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Miniature Turbine for Micro Rocket Engine

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ABSTRACT

Micro-satellites (from 10kg up to 100kg) have mass, volume, and electrical power constraints due to their low dimensions. These limitations lead to the lack in currently available active orbit control systems in micro-satellites. Therefore, there is a necessity of miniaturized rocket engine components development.
Mechatronic’s team is presently working on a liquid bipropellant micro-rocket engine under contract with ESA (Contract No.16914/NL/Sfe – Microturbomachinery Based Bipropellant System Using MNT). The advances in the project are to realise a complete micro-rocket engine consisted by miniaturized components in the mesoscale level. The system is composed mainly by two pumps, a turbine, a combustion chamber, and a nozzle. Small cooling channels around the nozzle are going to be employed in order to maintain the wall material below its maximum operating temperature. A mass budget comparison with more traditional pressure-fed micro-rocket engines shows a real benefit from this system in terms of mass reduction.
In particular, the miniature gas turbine is one of the key components of the system. In fact, it has to provide the required power for supplying the propellants micro-pumps which are electrically connected to the turbine – generator assembly. The miniature turbine is driven by hot gases produced by the decomposition of hydrogen peroxide and it is magnetically coupled to an electrical generator. Different turbines have been manufactured and tested.
In the paper, after a description of the rocket engine, the first experimental results obtained from the turbine prototypes will be presented.

1 PROJECT MOTIVATIONS

Recently, a micro-satellites market is emerging providing affordable access to space. Micro-satellites have very strong constraints in mass, volume, and available electrical power. Therefore, all the subsystems have to be miniaturised, in particular the propulsion systems for orbit control and attitude control. Moreover, actual micro-satellites have not suitable propulsion systems for orbit control. This capability lack, together with the usual “piggy back” launch condition, leads to the impossibility in a precise orbit injection of micro-satellites. It is thus clear that there is a current need of miniaturised propulsion systems for widening the capabilities of low cost micro-satellites.

2 ENGINE CYCLE

In order to realise micro-thrusters suitable for micro-satellites which are characterised by low costs, it is mandatory to use so-called “green propellants”. In fact, a large fraction of launch costs consists of safety procedures required for handling toxic and sometimes also carcinogenic propellants (i.e. hydrazine). The use of green propellants would also lead to a simplification and time reduction of propulsion tests, and in turn of thrusters development. In addition, storable propellants at standard conditions would be preferable for micro-satellites applications. In fact, the use of cryogenic propellants is not suitable for small spacecrafts due to the high complexity of cryogenic tanks which would completely pull down the benefits brought by miniaturised thrusters. Therefore, hydrogen peroxide and ethanol have been selected as propellants for the micro-rocket engine due to their heritage and successful use in past propulsion systems.
The project objectives is to exploit two micro-pumps for pressurizing the propellants before being injected in the combustion chamber. The propellants will be stored at low pressure in the tanks which will bring to high mass

savings in the storage system. After the pump, hydrogen peroxide will be decomposed in a gas generator and the produced oxygen and steam will be used for driving a miniaturised turbine (see Figure 1). The turbine power output will be exploited for powering the micro-pumps in a self sustained cycle. In order to overcome the thermal management issues, which are usually involved in micro-systems [1], an electrical connection between the relatively hot regions of the turbine to the cold parts of the micro-pumps has been designed. With this solution, the micro-pumps can have higher efficiency because they operate at low temperatures. In addition, they can be placed very close to the propellant tanks without any safety issues.

