Solid Propellant MicroThruster : an alternative propulsion device for nanosatellite

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Abstract

With the aim to reduce cost and increase reliability, MEMS technology penetrates space market and new MEMS components are developed for Space application. Among them, micropropulsion devices is an active field of technological research. Several micropropulsion option are under development : the thrust is produced either from a electrical source (electrical thruster) or a chemical source (chemical thruster). LAAS has selected a chemical thruster option and develops arrays of solid propellant thruster. The concept of solid propellant thruster is based on the high rate combustion of one single propellant stored in a combustion chamber. The lack of restart ability is compensated by the fabrication of 2D addressed arrays of microthrusters. Each single thruster consists of an assembling of 3 parts. The main application of the microthruster is the micropropulsion for microsatellites (20-100kg) or nanosatellites (<20kg). In this paper, after an overview of the different micro thruster option under development, we present the design and fabrication of the solid propellant microthruster arrays. We give ignition and combustion test results. In this paper, we also present the microthrust balance used for thrust characterization.

1. Introduction

Building a cluster of small satellites should be cheaper, more robust and more versatile than building a single huge satellite. Typical science missions well adapted for microspacecraft application are space science, asteroid mission, or multi spacecraft observer clusters [1] [2]. Propulsion are a key point in the miniaturization of spacecrafts because micro and nanospacecrafts would need very small (below the Newton) and very accurate force to realize the stabilization, the pointing and the station keeping [3] [4] [5]. The level of thrust and the impulse precision required for nanospacecrafts maneuver can not be reached with conventional propulsion systems. For more than ten year micropropulsion field has been a world-wide active field of research. The technological efforts are made on two directions [6] [7]: the miniaturization of conventional thrusters, the development of new concept and devices. Most focuses on the use MEMS technology and hybrid technology for cost and weight gain. Among the propulsion systems under investigation for the miniaturization and using MEMS technology or hybrid technology are the cold gas thrusters , the subliming solid thrusters, bi-propellant thruster, the vaporizing liquid thruster, micro ion thrusters, FEEP, pulsed plasma thrusters, hall effect thruster.

The solid propellant microthruster based on MEMS technology could also bring an alternative to nanosatellites propulsion needs. The paper describes the design of solid propellant microthruster and gives initial results obtained for the fabrication, assembling and testing.

2. Overview of micropropulsion developments

2.1. Cold gas system

In 1997, cold gas system constituted smaller engines available [6] . An example of these motors has been made by MOOG [8], it has got a length of 4.3 cm, a diameter chamber of 1.4 cm and deliver 4.5 mN thrust with specific impulse of 65 s. The use of four piezoelectric microvalves to regulate gas flow enabled the reduction of leakage rate below 10^{-5}scc/s He. This new device is realized by hybrid technology [9] where highly integrated micromechanical parts are associated to conventional components. The hybrid system using nitrogen propellant delivers maximum thrusts ranging from 0.1 to 10mN with specific impulse of 45s. These characteristics enable short term nanoprobe missions or attitude control.

2.2. Bi-propellant thruster

This type of propulsion is commonly used for spacecraft, it is a complex structure resulting in separated parts. Since 1997 american team [10] [11] has been developing a bi-propellant engines. The motor consist of two tanks, two sets of valves and lines. The objective is to build a rocket engine able to generate a 15 N thrust and featuring a specific impulse of 300 s. First tests with a 1.2g device delivered thrust power of 750W and produced 1N thrust at a chamber pressure of 12 atm. A concept will be adopted with liquid oxygen and ethanol with the aim to have easiest cooling walls. The main advantages of bi-propellant systems are their high specific impulse, their ability to be restarted, and their high thrust-weight ratio. The main disadvantages are the power consumption required for pumps and valves, the necessary cooling system to avoid high temperature.

