Development and Test of a 3D printed Hydrogen Peroxide Flight Control Thruster

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Today's orbital propulsion hot gas thrusters are often based on hydrazine as a storable propellant and are manufactured in a classical way with many single pieceparts. The need for non-toxic or green propellants is increasing not only due to the fact that classical, hydrazine based propulsion systems are facing legislative regulations but also because non-toxic alternatives may offer significant economic benefits.

Further, innovative new manufacturing technologies such as 3D printing (ALM) can offer significant cost savings by reducing the number of parts and subsequent joining processes.

The development and test of a fully ALM manufactured control thruster is a part of Airbus Defence and Space Alternative Propellant Initiative where non-toxic propellants for various applications are being investigated.

Nomenclature

ADN	=	Ammoniumdinitramide
ALM	=	Additive Layer Manufacturing
Cd	=	Coefficient of discharge
FCS	=	(Spaceplane) Flight Control System
GEO	=	Geostationary Orbit
H2020	=	Horizon 2020 EU programs
LEO	=	Low Earth Orbit
MMH	=	Monomethlyhydrazine
PMF	=	Pulse Mode Firing
RACS	=	Roll and Attitude Control System
REACh	=	Registration, Evaluation, Authorisation and Restriction of Chemicals
SSF	=	Steady State Firing
UDMH	=	Unsymetric Dimethylhydrazine

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I. Introduction

Airbus Defence and Space as the main European supplier for launcher and orbital propulsion systems invests in the development and industrialization of components and propulsion systems using classical and non-toxic propellants. Therefore an internally funded R&T project called “Alternative Propellants” is funded that investigates alternatives to the currently used propellants for orbital and launcher propulsion.

In **Figure 1** propulsion technologies for the companies various applications are shown with actual state of the art, mid term and future technologies. Technologies which are considered as "green" are marked in green colour. The figure shows that Hydrazine based propulsion technologies (including derivatives like MMH and UDMH) are widely spread in orbital and launcher propulsion [1].

Hydrazine (also called diazane) is an inorganic compound with the chemical formula N_2H_4. It is a colourless flammable liquid which is toxic, carcinogenic and mutagenic. Hydrazine is a standard component in the chemical industry with a big production volume (260.000 tons in 2002 ; less than 5% used for space applications [2]). Currently pure hydrazine is used on European launchers (Ariane 5 and VEGA roll control) and in most of the LEO satellites as a reliable and robust monopropellant with a significant space heritage and a great variety of commercially available components.

Due to its toxicity hydrazine was included in Europe's ECHA’s candidate list in 2011 and may be prioritized for inclusion in Annex XIV of REACh at any time, which means that the worst case scenario is that hydrazine cannot be used from 2020 onwards [3]. Therfore the search for a replacement candidate for Hydrazine as a monopropellant has first priority in the R&T project "Alternative Propellants".