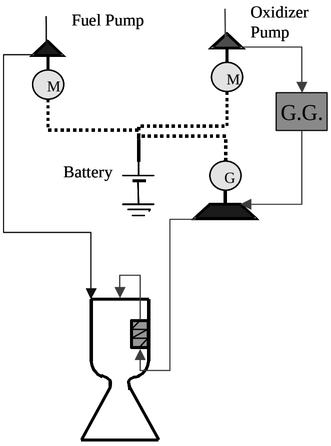

Figure 1 Engine Cycle

After the turbine, the decomposed hydrogen peroxide will be utilized for regenerative cooling the nozzle wall. The cooling of the nozzle wall has two main effects. The first is to keep the material maximum temperature below its maximum threshold. The second is to realise an almost adiabatic combustion chamber. In fact, micro-combustors suffer by a proportional increment in the surface / volume ratio as the dimensions are reduced. This corresponds to higher heat dissipation fluxes through the chamber walls that might leads to flame quenching [2]. In fact, in the case of no regenerative cooling the nozzle, a big fraction of the propellants inlet power would be dissipated by thermal radiation to space due to the high temperature of the nozzle wall. For the selected propellants and for a combustion chamber pressure of 0.6 MPa, the useful thermal power in the combustion chamber would be reduced to only 60/70 % of the inlet propellants power.
The main rocket engine cycle properties are summarised in Table 1.

Table 1. Rocket Engine Properties

Thrust	N	0.7 - 1
Combustion chamber pressure	bar	3
H_2O_2 concentration	%	70
H2O2 mass flow rate	g/s	0.32 – 0.37
Ethanol mass flow rate	g/s	0.06 – 0.07
Pump electrical power (Pump efficiency 15 %)	W	1.5

3 DESIGN AND SIMULATION

The miniature rocket engine encompasses a Propulsion Unit, an Electric Conversion Unit, and finally a Pump Unit. The Electrical Conversion Unit is further composed by a decomposition chamber, a turbine, and an electrical generator, whereas the Propulsion Unit is realised by a combustion chamber, a nozzle, and cooling channels.
The main components of the rocket engine are hereby described.

3.1 Decomposition Chamber

Hydrogen peroxide will be decomposed in a catalyst bed. The decomposed products of hydrogen peroxide is a mixture of oxygen and steam at a temperature which depends on the hydrogen peroxide concentration.
A monolith structure made of mullite and having square channels of 1 x 1 mm² has been designed for the decomposition chamber. In comparison with more traditional systems using gauzes or beads, this solution is preferable because it is characterised by the lowest pressure drop of the fluid. This is particular important due to the limited capacity of the micro-pumps to deliver high differential pressures.
As catalyst active phase, either metallic silver or manganese oxides could be utilized due to the limited concentration of hydrogen peroxide which is being used (70 % by weight) [3]. In Figure 2, there is one of the produced monolith impregnated with silver precursor on alumina.

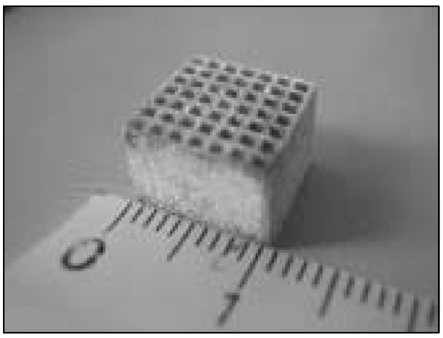

Figure 2 Monolith with Silver as Catalyst

The overall length of the decomposition chamber will be 15 mm and it consists of two monoliths having half the length. A modular system is thus realised where two monoliths with different catalyst materials might be used.
A proper injector has been also designed and manufactured in order to equally split the hydrogen peroxide mass flow rate through each channel composing the monolith.
The catalysts have been tested in order to assess their decomposition efficiency. In the test bench, the monoliths have been placed just before a small nozzle, thus realising a pressure fed monopropellant thruster. Different monoliths have been tested and the best results have been obtained by catalyst using MnO_x as active phase. The

evolving gas temperature of the decomposed products during one firing test is shown in Figure 3.

