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Highlights of Nanosatellite Propulsion Development Program at NASA-Goddard Space Flight Center

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Abstract. Currently the GN&C's Propulsion Branch of the NASA's Goddard Space Flight Center (GSFC) is conducting a broad technology development program for propulsion devices that are ideally suited for nanosatellite missions. The goal of our program is to develop nanosatellite propulsion systems that can be flight qualified in a few years and flown in support of nanosatellite missions. The miniature cold gas thruster technology, the first product from the GSFC's propulsion component technology development program, will be flown on the upcoming ST-5 mission in 2003. The ST-5 mission is designed to validate various nanosatellite technologies in all major subsystem areas. It is a precursor mission to more ambitious nanosatellite missions such as the Magnetospheric Constellation mission.

By teaming with the industry and government partners, the GSFC propulsion component technology development program is aimed at pursuing a multitude of nanosatellite propulsion options simultaneously, ranging from miniaturized thrusters based on traditional chemical engines to MEMS based thruster systems. After a conceptual study phase to determine the feasibility and the applicability to nanosatellite missions, flight like prototypes of selected technology are fabricated for testing. The development program will further narrow down the effort to those technologies that are considered "mission-enabling" for future nanosatellite missions. These technologies will be flight qualified to be flown on upcoming nanosatellite missions.

This paper will report on the status of our development program and provide details on the following technologies:

- Low power miniature cold gas thruster
- Nanosatellite solid rocket motor
- Solid propellant gas generator system for cold gas thruster
- Low temperature hydrazine blends for miniature hydrazine thruster
- MEMS mono propellant thruster using hydrogen peroxide

Introduction

Recently, there has been an increased interest in using nanosatellites in space science missions due to many unique science mission architectures that are possible with a nanosatellite constellation. Hundreds of small and light weight nanosatellites can form an intelligent constellation of a distributed network of instruments to obtain measurements that are not possible with traditional single spacecraft architectures. Simultaneous, in-situ measurement of dynamic phenomena in the Earth's magnetosphere is one area in which a distributed instrument network concept of nanosatellite constellation makes the mission feasible. Such scientific data are considered to be critical elements in the NASA Sun-Earth Connection (SEC) roadmap. Currently, the SEC roadmap features several nanosatellites constellation missions under consideration as potential future missions. One such mission currently in a study phase at NASA's Goddard

Space Flight Center (NASA-GSFC) is the Magnetospheric Constellation (MagCon) mission.

The Magnetospheric Constellation Mission Concept

The MagCon mission architecture calls for a constellation of about 100 nanosatellites launched from a single deployer ship. The nanosatellites are placed into elliptical orbits with apogees from 3 to 50 R_e to provide simultaneous, multi-point observations of the Earth's magnetospheric environment. Figure 1 shows the current configuration of the nanosatellite as an octagonal disk of 30 cm in diameter and 10 cm in height. Each nanosatellite is spin-stabilized about its major axis with a primary attitude control mode of spin axis precession. Figure 2 illustrates the current deployer ship concept. While each nanosatellites has mass of no more than 10 kg, it is designed to carry a multi-instrument suite of particle and field detectors to perform "research quality science". In addition, nanosatellites are designed to form an intelligent constellation of a distributed instrument network, enabling nanosatellites to autonomously reconfigure the network to quickly respond to dynamic magnetospheric events. The constellation autonomy requires that each nanosatellite be capable of space to space communication. In addition, each nanosatellite requires a propulsion system providing both the attitude control and orbit changing capabilities to be able to reconfigure the constellation.

