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Publication date
2020

Document Version
Accepted author manuscript

Published in
Aerospace Europe Conference 2020; BORDEAUX, FRANCE, 25-28 February 2020

Citation (APA)
Franken, T., Valencia-Bel, F., Jyoti, B. V. S., & Zandbergen, B. (2020). Design of a 1-N monopropellant thruster for testing of new hydrogen peroxide decomposition technologies. In *Aerospace Europe Conference 2020; BORDEAUX, FRANCE, 25-28 February 2020* Article AEC 2020-0237

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DESIGN OF A 1N MONOPROPELLANT THRUSTER FOR TESTING OF NEW HYDROGEN PEROXIDE DECOMPOSITION TECHNOLOGIES

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KEYWORDS: Alternative technologies, Catalyst, Decomposition, Green Propulsion, Hydrogen Peroxide, Monopropellant

ABSTRACT:

Since there is a high interest in the use of green propellants, hydrogen peroxide is coming back after once making place for the rise of Hydrazine in monopropellant propulsion systems. Typically, these thrusters are outfitted with catalyst beds. A fully modular 1N thruster is designed to provide the capability of testing and comparing the performance of different concentrations of hydrogen peroxide, different catalysts as well as new technologies in an attempt to resolve the disadvantages associated with the use of catalyst beds. A preliminary baseline design of a catalytic thruster has been created. This will be followed by the design of a secondary decomposition chamber for new technologies, a propellant feed system, a test setup and a test plan.

1. INTRODUCTION

Hydrogen peroxide has been around as a rocket propellant for several decades. It was popular as a monopropellant in reaction control thrusters until the discovery and technical viability of Hydrazine [1], which has an improved performance over hydrogen peroxide. However, in 2011 the European Commission has added Hydrazine to its candidate list of "substances of very high concern" in its Registration of Evaluation Authorisation and Restriction of Chemicals (REACH) framework [2], as Hydrazine is very toxic. This means that there is a risk that the use of Hydrazine will be prohibited in the near future. As this causes an interest in low-toxicity "green" alternatives, hydrogen peroxide is considered again [3], though it struggles with a bad image based on anecdotal collections of accidents as well as the way it was treated in textbooks, emphasizing the disadvantages such as its stability according to [4]. Using a green propellant reduces risk to personnel handling the propellant, simplifies and reduces the duration of overall system operations and reduces recurring costs associated to the system and handling of the propellant [5]. Additionally, other advantages are recognized such as its high density. Though a vast amount of research already exists on the topic the European Space Agency (ESA) and staff from the Space Systems Engineering department at Delft University of Technology have expressed an interest in the development of new, more efficient technologies to improve the performance of hydrogen peroxide based systems to compete with existing Hydrazine thrusters [3]. This includes research into increasing the propellant concentration for a higher density and specific impulse, performance effects due to stability and storability issues of hydrogen peroxide and the possibility of decomposition without the use of a catalytic bed, possibly allowing for the simplification of thrusters. To support this goal a 1N class propulsion system and test setup is designed capable of using 87.5% and 98% hydrogen peroxide as its propellant, in which parts can be exchanged to test different technologies and to compare the performance when using different concentrations of propellant. This approach will help to develop and demonstrate a re-usable system for different propulsion applications.

2. THEORETICAL STUDY

To support the design of the propulsion system and test setup a literature review was performed on the use of hydrogen peroxide in monopropellant thrusters. This includes the review of technology concepts for the decomposition of hydrogen peroxide.

2.1. Hydrogen peroxide as a monopropellant

Hydrogen peroxide is generally available in the form of a solution with water. For propulsion purposes high concentrations are used, commonly referred to as High Test Peroxide (HTP). Usage of HTP as a monopropellant relies on its exothermic decomposition into water and oxygen to form a hot gas as represented in Eq. 1 [6].

$$H_2O_{2\,(l)} \rightarrow H_2O_{(l)} + \frac{1}{2}O_{2\,(g)} \qquad (1)$$

This decomposition reaction occurs naturally at a low rate. The reaction is accelerated with increasing temperature or in the presence of contaminants [7]. These properties can be utilized to generate the hot gas flow required for the functioning of a monopropellant thruster. Unfortunately, these properties affect the stability of the propellant during storage as well. Decomposition during storage can be minimized by choosing compatible materials as well as proper passivation of these materials.