2.3. Mono-propellant thruster

Monopropellant thrusters offer a less complex technology than the bi-propellant motor. Two kinds of mono-propellant thrusters can be found in the litterature: hydrazine and hydrogen peroxide thrusters developed for nanosatellite attitude control application. Ordinary spacecraft use hydrazine propulsion systems for attitude control, primary and secondary propulsion systems. Hydrazine thrusters fabricated by PRIMEX, Kaiser-Marquardt, Daimler Benz operate in the 0.9-4.45N thrust range [6]. ASTRIUM has also set up hydrazine propulsion systems (Length: 113 mm; Nozzle diameter:4.8mm) for satellites application that can deliver 0.5N thrust.

The Micro Aerospace Solutions, Inc. of Melbourne has developed a MEMS based monopropellant thruster [12] operating in the milli-newton range. This device addresses micropropulsion systems for micro and nanospacecaft used for the exploration of asteroids, comets, Mars and moons. The disadvantages of hydrazine systems are the special required handling and the toxicity of the peroxide hydrogen. Hydrazine thrusters don't suffer of leakage, have greater propellant densities and feature good specific impulse. More over they require relative low input power. In consequence, the development of MEMS based mono-propellant thruster could be an interesting alternative for microspacecraft propulsion systems. Next generation of devices aim the following performances (fixed by NASA [13]): thrust force in the range of 10-500 µN, specific impulse of 145 s, impulse bit of 1-1000µN-s, adiabatic flame temperature lower than 1700 K.

2.4. Colloid thruster system

Before the apparition of emerging technologies, colloid thruster have been extensively studied but it was not possible to obtain sufficient thrust level with a reasonable input power to think eventual application for microspacecraft. The principle relies on the electrostatic acceleration of droplets from a capillary providing thus a thrust. The separation of the charged liquid propellant is possible with a high electric field. Colloid thrusters option can be an alternative to chemical propulsion. Indeed, this technology could give good specific impulse. Several American research group investigate this option. The university of Stanford and Santa Clara [14] [15] started a colloid microthruster research project named EMERALD[1]. They plan to test two kind of structures, one with single jet and a second with multi jet. Actually thrust delivered is ranging from 20 to 189N with a specific impulse of 389s . The goal is to reach 1-20µN and 500s [16].

A new colloid microthuster system [17] is elaborated integrating a PCB based

[1] http://ssdl.stanford.edu/Emerald

microthruster and a silicon micropump (200mW actuation power needed) which provides an accurate flow control of about 1µl/min. The obtained thrust depends on the driving potential : for 3500V the system delivers up to 2.56µN.

2.5. Field Emission Electric Propulsion (FEEP)

The FEEP thrusters use the acceleration and emission charged ions of a liquid metal with a high voltage between 5-12 kV. A tube or capillary represent the tank for the metal liquid. An electrical field of about 10^9 $V.m^{-1}$ is applied between both extremities of the capillary. Therefore, electrons are rejected into the bulk of the liquid metal while ions are accelerated by the same electric field to produce thrust. The emission of ions (LMIS: Liquid Metal Ion Source) is preferred to electrons source (LMES: Liquid Metal Electron Source) because of a inherent stability. Two types of metal liquid is used for these thrusters: Cesium [18] and Indium [19]. Nowadays, Cesium FEEP thrusters feature a specific impulse ranging from 6000 to 12000s while Indium FEEP thrusters give specific impulse in the range of 4000-11000s. In comparison with other electric propulsion systems, FEEP has the highest thrust-to-power ratio (182µN/W) and best specific impulse. FEEP system respond the best to fine attitude and orbit control requirements which is necessary for scientific mission as astronomy missions, interferometry in space, and microgravity experiments.
The miniaturization of FEEP thrusters [20] [21] are envisioned using Cesium and Indium propellant. The microfabrication techniques should produce micron sized cones (tubes or capillary) and microvolcanos to obtain emitter arrays. A MEMS based device has already given specific impulse higher than 50s and a thrust of 500 µN in vacuum with a tank pressure of 6 bar [20].