NASA-GSFC Nanosatellite Propulsion Development Program

To build such small, lightweight and intelligent spacecraft poses tremendous challenges. Existing spacecraft components designed for larger size spacecraft would not work with nanosatellites due to severely limited power and volume constraints imposed by nanosatellites. The MagCon study results show that virtually every spacecraft subsystem needs breakthroughs in fully functional miniaturized components in order to make the intelligent nanosatellite constellation feasible. As a result, the MagCon mission study is also focusing on the identification and development of spacecraft component technologies that are suitable for nanosatellite missions. A significant part of the nanosatellites component development effort deals with developing suitable propulsion components. Currently, there are very few propulsion components that are designed for nanosatellites. As nanosatellites evolve, greater demand will be placed on the propulsion subsystem to provide complex maneuvers required to maintain an autonomous, intelligent constellation. Recently the GN&C's Propulsion Branch of NASA-GSFC embarked on a program to develop nanosatellite propulsion components that are designed to meet the requirements of both current and future nanosatellite constellation missions. Our approach is to pursue a multitude of nanosatellite propulsion options simultaneously by teaming with the industry and government partners. After a conceptual study phase to determine the feasibility and the applicability to nanosatellite missions, flight like prototypes of selected technologies will be fabricated for testing. The development program will further narrow down the effort to those technologies that are considered "mission-enabling" for future nanosatellite missions. These technologies will be flight qualified to be flown on upcoming nanosatellite missions. This paper describes the status of our program and provides details on the following propulsion technologies that are currently being studied by NASA-GSFC.

- Low power miniature cold gas thruster
- Nanosatellite solid rocket motor
- Solid propellant gas generator system for cold gas thruster
- Low temperature hydrazine blends for miniature hydrazine thruster
- MEMS mono-propellant thruster using hydrogen peroxide

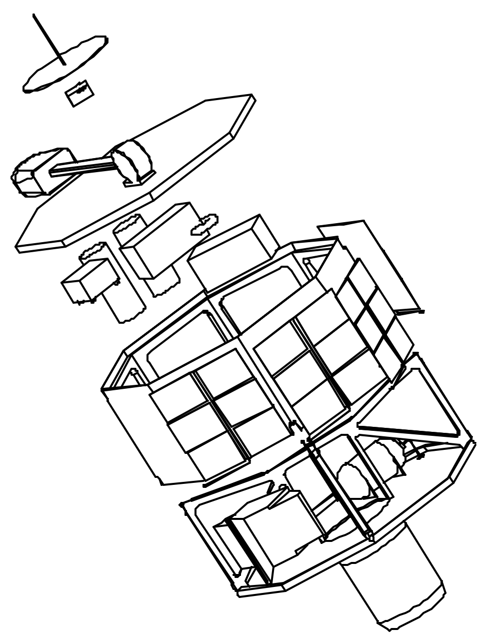
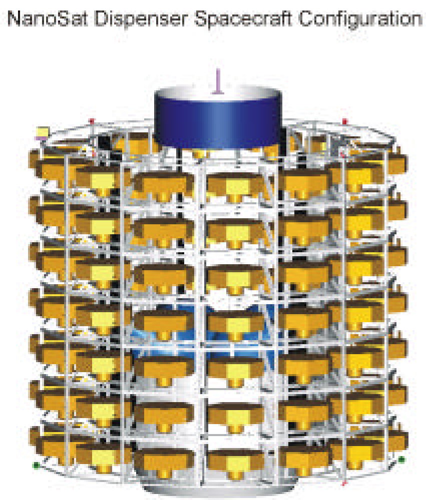


Figure 1. Exploded view of the nanosatellite for the MagCon mission.

Figure 2. Nanosatellite deployer ship concept.



Propulsion Requirements for MagCon Mission

Table 1 shows the propulsion requirements for the MagCon mission. Since each nanosatellite is spin-stabilized, the requirement calls for relatively high thrust devices as compared to three-axis stabilized nanosatellites. The likely candidates are miniature chemical thrusters rather than miniature electric thrusters. The current focus of our development program are miniaturized chemical and cold gas thrusters. In addition, several MEMS propulsion technologies are also of interest to NASA-GSFC as candidates for formation flying missions where very low thrust and precision impulse are required. Potential MEMS propulsion devices for formation flying missions are also currently being developed by NASA-GSFC as discussed later in the paper.

Table 1. MagCon propulsion requirements.