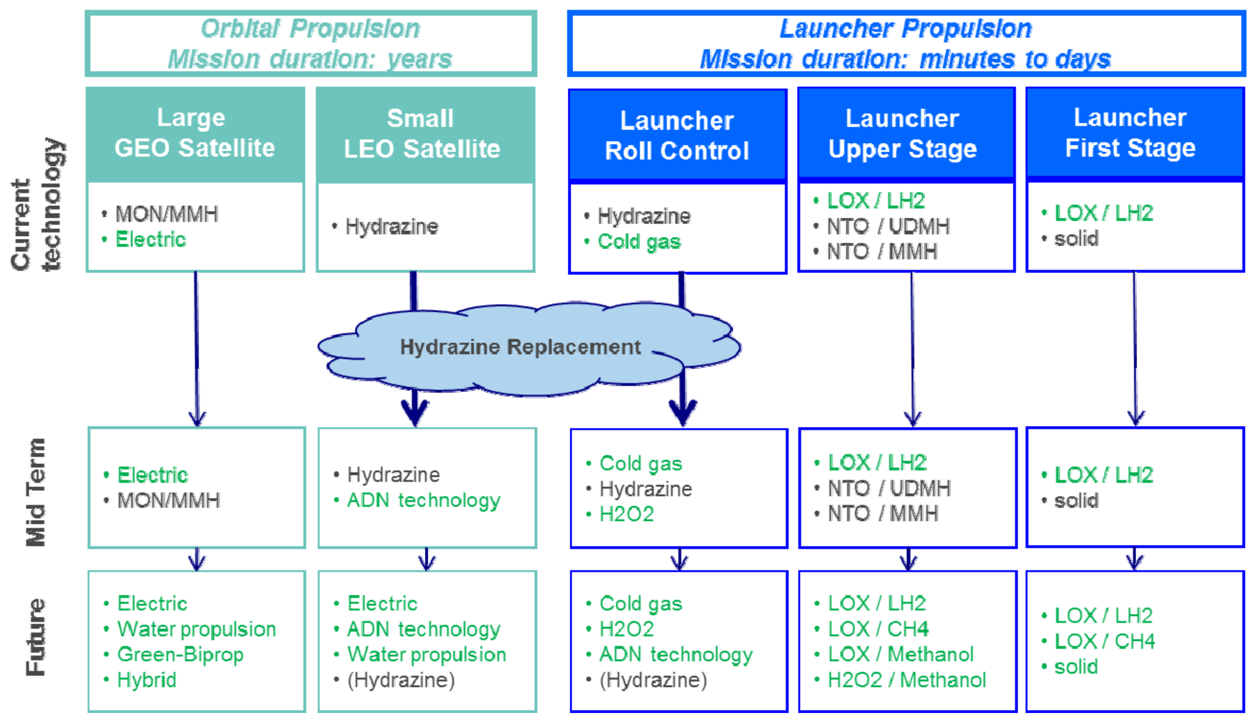

Figure 1. Propulsion technologies

A. Hydrazine replacement technologies options for long term applications

For orbital propulsion the life requirement is up to 15 years therefore technologies are considered which have a similar storage capability as hydrazine:

- Electric propulsion is green and offers significant benefits w.r.t. performance (ISP) and thus a reduction of the overall propellant mass
- the ADN technology that has successfully demonstrated in orbit operations with the ECAPS built PRISMA satellite and which is now on its way for a commercial application in the Airbus DS built MYRIADE platform. The drawbacks of this technology (expensive chamber material and high preheating power) are currently investigated in the H2020 program *RHEFORM* [4]
- The HAN technology as foreseen by Aerojet-Rocketdyne on the NASA GPIM mission [5]
- Water propulsion as a low cost alternative where the propellant is being produced via electrolysis as proposed e.g. in the HYDROS module for cubesats [6].
- Hydrogen Peroxide as a monopropellant and as an oxidizer for hybrid systems as currently investigated in the H2020 program *HYPROGEO* [7].
- Mixtures of nitrous oxide with fuel (NOFBX) as e.g. proposed by Firestar [8]

B. Hydrazine replacement technologies options for short term applications

For short term missions hydrogen peroxide is an interesting low cost non-toxic alternative propellant. This propellant is used in rocket engine applications since the 1940s and is still in use for the Soyuz launcher as a gas generator and for the manned Soyuz capsule for flight control during re-entry. Since the 1960s hydrogen peroxide was mainly replaced by toxic hydrazine because hydrazine has a higher performance, a better storability and robustness and toxicity at that time was not a major issue.

But times have changed and toxicity and environmental friendliness are becoming more and more important not only because they are directly linked to operational cost.

C. Technology Summary

The following table summarized advantages / disadvantages compared to classical hydrazine of the various non-toxic technologies that are currently emerging:

Technology	Advantage	Disadvantage
Electric Propulsion	▪ Significantly higher ISP	▪ Low thrust level ▪ System is complex
ADN Technology (ECAPS)	▪ Slightly higher ISP and density ISP	▪ High combustion temperature requires exotic materials ▪ Preheating is required
HAN Technology (Aerojet)	▪ Slightly higher ISP and density ISP	▪ High combustion temperature requires exotic materials ▪ Preheating is required
Water Propulsion	▪ Higher ISP	▪ System is complex
Hydrogen Peroxide	▪ Simple technology	▪ Slightly lower ISP ▪ Self decomposition (storage)
NOFBX	▪ Self pressurizing ▪ Higher ISP	▪ Large and heavy tanks due to high pressure and low density propellant ▪ Igniter necessary

II. 3D printed Thruster Development

In the frame of an investigation of a flight control system (FCS) for a spaceplane, different propulsion concept were investigated (see also separate AIAA paper [9]) and hydrogen peroxide was found to be one of the promising candidates. As spaceplane FCS requirements and launcher RACS requirements are similar it was decided to design, manufacture and test a 250N class thruster demonstrator that fits in both applications. Target was to gain detailed first-hand experience about design parameters, performance and handling of this technology. In order to speed up the development process and to lower the overall manufacturing cost it was decided to build the thruster hardware via Additive Layer Manufacturing (ALM).

