The Fifth International Workshop on Micro and Nanotechnology
for Power Generation and Energy Conversion Applications
PowerMEMS 2005
Nov. 28-30, 2005
Takeda Hall, The University of Tokyo, Tokyo, Japan

A Bipropellant Liquid MicroRocket Engine System

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Abstract

This paper is an overview of the research at MIT on MEMS liquid bipropellant rocket engines. Sized to provide 10-15N of thrust at a thrust-to-weight ratio of about 1000:1, these engines have specific impulse approaching the theoretical value achievable for the propellants. The motivation for this effort is twofold: first, to explore the high power density boundaries of microrocket propulsion systems, and second, to demonstrate a useful propulsion capability. The basic approach is to engineer miniature versions of rocket engine subsystems – thrust chamber, exhaust nozzle, turbopumps, valves – redesigned to be compatible with conventional MEMS silicon processing technology. Work has also been done to characterize the thermo-fluid behavior of rocket propellants at conditions representative of the MEMS rocket engine operating environment.

Keywords: PowerMEMS, microrocket engine, micropump, microvalve

INTRODUCTION

MIT has been working to realize a complete MEMS bipropellant, high pressure liquid rocket motor – including propellant turbopumps, control valves, injectors, thrust chamber, and nozzle – as shown in Fig. 1. At the millicentimeter size scale of interest here, the fuel economy (specific impulse, Isp) of such an engine need not be different than that of current larger scale engines with similar cycles. What is changed is the manufacturing technology which in turn sets both the geometric constraints and the material system. The geometry is constrained to be largely two-dimensional while the materials change from nickel-based superalloys to single-crystal silicon or silicon carbide. The T/W capability of a MEMS engine is superior due to the scaling advantage of small size (cube-square law) combined with the strength-to-density advantage of silicon over nickel superalloys (3:1 better). This paper presents an overview of the work to date at MIT on this MEMS propulsion system.

BASIC DESIGN PRINCIPLES AND CONSTRAINTS

At fixed size, engine thrust scales with combustion chamber pressure so high chamber pressure is the key to realizing high thrust-to-weight ratio. Also, there is a minimum chamber pressure required to avoid vibrational freezing of the exhaust gas which would reduce Isp. This minimum pressure is on the order of 10 atm. The superior mechanical properties of single crystal silicon, 4 GPa yield strength [1], are sufficient to overcome the disadvantages of a planar pressure vessel geometry shown in Fig. 1 demanded by MEMS fabrication limitations and of a brittle material in tension. The structure of such engines can be designed to operate at pressures comparable to those of large engines, 100-150 atm, so that vibrational freezing need not be a problem.

As size is reduced, the heat transfer/thrust grows so there is proportionately more heat transfer in a small engine. So long as the engine is cooled by the propellants ("regeneratively" cooled), this heat transfer has little effect on the engine performance. It does, however, limit the minimum size of the engine since, as the engine shrinks, the heat capacity of the coolant drops faster than the heat load [2].

A reduction in size does not change the functionality required of the chemical propulsion system. A complete liquid rocket propulsion system must include a thrust chamber and nozzle, controls, and, when high-thrust–to weight is desired, high pressure pumps and power for driving the pumps. As for large engines, many pumping cycles and system configurations are feasible. The evaporator cycle circulates the propellants around the chamber and nozzle to cool them and

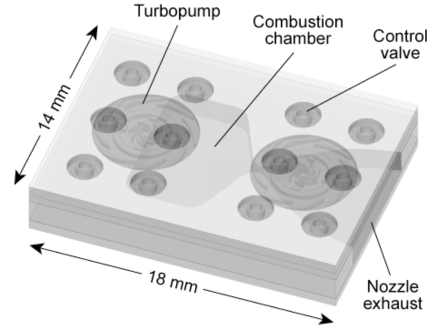


Figure 1. Micro-bipropellant MEMS rocket engine concept sized for 10-15N thrust.

