The development and test of a hydrogen peroxide monopropellant microrocket engine using MEMS technology

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ABSTRACT

Given the present, relatively limited deployment of low cost and mass space missions, there are clear opportunities for the application of small-scale propulsion systems in further enabling these small satellite missions.
With this situation in mind, a team comprising ASTC, SSTL, TNO and QinetiQ – under funding from the European Space Agency – has undertaken the development of a MEMS-based micro-rocket engine concept with the intended capability of providing a specific impulse of greater than 100s.
Both turbo-pump fed bi-propellant as well as mono propellant concepts were investigated. For demonstration a mono propellant design was selected of which an initial design concept was developed, based on hydrogen peroxide decomposition.
Identified as the critical component in the mono-propellant system, several batches of the honeycomb wafers – upon which the decomposition occurs – have been manufactured. As a proof of concept, the wafers have been subjected to a set of structural and functionality tests.
Given the results of the latest testing initiative, it is envisaged that, with adequate refinement and development of the current design, a reliable and deployable monopropellant micro-rocket engine solution may be realised.

INTRODUCTION

Given the present, relatively limited deployment of low-cost, low-mass space missions, there are clear opportunities for the application of small-scale propulsion systems in further enabling these small satellite missions. Requirements from Micro and nano satellites for constellation forming, LEO (low Earth orbit) drag compensation, manoeuvring for satellite inspection, formation flying, and end-of-life de-orbiting all indicate a need for significant ΔV (velocity change) capability from such a propulsion system, ideally coupled with high thrust levels, thus minimising energy losses. The requirement for such a technology is particularly evident when considering a problem commonly faced by low-cost missions; securing a precise orbit may often be compromised as a consequence of a shared or secondary launch.
It is difficult to provide a suitable solution with conventionally engineered, miniature propulsion systems. Typically, these are expensive solutions, using toxic propellants that, in many cases, do not scale favourably to the sizes demanded by small satellite technology [1].
Micro System Technology based chemical propulsion is a candidate technology with the potential to fill the near term performance gap in the market for small satellites, defined by the lack of proposed systems providing an I_{sp} in the range 100-300s. Such a system could conceivably be used for such tasks as orbit modification in micro- and nano-satellites, end-of-life de-orbit manoeuvring, and so on.

Overview of MST propulsion research work

Despite an increasing interest in miniaturised propulsion systems [2], very few MEMS-based chemical propulsion systems exist beyond the concept or breadboard stage. For example, the Ångström Space Technology Centre (ÅSTC) at Uppsala University has built and tested a MEMS hybrid system with a specific impulse (I_{sp}) of 45s, with up to 100s projected for future developments [3]. Further, the Gas Turbine Laboratory at MIT has undertaken an in-depth development programme with the objective of demonstrating a high performance bipropellant system with a target I_{sp} of 300s, a thrust of 15N, and with propellant feed achieved using a MEMS-

based centrifugal turbo-pump [4]. These two concepts demonstrate the potential to be offered by such technology, and also serve to illustrate the wide gap in performance which exists between those systems currently being considered.
NASA’s Goddard, together with Vermont University has been working on a HTP mono propellant engine. Problems with modelling were encountered and testing was performed on the engine, but no complete decomposition was reached [8]. In Austria, a team led by Mechatronics is working on a turbopump fed micro engine bi-propellant micro engine. This engine is small, but built out of standard materials [9]. Other work on the field of liquid micro propulsion is being performed in China and Japan. Different teams in Europe and the United States are also working on micro solid propellant engines. An example is the work performed by a team lead by CNRS [10].

REQUIREMENTS AND TECHNOLOGY SELECTION

To establish a baseline for the micro-rocket engine concept, a top-level specification was derived from perceived market requirements (space mission level), and the current performance of state-of-the-art 1N hydrazine monopropellant thrusters. The key features of this baseline specification are:

- I_{sp} 100-150s, scalable in bipropellant mode to ~230s;
- Propellant mass throughput of 200g (minimum) and 6000g (target), thus establishing the useful lifetime for the engine;
- Power draw of 1W (nominal) and 3W (maximum), and with the possibility to extend this to 10W for small satellites in the >1kg category;
- Minimum of 100 restarts / firings (a target of 1000 would be desirable to achieve competition with an industry standard hydrazine system);
- Minimum continuous burn duration of 3 minutes;
- Maximum mass of thruster and integrated propellant flow control system of 100g (system dry mass excluding tankage). A future target would be to achieve a 100g total dry mass including integrated propellant tank (i.e. total propulsion system mass);
- Decomposition chamber pressure of 0.4MPa minimum, with 1.0MPa target for bipropellant mode.
- An achievable thrust range of 20mN, through 200mN (nominal), and up to about 0.5-1.0N (preferred);
- A minimum auto-ignition delay time of 50ms;
- A peak decomposition temperature of around 900K assuming an uncooled silicon construction;