Target was also to answer the question if a thruster can be manufactured in a low cost technology where the rework after ALM manufacturing is limited to the sealing and welding interfaces but also to assess what the cost and schedule impacts compared to a classical manufacturing are.

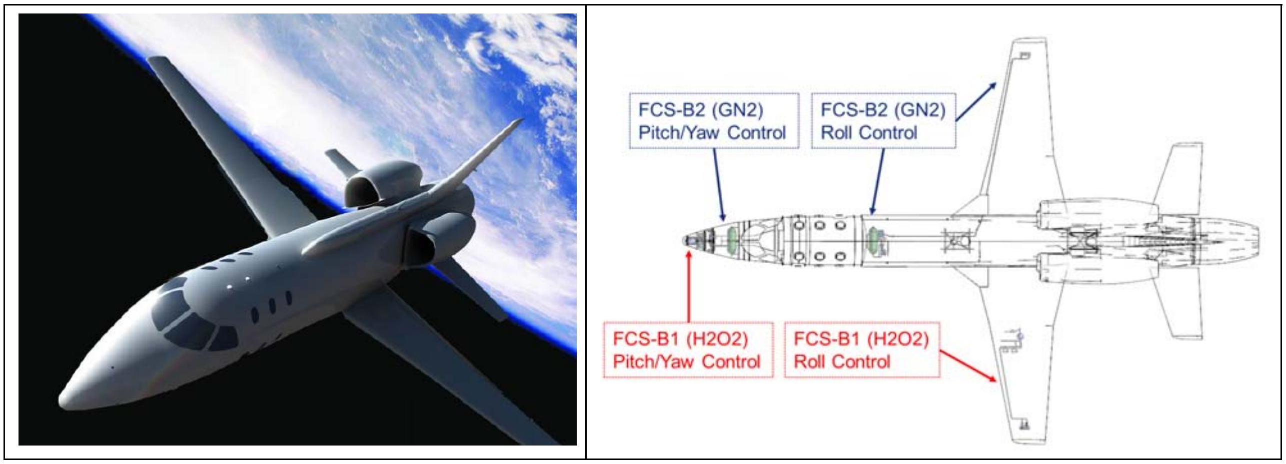

Figure 2. Airbus Spacplane (left) and its two FCS systems (right)

This hydrogen peroxide Thruster Demonstrator development was realized in less than one year from project initiation, objectives and requirements definition to first full flight-like thruster hot firing demonstration tests. All activities were performed in close contact between the different Airbus DS sites in France and Germany as well as with the support of partners like Heraeus Hanau and DLR Institute in Braunschweig / Trauen.