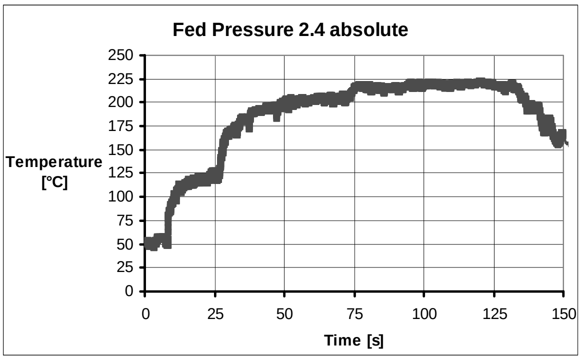

Figure 3 Gas Temperature During Firing Test

The picture reveals that the maximum decomposition temperature was 225 °C very close to the theoretic adiabatic decomposition temperature of 230 °C for 70 % hydrogen peroxide. In addition, there is a plateau zone at a temperature of about 120 °C. This plateau zone is most probably caused by the evaporation of trapped water in the porosity of the monolith. The trapped water derives from the initial transient phase when the gas temperature is lower than the water boiling point at the pressure conditions of the decomposition chamber.
Another issue that has been recognised during the test is the high pressure and mass flow rate oscillations produced during the firing tests. This might be particularly dangerous both in monopropellant thrusters applications as well as in driving the turbine. From a first analytical study, the oscillations seem to be produced by a coupling between the decomposition chamber with the propellant feeding system. According to the theoretical study and among different solutions which can be adopted, smaller feeding pipe diameter might be used to overcome the oscillations.

3.2 Turbine

Oxygen and steam produced by the decomposition of hydrogen peroxide are then used for driving a miniaturised turbine. The turbine coupled with an electrical generator should produce an electrical power of at least 1.5 W for supplying the propellants micro-pumps.
A radial impulse miniature turbine has been designed with one gas injector. The limited mass flow rate calls for the use of a single gas injector in order to limit the viscous losses. A single miniature ball bearing has been selected for supporting the rotor due to their commercial availability and long heritage in similar applications such as turbine for dentist hand-piece.
The turbine has been manufactured using a double step process. At first, the LIGA process has been used for the realisation of an electrode made of copper which has the negative pattern of the turbine blades. Then, the electrode is utilised in an EDM process for the replication of many turbines. LIGA is well adapted to the production of high aspect ratio microstructures with very steep and well polished side walls and it does not limit the choice of material to silicon and derivative, but offers the possibility to fabricate the turbine in metallic alloys. Moreover it has been found that LIGA, when coupled with EDM, is the best technique to obtain rotors with small diameter and high aspect ratio blades.

A picture of one of the first prototypes is shown in Figure 4, where there is a turbine with a rotor diameter of 10 mm. With the same process smaller turbines can be realised in different materials (i.e. aluminium, stainless steel, titanium, etc.).

Figure 4 Turbine Rotor Prototype

From the first tests it appears that the friction losses associated to the ball bearing are too high. In fact, the turbine idle speed was found to be not greater than 120000 rpm, which is far slower than the designed operational speed of about 230000 rpm.
In fact, at a rotational speed of 100000 rpm the bearing losses ranges from 60 % to 80 % of the inlet power and they are much greater than any other type of losses. These values have been obtained from the test results using equal ball bearings (ball bearing diameter 1.588 mm, balls number 12, internal bearing diameter 9 mm, external bearing diameter 14 mm, bearing width 3 mm, standard cage).
As a consequence, a new turbine design has been prototyped where the turbine rotor has been increased from 10 mm to 23 mm in order to reduce the rotational speed, but keeping the same power output of the previous turbine.
In the new version, a different bearing has been used which has balls made of Silicon Nitride instead of stainless steel. As rough estimation this modification can produce a friction reduction to about 20 % due to the lower material density. The ball diameter would be the same as before, but the bearing has only 7 balls instead of 12, thus reducing the contact points and therefore the friction. Moreover, the internal diameter of the bearing has been reduced to about 3.2 mm in order to reduce the centrifugal forces acting on the raceways and in turn the friction. For the bearing lubrication, ribbon type stainless steel retainer has been selected which is coated with PTFE and can work up to 200°C. In addition, the bearing is provided with stainless steel shields in order to avoid that the high temperature gas flow might disturb the balls motion.
In Figure 5, the electric conversion unit is shown, which includes the second turbine version and the decomposition chamber.