2.2. Decomposition technologies

I. Catalyst beds

Traditionally monopropellant thrusters are outfitted with catalyst beds. Most commonly they are of the so-called particulate type. A support structure is used onto which an active catalytic material is deposited. These catalyst beds are usually held in place by a perforated retainer plate and made mainly by using pellets or stacking screens [8]. Some of the major disadvantages of these catalyst beds are related to crushing and abrasion, which leads to attrition [9]. In an attempt to improve performance of catalyst beds, monolithic versions have been developed in the form of foams as well as more advanced structures. The manufacturing process of these structures has been improved significantly with the rise of additive manufacturing as has been shown in [10] and [11]. For the purpose of this project in the baseline design a catalyst bed is used based on pellets, because of its simplicity and the availability of information relative to other types of catalyst beds.

II. Thermal decomposition

An alternative to the use of catalyst beds is a concept that uses the thermal decomposition properties of HTP. This concept is not new, as several attempts have been made to build or model a thruster with this capability. In [12] a model was created using 90% HTP, in which a secondary flow was injected after the main flow passed a traditional catalyst bed. It was concluded that, due to the speed of vapor-phase decomposition at the expected chamber temperatures and evaporative cooling effects, large reaction lengths were required, limiting the appeal of such a thruster. For a similar concept a CFD study was performed in [13], where it was found that for high concentrations of HTP the decomposition is vigorous and uncontrollable. In case of this design project however, the concept of interest is one that completely removes a catalyst bed from the thruster. Such a thruster was designed in [14] and uses a heating coil in the decomposition chamber to heat the propellant flow. In testing, a steady state operation was achieved and maintained.

A preliminary, first order assessment has been performed to estimate the required energy input into the propellant to achieve a successful decomposition. According to [15] thermal decomposition of hydrogen peroxide starts when a temperature of approximately 150 degrees Celsius (boiling point at atmospheric pressure) is achieved. At a pressure of 5 bar (design minimum inlet pressure) the boiling point of water is at 151.8 degrees Celsius [16]. Therefore, no phase changes were considered in the calculation. Enthalpy values for water have been taken from [16] and Enthalpy values for hydrogen peroxide have been taken (and linearly extrapolated for temperatures over 400 degrees Kelvin) from [7]. The required power is calculated using Eq. 2.

$$P = \Delta H \cdot \dot{m} \qquad (2)$$

In which P is the power in Watt (W), ΔH is the enthalpy change in Joules per gram (J/g) and $\dot{m}$ is the mass flow in grams per second (g/s). Mass flows have been calculated from specific impulse values calculated using RPA Lite using a nozzle designed for optimum expansion at a pressure of 1 *atm*. The results of the calculation have been displayed in Fig. 1. This shows that to heat the complete propellant flow from 0 to 150 °C with 100% efficiency, a power requirement of approximately 280 to 425 Watt is expected, depending on the chosen chamber pressure and HTP concentration. The power requirement could be reduced if heating is focused on a small part of the flow, of which the decomposition heat could cause complete decomposition of the propellant flow.

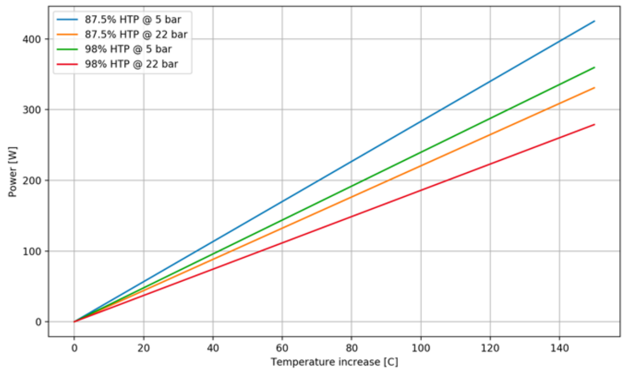

Figure 1. Power requirement for thermal decomposition

III. Laser decomposition

Already in 1994 NASA performed experiments on laser ignition in liquid rocket engines [17]. One of the advantages of using a laser ignition identified by [18] is its potential for a very small ignition delay. Lasers are interesting for reaction control systems as they are capable of operating in a cycle pulse mode according to [19]. They identify two ignition methods: laser breakdown and laser ablation. If the energy released in a laser spark is large enough this could be used to start a self-sustaining HTP thermal decomposition reaction.

