, two main supply lines for Oxidizer and Fuel and two igniter supply lines. The two main supply lines are equipped with a dome loaded pressure regulator (DDR) which allows to control the pressure of up to 45 bar. With these regulators, the characteristic pressure profiles for the different operating modes (steady-state or blowdown) can be realized. Behind the regulator a pressure sensor (Wika S-20) and Thermocouple (Type K) is implemented to monitor the pressure (P) and temperature (T) during operation. The fuel main supply line (red) is equipped with a Coriolis mass flow sensor (Krohne OPTIMASS 6400) to monitor the mass flow (MF), which is set by the pressure regulator and the metering valve (MFMV). Behind the main valve (MV) of the fuel line another pressure sensor (PCB 102B11) combined with a Thermocouple is placed before entering the thruster. The fuel supply line is designed to work primarily with Hydrogen and methane as propellant. All components of the fuel supply line are also designed to work with the fuels (see Table 1), where only minor adjustments needed to be set to convert the supply line for the desired configuration.

The Oxidizer line (blue) has a similar structure like the fuel line but is additionally equipped with a cryogenic cooling unit (CCU), which is located in the middle of the LOTUS test bench. The CCU is a liquid nitrogen cooling bath, where liquefaction of Oxygen is achieved in a copper cooling coil running through the liquid nitrogen bath. To monitor the production of LOX, measuring points are placed up- and downstream of the CCU. When testing with gaseous oxygen is desired the cooper coil can be removed and a metering valve for tuning the mass flow can be implemented. The different Oxidizer configuration are listed in Table 1.

Additionally, the LOTUS test bench contains two igniter feed lines in parallel to the main feed lines. These igniter feed lines provide the same gases as in the main supply lines, but at a reduced mass flow of 0,77 g/s. The igniter feed lines are adapted from the PICO test bench and can be used to implementing and analyse different torch ignition systems in combination with the main thruster to enable research activities across the test facilities. For the secure operation LOTUS is equipped with several nitrogen lines to flush each feed line individually.

The measurement and control system of the LOTUS test bench is based on the Compact DAQ System from National Instruments and is controlled with LabView. All cDAQ components are embedded in actuator and sensor racks which provide the link for energy and data communication between computer and test bench. Furthermore, all actuator and sensor racks are modular and can be exchanged between different test benches to allows maximum flexibility at the GLR test infrastructure.

Table 1 Specifications of test infrastructure PICO and LOTUS

Parameter	**PICO**	**LOTUS**
Massflow	0,77 g/s	30 g/s
Pressure	20	5-45 bar
Fuel	GH2, GCH4	GH2, GCH4, (GC2H6, GC3H8, Ethanol)
Oxidizer	GOX	GOX, LOX, (GN2O)

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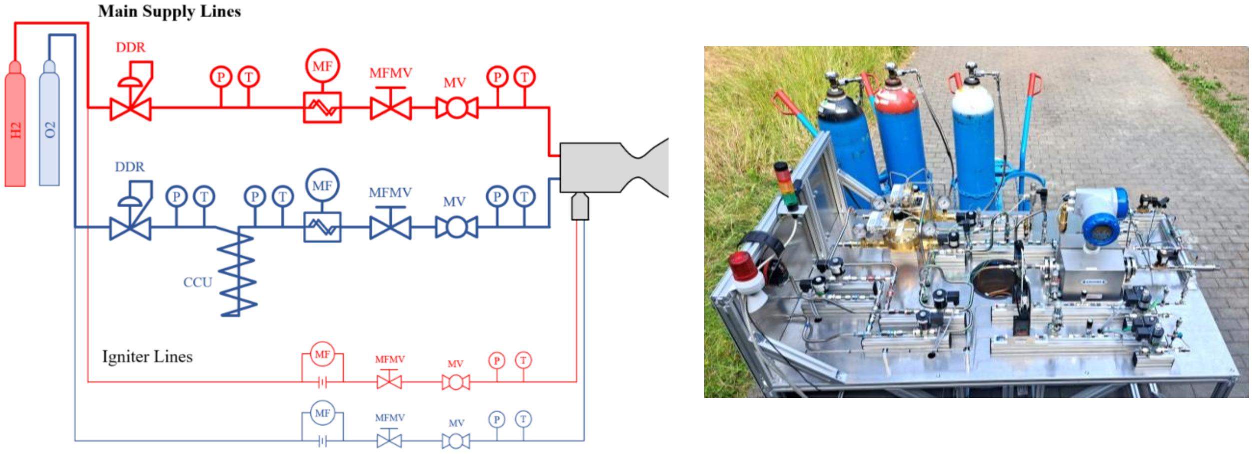