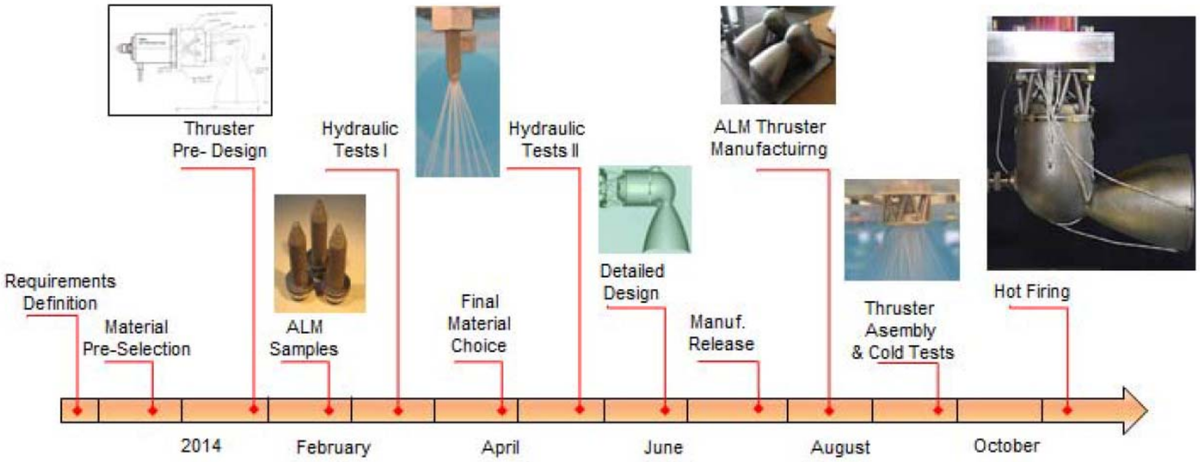

Figure 3. Development Timeframe

The following main development steps were performed:

- Harmonization of requirements between spaceplane FCS and launcher RACS leading to a common thruster development specification
- Layout and definition of a thruster with a “near flight” design.
- Selection of ALM base material and manufacturer
- For each selected configuration multiple injector test sample elements were manufactured and inspected via optical means and via computer tomography.
- In a hydraulic lab these injectors were tested for pressure drop, spray distribution and reproducibility. Result was that classical design rules had to be adapted for ALM manufacturing process. Based on the results a second design loop was performed in order to take into account the differences in hydraulic characteristics
- Selection of the final catalyst via test by comparing different catalyst types. Besides different classical silver screen catalysts various pellet based catalysts were tested
- Layout of the thruster with respect to initial thermal and mechanical environment and safety rules, selection of flow control valve, definition of pressure loss cascade, detailed injector design based on initial hydraulic sample test results and layout of the catalyst chamber.
- Final design, manufacturing and hydraulic test of two thruster parts with a variation in catalyst bed configuration. Each thruster consisted of 2 parts only: in one part the injector, heat barrier and mounting interface and in the second part the chamber with the canted nozzle.
- Design and adaption of the manufacturing and test environment such as jigs and tools for manufacturing and cold test, hot firing test rig, propellant supply and measurement and control system
- Hot firing test of two thrusters with detailed data evaluation. The thruster was tested in steady state and pulsed mode at various inlet pressures and duty cycles.

A. Establishment of Requirements

In a first step development requirements were defined. The basis for the requirement definition were the two mission cases Spaceplane FCS and launcher RACS. A vacuum thrust level of 250N with an inlet pressure range between 8 and 26 bar (116 to 377 psi) and a set of duty cycles derived from launcher mission needs was defined as design driving requirements. Also a canted nozzle was selected to have full mounting flexibility.

B. Initial thruster design

The initial thruster design was a "near flight” design with a canted nozzle. An expansion ratio of 30 was selected to fit into a predefined envelope. The further elements were a classical multi showerhead injector, a flight like heat barrier, a single seat flow control valve and a classical catalyst bed.

C. Material and manufacturer selection

In a first step several potential combinations of material and manufacturer were defined based on material compatibility with liquid propellant, gaseous decomposition products and hydrogen peroxide decomposition temperature. The selectected materials were Inconell 718, CoCr and Stainless steel.

D. Sample injector manufacturing and establishment of 3D printed hydraulic parameters

With the selected materials and manufacturers sample probes of the injector were printed, optically inspected and the pressure drop measured. The following figure shows various injectors that were printed to fit easily into a hydraulic lab to measure their hydraulic performance.

	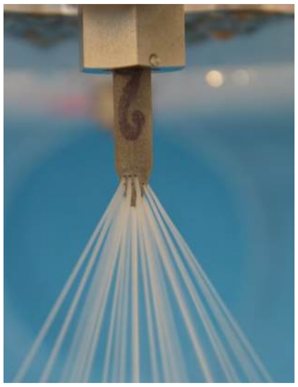
Printed injector samples in different materials for hydraulic tests	Spray test

Figure 4. Sample Injectors

Based on the hydraulic tests the design parameters of the injector (cd value) were iterated. Due to differences of ALM manufactured parts and classically manufactured parts in the area of surface roughness, dimensions (designed versus printed) and edge configuration these iteration steps had to be performed until the required total pressure drop of the injector was reached with a good repeatability and predictability.

