FABRICATION OF LTCC BASED MICRO THRUSTER FOR PRECISION CONTROLLED SPACE FLIGHT

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Abstract: The paper at hand presents the initial investigations on the development and fabrication of a micro thruster based on LTCC technology, delivering a thrust in the micro Newton regime. Using smaller segments of an observation system distributed on two or more spacecrafts, one can realize an observation platform with a huge effective aperture. However, when having instruments on several spacecrafts cooperating, a demand arises for highly accurate positioning of the formation flying spacecrafts. Consequently, the purpose of the micro thruster being developed here is to facilitate constant elimination of the disturbance forces perturbing the orbits and attitudes of formation flying spacecrafts. In the flowing the investigations on the dimensioning and the calculations on the somewhat predictive performance are presented in short form.

Keywords: Disturbance forces, micro propulsion, LTCC technology, micro milling, capillaries, de Laval nozzles

INTRODUCTION

In the past few years, the space born observation systems have encountered a problem due to the demand for larger coherence lengths or apertures. The problem is the practical limitations on the size and mass of a single spacecraft, as well as the maintaining of mechanical stiffness and simplicity while increasing apertures. Consequently, in order to move forward with the improvement of the sensitivity and resolution on the space born scientific instruments, it can be necessary to distribute smaller segments of the instruments on several cooperating spacecrafts. However, when loosing the mechanical stiffness between these different segments, one needs the ability to control the relative position and orientation of the spacecrafts with extremely high precision. Hence, the minute disturbance forces – like atmospheric drag and solar radiation pressure – suddenly becomes significant due to their diverse influence on the attitude and orbit of the spacecrafts flying in formation.

Investigations show that the disturbance due to atmospheric drag is highly dependent on the solar activity, as seen on Fig. 1, and that it dominates the micro Newton regime for LEO. On the Fig. 1 the force acting due to the solar radiation is plotted also, and likewise this force reads 4.5 µN at the earth's distance to the sun. Hence, to be able to continuously eliminate these disturbances, it is intended to build a thruster with a smallest continuous thrust of a few micro Newton. When reaching such a low thrust, the main challenge becomes the reliability and as a result one may have to sacrifice efficiency in order to have a reliable working principle. Consequently, the thruster being developed here relies on the very simple principle of evaporating a fuel in a chamber instead of combusting the fuel – which on one hand would significantly increase the efficiency, but on the other hand also provide a larger thrust.
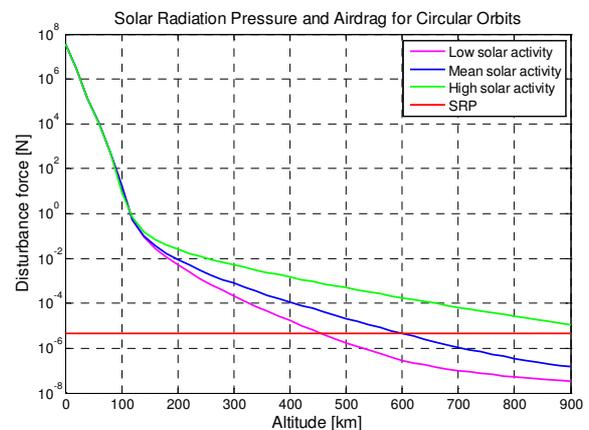

Fig. 1: Magnitude of the solar radiation pressure and atmospheric drag, for at satellite with a cross section of 1 m^2 and a drag coefficient of 1.

DESIGN

The body of the micro thruster developed here is made from LTCC material (Low Temperature Co-fired Ceramic). "Low Temperature" since the ceramic is sintered at a peak temperature of only 850°C and "Co-fired" since any present conductors and passives can be sintered in the same firing process as the substrate. The thruster is designed as a sandwich with two identical top and bottom parts constituting the two halves of the nozzles, as seen on Fig. 2. In between, the heater element is placed, which is a single sheet of LTCC with screen printed resistors located at the end

of every capillary tube. The connections to the resistors are made by filled via holes going through the two halves of the body.

