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A MEMS-based solid propellant microthruster array for space and military applications

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Abstract. Since combustion is an easy way to achieve large quantities of energy from a small volume, we developed a MEMS based solid propellant microthruster array for small spacecraft and micro-air-vehicle applications. A thruster is composed of a fuel chamber layer, a top-side igniter with a micromachined nozzle in the same silicon layer. Layers are assembled by adhesive bonding to give final MEMS array. The thrust force is generated by the combustion of propellant stored in a few millimeter cube chamber. The micro-igniter is a polysilicon resistor deposited on a low stress SiO₂/SiN_x thin membrane to ensure a good heat transfer to the propellant and thus a low electric power consumption. A large range of thrust force is obtained simply by varying chamber and nozzle geometry parameters in one step of Deep Reactive Ion Etching (DRIE). Experimental tests of ignition and combustion employing home made (DB+x%BP) propellant composed of a Double-Base and Black-Powder. A temperature of 250 °C, enough to propellant initiation, is reached for 40 mW of electric power. A combustion rate of about 3.4 mm/s is measured for DB+20%BP propellant and thrust ranges between 0.1 and 3,5 mN are obtained for BP ratio between 10% and 30% using a microthruster of 100 μm of throat wide.

1. Introduction

The idea of powering MEMS has become feasible thanks to propellant integration in micromachined silicon structures. But things are not that easy. At this small scale, heat loss phenomena are so dominant that stifles the chemical reaction of combustion and prevents the reaction-control. This result a non-sustained reaction, not stable over time and not reproducible. Technological solutions are therefore necessary to remove these locks. Micro-combustion is became a vibrant research field and researchers are studying the stability and performance of micro-combustion in different applications such as drug delivery, Swiss roll combustor [1], micro-gas turbine engines [2], micro-thermo-photovoltaic system [3], a free piston knock engine [4], micro-combustor tube [5], radial channel combustors [6], micropropulsion [7, 8, 9] and in various kinds of other micro-combustor [10].



Recently, micro-spacecraft and many other low-cost miniature space systems have got a lot of attention due to the reduction of launch cost and various mission capabilities, and the development of new miniaturized systems is required. The propulsion system must be capable of providing extremely low levels of thrust and impulse, in order to satisfy micro-spacecrafts and miniature space systems propulsion need. However, there is almost no propulsion system appropriate in the market, and the development of new miniaturized systems is required. In this context, many micropropulsion systems based on MEMS technologies have been developed in the world. A micro-thruster is composed basically from three parts: a fuel tank of few cubic millimeters, an igniter and a micro-nozzle. The thrust force is generated by the combustion of fuel stored in a chamber and thus the gas is accelerated through the micro-nozzle. Propellant ignition is achieved by the Joule heating effect thanks to a micro-resistor. The use of solid propellant is an interesting choice because combustion is an easy way to achieve large quantities of energy from a small volume. Moreover, the system is simple, has no moving parts and presents a good efficiency (generated thrust to input power). Furthermore, there could be the perspectives of monolithic integration of electronics in order to realize the command. However, it is difficult to achieve a successful, reproducible and continuum combustion of the propellant in a millimeter scale device due to quenching effect [11]. Within a European project, Rossi et al. [7] fabricated a solid propellant microthruster to deliver thrust impulse of a few tens mN.s and a thrust force ranging from μN to few mN required for micro-spacecraft station keeping needs. It consists of a stack of four micromachined wafers: a chamber, an igniter, a nozzle and a seal closing the structure. Igniters and nozzles were realized on bulk silicon using MEMS technologies; chambers were fabricated in Foturan (photosensitive glass) and the seals were in Pyrex glass. Due to their low thermal conductivity, Foturan and Pyrex were used in order to minimize the thermal cross-talk between neighboring cells. The igniter and the Foturan chamber of $1.5\text{mm} \times 1.5\text{mm}$ of area dimensions were filled with Zirconium Perchlorate Potassium (ZPP) mixture propellant, and the Glycidyle Azide Polymer mixed with Ammonium Perchlorate and Zirconium (GAP/AP/Zr) propellant respectively. This design can not resist very high combustion pressure because the nozzle part is glued to the reservoir using polymer glue. Furthermore, the microthruster is powered from the front side, which leads to some difficulties for a future matrix addressing. Zhang et al. [8] proposed a flexible design of solid propellant microthruster where both reservoir and the nozzle were etched on the same silicon wafer, and its geometry can be changed at the same time using only one step of DRIE. But the use of igniters deposited on bulk material does not allow low electric power consumption.