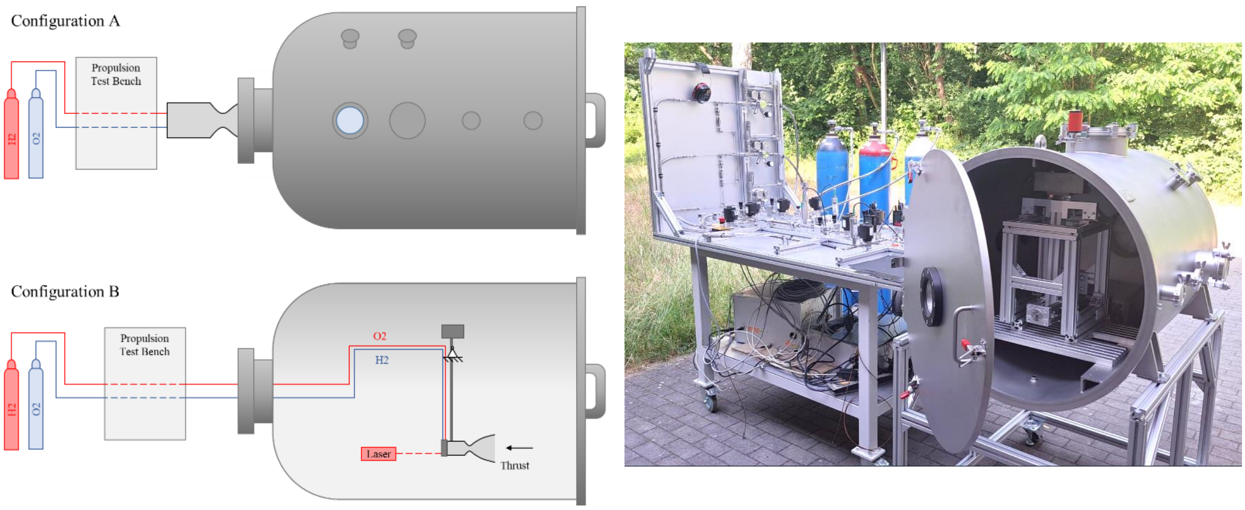
(a) (b)

Figure 2 (a) Schematic of feed system architecture and (b) Picture of LOTUS test bench

2.3 EVA

The EVA vacuum chamber allows for testing propulsion systems and individual components under vacuum conditions, thus simulating operation in space environment. The EVA vacuum chamber has a volume of 0.4 m^3 and can achieve a vacuum level of <10 mbar. The chamber is evacuated by a HiScroll 18 manufactured by Pfeiffer Vacuum.