Nozzles

For increasing the exhaust velocity and thereby improving the efficiency of the thruster, convergent-divergent de Laval nozzles are designed. Basically the purpose of the nozzle is to provide a fully expanded jet after the converging part has increased the exhaust velocity. However, if an overexpansion occurs – meaning that the exhaust pressure is lower than the ambient – the pressure will recover to ambient through a series of shock waves hindering the flow. On the other hand, if the jet is under-expanded the thrust is also reduced, since the ambient pressure will create a neck on the jet. However, in practice this thruster will always operate with some level of under-expansion, due the extremely low ambient pressure out in space.
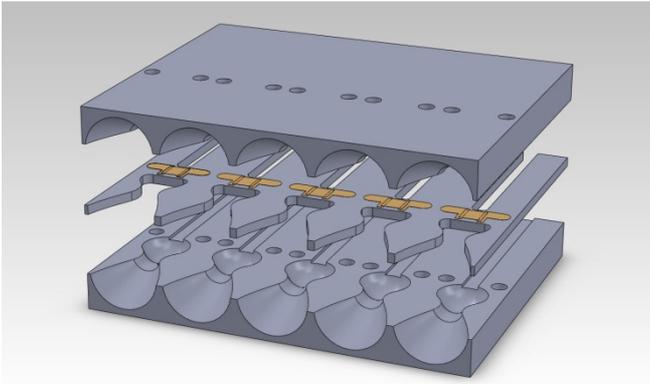

Fig. 2: The thruster is divided into three parts with the heater element in the center. Fuel will due to capillary forces flow through the 0.5 mm tubes towards the chambers and evaporate from the surface of the heaters.

The most important dimensions of the nozzles are the throat and exit diameters and the half angle α, see Fig. 3. The throat area determines the mass flow through the nozzle, as seen on Equation 6, and the ratio between the throat area and the exit area determines the exhaust velocity and exhaust pressure. To reduce the fluid boundary layer, the throat is rounded with a radius of curvature R_t equal to the throat diameter $2r_t$. In general, the bigger the nozzle and the lower the half angle is, the better expansion and axial thrust is achieved. But of course practicalities limits the dimensions on the nozzles at some point. The axial thrust may be shown to be approximately [1]:

$$F = \tfrac{1}{2}\dot{m}V_e\bigl(1 + \cos(\alpha)\bigr) \qquad 1$$

Choosing an α of 15° a good compromise between limiting nozzle dimensions and optimizing the axial thrust is achieved (for $\alpha = 15°$ the factor ½(1+cos(α))

reaches 0.986). The angle β of the converging part is typically 30°, but is of less importance.
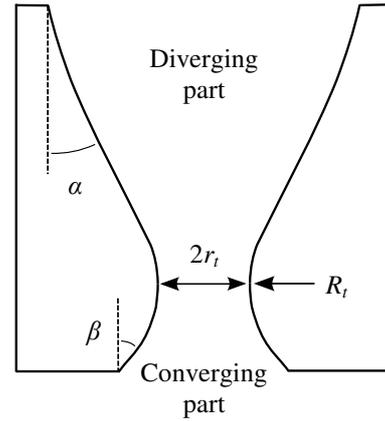

Fig. 3: Classic de Laval nozzle, with the converging part and diverging part.

Fuel tubes

The fuel is transported to the evaporation chambers through capillary tubes. Accordingly, these tubes are dimensioned to facilitate a fuel flow which continuously can supply the mass flow exiting the nozzles. The version described in this paper has fuel tubes with a square cross section. In this case the mass flow rate can be approximated by:

$$Q = \frac{\rho w \Delta p h^3}{12\eta L}\left(1 - 0.630\frac{h}{w}\right) \qquad 2$$

where η is the viscosity of the fuel, ρ is the density L is the length of the channel and Δp is the pressure difference given by:

$$\Delta p = 2\gamma_s \cos(\theta)\left(\frac{1}{h} + \frac{1}{w}\right) \sim \frac{\gamma_s \cos(\theta)}{w} \qquad 3$$

where γ is the surface tension of the fuel and θ is the contact angle between the fuel and the ceramic material. This approximation is valid for a very wide channel, but even for a square channel (with h = w) it is still correct with a 13% error. For the two fuels considered here, we find that the maximum possible mass flow rates are 2.54e-5 kg/s for ethanol and 1.85e-6 kg/s for rapeseed oil. Calculating the thrust vs. power curves later, we see that the mass flow needed for a maximum thrust of 100 µN per nozzle is only 1.2e-7 kg/s for ethanol, which is well below what is possible to transport through the capillaries.