2.6. Plasma Pulsed Thruster (PPT)

The system of PPT consists to ablate and ionize a solid bar of Teflon (PTFE: PolyTetraFluoroEthylene) held between two electrodes by means of a spring and a shoulder anode. A capacitor is connected to electrodes and is discharged once the Teflon ionization obtained by spark plug firing (disposed near Teflon surface). The capacitor is charged by a Power Processing Unit (PPU). A current circulates along ionized Teflon between the electrodes and the discharge generates high field magnetic near it. The neutral plasma created by fluorocarbons particles is exhausting outside the thruster with the Lorentz forces. The PPT systems are able to provide specific impulse of 300-1200 s and Thrust-to-mass ratio ranging between 50µN/kg and 310µN/kg. The university of Washington works on a project named Dawgstar that consists in a formation. The nanosatellite Dawgstar [22] uses a PPT propulsion system delivering impulse bits of 70 µN-s with a specific impulse of 500s and needs voltages from 500-2000V to charge the capacitor. The total weight of the nanosatellite is 15 kg while the mass of the thruster is 3.8 kg.
A miniaturized propulsion systems weighing of about 100g is setting up by Air Force Research Laboratory[2] (AFRL) to addressed station keeping and primary propulsion of 25kg class microsatellites.

2.7. Micro Ion Thruster (µIT)

The µIT is a miniaturization of a conventional ion engine in which a low-pressure gas discharge is created through bombardment by electrons generated by a cathode. Ions are extracted from the gas discharge and are accelerated electrostatically to high velocities (about 30,000 m/s) in a set of accelerators grids being at a voltage of 1kV. Currently, a 3 cm diameter micro engine is under investigation at Jet Propulsion Laboratory (JPL) for microspacecraft. Micro ion engine development has not yet been performed. Many technical difficulties have been encountered in miniaturizing the device subsystems such as the cathodes, the neutralizers, and the grids [23]. Each of these subsystems requires extensive technical investigations.

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2 http://www.afrlhorizons.com/Briefs/Dec01/PR0109.html

Micro ion thruster concepts may be proposed for primary propulsion devices for micro and nano spacecraft because of their very high specific impulse (~3000s). The high level of the specific impulse results in a decrease in the propellant consumption to achieve a high ΔV requirement. μIT could also find applications in the field of attitude control for large spacecraft.

2.8.Hall Effect Thruster (HET)

The same propellant (Xenon) as μIT is used and principle relies on electrons source emitted from hollow cathode placed outside thruster. These electrons follow the opposite way of the electric field created between the anode through xenon is injected and the cathode. A coaxial channel placed between the two electrodes is fed with xenon. Inside this channel is realized an ionization of the neutral gas, then ions go on the direction of the hollow cathode and are exhausted outside the thruster providing propulsion. A radial magnetic field is applied across the channel to contain electrons and reduce their velocity increasing number of collisions and ionizations. This type of thruster is built by Busek company[3] (50-70mN/kW).
Models of HET[4] [24] lead to improve the system and also to see the behavior of the plasma for a future miniaturization of the structure. The interest of HET is mainly due to its efficiency and its optimum range of specific impulse (1000-2000s) enabling nanosatellites station-keeping application [25]. A Stationary Plasma Thruster calculations and characterization (SPT100-ML) gave the following characteristics [26]: for a voltage of 300V and a anode mass flow rate of 5 mg/s, the thrust reached is about 70 mN for Xenon and Krypton with a global efficiency respectively of 40% and 30%. This thruster will be used for the launching of an experimental satellite STENTOR by the French Agency CNES. In 1998 the Hokkaido Institute of Technology decided to build and launch a 3kg nanosatellite [27] fed with a maximum power of 30 W and having a hall thruster on board.
The high magnetic field used to contain electrons is the main drawback of the HET as for μIT.