E. Thruster final design and printing of Hardware

Based on hydraulic results the final thruster was designed. The design files were directly used for ALM manufacturing without need for additional piece part manufacturing drawings. The following picture shows the designed parts and the finally printed thruster hardware. In the lower thruster part additional elements were printed not necessary for the function in order to allow the implementation of sensors like chamber pressure and temperature.

After the printing the parts were reworked in interface areas (FCV sealing interface and welding interfaces) because the qualiy of the printed surface was not adequate.

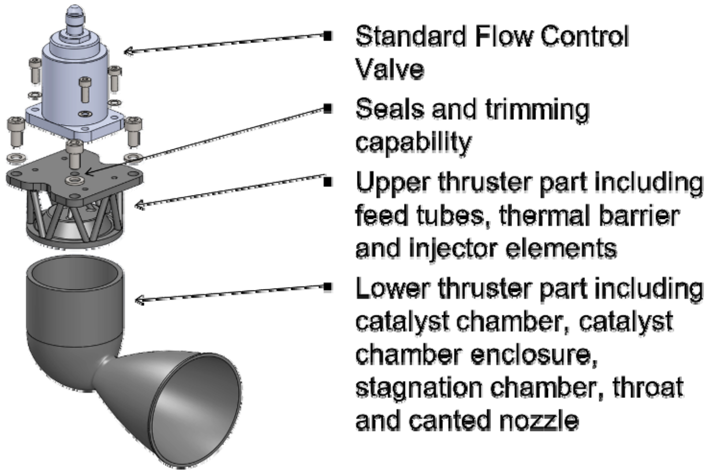

Figure 5. Final Thruster Design and printed Hardware

F. Thruster final assembly and preparation for hot firing test

Before filling the thruster with catalyst and welding it together the entire injector head was verified to have the required pressure drop. The difference in pressure drop of the two manufactured injectors was less than 0,1 bar.

III. Catalyst Selection

The catalyst was developed and manufactured by Airbus DS parter Heraeus. Heraeus, a technology group headquartered in Hanau, Germany, was formed in 1851 and is focused on themes such as the environment, energy, health, mobility and industrial applications. Heraeus Deutschland GmbH & Co. KG is a part of the leading international family-owned company with more than 30 years expertise in catalysts development for space applications. The aim of the work presented below was to develop a new catalyst for the decomposition of H_2O_2, which is currently considered as a promising green propellant for low and medium thrust applications. The impact on catalytic activity of several parameters such as PGM precursor, Al_2O_3 support, and annealing temperature were studied and more than 30 catalysts were tested in house during this project. Pt catalysts supported on the flight-proven Al_2O_3 granules used for our hydrazine decomposition catalyst H-KC12GA showed promising results both in terms of activity and stability for the H_2O_2 decomposition. The most relevant results obtained for these Pt catalysts are presented below.

A. Catalyst Preparation

Pt/Al_2O_3 granules catalysts were prepared by a conventional impregnation method using aqueous solutions of three different Pt salts (A, B and C), followed by drying and calcination. Two different Pt loadings (5 and 10 wt%) were prepared for precursor C and the catalysts are denoted Pt (x, y)/ Al_2O_3, were x is the Pt loading in wt% and y is the Pt salt used for the impregnation. Prior to the activity tests, the catalysts were reduced in forming gas.

B. Catalytic activity Measurement

The catalytic activity was evaluated using a simplified test designed in house. Before each test, a known mass of reduced catalyst was placed in a reaction flask. This flask was connected to a device that monitors the gas release during the reaction. The test started when a given volume of hydrogen peroxide solution was introduced in the reaction flask. We recorded both the reaction time as well as the volume of the hot gas mixture generated by the decomposition reaction.

C. Results

Each catalyst was tested several times and the average values for the gas release and the reaction time are presented in the following table:

Catalyst	Gas release [ml]	Time [s]
Pt (5%, A)	40	68
Pt (5%, B)	54	33
Pt (5%, C)	54	16
Pt (10%, C)	48	33

The catalyst prepared with 5% Pt using precursor C shows the most promising results in term of activity for H_2O_2 decomposition. This catalyst reacts two times faster than the corresponding catalysts prepared with precursor B and almost four time faster than the catalyst prepared with precursor A. The catalyst prepared with 10% Pt using precursor C does not show any benefit compared to the 5% Pt catalyst probably because the dispersion of this highly loaded catalyst is rather low. Some characterization are currently being performed to better understand the impact of Pt precursor and loading on the catalytic activity.