	Orbit Raising	Attitude Control
Total Impulse	3000 N-s max.	2.4 N-s
Thrust	445 N max.	1.0 N
Input Power	< 1 Watt peak	< 1 Watt peak
Isp	280 sec.	60 sec.
Minimum I_{bit}	-	0.044 N-s
Pulse rate	-	1 Hz.
Cycle life	-	>1000 cycles

Thruster technologies for orbit raising

Two candidate technologies suitable for the orbit raising requirements are the miniature solid rocket motor and the miniature mono-propellant hydrazine thruster. Miniaturized bi-propellant thrusters, while offering better performance that either the solid or the liquid mono-propellant thrusters, require a complex feed system and occupy more volume than the other options. This makes them less attractive for nanosatellites uses. Typically, solid rocket motors can provide an efficient way to perform orbit raising operations for a spin-stabilized nanosatellite However, even the smallest currently available solid motor would greatly exceed the mass, volume, and power requirements of a 10 kg class nanosatellite. Monopropellant hydrazine thrusters can also meet the total specific impulse requirements within the limited volume allowed by a 10 kg class nanosatellite. However, the power required to actuate the thruster valve is far greater than 1 w allocated by nanosatellites. In addition, hydrazine systems require heaters to prevent the propellant from freezing. Nanosatellites, with their limited power output, cannot support such active thermal regulation. An ideal miniature hydrazine thruster would use a hydrazine propellant blend with very low freezing point that can still deliver the I_{sp} of 220 seconds. Based on this reasoning, the following technologies have been selected for development: the miniature nanosatellite solid rocket motor and the low temperature hydrazine blends for miniature hydrazine thruster.

Thruster technologies for attitude control

As mentioned earlier, the primary attitude control mode of the MagCon nanosatellite is the precession of its spin axis. This requires that a thruster operate in a pulse mode. The pulse rate specification in Table 1 accommodates spin rates up to 60 rpm, which may occur during solid rocket motor firing. During science operations, the MagCon nanosatellite spin rate is 20 rpm. Thus, an additional attitude control mode of spin rate control is also desirable. These requirements dictate thrusters that can deliver short, accurate pulses. Within the volume limitation found in a 10 kg class nanosatellite, both the miniature nitrogen cold gas thruster and the miniature mono-propellant hydrazine thruster can satisfy the requirements. Existing cold gas thrusters can meet the thrust and minimum I_{bit} requirements. However, the power needed to actuate the valve far exceeds the less than 1 w peak power allocation of the MagCon nanosatellites. Another limitation of the cold gas thruster system is its low propellant density. Only a small amount of gaseous propellant can be stored in a given volume as compared to liquid or solid propellants. Thus, cold gas thrusters are less attractive options when the attitude control total impulse requirement is large. In the case of MagCon nanosatellites, the attitude control total impulse requirement is small enough for a cold gas thruster to be competitive. However, future nanosatellite constellation missions may require a significantly higher attitude control budget. In this case, other options need to be explored.

One way to overcome the low propellant density problem of a cold gas thruster system is to store the propellant in solid propellant pallets and generate gaseous propellant by igniting solid propellant pallets inside a plenum. Multiple actuation cycles allows the plenum to go through several charging-discharging cycles, enabling the system to generate a significantly higher total propellant load as compared to the single pressurization cycle common with existing cold gas systems.

Another option is to use a liquid mono-propellant system. A combination of higher Isp and propellant density means significantly greater total impulse capability for a liquid mono-propellant system. In addition, a single mono-propellant system to handle both the orbit raising and attitude control requirements could simplify the propulsion system and reduce the dry mass. However, as in the case of existing cold gas thrusters, current mono-propellant hydrazine thrusters are not suitable for nanosatellite application due to limited power and volume constraints of nanosatellites. In addition, the freezing point of currently available hydrazine propellants remains a problem. These issues need to resolved before a mono-propellant liquid system can be used on nanosatellites. The most promising option is found to be miniaturizing a cold gas thruster. In addition, a solid propellant storage system as a replacement for a cold gas propellant is found to be promising enough to warrant a feasibility study.