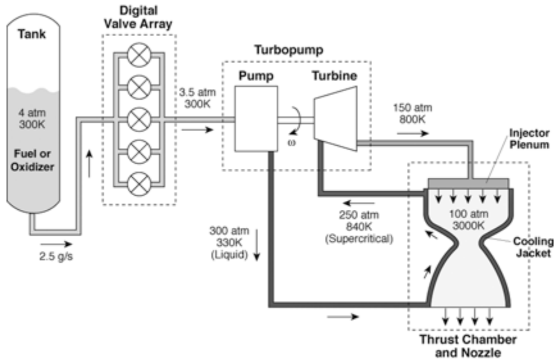
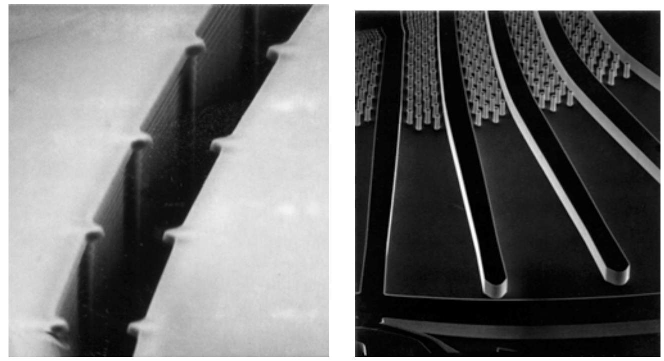


Figure 2. Evaporator cycle, pump-fed microrocket engine system.



(a) (b)

Figure 4. (a) 9 μm wide side wall cooling channel near the nozzle throat with turbulator ridges. (b) Nozzle endwall cooling channels with 10 μm dia cylindrical turbulators.

then uses the heat absorbed to provide the enthalpy necessary to drive the pump turbines. This is perhaps the simplest approach in that it has the fewest components. MIT adopted such a design for liquid oxygen-hydrocarbon [3] and storable propellant engines [4], Fig. 2. A digital valve array is fed from propellant storage tanks at sufficient pressure to prevent cavitation (2-4 atm). The valves modulate the flow to the turbopump. The following sections describe the individual system components.

THRUST CHAMBER AND NOZZLE

The regeneratively cooled, micro-thrust chamber and nozzle assembly was first described in [2]. It consists of six layers of silicon which are deep reaction ion-etched (DRIE) and then diffusion-bonded together, Fig. 3. The thrust scales with motor thickness which is determined by the number of layers (constrained by precision-aligned bonding technology) and by the layer thickness (limited by the aspect ratio capabilities of the DRIE process for producing narrow cooling passages) [5].

Silicon has similar strength versus temperature characteristics as nickel superalloys so the chamber and nozzle must be cooled. Cooling in these MEMS devices is aided by the high thermal conductivity of silicon and by the sophisticated cooling geometries enabled by the flexibility of micromachining. Figure 4 shows a 9-micron-wide cooling channel in the nozzle throat side wall and the 10-micron-dia cylindrical turbulators arrayed throughout the top nozzle wall cooling passages. Figure 5 shows a detailed cross-section of a laminated 6-wafer combustion chamber with 14 (oxidizer) and 18 (fuel) micron dia propellant injectors and cooling passages. This oxygen-ethanol device was designed at MIT in the late 1990’s and has produced about 3N of thrust at an estimated vacuum Isp of 300 sec operating at a chamber pressure of 30 atm [4]. The thrust has been limited by packaging failures, which are now thought to be resolved [6,7]. The cooling channels have survived pressures of over 150 atm and the chamber 60 atm [4]. As built, these are capable of producing thrust of 9N. Design refinements and DRIE advances since then permit higher chamber pressures and larger sizes so that evolved designs are expected to produce thrusts up to 25-50N. Also, the design space has been expanded by experimentally determining the coolant properties of alternate propellants at microscale [8,9].