D. Conclusion (Catalyst selection)

Among the various Pt salts tested during this project, precursor C was found to be the most promising in terms of catalytic activity for the H_2O_2 decomposition. A catalyst with 5% Pt prepared with precursor C was sampled to Airbus for evaluation in their firing test facilities. This catalyst was supported on Al_2O_3 granules with a grain size between 10 and 14 mesh for backpressure considerations but other grain sizes are also available (same as the ones available for H-KC12GA).

IV. Hot Firing Test and Results

The final hot firing demonstration were performed in November 2014 at the Trauen test site, where DLR has set up a test environment for hydrogen peroxide technology for the AHRES (Advanced Hybrid Rocket Engine Simulation) project [10]. The thruster was equipped with various sensors in order to measure the performance and the temperature distribution at the outside of the thruster as well as in the catalyst bed. Two thruster demonstrators have been tested with different catalyst bed configurations.

High definition and infrared cameras were installed in order to get optical and infrared images but also to measure the temperature distribution of the entire thruster. The following figure shows the test setup and the instrumented thruster in the thrust meadsurement rig.

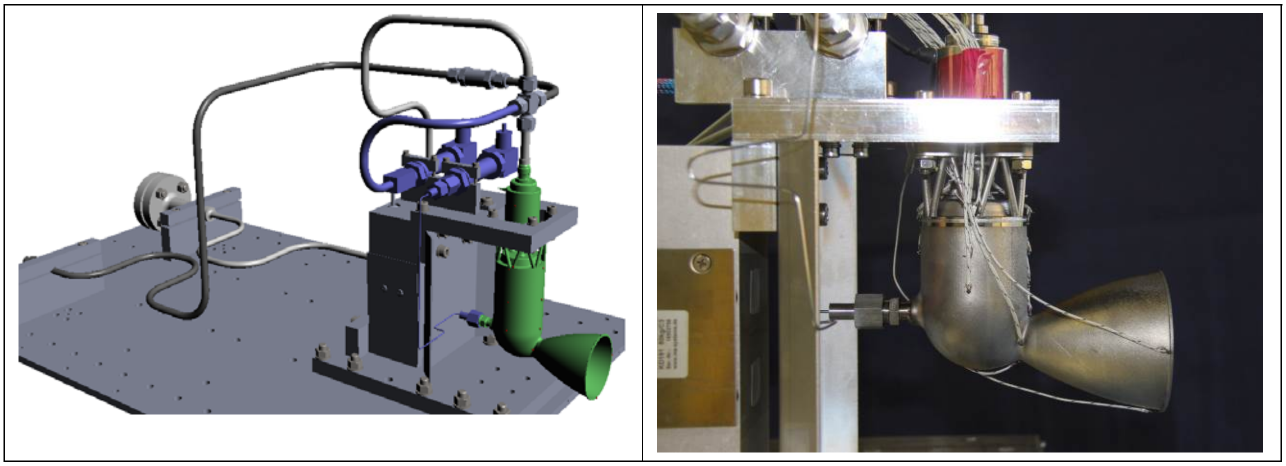

Figure 6. Thrust rig configuration with integrated thruster

A. Thruster Hardware and Test Program

Two different thruster hardware were tested with a variation in catalyst bed configurations, named thruster #1 and thruster #2. The difference in catalyst bed pressure drop was compensated via an adjustment of the feed pressure for thruster #2. To allow a direct comparison in the diagrams a theoretical trimming was applied in **Figure 7**.

An actual propellant concentration of 86,45% leading to a theoretical decomposition temperature of 661°C was used for the tests.

Each thruster was nominally tested at a tank pressure range between 8 / 12 / 16 / 20 / 24 bar in SSF (20 sec) and PMF (100ms ON / 1s OFF and 30ms ON / 170ms OFF) mode. Based on the good results of the the initially foreseen test program the following additional tests could be performed without any problems:

- Operation (SSF and PMF) at high (26 bar) and low (5,5 bar) feed pressure
- Cold start (7°C) at 24 bar inlet pressure going directly into SSF
- Long duration firing (60s for thruster #1 and 50s for thruster #2)
- Depletion test through the thruster (propellant followed by nitrogen as pressurizing gas)

Before and after the test program SSF and PMF health checks were performed in oder to indicate a potential thruster degradation. The health checks showed a difference between begin and end of test of max. 1,3% in thrust indicating that the both thrusters suffered no measureable degradation during the entire test.