MEMS propulsion options

While the MagCon propulsion requirements lead to technologies that are more suited to spin stabilized nanosatellites, three-axes stabilized nanosatellites require propulsion technologies that can generate very low, accurate and repeatable impulse bits. In this field, the MEMS technology holds much promise for miniaturization. Currently in development at NASA-GSFC is the MEMS mono-propellant thruster using hydrogen peroxide. Current status of this program is described later in this paper.

Technology Status and Description

Low Power Miniature Cold Gas Thruster

In August '99, the Propulsion Branch took delivery of the prototype Miniature Cold Gas Thruster (MCGT) developed by Marotta Scientific Controls, Inc. Figure 3 shows the prototype MCGT. The MCGT is a breakthrough design that can provide full attitude control capabilities in a package weighing only 50 grams. Furthermore, the innovative solenoid valve design requires less than 0.4 w continous operation power through out its operational pressure and temperature ranges. Table 2 lists the full performance data for the MCGT.

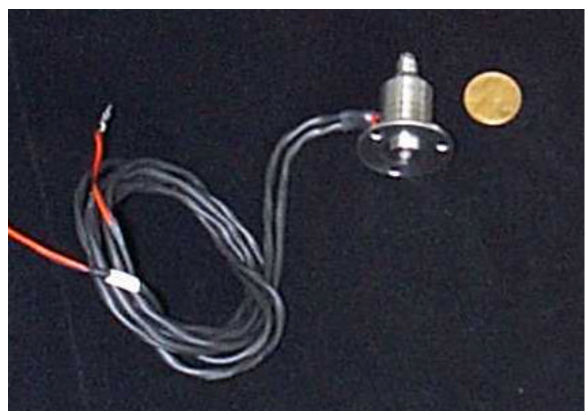


Figure 3. The prototype Miniature Cold Gas Thruster (MCGT) shown next to a penny.

Due to its advanced features, the MCGT has been selected to fly on the NASA-GSFC's ST-5 mission, which was recently awarded funding as a full spaceflight project. The ST-5 mission features three nanosatellites in a constellation to map the Earth's plasma environment and it is essentially a precursor mission to the MagCon mission. The goal of the ST-5 mission is to validate various nanosatellite technologies that will enable more ambitious future nanosatellite missions. Each 20 kg class nanosatellite is spin-stabilized and features one MCGT that provides full attitude control and stationkeeping capabilities. Figure 4 shows the current configuration of the ST-5 nanosatellite as an octagonal disk of 40 cm in diameter and 20 cm in height. More details on the MCGT can be found in Ref. 1.

Table 2. Performance data of the MCGT.

Power	1 W Peak, < 0.4 W continuous
Thruster Mass	50 g, w/o lead wires
Response Time	5 msec.
Leakage	< 1 x 10^4 sccs GHe
Vaccum Thrust	0.445 N at 500 psi inlet
Minimum Impulse Bit	44 mN-sec
Pressure-MEOP	1000 psia
Pressure-Proof	3000 psia
Pressure-Burst	4000 psia
Minimum Pulse Rate	< 1 Hz
Operational Modes	Pulsed & continuous
Cycle Life	> 1000 cycles
Duty cycle	10%-90% at 1 Hz

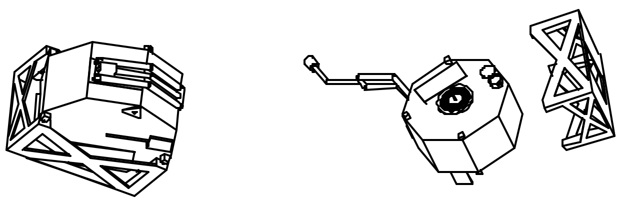


Figure 4. Stowed and deployed configurations of the ST-5 nanosatellite and its deployment mechanism.