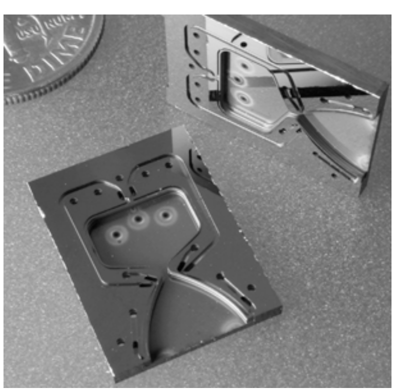


Figure 3. Cooled Si combustion chamber-nozzle.

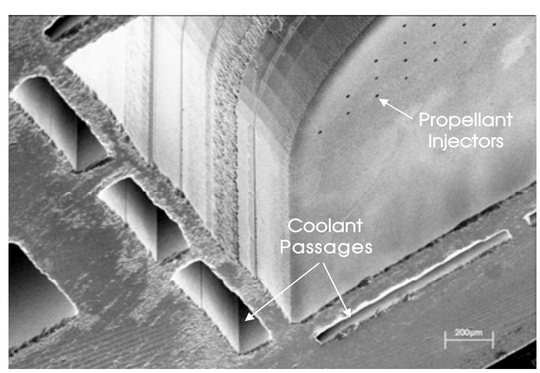
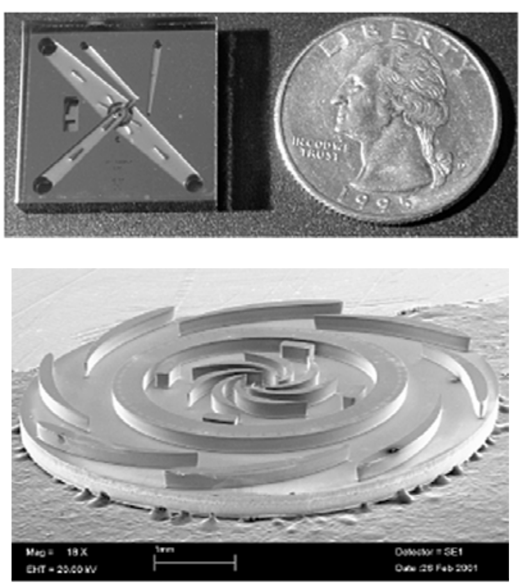


Figure 5. Cutaway TCA head showing cooling passages and propellant injectors.

TURBOMACHINERY

The turbopumps are the most challenging component in a microrocket engine system, just as they are for large-scale engines. The engineering concerns of MEMS micropumps are those familiar to large pump designers and include: cavitation, fluid dynamic efficiency, mechanical stress, bearings and rotor dynamics, and sealing. A study of cavitation in microcascades showed that cavitation behavior is independent of airfoil size, so that standard techniques can be used to predict and avoid cavitation in micropumps [10]. Pump and turbine efficiency are lower at microscale, but this has only a small impact on engine system level performance. Mechanical stress concerns are reduced by the high strength-to-density ratio of silicon which permits relatively simple rotating part geometries. Fluid bearings can use the same working fluid of the pump or turbine to minimize leakage and sealing challenges. Rotor dynamics are always a concern for high speed fluid bearing systems and remain so at microscale [11].

Pressure rises of 100-200 atm in a single centrifugal stage can be difficult to realize with many rocket propellants due to one or more of the above concerns so that inducers or boost pumps are often used. To demonstrate the feasibility of a MEMS silicon turbopump, MIT designed and built a 0.5 grams/sec, 20 atm pressure rise “demo” pump [12]. This is sufficient head to suppress cavitation in a 120 atm pressure rise MEMS pump. The demo pump is shown in Fig. 6. The top figure is the chip seen through its glass upper layer (used for flow visualization). The lower figure is an isolated 6 mm dia rotor. Its unusual planar arrangement was adopted to simplify microfabrication. The 1 mm dia centrifugal pump is at the center, with the radial inflow turbine rotor blades around the periphery. The raised ring between them is a combined axial fluid bearing and seal. The rotor is supported for axial loads by this bearing and its mate on the rear of the disk. The disk rim forms the journal of a gas bearing which supports all radial loads. Designed for 750,000 rpm, this machine has spun to date to about 100,000 rpm [13], with the measured flow and pressure rise consistent with the theory. Turbine rotors of similar geometry have spun to 1.4 M rpm [14].