This paper focuses on our progress toward the development of a micropropulsion array composed of 7 solid propellant units. Based on a previous work, we propose here a better adapted design, which allows flexibility to obtain a large range of thrust forces. The thrust force can be easily scaled by varying the geometry, the number of micro-thrusters and the type of propellant in order to satisfy the specific space/aeronautic applications. The key point of our design is the concept of initiator of thin membrane. This requires a very low power for ignition but a special care in microfabrication, loading and assembly processes. The design, fabrication process as well as the assembly process are described here in detail. After that, ignition tests and micro-combustion near extinction limit, employing several propellants, are also presented.

2. Solid propellant microthruster design

A microthruster is composed of three layers: a micromachined silicon layer containing initiator on thin membrane, chamber and nozzle, a ceramic (Macor) layer consists of chamber extension glued on the top side of the silicon layer and a glass (Pyrex) layer glued on the silicon bottom side (Fig. 1). This structure has been designed to work with a double base propellant at high combustion pressure (60bar) to reach $1\mu\text{N}$ to 50mN of thrust forces. A schematic top view of micro-thruster is shown in Fig. 2. Two types of thrusters have been designed in the same silicon wafer: $1.5\text{mm} \times 5.5\text{mm}$ and $1.5\text{mm} \times 3.5\text{mm}$ of $W_c \times L_c$ parameters with membranes of $\text{SiO}_2/\text{SiN}_x$ in order to have a large range of specific impulses. From computation, the chamber to throat section ratio must be of 100 to obtain the 60bar with a double base propellant. Three types of nozzle throats (W_t) have been realized: $100\mu\text{m}$; $150\mu\text{m}$ and $200\mu\text{m}$. The silicon wafer is a $545\mu\text{m}$ of thickness.

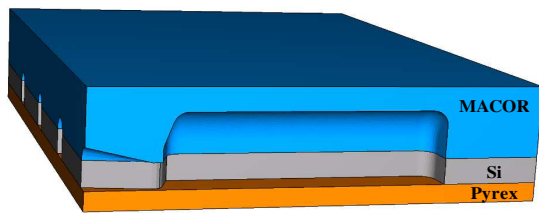


Figure 1. Schematic of the solid propellant structure array.

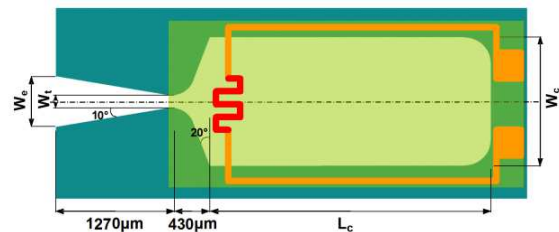


Figure 2. Schematic top view of a single microthruster with the geometric dimensions.

3. Summary of the fabrication and assembly processes

The microthruster process flow is simple but the challenge lies in attaining a highly variable geometry that maintains anisotropy over the full height of the structure. The chambers and the micro-nozzles are designed in the same silicon wafer. We know that large features etch faster than smaller features. This causes a residue of silicon on the nozzle throats when membranes are free. To completely perform nozzle throats, etch time will be extended that causes over etching of SiO_2 and destroy membranes. To provide over etching of the $\text{SiO}_2/\text{SiN}_x$ thin membrane, we realized SiO_2 layer thicker than calculation to be slim down by DRIE during performing the nozzle throats. Fig. 3 shows a SEM picture of the fabricated convergent-divergent nozzle. The throat wide is of $150\mu\text{m}$. The etched sidewalls are vertical in spite of the high chamber over nozzle area ratio (about 15). Micro-igniter layer is composed of a thin film polysilicon deposited on $\text{SiO}_2/\text{SiN}_x$ low stressed membrane and then etched for obtaining the resistor design. Fig. 4 shows a 3D optical microscopy of the polysilicon resistor design. The measured value of resistance is about 500Ω and the heating surface is of $550\mu\text{m}\times 550\mu\text{m}$. A maximum temperature of 100°C is reached for only 15mW and 450°C for 80mW .

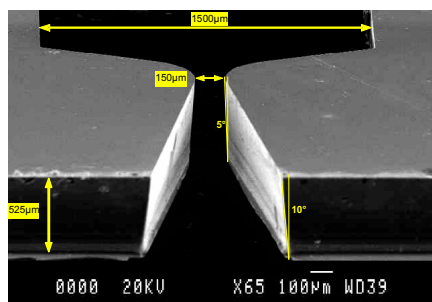


Figure 3. SEM photo of a micro-nozzle etched by DRIE on a $525\mu\text{m}$ thickness of silicon wafer.