2.9. Laser Plasma Thruster (LPT)

A LPT project is under investigation at the university of New Mexico [28] to fabricate a prototype engine of 5 kg for low earth orbit. The system is working with recent commercial diode lasers enable to do 100% duty cycle with sufficient brightness. The diode lasers are producing an ablation of polymer as PVC or Kapton by heat exchange to extract atoms from the surface, then a spark is created involving vapor and plasma formation. The miniature plasma jet generates the thrust. Two configurations are tested : either by reflection or by transmission. The thrust-to-power ratio is 50 to 100 N/MW and the specific impulse is 200-500 s with 1 W laser.

2.10. Vaporizing Liquid Thruster (VLT)

An typical example of MEMS based micropropulsion device is the vaporizing liquid thruster. The change of phase liquid-gas is exploited to produce a thrust. A structure proposed by the university of California [29] is composed of a polysilicon heater realized on silicon top side substrate. The backside is micromachined to form a cavity named vaporization microchamber. This chamber is sealed by anodic bonding with a pyrex wafer also micromachined. The different parameters influencing the thrust are the chamber length, the nozzle geometry, the heater power and the liquid flow rate. The structure requires 5 W to heat the water until its vaporization. Then the vapor goes into the microchamber at a maximum flow rate of 0.09 cc/s. The obtained thrust ranges between 0.15 mN and 0.46 mN. A set up of dry etching (hot and cold supersonic microjets [30]) enable to provide highest thrust. The main drawbacks of the vaporizing liquid thruster is the low thrust-to-power ratio due to the high thermal conductivity of silicon.

[3] http://cpa94.ups-tlse.fr/operations/operation_03/POSTERS/SPT/
[4] http://www.busek.com/bpt.htm

2.11.Micro Solid Propellant Thruster

For the beginning of space industry, solid rocket motors are among the most powerful propulsion systems. They have a better reliability than the liquid propulsion and can be also are used during the boost phase of space launch due to their high thrust. The miniaturization of solid motor are attractive for MEMS researches group. This has started in 1997 at the Laboratory for Analysis and Architecture of Systems (LAAS-CNRS, France) and numerous solid state thruster are under investigation in Europe, USA and Japan ([31] [32] [33]).

The concept is based on the high rate combustion of one single propellant or explosive stored in a combustion chamber. The lack of restart ability is compensated by the fabrication of arrays of microrockets. The gas generated by the combustion of the propellant is accelerated in a nozzle thus delivering a thrust. This concept of solid propellant thruster offers several advantages: there is no leakage because there is no liquid fuel, there is no moving parts and the solid propellant ignition requires low power consumption. First theoretical thrust evaluation gave results ranging from a few mN to a few hundreds of Newtons depending on the design of the thruster (section ratio between chamber and throat area) for a given fuel. These solid micro-thruster could find applications for small satellite station keeping. The main drawbacks are their one-shot characteristics and their relative low specific impulse.

3.LAAS solid propellant microthruster array

In 1997, the LAAS-CNRS in collaboration with CNES have initiated a research program aiming to develop micropropulsion system based on the use one solid propellant stored into a silicon microfabricated system. The project is now funded by the European community[5] and aim the fabrication of module of solid propellant thrusters using solid propellant.

[5] http://www.laas.fr/Micropyros

3.1.Structure

Figure 1 gives a schematic view of one single thruster. It consists of 4 parts of silicon :

- A silicon micromachined igniter with a polysilicon resistor patterned onto a very thin dielectric membrane.
- A propellant reservoir which defines the combustion chamber.
- A nozzle wafer added on the top of the structure.
- A seal wafer.

Chambers are fabricated using Foturan® or silicon and the nozzle and igniter parts are realized by micro machining the silicon.

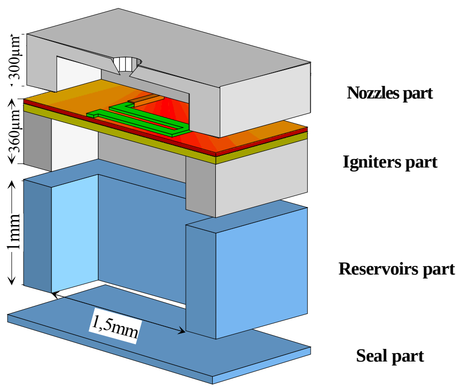


Figure 1. Schematic view of one single microrocket

Igniters are fabricated by silicon micromachined. Chambers are fabricated using Foturan® glass or silicon and the nozzle part is realized from silicon.