EVA can be run in two different configurations for thruster testing, configuration A and configuration B. Figure 3 shows a schematic overview of the test setup and Figure 3 shows a picture of the PICO test bench with the EVA chamber arranged in configuration B. EVA can be combined with both available propulsion test bench, the PICO and the LOTUS test bench. During configuration A, the propulsion test bench is directly attached over the thruster, which is flanged to the EVA chamber. In this configuration, the thruster remains outside of the chamber, thus making this a simpler setup, as no fluid- or electrical feedthroughs into the vacuum chamber are required. In configuration B, the thruster is placed into the chamber, either on its own, or mounted on a thrust balance as shown in Figure 3. This setup requires feedthroughs (fluid supply, electrical wiring, thermocouples) into the vacuum chamber to supply and control, and measure different quantities on the thruster inside the chamber. Only configuration B allows thrust measurements, for performance analysis and determination of the Isp of the propulsion system under operating conditions. An additional feature of EVA is the optical access for cameras and schlieren imaging, in order to obtain more data and a better controllability of the test setup.



(a) (b)

Figure 3 (a) Schematic overview of both configurations A and B for the test setup, (b) Picture of the PICO test bench with the EVA chamber and the thrust balance

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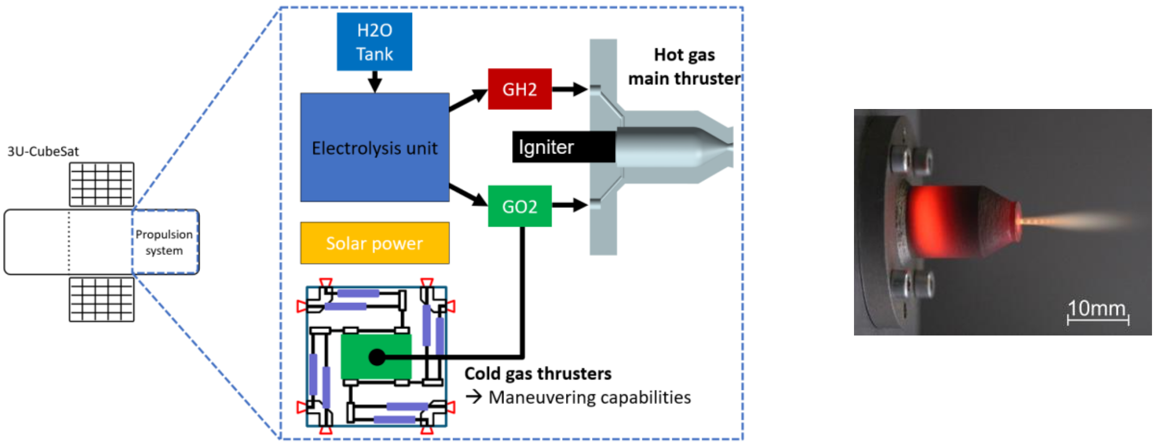
3. Research activities

The Technical University of Darmstadt's Institute of Gas Turbines and Aerospace Propulsion (GLR) is taking action to develop innovative solutions for space propulsion systems and sub-components. The main objective is to focus on sustainable solutions by using green and environmentally friendly propellants, such as hydrogen and oxygen.

3.1 Water Electrolysis Propulsion

Water propulsion has emerged as a promising technology in the field of space propulsion, with several different technologies and propulsion concepts being demonstrated [5], [6], [10]. The primary approach to use hydrogen and oxygen as a propellant, generated through an onboard electrolysis process is not new. However, in view of a search for an alternative to toxic and hazardous hydrazine or hydrazine-like propellants, water propulsion has gained more and more relevance in recent years. Water Electrolysis Propulsion systems come with additional complexity, mass and power demand, but offer safety, easy storability, high energy density and a good performance in terms of Isp. The multitude of concepts being developed currently highlights the potential benefits that this technology can bring to the space sector. These efforts showcase the growing interest and research in water propulsion technology, and the potential it holds for advancing space propulsion capabilities.