The total propellant consumption was approx. 36 kg for thruster #1 and 25 kg for thruster #2.

B. Performance Calculation and Steady State Test Results

Performance calculation was performed with measured throat diameter, mass flow and chamber pressure. The measured sea level thrust was taken as an additional reference.

All channels were corrected w.r.t. sensor offset and drift and the effective throat area was corrected due to thermal heat up with the mean value of throat temperatures. In a first step chamber pressure correction due to difference between static / dynamic pressure related to mach number effects were not regarded due to the high contraction ratio of stagnation chamber diameter versus throat diameter. Also boundary layer and curvature effects were not regarded whicht might influence the results due to the unknown effects of ALM manufactured surface.

The following figures show the key performance parameters as a function of inlet pressures. The figures show that the 2 thrusters behave similar with the c* performances slightly increasing with lower inlet pressure. The temperature distribution was as expected with a maximum temperature at the end of the catalyst bed that was close to decomposition temperature with the actual H_2O_2 concentration. This indicates the good performance of the thruster design.

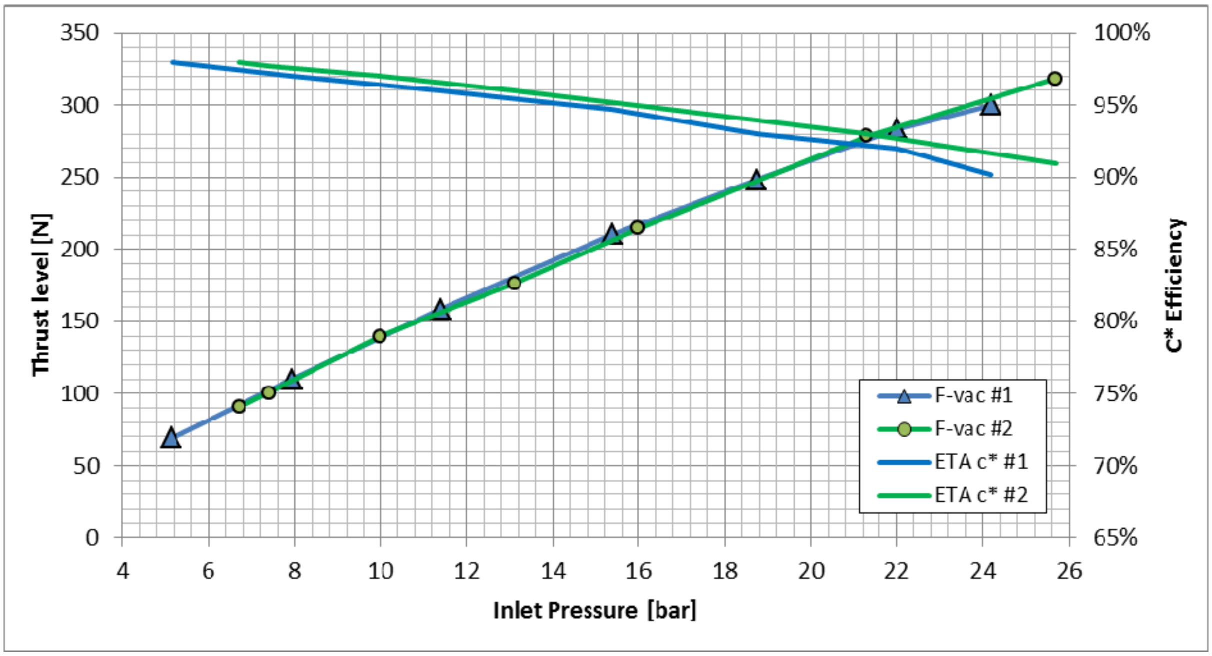

Figure 7. Steady State Fring Results

C. Pulse Mode Operation

All defined PMF tests were performed without any problems. The following figure comapres the chamber pressure evolution (first 100ms pulse) of the tested H_2O_2 thruster with the chamber pressure evolution of a classical 400N class Hydrazine thruster.