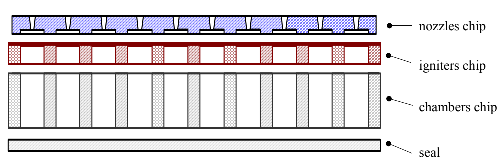


Figure 2. Cross-section of a chip.

The lack of restart ability is compensated by the fabrication of 2D addressed arrays of microthrusters. The cross-section of a chip is shown on Figure 2. It is constituted of a matrix of 10 x 10 single thrusters.The 2D addressed array contains 100 separate micro-thrusters on 24mm × 24mm silicon chip. The interconnection techniques and assembling procedures are strongly related to the electronic system architecture definition compatible to microsystem assembling. Each thruster could be addressed independently on the order of a computer.
The addressing is realized by placing PN junction in series with the heater filament as shown in Figure 3.

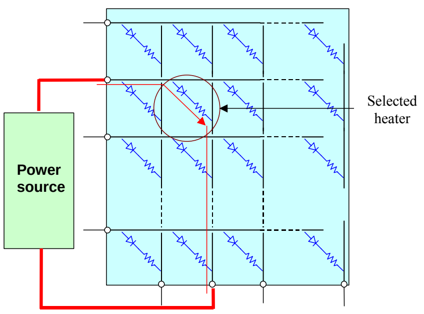


Figure 3: Schematic view of a 2D addressed matrix of resistors.

3.2.Assembling process

Because of the simplicity of the process and probable problems because of contamination after filling of the chambers with propellant, adhesive bonding was chosen for the assembling of the device.
At first the nozzle part is bonded to the igniter part by a thermal epoxy (H 70 E, Polytec) and the seal wafer to the chamber wafer by a UV sensitive polymer (NOA 88, Norland) that can be spun-on the seal wafer. Since for the seal wafer Pyrex is used, the adhesive can be polymerized UV-light. Then the igniter part is filled with the ignition propellant and the combustion chambers filled with the GAP based propellant. Finally the two wafer stacks are again bonded by thermal epoxy (H 70 E, Polytec). The annealing temperature of 60°C ensures that no ignition of the propellant occurs. Table 1 summarizes the assembling process.

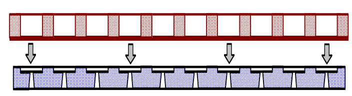
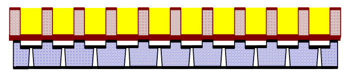
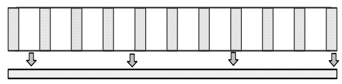
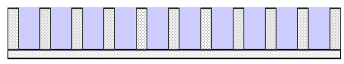
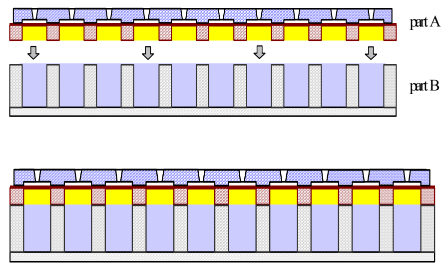
Step		Scheme
#1	Assembling by gluing ►Part A	 igniter nozzle
#2	Filling with propellant	 Part A
#3	Assembling by UV gluing ►PartB	 chamber seal
#4	Filling with propellant	 Part B
#5	Assembling by low T epoxy gluing	 part A part B

Table 1 : Assembling process

3.3.Ignition tests results

Arrays of 16 non-addressed heaters have been realized to characterize the propellant ignition. For ignition process, we used a Zirconium doped composite propellant. The filament is powered with an electrical current via a programmable power supply. When the ignition propellant is