(a)

(b)

Figure 6. Demonstration microturbopump: a) 2 cm sq turbopump chip; b) 6 mm dia turbopump rotor.

CONTROLS

High performance liquid rocket propulsion systems require sensors and control valves for startup, shutdown, and thrust modulation. There are hundreds of MEMS valves and flow sensors in the literature but none met the simultaneous, exacting space propulsion requirements for environmental compatibility, flow rate, pressure drop, leakage, time response, and chemical compatibility. Consequently, MIT designed and built valves and sensors suitable for microrocket engine application. A two-stage pilot-servo valve architecture was adopted in which the flow from an electrostatically operated pilot valve opens a servo valve with about 10 times its flow rate (about 0.5 g/s of $H_2$0) against 10 atm [15]. Scaling of these actuation force constrained valves is such that an array of small valves has more flow capacity than a single valve occupying the same total area, so ten such valve pairs are used in parallel to give the throttle capability needed for startup and thrust modulation.

The technical advantages of the electrostatic actuation are very low power consumption (less than a microwatt full open), fast response (a few milliseconds to full open), wide temperature range (cryogenic to 700 K), and compatibility with standard MEMS processing approaches. The major disadvantage is the low force per unit area of electrostatic actuation (about 0.5 atm), requiring a relatively large actuator area ($1mm^2$) and a pressure balance approach (which adds to the fabrication complexity). Both a servo and a pilot valve have been built and tested separately (Fig. 7), but have not yet been operated together.

There are a large number of MEMS flow sensors which can measure pressure, temperature, and velocity. As with valves, selection is constrained to approaches which meet the stringent requirements of the rocket engine system while being process-compatible with the fabrication of the other components. One relatively simple approach which fulfills these requirements is the 2 microns thick, thermally isolated thin-film resistor shown in Fig. 8. Designed as an rpm sensor for a microturbine [16], the sensor can also measure fluid temperature while two together can measure flow velocity.

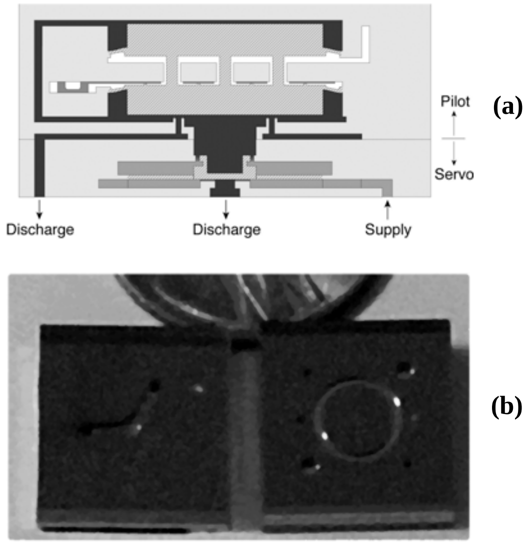


Figure 7. Rocket propellant valve: a) valve cross-section, b) valve prototypes, servo (left) & pilot (right).

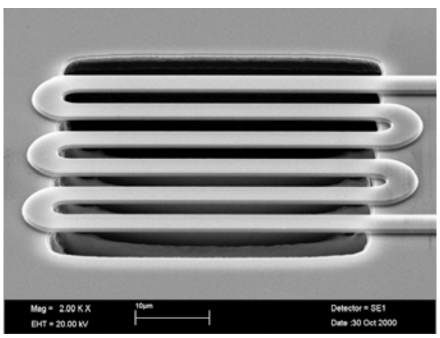


Figure 8. 50 μm sq. flow/rpm sensor.

FUTURE WORK

The work to date at MIT has focused on demonstrating operation of the individual components of a MEMS bipropellant chemical rocket engine system. Once this is completed, the next step is to demonstrate the complete system, with components integrated with external packaging. Flight systems will require wafer or die level component integration as well as multi-engine interconnection schemes to enable groups of engines to work together and be robust to individual failure [17].

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