The figure shows that for the H_2O_2 thruster a slightly delayed reaction has to be expected which has to be considered in the system design.

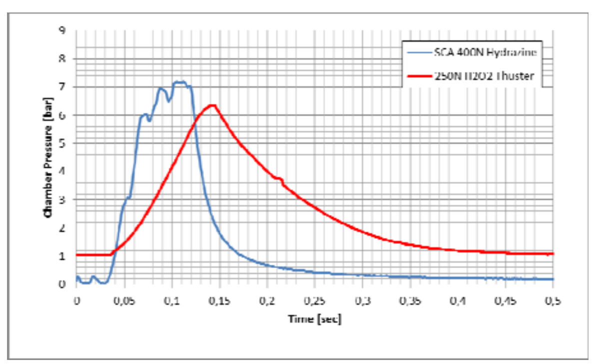

Figure 8. Pulse Mode Fring Results

D. Image Recordings

Optical (HD video) and thermal video recordings were performed for several positions. The following figures show sample images. The optical image shws a thruster operation during the start of a SSF when visible steam is produced. The thermal image from the outside of the thruster was taken at the end of a SSF when the thruster reached nearl staty state temperatures. He red area indicates the position of the catalyst bed where the energy is released.

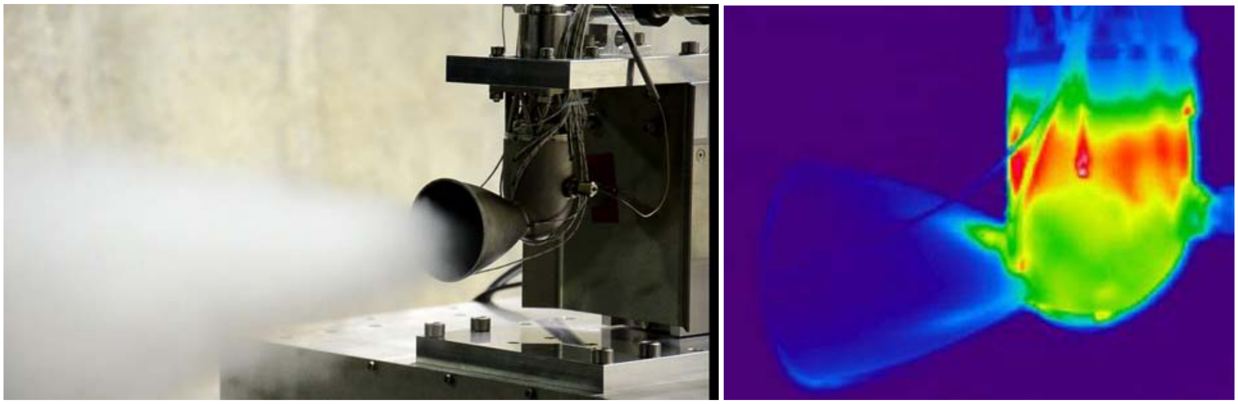

Figure 9: Example of optical and thermal image

V. Conclusion

In the frame of a Franco-German Airbus Defence & Space R&T program a hydrogen peroxide thruster dedicated as a FCS thruster for spaceplane was designed, manufactured and tested. Manufacturing was performed using ALM technique. After the selection of material and manufacturer several injector samples were printed to establish hydraulic parameters for final injector layout. Based on sample test results the final thruster was finally designed and manufactured.

In parallel a dedicated catalyst was manufactured by Airbus DS partner company Heraeus and the hot firing test cell was prepared using a test environment that is used by DLR Braunschweig at the Traue test site.

The test is considered as a full success. The nominal foreseen program and margin demonstration could be performed without any problem. This test clearly demonstrated that a thruster based on ALM manufacturing technology is feasible and thus offers significant potential for design optimization and cost decrease. No critical operational conditions occurred. The maximum temperatures during heat soak (after a SSF with max. inlet pressure) were below acceptable limits and did not lead to a critical situation in the thruster.

The measured performances and the comparison of the two different chamber configurations gave valuable results for a further thruster development.

Acknowledgments

The author wants to thank all the involved parties for the good and effective contribution to the project, namely the different departments in France and Germany, Heraeus and finally the DLR team that did a brilliant and efficient job to conduct the hot firing tests.

Figure 10: The test team at the test cell in front of the test hardware

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