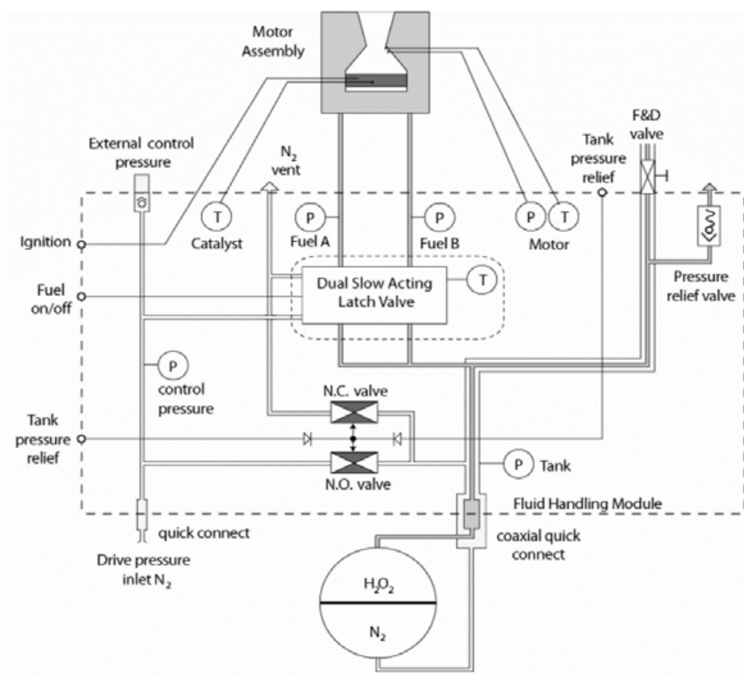

Figure 1: A system block diagram for a pressure-fed monopropellant micro rocket engine, readily transferable to a bipropellant concept as part of a future development programme

The micro-rocket design being pursued should have the potential to be manufactured in silicon-based materials that are traditionally used in MEMS fabrication (i.e. good processing heritage exists).

A technology selection roadmap, covering miniature pump technologies (for pressurised feed of propellant), micro-ancillary components (principally valves and environment sensors), and materials selection criteria, has been drawn up to aid in the monopropellant design concept development.

Propellant selection

The selection of a propellant well suited to meeting the defined requirement specification was accomplished through a semiquantitative trade-off, in which potential propellant options were scored according to different, weighted criteria. The criteria considered in this exercise were: chemical and physical properties, storage and handling properties, performance properties, and technology and hardware.

For a micro-rocket engine the process of selecting a suitable propellant differs substantially from that which would be used for a larger scale system. In a micro-engine the different criteria are much more interaction with each other Thus favouring systems that have an overall good trade-off performance instead of systems that are very strong in only one or two points.

A number of possible monopropellants were examined in this trade-off, the three most promising candidates being concentrated (90%) H_2O_2, otherwise known as High Test Peroxide (HTP), N_2O and high performance HNF (hydrazinium nitroformate). Out of this down-selected list, HTP was selected as offering most promise for the following reasons:

- N_2O: Techniques for achieving effective catalytic decomposition of this monopropellant are insufficiently mature;
- HNF (and other propellants based on energetic salts): Considerable propellant development is required;
- HTP: A well understood propellant from the point of view of achieving catalytic decomposition in a propulsion system. Other factors influencing this choice include prior experience, performance (relative to cold gas), low toxicity and applicability to a bipropellant system when used as an oxidizer.

For the bi-propellant motor a hybrid motor using HTPB as a fuel and oxygen or HTP as oxidiser gained the highest score. This was mainly due to the ease of handling and storage and technology status of this combination. The second best was HTP in combination with kerosene or pentane, essentially for the same reasons. As the assignment was to develop a liquid motor and not a hybrid, the last combination was selected for the bi-propellant study. Kerosene was selected over pentane as more knowledge and experience was available on kerosene.

Propellant feed strategies

For the micro propulsion systems, the decomposition chamber pressure (P_c) and the resulting feed requirement were reviewed in detail. High P_c is desirable in micro-rockets principally to increase residence time of the propellant(s), compensating for small chamber sizes and high heat losses and maintaining c* (characteristic velocity) despite fixed chemical reaction times. However, high P_c in a pressure fed system poses challenges for MEMS fabrication, principally due to limited bond strengths and planar designs ill-suited to act as pressure vessels. Pumped propellant would be attractive if small high power density pumps can be fabricated.

Analyses considered 1-100kg wet mass missions with 0.1-10kg of propellant respectively, with P_c values between 0.2 and 2.0MPa and a flow rate envelope of 0.01 to 0.5g/s (equivalent to a thrust of 20-500mN for an I_{sp} of 100-150s). A pump fed propulsion system was considered to be competitive at low chamber pressures for the smallest (1kg) missions, a fully packaged pump mass of 15-50g being required. 10-100kg missions would benefit from 0.1-1kg mass pumps, which might also be able to meet the system requirements (head, flow rate) if based on miniaturized conventional materials. Several such pumps with 1-5kg mass are under test, for example [5], and provide a basis for future development. However MEMS micropump technologies appear unable to meet the requirements in terms of flow rate *and* pressure rise, at present.