The goal of this technology is to design a water electrolysis propulsion system for a 3U CubeSat, showcasing the broad application cases of this technology. This includes a main thruster providing thrust for a de-orbit event, while several smaller cold agas thrusters provide manoeuvring capabilities. Figure 4 (a) shows the schematic of the sub-newton propulsion system design, which consists of an electrolysis unit feed from a water tank to generate gaseous hydrogen and oxygen with the onboard available solar power. Both hydrogen and oxygen are fed to the main hot gas thruster with near stoichiometric mixture ratio, whereas small portions of oxygen can be used for of cold gas thrusters as an attitude control system (ACS). The main focus of our current research is on the main hot gas thrusters. In addition, subsystems such ignition concepts need to be carefully considered in the development of a thrusters. Our activities in this field, with current design and first results are described with more detailed in chapter 3.2.



(a) (b)

Figure 4 (a) Schematic of a Water Electrolysis Propulsion System on a 3U CubeSat (b) Hot fire testing of the thrusters

The current thruster design is focusing on a simple impinging injector design, using a spark plug as ignition method, see Figure 4 (a). The injectors orifice for the H2 and O2 propellant injection have a diameter of 0.6 mm and 0.8 mm, respectively. Especially the challenges posed by a near stoichiometric combustion, with hot combustion temperatures of up to 3000 K need to be addressed. It is design as a capacitively-cooled thruster, 3D printed using a high-temperature material, Inconel 718. During an initial test campaign aiming at a thermal analysis of the outer wall temperature field of the 3D printed thruster, tests were carried out using thermocouple measurement at various axial, circumferential and radial positions on the thruster. Figure 5 (a) shows the thruster with the varying circumferential positions, consisting of 8 equally spaced positions. Additionally, the figure shows the injector position for oxygen and hydrogen, with the oxygen injector orifice being positioned at 90° and the hydrogen injector orifice being positioned at 270°, with 0° being the top position as reference.

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Figure 5 (b) shows the temperature distribution of two identical 3D printed thruster (Thruster 1 and Thruster 2), where a varying temperature distribution can be seen. Thruster 1 predominantly shows higher temperatures near the oxygen injector orifice, whereas thruster 2 has increased temperatures near the bottom region and the hydrogen orifice. This difference is likely caused by an imbalanced injection in the combustion chamber. Minor geometrical changes in the injector holes between thruster 1 and 2 can be caused through local imprecision or roughness of the 3D printed holes. These can already result in a change of the alignment of the impinging injector design and thus in a changed flow pattern of the propellants.

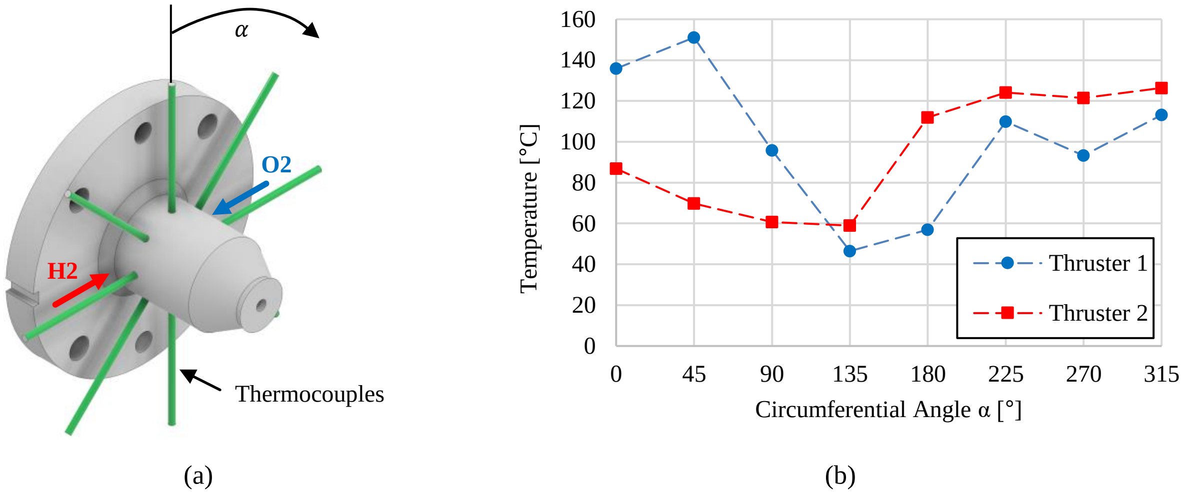

Figure 5 (a) Thruster with circumferential positions for temperature measurement, (b) Outer wall temperature distribution for thruster 1 and 2