THRUST VS. POWER

Knowing the specific latent heat L of the fuel, the amount of fuel m evaporating is related to the heat input Q by:

$$Q = mL \qquad\qquad 4$$

which translates into:

$$P = \frac{dQ}{dt} = \dot{m}L \qquad\qquad 5$$

where P is the power needed to maintain a mass flow of *dm/dt* going out of the evaporation chamber.

Analyzing the flow through the nozzles, it is assumed that the fuel obeys the ideal gas law, the specific heat is invariant with temperature and pressure and that the flow is one dimensional, steady and isentropic. The mass flow through the nozzle is then found by [1]:

$$\dot{m} = \sqrt{\gamma} \cdot \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \frac{p_c A_t}{\sqrt{RT/W}} \qquad 6$$

where γ is the ratio of specific heats, p_c is the pressure in the evaporation chamber, A_t is the throat area of the nozzle, R is the gas constant, T is the temperature and W is the molar mass of the fuel.

The conservation equations for mass and energy may be used to find the exhaust velocity [1]:

$$V_e = \sqrt{\frac{2\gamma RT}{(\gamma - 1)W}\left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} \qquad 7$$

where p_e is the exhaust pressure at the end of the nozzle. Assuming that the nozzle flow is ideally expanded, the thrust reads:

$$F = \dot{m}V_e \qquad\qquad 8$$

Combining Equation 5 to 8 the power consumption vs. thrust is calculated and plotted for two different fuels, as seen on Fig. 4. This is to show the difference on the sensitivity. The real thrust vs. power curves will look very different, due to the fact that much power will be lost due to black body radiation from the body of the thruster. The amount of heat lost due to radiation can be calculated from [1]:

$$P = A\varepsilon\sigma T^4 \qquad\qquad 9$$

where for the rapeseed oil we have a boiling temperature of 360: $P = 3.58 \cdot 10^{-4}\text{m}^2 \cdot 0.69 \cdot 5.67 \cdot 10^{-8}\frac{W}{m^2 K^4} \cdot 633\text{K} = 2.2\text{W}$ This is a significant amount of power compared to the milli-Watts used for the thrust. If power is very limited it is thus necessary to modify the design, so that the heating element is thermally decoupled from the thruster body.

As mentioned, much fuel efficiency is sacrificed when the fuel is not combusted. This is seen on the plotted specific impulse of the thruster, seen on Fig. 5. Normally for combusting thrusters the specific impulse is 10 times higher, meaning that a set of conventional thrusters is necessarily to be added for the high thrust applications of a satellite mission.
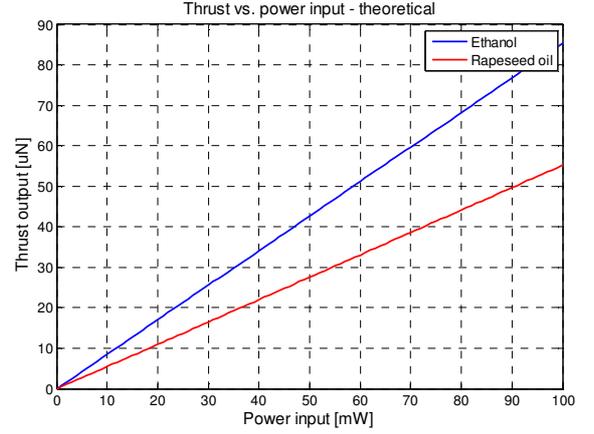