The team is presently evaluating in detail the problems of pumping propellants on a very small scale with a view to determining the best design options for a future programme to develop a MEMS micro-pump suitable for space applications. Options such as piston and Wankel rotary pumps in addition to centrifugal turbo-pumps have been compared. The long term aim would be to integrate such a pump with the core monopropellant engine developed in this programme.

Materials selection

The selection of materials suitable for application to a micro rocket engine is a critical part of the design process, being heavily influenced by factors such as maximum system operating temperatures and the important requirement to control thermal losses from the system. From a MEMS fabrication point of view, silicon is a well-understood material with a mature status in terms of micromachining processes having been developed (masking, etching, diffusion bonding etc). However, silicon is known to suffer from reduced strength at elevated temperatures (above about 675°C), which limits its applicability to high temperature applications such as rocket engines. In general ceramic materials have excellent potential for application to such a demanding system as that represented by a micro rocket engine. Key physical factors that influence the selection of materials include:

- Thermal endurance (strength retention and creep resistance at elevated temperatures);
- Thermal conductivity (managing heat losses);

– Corrosion resistance (propellant compatibility);
– Oxidation resistance at elevated temperatures;
– Low density, which is crucial to achieving the low mass solution for small satellites.

The primary limitation associated with the use of ceramic materials in macro-scale systems is their relatively low strength and inherent brittleness. However, the strength and toughness of these materials is scale-dependent, with significantly reduced probability of defects being present in material samples on the microscale. Several refractory ceramic materials have been considered for application to a micro rocket engine, resulting in the selection of potentially suitable candidates. These include silicon nitride, silicon carbide, sialon, molybdenum disilicide and zirconia. Table 1 presents candidate materials, together with key physical characteristics, that were considered during the early stages of the technology selection phase of the programme.
Detailed thermal analyses conducted during the study, have indicated that $MoSi_2$, Al_2O_3, and ZrO_2 do not lend themselves to micro-propulsion concepts, primarily due to their poor performance in terms of thermal shock. Sialon, SiC and Si_3N_4 materials offer attractive combinations of properties (high failure stress level coupled with relatively low thermal conductivity) and may well be ideal choices for future bipropellant concepts where thermal endurance is critical. Chemical vapour deposition of SiC coatings on Si may also hold promise.
Ultimately, the major challenge is in defining suitable micro-fabrication processes for these materials. Current micro-fabrication capabilities are heavily focussed on silicon, with its fabrication process heritage from microelectronics. The development of non-silicon micro-systems is a much less mature area of technology, although certain techniques, such as focussed ion beam (FIB) fabrication, have promising potential.

MEMS-based ancillary components

For an operational micro-rocket engine, MEMS based pressure sensors and temperature sensors will provide crucial telemetry points for monitoring performance of the system, whilst pressure regulators, fluid flow rate sensors and micro-heaters will all be important system components.
Given the developmental status of many micro-rocket engine systems, it is difficult to accurately define the physical conditions to which such components will be subjected. It is clear, however, that the majority of existing micro-fluidic MEMS components have been developed for terrestrial applications, and are thus likely to be unsuitable for direct application to the demanding environment posed by a microrocket engine. Improvements to existing MEMS component technologies will be necessary if their application is to become a reality (this is particularly the case when one considers highly demanding bipropellant concepts).
Taking the case of MEMS sensor technology, the performance requirements can be divided into two distinct areas:
1) The requirement to be able to measure the parameters of interest to the level of accuracy, stability, etc., required; and
2) The requirement to be able to survive the harsh environmental conditions to which the technology will be subjected (i.e. pressure and temperature).

For example, MEMS pressure sensing techniques employing corrugated silicon diaphragms with capacitive pick-offs appear highly applicable, provided that sufficient thermal isolation can be achieved with respect to the hot decomposed gas. Similarly, thin film temperature sensors (platinum / titanium or possibly platinum / tantalum, if eutectic formation proves to be a key concern) can be used to monitor thermal performance or permit pre-heating of the system to reduce thermal losses at start-up.
A complete integrated micro-rocket engine system also requires micro-valve technology development and integration efforts. Phase-change material valves [6] separated thermally from the hot engine parts need particular attention, as do compact and dismountable fluidic and electric interfaces between system parts. At the micro-scale, surface energy dominates over body forces, causing unexpected problems when surfaces are not intended to ‘stick together’ (e.g. in dismountable interfaces). This could be a particular problem where surfaces are required to seal tightly (fuel feed interface and valves).