These simple/early test results already highlight the importance of precise analysis of miniaturized propulsion systems, such as needed for CubeSat operations. In these systems geometrical alignment and precision becomes more important to achieve an efficient propulsion system. Our group will investigate further studies on these miniaturized design aspects such as injector or novel cooling concepts, utilizing 3D printing.

3.2 Small-scale rotating detonation engine

Rotating Detonation Engines (RDEs) have gained significant attention in recent years for their potential to deliver high thrust and efficiency in a compact form factor. Unlike traditional engines that use deflagration combustion, RDEs utilize detonation combustion, which theoretically enables them to achieve significantly higher Isp values. Detonation combustion is a pressure-gained process that results in substantial pressure increases and elevated temperatures, making RDEs promising candidates for airbreathing engines and space applications alike. When paired with an aerospike nozzle, RDEs can offer additional performance gains for launch vehicles [11], [12], while smaller versions can be utilized for in-orbit applications. The increasing interest in RDEs over the last years has fuelled ongoing research into the fundamental physical processes of detonation technology, and also one successful sub-orbital launch utilizing RDEs has been reported [13].

However, achieving RDE applications across a wide range of sizes necessitates a comprehensive understanding of the underlying physical processes occurring in RDEs, to develop an engine with a controlled detonation wave characteristic. Only when these processes can be comprehensively understood and effectively controlled, can the full efficiency gains of this technology be realized. These processes need to be understood across various size scales to cover a wide range of mission requirements, from launcher applications to smaller in-orbit propulsion systems. For instance, especially small-scale RDEs pose challenges such as high heat loss and increased frequency of the rotating detonation wave. Our current focus is on the design and development of a small-scale RDE to investigate its efficiency. Specifically, our contribution will centre around assessing scalability effects, as we conduct research and development activities on small-scale RDEs and strive to understand the technical challenges involved, such as high heat losses and increased frequency of the rotating detonation wave, which pose additional control challenges.

The GLR institute is currently working on the design and development of a small-scale RDE to explore their use for space missions where their compact size can provide significant benefits. In particular, we are pursuing a modular engine using a hydrogen/oxygen propellant combination with low mass flows of 5-20 g/s. The aim is to identify key parameter that dominate the physical processes in small-scale engines and investigate their efficiency. Figure 6

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shows a schematic of the designed RDE, consisting of three main components, a chamber, the injector plate, and a fuel dome. A modular approach allows exchange of individual components to offer maximum flexibility for a research engine. The chamber is designed with a diameter of 19 mm and a length of 55 mm, which are estimated from the critical equations for annular RDEs from Bykovskii et al. [14]. The chamber is primarily hollow, but allows the option of a small centerbody, which makes it a semi-annular configuration. Figure 6 shows those two possible configurations. The hollow chamber provides a lighter and simpler design. Although it has reduced cooling needs, as no centerbody exist, it can come with the drawback of slightly lower performance [15]. Implementing a small removable centerbody can balance these effects, as is creates a potential reduction of deflagration combustion in the centre of the chamber [16] and acceleration of the exhaust of the burned gases , while being small enough to not require active cooling. The injector plate features an impinging injector design with unlike doublet impinging orifices to achieve good mixing. A spark plug is chosen as the ignition system of the small-scale RDE, due to its simplicity and successful use in other experimental research RDEs [18], [19].