Nanosatellite Solid Rocket Motor

For the past year, NASA-GSFC has been working with Thiokol corporation to develop a light weight, low cost, miniature solid rocket motor suitable for nanosatellites. So far, two prototype motors have been built and tested successfully, validating the concept. Figure 5 shows a dramatic illustration of the motor size. To minimize the inert mass, the motor uses a filament wound composite casing. Careful attention to material selection resulted in the prototype inert mass of 0.4 kg, which is 29% of the total mass of 1.38 kg. Figure 6 shows the sea-level live-fire test of the prototype motor No. 1. Table 3 summarizes the performance obtained during the sea-level test. The second prototype, shown in Figure 7, was tested in a vacuum chamber to simulate the actual operating condition. In addition, the second prototype was tested with a prototype low power electronic safe-and-arm (ESA) device to meet the limited power requirements of nanosatellites. During the arming phase, the low power ESA draws 100 mA at 3.3 Vdc, during its 5 sec charging duration. The fire command is sent via 20ms pulse at 3.3 Vdc. Figure 7 shows the low power ESA prototype circuit board mounted next to the motor. Complete details on the Nanosatellite solid rocket motor development program can be found in Ref. 2.

Table 3. Measured performance data for the nanosatellite solid rocket motor prototype No. 1.

Parameter	Value
Propellant Mass	0.98 kg
Maximum Thrust	290 N
Average Thrust	223 N
Total Impulse	2554 N-s
Burn Time	11.5 seconds
Isp	266 seconds
Max. Combustion Pressure	2600 psia

Figure 5. Nanosatellite Solid Rocket Motor Inert Model.

Figure 6. Sea-level live-fire testing of the prototype motor No. 1.

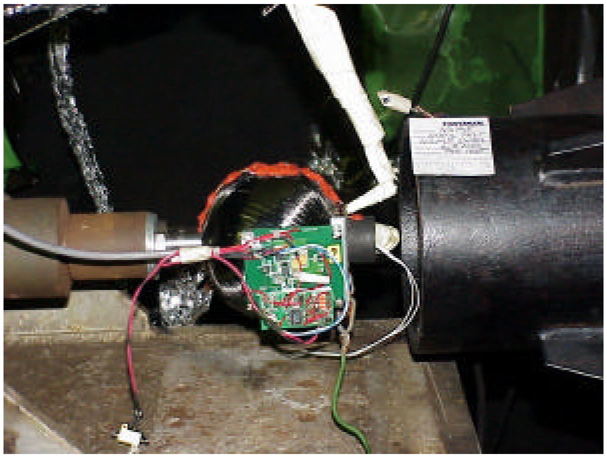


Figure 7. The second nanosatellite solid rocket motor prototype shown just prior to vaccum live-fire test.

Solid Propellant Gas Generator System

While the MCGT system is optimized for nanosatellite operations, the inert gas supply has not been optimized. By carrying the propellant in gas form, the amount that can be stored aboard a nanosatellite is severly limited. For example, a 150 mm diameter tank is the largest that can be installed inside a 20 kg mass nanosatellites of the ST-5 mission. The system provides about 130 N-s of total impulse when pressurized to 10.3 MPa (1500 psia). Although this is enough to meet the objectives of the ST-5 mission, it may not be enough for future nanosatellite missions where hundreds or more nanosatellites are actively maintaining a constellation. In addition, as nanosatellites get smaller, so does the volume available for propellant storage.

The idea behind the solid propellant storage system is to extend the total impulse capability of a cold gas system by storing the propellant as solids. In space, the solid propellant can be converted to inert gas inside a plenum and fed into a miniature cold gas thruster. Multiple charging cycles can provide far greater total impulse capability than the cold gas tank within the same volume constraints.