Table 1: Candidate materials for a mono- or bipropellant microrocket engine

	Si	**Sialon**	**Si_3N_4**	**SiC**	**$MoSi_2$**	**Al_2O_3**	**ZrO_2**	**Pyrex**
Thermal expansion coefficient, $\times10^{-6}$	2.6	3	3	4	8	8	10	3.25
Young's modulus, Gpa	170	300	310	400	430	350	200	64
Thermal conductivity, W/mK	148	22	25	100	20	22	2.5	1.1
Failure Stress at ~300 K, MPa	300	950	650	400	250	300	800	69

Concept development and preliminary design of a mono-propellant thruster

With all background data available the concept development of the actual thrusters could start. As propellant HTP was selected because it was well known to different members in the team. Furthermore, its decomposition temperature and chemical nature was compatible with silicon. The use of silicon as a material was desired due to the experience at ASTC with Silicon micromachining. A vertically stacked wafer design was chosen as this allowed more freedom for the design and modification of the decomposition chamber and catalyst bed. Furthermore it allowed for familiar design rules to be used as the combustion chamber was cylindrical and was more pressure resistant then a horizontally integrated design. It was decided not to include any sensors or valves in the design, but to place them separately from the thrusters in order to avoid the development of hot components. The concept is shown in figure 2.
Based on this concept a preliminary design was made using standard rocket engineering practices. Based on the requirements, the decomposition chamber and nozzles were sized and a simplified thermal model was created to check the thermal behaviour of the thruster. From this analysis it became clear that thermal losses would be significant and that the design had to be improved in order to reach reasonable performance.

Concept development and preliminary design of the bi-propellant thruster

A bi propellant thruster concept was established to allow further investigation of different aspects of this technology. The bi-propellant concept is based on the mono-propellant thuster, but adapted to accomodate the injection, mixing and combustion of a fuel. HTP and kerosne were selected as propellants, on the basis of experience within the team with these propellants. The fuel was injected after the catalyst bed to allow hypergolic ignition with the decomposed HTP. For the design of the injector holes conventional engineering rules of thumb were used that had proven themselves during the design and development of other larger rocket motors. The amount of injector holes and feed channels was limited to avoid excessive pressure drops.
The combustion chamber was enlarged relative to the mono-propellant thuster to allow a sufficient resident time. It was decided to adapt the mixture ratio to 2.7 (O/F) to limit the combustion temperature to a value of 1550 K so silicon could still be used a construction material. This led to a theoretical vacuum specific impulse of 260 s.
Just like for the mono propellant thruster a preliminary design was made to establish sizes of the different parts based on the requirements.

FLUID DYNAMIC ANALYSES

After completion of the preliminary design two CFD analyses were carried out. One was aimed at examining injection, mixing and combustion in the bi-propellant engine, while the other investigated micro turbo pump feed systems.
The reason to investigate micro turbines as feed system is based on the widely reported efforts of the MIT's Gas Turbine Laboratory with their micro gas turbine engine development programmes [4]. MIT have reportedly developed and tested a device which combines a centrifugal pump and a turbine on a single rotor. To date a flow