As a 1N reaction control system is rather small, a laser ignition device loses its appeal if it is too large. In [20] a successfully miniaturized laser ignition system called HiPoLas (by CTR) was used for a large number of reliable ignitions of a LOX/H2 combustion chamber. Moreover, it is suggested in [19] that a single centralized laser unit in combination with some kind of distribution device would be capable of providing an ignition source for multiple thrusters in a single reaction control system. Successful laser transportation has already been performed in [21], making this an interesting prospect for the future of reaction control systems.

IV. Spark and glow plug ignition

In [22] the explosive characteristics of HTP vapours were tested by using a hot wire or spark gap as ignition device. The result of these experiments was the determination of an ignition limit. This is the minimum concentration of hydrogen peroxide required in the vapour to successfully ignite it. It was concluded that for pressures higher than 2 *atm* (at least until 6 *atm*), the ignition limit remains constant at 20.7%. The temperature required to achieve such a vapor concentration was calculated at different (chamber) pressures using activity coefficients as calculated in [23]. As is seen in Fig. 2, the temperature required for 98% HTP to achieve the ignition limit is over 150 degrees Celsius at 5 bar, which is higher than the estimated temperature required to start a thermal decomposition reaction. The required temperature increases further with increasing pressure. The option of using a spark or glow plug for the ignition of vapours has therefore lost its appeal.

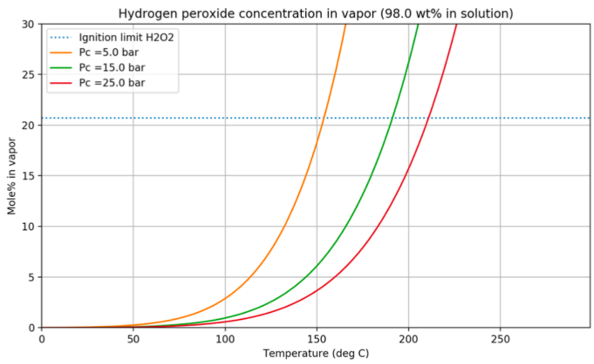

Figure 2. Hydrogen peroxide vapor concentrations

V. Other methods

Among other methods that were identified to have potential use in an HTP monopropellant thruster is the use of a hypergolic system. Though this is usually associated with a bi-propellant system, using the second fluid solely for the purpose of initiating the decomposition reaction could be considered a potential solution. Several attempts have been made to assess the performance of hypergolic propellant combinations with HTP. Experimentation using gelled ethanolamine fuel and HTP in [24] showed ignition delay times in a range of 1 - 5 ms. Small ignition delays are a major advantage in the case of pulsed thrusters, but the system may become complex as the fluid is required to be injected for every single pulse. For this reason, it is not considered for this project at this stage.

In [25] a concept was tested in which the decomposition reaction was started with a plasma arc. It was estimated that a minimum HTP concentration of 64% is required to maintain an auto-decomposition reaction. Based on the concept used, it is expected that a dedicated complex thruster design would be required to test this technology. For this reason, it is not considered for this project.

3. DESIGN SETUP AND APPROACH

Initially a baseline design is created for an HTP monopropellant thruster that uses a catalyst bed to achieve decomposition. A modular approach is used for maximum flexibility in operation. In this way, an alternative decomposition chamber can be designed for technologies to maintain a decomposition reaction, other than catalyst beds, without the necessity of an entirely new thruster design. This allows for easy and quick testing of new technologies. A list of requirements has been generated as input for the design process.

3.1. Key requirements

A summarizing overview of the key requirements that describe the general capabilities of the thruster envisioned is found in Tab. 1.

Table 1. Overview of key requirements

Parameter	Value
Propellant	87.5% & 98% HTP
Thrust	1N at 22 bar
Feed pressure	5.5 – 24 bar
Specific Impulse (98% HTP)	160 – 173 s (vacuum)
Valve open response	< 15 ms
Valve closed response	< 10 ms
Centroid delay time	< 150 ms
Minimum Impulse Bit	0.023 Ns at 5.5 bar 0.07 Ns at 22 bar

As is evident from Tab. 1, the thruster shall be capable of operating on both 87.5% and 98% concentrated HTP, producing a reference thrust of 1N. It is expected that this level of thrust is most relevant for attitude control systems and therefore the main mode of operation shall be pulsed mode. For this reason, requirements have been set for response times as well as the centroid delay time to guarantee accurate impulse delivery. The centroid delay time is the time that passes between the centroid of the electrical “on” signal and the centroid of the thrust that is achieved [26]. This is illustrated in Fig. 3. Nevertheless, requirements have been set for continuous operation as well, to allow for testing of the decomposition technologies in continuous mode. Several requirements related to the operating lifetime have been set, such as the minimum amount of cycles as well as accumulated on-time.