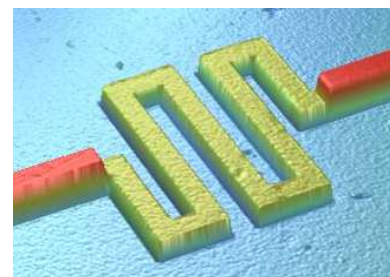


Figure 4. 3D optical profilometry picture of the thin film membrane with polysilicon resistor.

The fabrication and assembly process were described previously in detail [11]. A fabricated but not assembled silicon chip is shown in Fig. 5. It is composed of seven micro-rockets. Each single micro-rocket is separated with air grooves of 250 m width to prevent undesirable ignition from close neighbours. Silicon chip of $545\text{ }\mu\text{m}$ thick can be filled with energetic material and sealed from both top and backside using pyrex-7740 glass. But in this configuration the combustion is not continuum in the silicon chamber, and thus the generated thrust force is not reproducible. This is due to the extinction limit phenomena of the flame caused by heat loss in the silicon surrounding the combustion chamber. For this reason, we sealed the backside of the silicon microthruster array by a machined chip to add a supplementary chamber space. It leads to increase the ratio of chamber section to nozzle throat section, and by consequence increases thrust force. In a previous work, we have seen that silicon supplementary chamber causes unsuitable firing of the close neighbor of the tested chamber [7]. Macor has advantageous mechanical and thermal characteristics: coefficient of expansion very close to the silicon one, very low thermal conductivity ($1.46\text{ W}/(\text{m }^\circ\text{C})$) and high operating temperature ($1000\text{ }^\circ\text{C}$). Its drawbacks are: the high weight ($2.52\text{ g}/\text{cm}^3$) and the difficulty of machining.

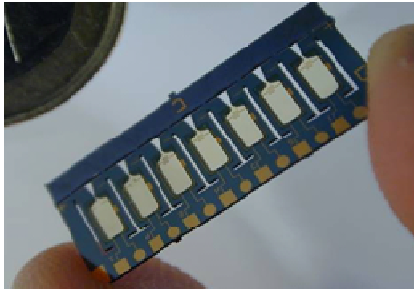


Figure 5. Photograph of 7x1 micromachined but non assembled silicon array

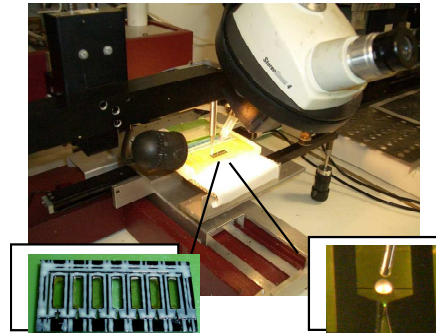


Figure 6. Photograph of the Flip-Chip setup for propellant loading and structure sealing.

There are three steps for the assembly process which are presented in Fig. 6. Firstly, the topside of silicon chip was sealed with Pyrex glass using H70-E epoxy glue. Then the silicon reservoirs were filled of propellant. This propellant was glued on the membrane by means of deposited micro-drops of the H20-E thermal conductive epoxy glue. Finally, the backside of silicon chip was sealed with machined Macor ceramic part using H70-E epoxy glue. Fig. 7 presents a micro-rocket array assembled and ready for test.

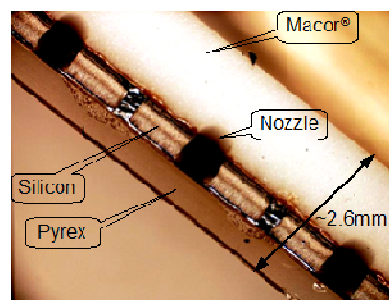
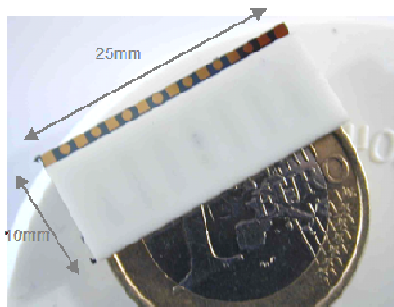


Figure 7. Photograph of a fabricated, filled with propellant and assembled 7x1 microthruster array.