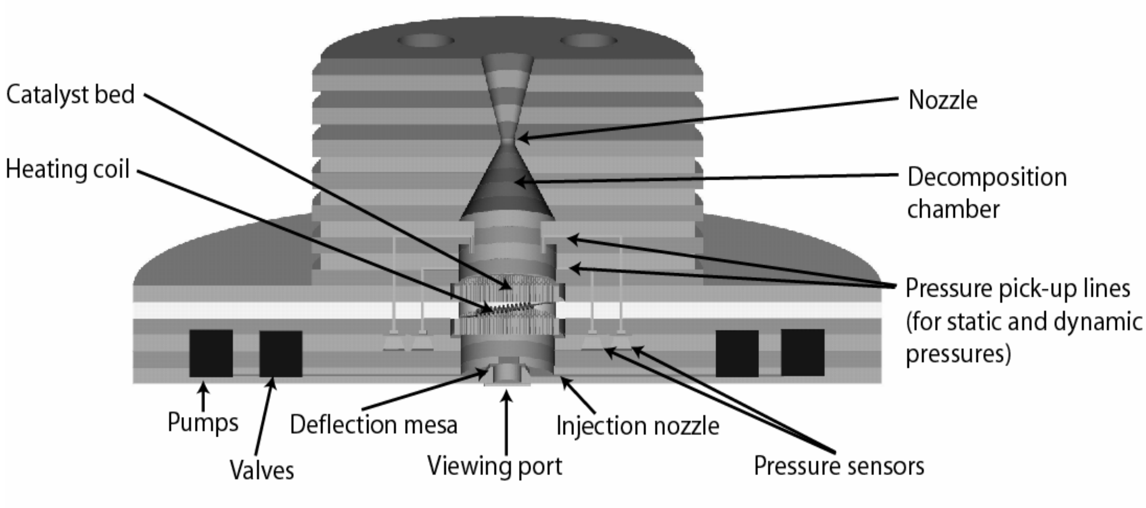

Fig. 2. Preliminary design concept

rate of 2.5g/s and 3MPa ΔP have been achieved, against a target of 5g/s and 30MPa ΔP, at a power of 75W. However, there is no clear evidence that microturbomachinery represents the most attractive solution to the provision of pump pressurisation of propellants in a microrocket engine.
For these reasons, it has been proposed by this project team to review the turbine activities of MIT and evaluate their design by means of Computational Fluid Dynamics CFD.
Attempts to replicate some of the design aspects of the MIT demonstration turbopump have been reasonably successful, supporting the idea that the aerodynamic aspects of the design are reasonably well understood. However, attempts to reproduce the design of the main engine turbopump suggest that great difficulty would be encountered in achieving an operable unit. The rotor relative exit flow angle is exceedingly high, which would give significant difficulties in the design and manufacture of such a unit. In addition, the absolute exit swirl angle is high, which would cause further problems downstream of the turbine (i.e. where the flow enters the combustion chamber). Apart from the design issues discussed above, the biggest problem areas with the operation of the turbopump are the seals and the rotordynamic stability of the unit. The problems with the seals may well be solved through further improvements in the design and manufacture of the components, but the rotordynamic problems appear to be fundamental. To obtain test results at even moderately realistic rotational speeds, it was necessary to adjust the bearing pressures. It will not be feasible to do this for a real micro-rocket engine and so it is necessary to achieve a fully stable rotordynamic system all the way from stationary to the design speed without any active control.
Currently, there is no evidence of the problems described above being overcome and so it must be concluded that the likelihood of micro-turbopumps for micro-rocket engines being successful appears slim.

For the bi-propellant motor a combustion analysis was performed in order to check whether a small combustion chamber would lead to problems in injection, mixing and combustion. The combustion analysis was carried out using the FLUENT software based on previous experience with combustion analysis at TNO with the Vinci and national programmes. The combustion was modeled using a PDF scheme. For turbulence the standard Reynolds Stress Model was used.
The main result of the analysis was that complete combustion seems to be possible in such a small chamber and that 90% of the theoretical performance can be reached (not including thermal losses to the wall). The analysis also showed that hot spots would occur on the walls, but these could be avoided by changing the injection speed and/or the injection angle.

DEMONSTRATION MOTOR DESIGN

It was decided to develop the mono propellant thruster concept further towards a demonstration motor. From the preliminary design effort it was clear that the thermal losses would be significant and that the design had to be improved.
Recommendations aimed at reducing the thermal losses were:

- The bulk of the nozzle section was reduced in order to remove unnecessary heat sink;
- The heat sink of the bulk material around the decomposition chamber was reduced by removal of unnecessary material to form internal cavities;
- The catalyst bed is suspended on very thin, perforated arms, etched in the silicon wafers, in order to minimise conductive losses between the catalyst bed and the bulk of the micro-rocket engine (Fig 3);
- A thermally insulating Pyrex wafer has been added to the design configuration, immediately below the

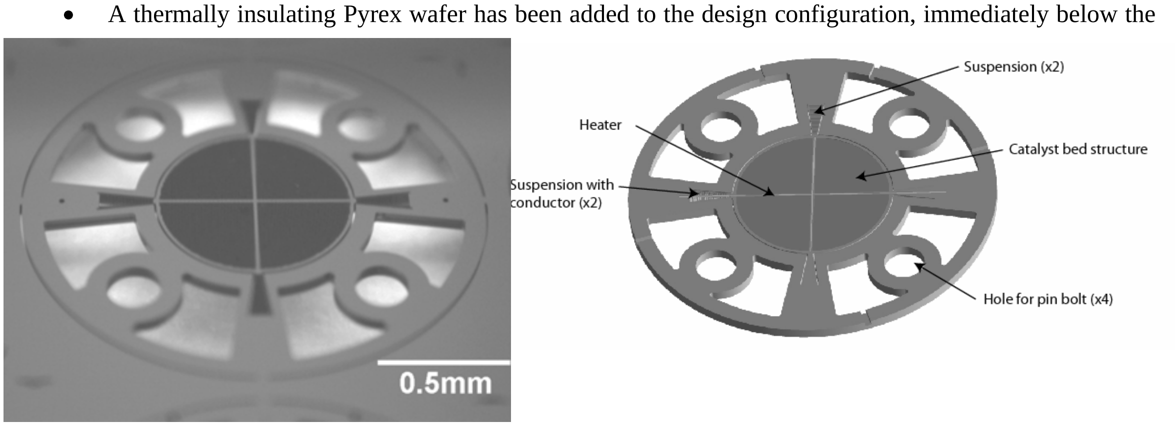

Figure 3: (Top) Layout of a single catalyst bed wafer. (Bottom)
A sample catalyst bed wafer from an early fabrication trial

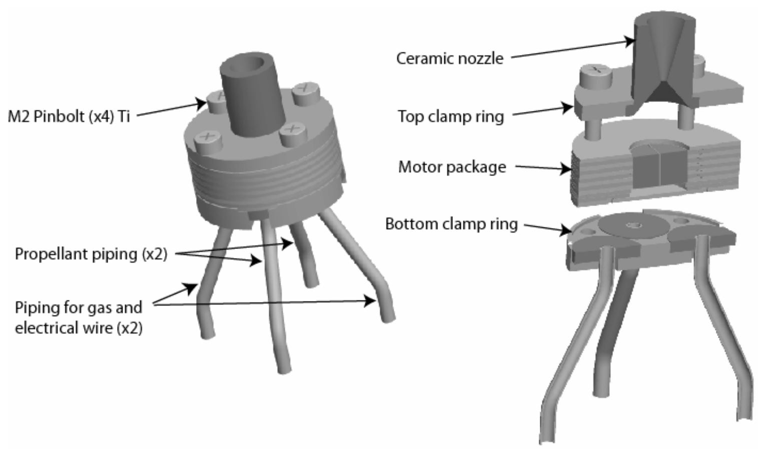

Figure 4: Identification of the key components of the updated DDM concept

catalyst bed wafers. This wafer is designed to reduce conductive heat losses between the catalyst bed and the propellant feed section of the microrocket engine.