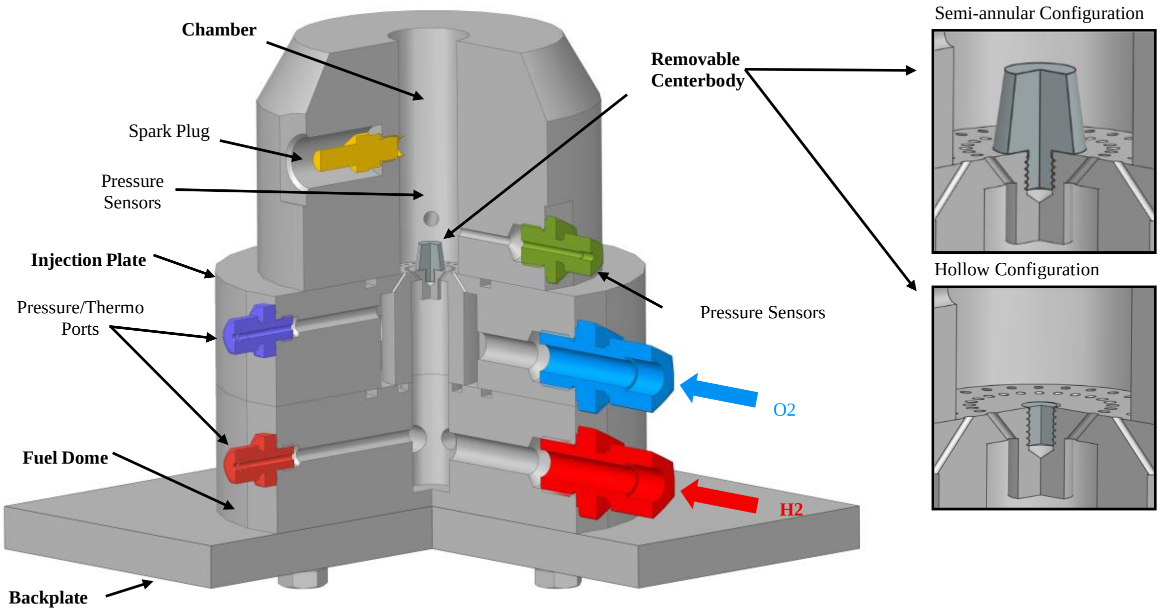

Figure 6 Small-scale RDE with Centerbody options for semi-annular and hollow configuration

The next stage in the design and development process focuses on the final assembly and demonstrate successful ignition and the proof of a detonation waves inside the chamber. The small-scale RDE is equipped with suitable diagnostic methods to analyse the rotating detonation waves with frequency of around 30 kHz. Using a pressure transducer in an Infinite Tube Pressure (ITP) configuration, the detonation wave frequency can be detected. Through the ITP configuration, the transducer is protected from direct exposure to the harsh environment inside the chamber, yet allows to identify the passing detonation wave [20]. A microphone with a frequency range up to 100 kHz is used as a secondary detonation detection method. The first feasibility test campaign will provide a qualitative proof of detonation in our small-scale RDE using mass flows of 5-20 g/s with a hydrogen/oxygen propellant combination. We aim to map the operating range of the small-scale RDE with reference to different mass flows, ROFs and centerbody configurations.

3.3 Miniaturized Resonance Ignition

As the demand for CubeSats and other micro-satellites continues to grow, the development of miniaturized ignition concepts will play an important role in the future of space exploration and satellite technology. The ignition system is a crucial part in the operation of a propulsion systems, as delay or failure of the ignition can directly affect the mission success, thus can directly affect the reliability of the propulsion system. Commonly used sparkplugs are not always reliable and designing efficient and reliable ignition systems especially for these small satellites poses significant challenges due to their size limitations. We are currently working on novel miniaturized ignition concepts, such as resonance ignition, catalytic ignition, and a non-thermal plasma ignition concept. In this study, first activities and results for miniaturized resonance ignition shall be presented.