4. Ignition and combustion and thrust investigations

Principle of an ignition test is based on the use of the polysilicon resistor on membrane both as heater part and piezoresistive sensor. The resistor is powered at specified electric current to heat the propellant, by Joule effect, until its ignition temperature. Then, the time of ignition is obtained when the membrane breaks (current =0) due to the gas issue of the starting of the micro-combustion. The voltage at the edges of resistor is measured. Data acquisition (voltage and current) are saved in computer and treated after. A double base propellant called SD1152 has been used for the ignition and combustion. A preliminary ignition test at atmospheric pressure enables us to find, for each configuration, the best ignition parameters such as minimum electric power and time of application. After that, we apply optimum ignition parameters for performing combustion tests for full assembled structure. Fig. 8 shows a successfully combustion test for an assembled microthruster. First results of combustion tests give a combustion rate of only 1.6mm/s. The combustion stops before the end of reservoir because the SD1152 propellant is not adapted for small sizes. Tests using DB propellant gave a successful ignition but not sustained combustion. Thanks to the home made (DB+x%BP) solid propellant, a complete combustions have been achieved for x more or equal to 10%.

For the thrust force measurement, we used a stand based on capacitive sensor [13]. The resulting thrust over the time is given in Fig. 9. Thrust level and combustion time are reported directly from the thrust curve versus time. The disturbances present in the curve during data acquisition are caused by the material of measurement and the environment air. To these disturbances are added fluctuations present only during the thrust application. They can be explained by inhomogeneity of propellant, down scaling effect (the

size of the propellant's grains become considerable compared to the chamber volume) and/or to approach towards the flame extinction limit diameter given of one millimeter by the literature.

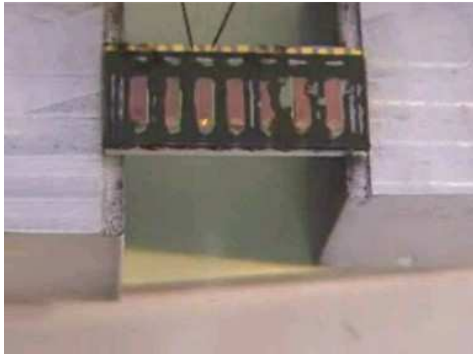


Figure 8. Photo of the combustion test of an assembled planar structure of micro-thruster

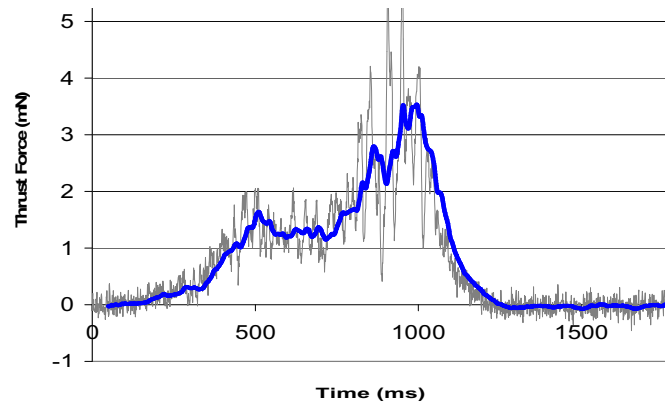


Figure 9. Thrust force vs. Time for thruster of $W_t=100\mu\text{m}$, loaded of (SD+30%PN) propellant.

5. Conclusion

In this paper, a summary of the fabrication and performance of MEMS solid propellant thruster array have been described. We improved the stability and low electric power consumption of the micro-igniter by using a thin film membrane. We developed a process to fabricate the micro-igniter on a thin dielectric membrane of $2\mu\text{m}$ thick instead of on bulk silicon or glass. A $0,5\mu\text{m}$ thick of polysilicon was used as the heater coil and the heating characteristic with the given electric power was estimated. The electric power required to obtain temperatures up to $250\text{ }^\circ\text{C}$, which is the initiation temperature of home made (DB+x%BP) propellant, was approximately 40 mW. The proposed concept of micro-igniter provide sufficient heat for the (DB+x%BP) propellant and allow the combustion of a large range of propellant because temperature of $450\text{ }^\circ\text{C}$ is obtained by only 80 mW of electric power. The micro-igniter, the chamber and the nozzle are realised on the same silicon wafer by the conventional MEMS technology. This study reveals a reproducible ignition process and stable combustion in a millimeter scale device thanks to the DB+x%BP propellant. The last one seems more adapted for MEMS than DB propellant. The use of Macor ceramic as chamber results in a better thermal insulation between the thrusters of the array, thus preventing from thermal cross-talk between each thruster. A combustion rate of about 3.4 mm/s is measured for DB+20%BP propellant and thrust ranges between 0.1 and 3,5 mN are obtained for BP ratio between 10% and 30% using microthrusters of $100\mu\text{m}$ of throat wide.

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