In order to check the design improvements, a more detailed thermal-mechanical analysis using a multi physics software package (FEMLAB) was performed. Due to the use of FEMLAB different physical phenomena and their interaction could be analysed in one model. In this case the heat-exchange between flow and structure as well as the mechanical stresses induced by the heating were analysed in one model. In the FEMLAB model relations for the heat transfer between the flow and structure were used that were developed at TNO. The model was first established in 2-D before going to a full 3-D model. The analysis gave a good understanding of the thermal-mechanical environment and stresses. It also showed that the design changes had the expected effects [11].

Description of the Demonstration motor

The final design concept (described in detail in [7]) is illustrated in fig 4 and 5, in which the key components (nozzle, motor package, clamp rings and feed pipes) are clearly shown. The nozzle, plumbing and clamp rings are all produced using conventional precision-machining techniques, whilst the key structure is the silicon micromachined motor package assembly.

The motor package consists of the catalyst wafer pack (8 bonded, silver coated catalyst bed wafers), a top sealing wafer, a bottom sealing wafer and a fluid-handling wafer. The catalyst bed wafers consist of a central honeycomb structure in which hydrogen peroxide decomposition takes place, whilst the surrounding structure is designed to minimise heat losses from the reaction chamber. The design incorporates perforation of the support beams and a 50 m trench on each wafer around the honeycomb catalyst bed (
). All wafers are etched using a DRIE process.

The MEMS-fabricated motor assembly is fixed between two precision-machined titanium clamp rings, the upper of which also fixes the nozzle (manufactured from boron nitride) to the integrated assembly.

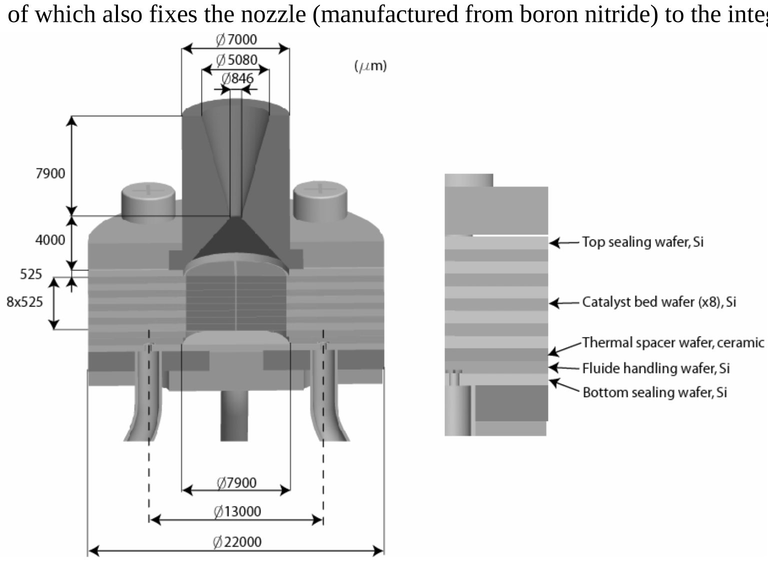

Figure 5: Basic geometry of the motor assembly in the monopropellant concept

Four pipes provide interface to the motor package, two of which act as propellant feed, another containing electrical interconnects to the integrated microheater elements in the catalyst pack, and the final providing a return pressure path for monitoring decomposition chamber pressures. The whole motor package is clamped axially by four pin bolts, which act to provide structural reinforcement, as well as allowing for a modular concept in which easy and rapid changes to the assembly's configuration can be made.

OVERVIEW OF TEST ACTIVITIES

The catalyst wafer stack containing the suspended catalyst bed was deemed the key-enabling feature of the DDM concept because its effectiveness and durability shall dictate the delivered performance and potential thruster lifetime. Some preliminary tests were conducted on a number of wafers in an attempt to briefly analyse their performance in isolation and conduct an initial evaluation of the following properties:

- Survivability and robustness of the wafers relative to the operating regime, i.e. a flow of HTP at around 10ml/min and an inlet pressure not exceeding 0.4MPa.
- Decomposition effectiveness and duration when subjected to the above flow conditions using 86% hydrogen peroxide.

The HTP firing rig at the SSTL propulsion test facility located at Westcott E-site was utilised to deliver the HTP to the cat-pack tester for the flow testing.