As a first step, NASA/GSFC, in partnership with Thiokol Corporation, conducted a preliminary design study for a solid propellant gas generator system. The goal was to design a system that fits within the same envelope as the ST-5 nitrogen cold gas tank while maximizing the total impulse and minimizing the inert mass. The results show that a system based on the embedded charge concept can deliver up to 500 N-s of total impulse while maintaining a wet mass of 3.2 kg. As a comparison, the MCGT system with a 150mm diameter tank has a wet mass of 1.25 kg and delivers 130 N-s of total impulse. This result indicates that the solid propellant gas generator system is a promising alternative to a single cycle cold gas thruster system. In the future, this concept will see further development from additional trades and optimizations to select a baseline design. Several prototypes will then be produced and tested to aid in further optimizations, culminating in a vacuum live fire testing of an optimized prototype.

MEMS Mono-Propellant Thruster

In addition to spin stabilized nanosatellite concepts, such as ST-5 and MagCon, NASA-GSFC is also interested in three axis stabilized nanosatellites for formation flying. Many formation flying concepts such as interferometry missions have unique propulsions needs with extremely low thrust levels (1 to 1000 uN) or extremely low minimum impulse requirements (1-100 uNsec) for both delta-V and attitude control [3,4]. Micro-electromechanical (MEMS) chemical and electro-thermal thruster concepts are particularly well suited for spacecraft that are very power limited, have stringent mass and volume constraints, demand moderate total delta-V requirements, and/or a relatively wide range of thrust levels. Currently both chemical and electric MEMS propulsion concepts are being pursued by a number of organizations[5,6,7,8,9], The possibility of MEMS monopropellant thrusters has been proposed; however no effort to date has been made to develop this technology [10,11].

MEMS monopropellant thrusters promise characteristics that have advantages over other chemical and electro-thermal concepts. These include:

- Greater propellant densities and higher specific impulses than cold gas

- Greater range of total impulse and thrust level/I-bit than discrete solid or bipropellant concepts with the benefit of fixed thrust vector

- Simpler than bi-propellant in both fabrication and propellant handling

- Lower power requirements than electrothermal devices

The GSFC Propulsion Branch in cooperation with the GSFC Detector Development Branch has undertaken an internal effort to determine the feasibility of developing

MEMS based monopropellant thrusters. Specifically, work is being done to design, fabricate and test MEMS catalyst beds and nozzles for use with High Test Hydrogen Peroxide (HTP) propellant. The performance goals of these reactors are:

Table 4. MEMS Performance Goals

Thrust	10-500 micro-N
Specific Impulse	145 sec
Impulse Bit	1-1000 micro-N-s
Adiabatic Flame Temperature	<1700K
Throughput	0.2 kg

This study will utilize 85-90% concentration HTP that is easy to work with, non-cryogenic, and non-toxic. However, HTP has is not commonly used as a space flight propellant because of several systems issues. The most problematic issue is that of auto-decomposition. Even without the presence of a catalyst, HTP naturally decomposes at around 1% per year at room temperature [12]. This rate nearly doubles for every 8.3 °C rise in temperature [13]. While this is not a concern from the standpoint of usable propellant, the auto-decomposition causes a build-up in pressure any place where HTP is not vented, such as in storage tanks or feed lines. For nanosatellites with relatively short total mission durations (1-3 yrs), the pressure increase due to auto decomposition may be acceptable. In-house material compatibility tests at GSFC are planned for in the near future to comprehensively address this issue.

Even if HTP is not the optimal propellant for specific satellite applications, the HTP MEMS reactor work will serve as a basis for understanding monopropellant catalytic reactions on the MEMS scale for other propellants such as hydrazine. This work is intended to advance the understanding of scaling principles of catalytic thrusters from current devices to micro-sized MEMS levels. By testing various catalyst bed designs, information on required residence times, decomposition instabilities, viscous flow effects, thermal response, and pressure dependences will be obtained.

Reactor Design

The major goal of designing the reactors has been to identify an approach that will allow for the efficient and full decomposition of HTP. For the first generation of reactors, propellant residence times were estimated by scaling macroscopic HTP reactors to the MEMS level using both catalyst contact time and pressure dependency. Silver, which is a highly reactive catalyst of HTP used in numerous macroscopic HTP reactors, was chosen as the catalyst. The range of thruster design variations is shown in Figure 8.