Fig. 4: The curves show the amount of thrust generated for a certain input of electrical power. Both curves are crossing the origin, i.e. it assumed that all electrical power is used for the evaporation.
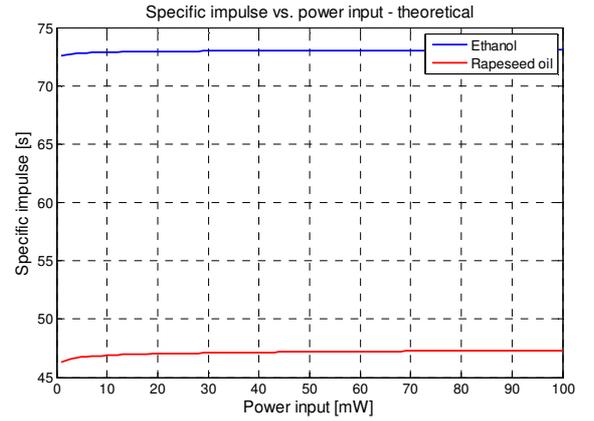

Fig. 5: The fuel efficiency is illustrated by plotting the specific impulse. Normally for chemical combusting thrusters the I_{SP} reads about 200-400s, so for many missions combusting thrusters would need to be added for the high thrust applications.

FABRICATION

The LTCC material used is constituted by an equal amount of Glass and Alumina, plus a binding agent keeping the material soft and suitable for mechanical processing. Before sintering, sheets of the material can be processed by laser, punch tool or milling. After sintering the fired substrate is very hard and ill-suited for mechanical processing, but resistant to high temperatures (below 850°C).

For mass production, techniques like laser cutting and punching are preferred due to the high throughput. However, the producible structures are very limited in terms of complexity, due to the 2D characteristics of these processing techniques. Consequently, the micro milling technique is used for the structures fabricated

here. With a 3D CAD program the structures are designed and drawn, and with a CAM extension for the program the required tool paths are defined. Using a 0.5 mm ball end mill tool the nozzles are then milled with a spindle speed of 7000 RPM. The nozzles are milled with constant Z and a step down of 0.1 mm in a roughing operation, leaving 0.05 mm of material for the final finishing operation. Subsequently the capillary fuel tubes are milled as seen on Fig. 6, and finally a 0.5 mm end mill is used to cut out the chip from the substrate.

Fig. 6: Micro milling during preparation of the capillary fuel tubes.

The resistors and conductors on the heating element, is applied using silk screen printing. In this process a thick film is printed onto the LTCC sheet and after drying the sheet can be placed in the stack with the two other parts of the thruster and thereupon laminated.

The three parts of the thruster is joined in a lamination procedure, where the parts are aligned in an aluminum fixture and exposed to a pressure of 270 bars while heated to 70°C. Finally the thruster is sintered in a 4 zone oven to reach a heating rate of 2-4 K/min, a dwell time at the peak temperature (850°C) of 20-30 min and a cooling rate of 3-6 K/min.

CONCLUSION

Initial investigations for fabricating an LTCC based evaporation thruster have been presented here. For use of continuous elimination of the present disturbance forces in LEO, it is necessary that the reaction force lies in the micro Newton regime for such a thruster. This can be achieved by evaporation of the fuel instead of combusting it, but with the price of lowering the fuel efficiency significantly.

Analyzing the flow through the nozzles for two different fuels, it is concluded that ethanol exceeds rapeseed oil on performance, both regarding the thrust to power ratio and the fuel efficiency measured on the specific impulse.

A very simple principle for transporting the fuel has shown to be effective. Instead of using a micro pump or similar, capillary tubes are used to transport the fuel to the evaporation chamber. In that way no moving parts are required and a much more reliable system is achieved.

For refining the design, calculations have shown that much power can be saved, by changing the design such that the heating element is thermally decoupled from the thruster body. In that way the amount of heat lost through radiation from the LTCC body – which is very heat conducting – could be reduced significantly. Finally, the micro milling technique shows to be well suited for processing LTCC, when 3D structures are needed.

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