The preliminary tests utilised an improvised test fixture (Figure below) to clamp single silver coated catalyst wafers in a fashion that was not entirely suitable for handling items of this delicate nature.

Later tests involved testing of a large number of catalyst wafer samples (coated with a Manganese Oxide based catalyst) using a bespoke test fixture designed after taking into account the lessons learned from the initial testing.
Unavailability of a complete (8 wafer) stack of bonded wafers, and a need to test individual wafers developed as part of the catalytic chamber development programme led to the need for a test rig design which could take between 1 and 8 wafers. Achieving this flexibility while sealing the rig and avoiding undue stressing of wafers was one of the major challenges of this test programme. With the experiences from preliminary tests in mind a new test fixture to accommodate multiple test wafers was designed .

Fig 6 shows a section view of the revised and final Catpack tester design. The device aims to deliver the filtered fluid flow to the wafers (location as shown) in a 'plenum feed' fashion that is, slowing moving flow with a uniform velocity distribution. The wafers benefit from being located up against a steel backing-ring that helps to alleviate any stresses induced in the wafer from the impinging fluid flow that may cause it to deflect. The chamber has a 0.4mm diameter exit orifice in line with the flow direction and 2 ports for transducer connections perpendicular to the direction of the flow.

Results from Preliminary Testing

Two thickly coated (unoptimised) samples were exposed to a flow of HTP estimated at <1g/s, in a simple stainless steel test fixture derived from a filter casing. Some decomposition occurred as evidenced by steam emerging from a 1mm diameter flow restriction at the downstream end. Decomposition improved when the fixture was preheated by a hot air gun (temperature not measured), although 'steady state' decomposition was not sustained for more than ~10s of flow. Removal of the sample from the casing revealed the wafer had cracked and the silver had been largely removed. The latter was confirmed by immersing the wafer shards into HTP, where little decomposition occurred.

A thinly coated (optimised) sample was immersed in HTP, decomposition occurred but at a slower rate than for the two thickly coated samples. A flow test under similar conditions to the previous sample revealed decomposition after pre-heating, sustained for a similar time of around 10s. After this time catalyst bed flooding appeared to occur and decomposition ceased. Removal of the sample from the casing showed the wafer had cracked, and immersion of the shards in HTP showed no reaction, suggesting the silver had been completely removed during the flow test.

Results from main test campaign

A comprehensive set of test articles were tested to flow rates of 86% HTP ranging from 0.1 to 0.5 g/s. None of the test articles experienced any macroscopic scale damage as observed in the preliminary testing. Observations through scanning electron microscopy showed that there had not been any damage to the honeycomb grid structure on the wafers at microscopic scale neither after the coating process nor after the flow tests.

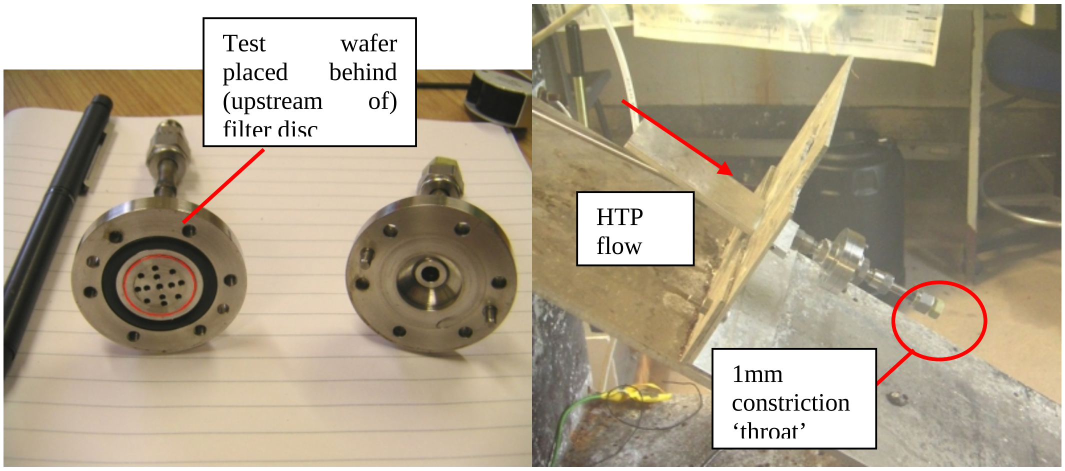

Fig. 6.: test equipment for wafer testing

Maximum decomposition temperatures of approximately 375ºC were observed for some short duration (in the order of 100 seconds) however this performance could not be reliably repeated.

The best result of HTP decomposition with the suspended catalyst bed was obtained with tests involving the four-wafer stack where there was an alkoxide based catalyst coating. With the use of pre-heat, a peak decomposition temperature of 350ºC was observed. Without pre-heat, temperatures of 170ºC were observed. The tests with this four wafer stack had subjected the wafers to a HTP flow for a duration of around 10 minutes and there was no indication of wash out.