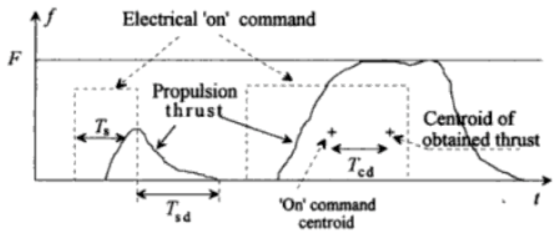

Figure 3. Thrust pulse in time domain (from [24])

3.2. Design method

In preparation for the design phase, a plan has been drafted containing the steps to be followed for the successful completion of the design. An overview is made of the envisioned thruster and its components. The complete pressure drop budget is divided over several parts. A concept overview of the thruster is found in Fig. 4. The main components from left to right are: Inlet valve, Thermal Stand-off & Capillary tube, Injector, Decomposition chamber and Nozzle. Five key components were identified in the pressure drop budget, namely: calibrating orifice, capillary tube, injector, catalyst bed and the catalyst bed retainers. The first step that is made in the design process is the inlet valve selection. This is followed by the preliminary design of interfaces between thruster components, followed by the selection of materials for both the structural components and the seals. The length of the thermal standoff is determined by means of thermal analysis. The final step is the mechanical design and analysis. It is expected that this complete process is of an iterative nature. Once the design is completed, technical drawings are created using CAD-software and a manufacturing plan is created.

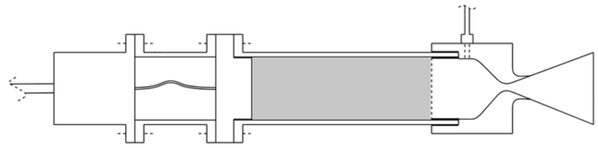

Figure 4. Thruster concept

4. RESULTS AND DISCUSSION

The design is executed according to the method as described before. Each of the outcomes of the main design steps is described individually.

4.1. Inlet valve

A trade-off has been performed on a range of available valves to find the one most suitable to serve as the inlet valve for this thruster. A number of factors were taken into account. Firstly, it is required that a valve of the correct size is chosen and that the materials of construction of the valve are compatible with HTP to prevent decomposition effects in the valve itself. For compatibility purposes, it was chosen to focus on stainless steel valves with compatible seals, such as those made out of PTFE or FFKM. The main performance parameter of relevance is the response time (open and close), the requirement of which was a show stopper for many of the candidate valves. Additionally, an estimate was made of the dribble volumes in the valves for comparison. Minimizing the dribble volume will reduce the time required for this volume to fill up before the propellant reaches the injector and will therefore reduce the thruster response times. The final choice fell on the Parker Pulse Valve, as is seen in Fig. 5, which is available in chemically compatible configurations and meets the response time requirements, whilst also possessing a minimal dribble volume.

Figure 5. Parker Pulse Valve [27]

4.2. Interfaces

Since the thruster has a completely modular design several interfaces exist. Some interfaces, such as the ones with the thermal standoff can be rather simple, as they are not in contact with the propellant or hot gases and no large loads are expected. The most interesting interfaces are the one between the injector assembly and the decomposition chamber, as well as the interface between the decomposition chamber and nozzle, as these experience high pressures and temperatures. In the current design the nozzle assembly is screwed onto the decomposition chamber as is seen in Fig. 6. The retainer plate of the catalyst bed is placed on a cylinder that fights tightly into the decomposition chamber and rests on the nozzle assembly. This way the amount of empty space after the catalyst bed can be increased, to allow for extra space for thermal decomposition, or decreased to improve the response time of the thruster. A seal will be applied between the two components in the form of an O-ring.

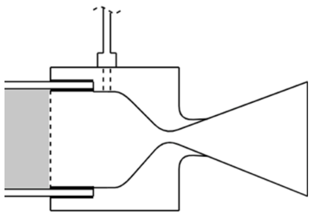

Figure 6. Decomposition chamber/nozzle interface

Fig. 7 depicts the interface between the injector and decomposition chamber. The injector assembly is screwed completely into the decomposition chamber. In this way the surface of the injector head is brought towards the start of the catalyst bed. This allows for injector designs in which the propellant is injected directly into the catalyst bed. A flange is created on the decomposition chamber so that a seal can be applied between the two components using a gasket.