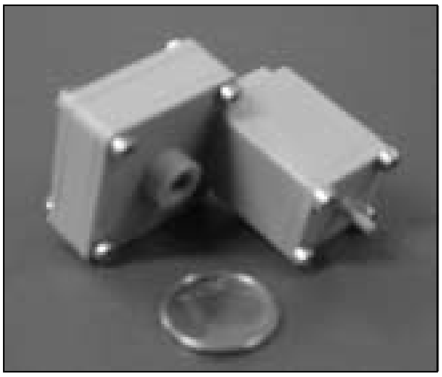

Figure 5   Electric Conversion Unit

The development of the miniature turbine is important not only for the micro-rocket engine purpose, but also for portable devices power generation.

3.3 Nozzle

The nozzle and combustion chamber will be manufactured together in one single part. The cooling channel will be realised by the gap formed between the nozzle and an external coaxial jacket. Additional cooling fins are designed at the nozzle throat in order to sufficiently cooled the nozzle. An injector plate with a glow plug will be used for mixing the propellants and igniting the mixture. For injecting the propellants in the combustion chamber a swirler injector has been adopted. This type of injector should realise a circulation region of hot combusted products in front of the injector in order to help the combustion of incoming fresh mixtures. The combustion chamber should achieve a power density of 1268 MW/m³ with a propellants residence time of about 1700 µs.

Different materials have been considered for the realisation of the nozzle. A stress analysis of the nozzle shows that some ceramic materials, even if characterised by a high temperature resistance, are not suitable due to their low thermal conductivity which generates too high thermal stresses at the nozzle throat. Among these materials which cannot be used is worth to mention alumina and zirconia. Instead, the best materials are found to be SiC, Si_3N_4, and Nimonic as metal. Due to easier manufacturing and welding, the first nozzle prototype will be realised in Nimonic 105, whereas a second prototype will be fabricated in SiC.

CFD numerical simulations have been carried out in order to assess the nozzle performance variations with some geometrical parameters. In particular, it has been found that the viscous losses are predominant at such small scale and therefore high convergent and divergent angles are preferable. For the same reason, also a small wall curvature radius at the nozzle throat gives better performance from a fluidic point of view, but for structural reasons it cannot be reduced too much.

In addition, CFD simulations have been carried out in order to check the cooling performance of the cooling channel around the nozzle. From the analysis it has been observed that the convective heat flow coefficients are much higher than the ones calculated using traditional semi-empirical relations. This results in lower nozzle wall temperature than calculated.

Thermal / Structural simulations of the nozzle show that the maximum wall temperature will be about 1100 K at the throat. The throat region is characterised by the highest temperature gradients and consequently by the highest stresses.

In Figure 6, the temperature and Von Mises stresses for the first 15 s of thruster firing are shown at the nozzle throat as obtained from the FEM analysis.

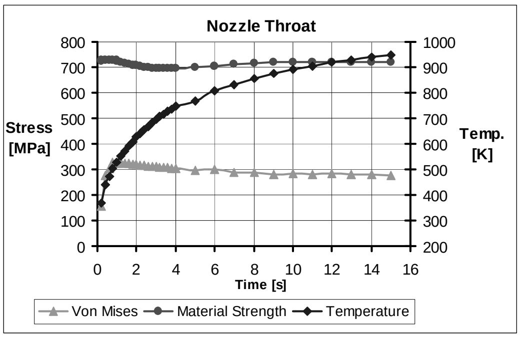

Figure 6   Nozzle Throat Transient Simulation

4 CONCLUSIONS

A miniaturised rocket engine has been designed for a thrust of 1 N. The main components have been manufactured and a test campaign is currently on-going in order to assess their performance.

In order to powering the pumps, the miniature turbine/generator assembly should produce an electrical power of 1.5 W. If the turbine would be able to produce higher electrical power, this could be exploited for powering not only the propellants micro-pumps, but also other electrical devices in a spacecraft.

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