Resonance ignition uses an acoustic principle in compressible fluids with an oscillating shock to achieve a temperature increase above the self-ignition temperature of a fuel-oxidizer mixture [21]. As it is a passive ignition

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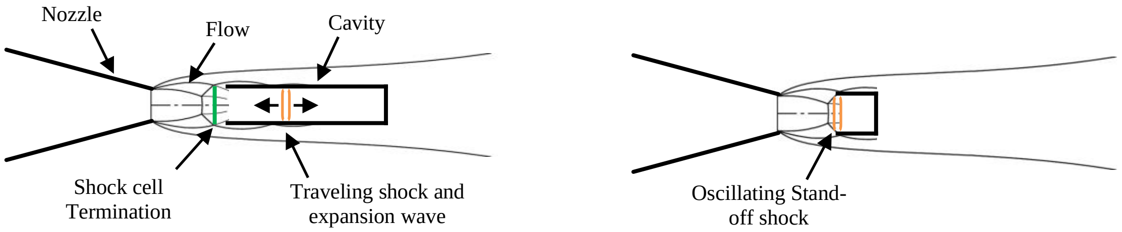
concept with a simple geometry, it has gained interest for aerospace applications. A resonance igniter consists of a nozzle and cavity configuration, with current research in this area is focusing on larger igniter systems with cavity diameters of 2.1 mm [22], [23]. For resonance igniters in propulsion systems for small satellites, such as CubeSats, smaller dimensions are required. However, with a reducing size of the system, the surface-to-volume-ratio is increased, leading to energy dissipation into the wall material of the resonance tube, which can affect the efficiency of the systems [24]. Our current activity is to characterise a miniaturized resonance igniter, for the use with green and environmentally friendly propellants, as used in the above presented water electrolysis propulsion system and RDE. The aim of our activities is to demonstrate a sufficient temperature increase above the auto-ignition temperature of a hydrogen/oxygen mixture of 540 °C [21]. Our research is aiming to identify the dominant physical principles, which result in a temperature increase and identify the relevant design parameters of the nozzle and cavity configuration.

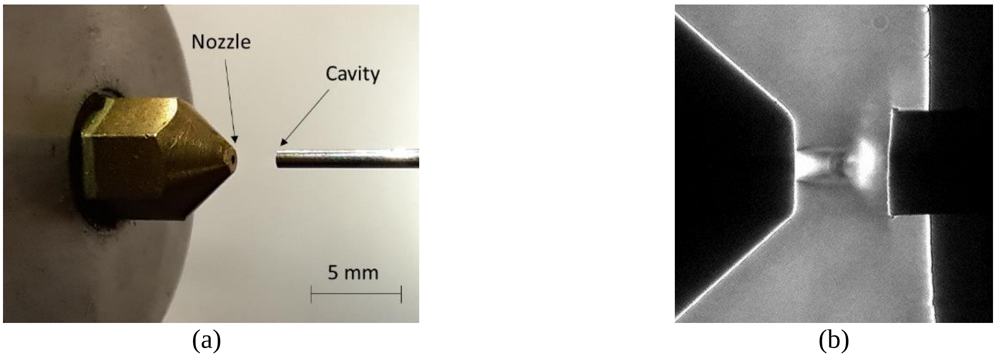
Figure 7 shows a simplified schematic for a resonance igniter system. Through the nozzle an underexpanded jet forms at a sufficiently high nozzle pressure ratio (NPR). The underexpanded jet consists of a structure of expansion waves, shock waves and Mach disks. Depending on the exact position of the cavity in this structure, different operation modes, namely jet regurgitant mode (JRM) and jet screech mode (JSM) can be achieved [25]. The JRM mode occurs when the cavity is position in the compression zone of the underexpanded jet, where the shock forms upstream of the cavity. Compression waves coalesce into shock if the cavity is sufficiently long, thus triggering a series of filling and emptying phases. The JRM mode is therefore characterized by a gas exchange within the cavity, where the frequency is defined by the quarter wavelength theory [25]. The JSM mode takes place when the cavity entrance is located at the shock cell termination in the jet structure, where a bow shock (stand-off-shock) forms in front of the cavity. However, this shock does not enter the cavity, but oscillates in front of it at a higher frequency, compared to the JRM mode.