Ignition Power P(mW)	150
Heating surface (mm²)	0.518
Power density (mW/mm²)	289
Ignition delay (ms) mean values	460
Ignition energy (mJ) mean values	69
Density of Ignition energy (mJ/mm²) mean values	134
% of Ignition sucess	100

Table 2. Ignition characterization results with a heating surface of 0.518mm^2.

heated up at its ignition temperature, it burns leading to the rupture of the membrane and resistor.
To determine the ignition energy, the time between the beginning of the ignition and the rupture of the membrane is measured. First results concerning the density of energy necessary to provide for the ignition have revealed a requirement of 134 mJ/mm²

Figure 4. Photos of one flame

A thermal model will be established in order to optimize the electrical power and therefore the heat surface area.

3.4.Thrust measurement

The thrust stand used to characterize micro-rocket is schematically given in Figure 5. It consists of a thin and rigid arm (100 µm thick) rotating freely around a pivot. The thruster is mounted at the end of the arm.

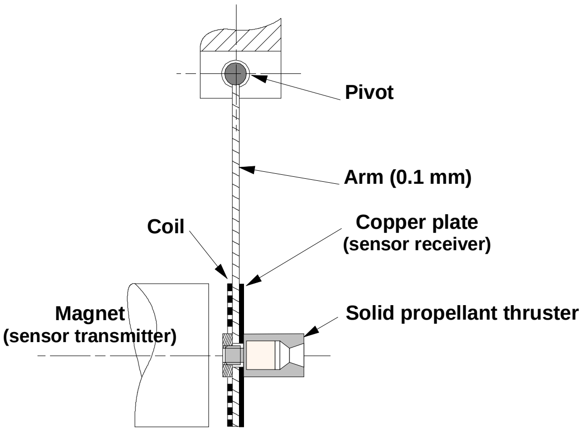


Figure 5. Schematic view of the thrust stand.

On the opposite face of the arm and aligned with the point of thrust application, a coil is deposited. The thrust stand is fixed on a marble block to minimize vibration disturbances during measurements.
The coil is immersed in the magnetic induction field produced by the permanent magnet. When the thrust force is applied, the pendulum rotates around the pivot axis. A displacement sensor feeds a voltage to the electronic control, which increases the current in the coil and thus the restoring force until the reference position of the pendulum. The displacement sensor is a high frequency transmitter and an antenna, respectively a magnet and a copper plate. The distance between magnet and copper plate modulates the amplitude of the high frequency signal. A feedback control is obtained with a PID (Proportional Integral Derivative) circuit. Measurement of the equilibrium force equal to the thrust is obtained by measuring the electric current. For static calibration purposes, the pendulum is laid horizontally. Calibrated masses from 1 mg (10 µN) to 2 g (20 mN) are placed coincident with the thrust axis. The entire loop gain is 2.549 mN/V and sensitivity is 25 µN. The natural frequency of the system is 42 Hz. This stand has been used for measuring the thrust impulse and the combustion rate. First thrust characteristics gave resulting impulse bit force of 6mN.s.

4.Conclusion

The paper overview the different micropropulsion devices developed for space application and detailed the solid propellant thruster option investigated at LAAS. The work described has been performed within an EC funded project (IST-99047). Partners are LAAS (CNRS laboratory-France), IMT (University of Neuchâtel-Switzerland), IMTEK (University of Freiburg-Germany), SIC (University of Barcelona-Spain), ASTC (University of Uppsala-Sweden), LACROIX (France).

In conclusions :

- Microthruster samples have been successfully tested with solid propellant.

- Flexible 2D addressing design has been realized.
- Study of epoxy and UV gluing has been performed. Even if one design has been presented in this paper, the structure is flexible and can incorporate different heater sizes, different chamber lateral dimensions. Different propellant material have been formulated and tested. Characterizations showed that the retained propellant required an ignition energy of 134mJ/mm².A thrust stand has been built and will be used for the ground testing of the micro-rockets array.

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