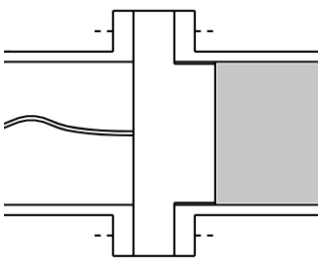

Figure 7. Injector/decomposition chamber interface

4.3. Materials of construction

As was mentioned before, HTP decomposition is accelerated by the presence of contaminants. Thus, in all parts of the thruster that touch the propellant, the material choice affects the stability of the propellant. In the case of the decomposition chamber, the stability of the propellant is not important, though the corrosion effects of the propellant on the material are still relevant. Several different alloys of stainless steel are common and suitable for use in combination with HTP.

Using RPA Lite an estimate was made of the maximum expected temperature in the thruster. For HTP with a concentration of 98%, the adiabatic decomposition temperature is near 1250 degrees Kelvin (± 977 degrees Celsius). This temperature was taken as a worst-case scenario. The material of choice for this thruster is AISI 310 Stainless Steel. It has a continuous service temperature of approximately 1150 degrees Celsius and an intermittent service temperature of about 1035 degrees Celsius [28], which means it is capable of handling the temperatures in the decomposition chamber. In [29] it is stated that it is especially suitable for high temperatures and corrosive atmospheres. It also shows that the compatibility (based on Active Oxygen Loss) of AISI 310 with HTP is lower than that of for example AISI 316, which has a lower service temperature. However, since no long-term storage of the propellant in the thruster assembly is required, for simplicity all structural components will be designed using the same material.

4.4. Decomposition chamber

The diameter of the decomposition chamber is determined using a typical catalyst bed loading determined as the average from a collection of available thrusters. The catalyst bed loading that was chosen is 10 $kg/m^2/s$. By using the expected mass flow as calculated from specific impulse, the catalyst bed diameter is calculated.

The required length of the catalyst bed as well as the pressure drop can be determined by using a model of the decomposition process. Several models have been created to describe this process. Since the models found are quite complex, a more simplified less accurate model was created based on stochiometric equations for isothermal packed bed reactors presented in [30]. As this book only describes gas phase reactors or liquid phase reactors, a hybrid version was created for the multi-phase decomposition reaction. This reaction is not isothermal, though the use of these equations is justified by using a small timestep in the model. As input for the decomposition model the reaction rate equations have been used from [23].

According to [30] the pressure drop within a packed bed reactor can be modelled using the Ergun equation. In [23] the performance of this equation is compared to that of several adaptations. One of these adaptations is the Tallmadge equation (Eq. 2), which can be used for flows with higher Reynolds numbers. This equation is used in combination with the model for two-phase flows through pebble beds from [31].

4.5 Nozzle

The thruster is designed with a simple conical nozzle optimized for operation at a pressure of 1 *atm*, as this is where testing will occur. Though the exact profile of the nozzle is not yet determined, it is designed according to empirical information available at ESA as well as in [32], which provides a collection of guidelines for the contraction half angle, throat longitudinal radius, contraction radius and divergence half angle. Since the nozzle assembly is a separate module, in the future a different nozzle design can be created and used with the existing setup, if so desired. The assembly is designed in such a way that there is an interface for a pressure and temperature sensor (as is seen in Fig. 6), to allow for measurements of the chamber pressure and temperature, which can be used in performance characterization and comparison.

4.5. Thermal standoff and capillary tube

During operation, heat from the decomposition chamber will transfer through the components and it will reach the inlet valve. In order to prevent decomposition effects from occurring at the inlet valve, even when the thruster is no longer in operation, a thermal standoff is included on the thruster (Fig. 8). This standoff is aimed at increasing the heat resistance between the hot gas in the decomposition chamber and the inlet valve. The length of the thermal standoff is determined using a simple thermal analysis. This analysis is performed by setting a maximum allowable safety temperature at the inlet valve and an assumed temperature at the injector face. The total heat dissipation at the inlet valve is estimated first. The heat flow from the injector face towards the inlet valve is calculated using an estimated heat resistance of the injector assembly and a variable resistance for the thermal standoff, which depends on the length. The latter is adjusted until the heat flow into the inlet valve is smaller than the dissipation out of the inlet valve. At this point it is concluded that the inlet valve will not heat above the maximum temperature.