(a) (b)

Figure 7 Operating modes for resonance ignition (a) jet regurgitant mode (JRM) (b) jet screech mode (JSM)

The final temperature increase of a resonance igniter is dependent on the exact position, but also on the length and geometry of the cavity. A tapered, instead of a cylindrical geometry can have a positive effect on the temperature increase [21]. However, this poses additional manufacturing challenges especially for miniaturized systems, thus this study is focusing on cylindrical geometries. Figure 8 (a) shows the nozzle cavity configuration, with a nozzle and cavity diameter of 0.5 mm. The cavity length is varied between 5 and 15 mm to identify the necessary length for a maximum temperature increase. The end of the cavity is formed by a thermocouple to measure the temperature increase of the gas. The cavity material is varied with stainless stell and glass. The material glass is chosen in this experiment to assess the dependence of a temperature increase on the thermal conductivity of the cavity material. The tests are carried out on the PICO test bench, with an added cavity holder allowing precise alignment variation of the nozzle cavity configuration. The first testing stage uses nitrogen as a gas to identify optimal operating and geometry parameters for a miniaturized resonance igniter. The consecutive second test phase is carried out with hydrogen, to assess the actual temperature increase with the optimised parameter for the ignited system from the first phase.



(a) (b)

Figure 8 (a) Nozzle and cavity configuration for resonance ignition, (b) Schlieren image with shock structure during operation of resonance ignition system

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During initial test campaigns we were able to realise the precise alignment of the nozzle and cavity configuration using a micrometer linear precision setup in combination with a Schlieren setup. The build Schlieren setup allows optical analysis on the GLR test facilities for thruster and sub-components and can achieve a resolution of 11 µm. For the resonance ignition activities, it is used to visualise the underexpanded jet structure and precise positioning of the cavity, as can be seen in Figure 8 (b). With this diagnostic, the cavity can be positioned such that the two different operation modes, JSM and JRM, can be identified. Using a microphone with a frequency range up to 100 kHz, the acoustic characteristic of the resonance ignition system is measured. Overall, using glass has cavity material has yielded to larger temperature increases. Further the optimal cavity length could be identified to be 12.5 mm. The JSM mode is identified by larger frequency with 93 kHz, however only a temperature increase of up to 100 °C was achieved. The JRM mode is characterised through a lower frequency of 11.9 kHz, but with a distinct temperature increase of up to 420 °C, reaching close to 77 % of the auto-ignition temperature of 540 °C.

Further studies for the resonance igniter system will focus on varying cavity geometries, to achieve temperature increases above the auto-ignition temperature. This will involve a variation of the cavity geometry. Using tapered instead of cylindrical cavity and an assessment of the cavity inlet geometry. The cavity inlet geometry has a significant influence on the initiation and sustenance of oscillation, therefore further optimisation can yield to larger temperature increases [26]. In addition, numerical studies on resonance ignition systems are currently carried out at the TU Darmstadt as well [27].

4. Conclusion

This paper presents the current research activities on sustainable space propulsion systems at the Technical University Darmstadt. The available test infrastructure for cold and hot gas propulsion systems for satellites has been presented. This test benches can be used for testing of thruster systems using H2/O2 as a green propellant combination, as well as testing of individual components, such as the igniter systems. The test infrastructure is used to advance the research activities in the field of water electrolysis propulsion, with focus on thermal management of 3D printed thrusters. A small-scale rotating detonation engine was presented which aims to demonstrating successful ignition and detonation for very low mass flows using a hydrogen/oxygen propellant combination. The activities focusing on resonance ignition as a miniaturized ignition method for space propulsion system has the potential to be applied to both a water electrolysis propulsion system and a rotating detonation engine.

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