The lack of repeatability is suspected to be a result of inconsistent catalyst coating deposition amongst wafers i.e. heavy coating on wafers resulting in either a greater number of channel blockages on some wafers or porous channels with layer-coated-channel-walls on other wafers. As a consequence there may exist intricate flow & decomposition mechanisms in the geometry of the honeycomb structure that are not yet understood and require further considerations supported by greater testing.

The recommended follow on work suggested after the tests was as follows:

- Consideration should be given to the flow dynamics occurring in the channels of the suspended catalyst bed and the channel size optimised if required. This should be done by simple fluid dynamic calculations & test if not by detailed computational fluid dynamics.
- Coating technique- the silver coating technique applied to the wafers for test should be optimised to allow an even coating to be reliably produced on the wafers whilst allowing the light and heavy loading variations to be investigated.
- The difference in thermal expansion coefficients between the catalyst and substrate shall be a key factor affecting the maximum firing duration or life of the catalyst bed. By way of test of numerous silver coated wafers the metal coating on non-metal substrate should be addressed.

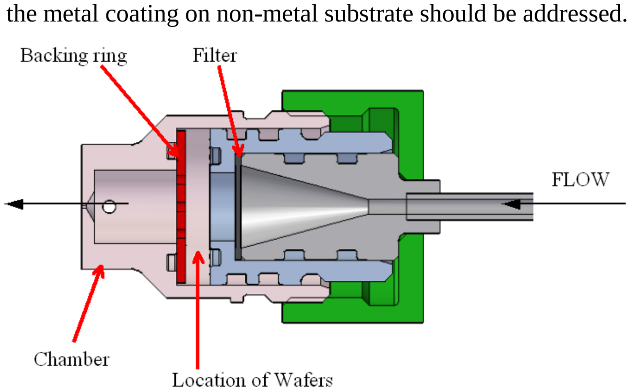

Fig.7. Test equipment designed for wafer and waferstack testing

- Further testing using the catpack tester investigating for repeatable and effective decomposition activity using an optimised set of test articles. Investigate the effect of lengthening of the suspended catalyst bed by using a stack of more than 4 wafers once reliable catalyst wafers are produced.
- The following factors require consideration before the manganate or silver based catalysts can become a viable option:

- Coating surface roughness.
- Chemical surface area.
- Adhesion between manganate catalysts and substrate.
- Inconsistency and variability between test wafers

CONCLUSIONS

This work shows that it is possible to design, manufacture and (partly) test a silicon based mono propellant micro-thruster. The team is confident that testing will advance to motor testing within the coming months.
Techniques and tools have been developed to design and analyse micro rocket motors and they are available for the development of future micro propulsion systems.
The technology of micro bi-propellant engines has been explored by theoretical analysis and this has shown that the technology developed can be extended in this direction.

It is recommended to continue the development of micro rocket motors in Europe and to specifically focus on components and feed systems.

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REFERENCES

[1] Salome R, “The propulsion system of the CNES microsatellite product line”, in Proc. 3rd Internat. Conf. On Spacecraft propulsion ESA SP-465, Dec. 2000, p. 99-105.

[2] Helvajian H (ed), “MicroEngineering Aerospace Systems”, the Aerospace Press / AIAA, 1999.

[3] Stenmark, L., A. B. Eriksson, “Cold Gas Micro Thrusters - Final Edition”, NanoTech 2002, Houston, TX, 2002. pp1-9.

[4] London, AP, Ayon, AA, Epstein, AH, Spearing, SM, Harrison, T, Peles, Y, Kerrebrock, JL, “Microfabrication of a high pressure bipropellant engine”, Sensors and Actuators A **92**, 2001. pp351-357.

[5] Dugue F, Roux F, Bonafos B, Billard J Y, Fruman D H, Gibek I, “Development of a micropump for hydrazine propulsion systems”, AIAA 2001-3831; presented at the 37th AIAA / ASME / SAE / ASEE Joint Propulsion Conference, July 2001, pp. 8.

[6] Klintberg L, Karlsson M, Stenmark L, Thornell G, “A thermally activated paraffin-based actuator for gas-flow control in a satellite electrical propulsion system”, Sensors and Actuators A, **105**, 2003, pp. 237-246.

[7] Köhler J, Moerel JL, Halswijk W, Baker A, Hebden R, Stenmark L, ”Detailed Design of a monopropellant microrocket engine using MEMS technology”, Journal of Micromechanics and Microengineering, 2005

[8] Hitt DL, Zakrzwski CM, Thomas MA, “MEMS based satellite Micro propulsion via catalysed hydrogen peroxide decomposition” Smart. Mater. Struct. 10; 2001 p 1163 - 1175

[9] Motti P, Tajmar M et al. “Bi-propellant Micro-Rocket Engine”, CANEUS 2004, 1-5 November 2004, Monterey, California, AIAA 2004-6707

[10] Larangot B, Rossi C et al. “Solid propellant microrockets – Towards a new type of power MEMS“, Nanotech 2002, Houston, 9-12 sept. 2002.

[11] Sanders HM, Halswijk W, Moerel JL, “Thermal design of a HTP Microthruster”, ICMENS 2005, 25-27 July 2005, Banff, Canada