The propellant passing from the inlet valve to the injector flows through a capillary tube. Immediately after operation the capillary tube will be empty and the temperature at the end of the capillary tube (at the injector face) is high. This means that once the propellant starts flowing the heat flow into the propellant is large. It is key that the propellant speed in the capillary is sufficient to prevent heating of the propellant to the point of rapid decomposition, as this could cause a thermal choke in the capillary tube according to [33]. It is stated can be partially mitigated by use of a thermal shunt on the capillary tube. With an estimated heat flow a transient simulation is performed to calculate an acceptable diameter of the capillary tube. The pressure drop increases with increasing flow velocity as well. These parameters need to be balanced.

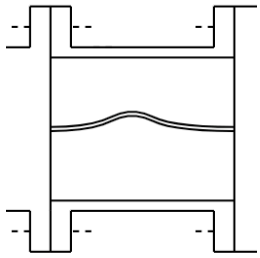

Figure 8. Thermal standoff and capillary tube

4.6. Injector

In the current design, the complete injector assembly is a single module. This means it is easily exchangeable with an alternative injector, providing the capability to test and compare the performance of several injectors. In the baseline design of the thruster a simple showerhead injector is included.

5. FUTURE WORK

The end product of this work is the complete detailed design of a fully modular 1N class monopropellant thruster suitable for testing of several methods of sustaining an HTP decomposition reaction, as well as the associated feed system, test setup and test plan. This paper describes the preliminary baseline design of such a thruster. This design is based on decomposition using a catalyst bed. To finish the baseline design a mechanical analysis still has to be performed, which may be followed by an iteration of the previous design steps. The design will be completed by creating detailed technical drawings of all thruster components as well as a manufacturing flow chart.

The next step in the project is to prepare for the design of a secondary decomposition chamber for use with alternative technologies. This will be preceded by laboratory experiments to test the feasibility of some of these concepts. Additionally, measurements from these experiments will help determine design inputs with respect to sizing and power requirements of the components of these alternative technologies. Thermal decomposition experiments will be performed by the means of a drop test using an aluminium covered hot plate. Additionally, some tests will be performed regarding the possibility of initiating decomposition using a spark gap. Though it was shown not to work for the ignition of vapours, the tests will show whether decomposition is possible due to local heating by the spark or the generation of radicals in the flow. Due to restrictions related to time as well as the availability of a high power pulsed laser such as an ND:YAG laser, unfortunately no experiments will be performed for laser induced decomposition during this project. The design of the secondary decomposition chamber will however take into account the option of attaching a laser ignition system. Again, the design is completed by means of technical drawings and a manufacturing flow chart. Should a different system become of interest in the future, the thruster is easily adapted as simply a single new decomposition chamber can be designed.

Once all modules of the thruster have been designed, a simple propellant feed system as well as a test setup will be designed. The test setup will be designed in such a way that all information vital for the comparison of the performance of different catalyst beds as well as the performance of different technologies is measured. For the quality of the measurements a list of requirements has already been created. Once all designs are finished and the documentation is completed, the design project is finished. Manufacturing will be done in the form of a new project.

6. CONCLUSION

It was identified that there is an interest in comparing performances of different types of catalyst beds as well as comparing the performance of these traditional methods with that of alternative technologies. Several technologies have been identified to have potential. A preliminary design is performed for a fully modular 1N monopropellant thruster for use with catalyst beds. Some work still has to be performed to complete the "baseline" design. In future work laboratory experiments will be used to determine the feasibility of alternative technologies. These technologies will be included in a secondary decomposition chamber. Unfortunately, due to the complexity of some technologies as well as time restrictions not all of these can be included. The design project will be concluded with the design of a propellant feed system and a test setup.

7. ACKNOWLEDGEMENT

This work is performed by Thim Franken in the form of an end thesis to obtain the master's degree in Space Systems Engineering at the Delft University of Technology in the Netherlands, in cooperation with the European Space Agency in Noordwijk (ESA/ESTEC). On the side of the university there is support from the two supervisors, Dr. Jyoti and Mr. Zandbergen. The main guiding supervisor of the project is Mr. Valencia-Bel from ESA/ESTEC.

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