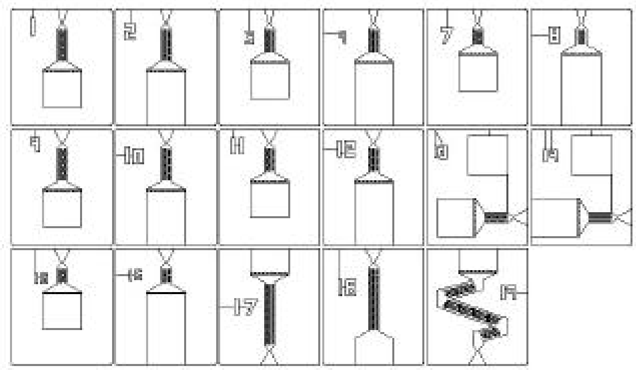


Figure 8. MEMS Design Variations

The design for these reactors incorporates a plenum, a filtering grid, four catalyst channels, and a nozzle in a silicon substrate about 2.5 mm x 3 mm and 500 microns thick. Reactors have been designed to include a combination of five different catalyst lengths (varying from 500-3000um), two different nozzle exit diameters (300-500 um), two different propellant inlet port configurations (axial and lateral), and an optional reaction chamber pressure tap. A total of 328 individual thrusters will be etched on a single wafer. Flow rates are expected to range from 10-500 μg/s at a nominal chamber pressure near 34 kPa. This matrix of reactor designs will enable residence time and decomposition effects at the MEMS level to be categorized.

Reactor Fabrication

The fabrication sequence of the reactors is as follows:

Table 5. MEMS Fabrication Sequence

1	Photoresist mask for silicon created from design drawings
2	Features etched on silicon wafer using Deep Reaction Ion Etching (DRIE)
3	Silver vapor deposited on wafer using catalyst mask
4	Counter sunk inlet holes and catalyst relief feature ultrasonically etched on cover glass
5	Glass bonded to silicon wafer using applied voltage (anodic bonding)
6	Wafer cut to form individual thrusters
7	Propellant feed tubes integrated to thrusters

The DRIE process will be used to create channels in the silicon approximately 300 microns in depth. The catalyst mask will allow only the catalyst area of the reactor to be exposed to the silver vapor deposition. Initial silver deposition will be 1-3 microns thick. The cover glass will include a relief feature to accommodate the build up of silver on top of the catalyst fins. To anodically bond the cover glass to the silicon substrate the silicon-Pyrex stack will be heated to between 300-400 °C while a negative voltage of 700-1200 V is applied to the Pyrex with the silicon at ground potential. After the bonding, the wafer assembly will be sectioned by diamond saw to form individual reactor chips. Finally, the propellant inlet tubes will be attached to the individual chips with an epoxy bond (see Figure 9).

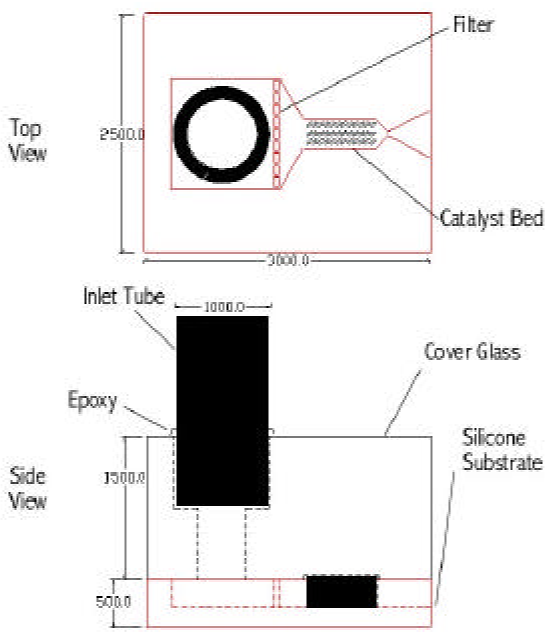


Figure 9. MEMS Thruster Integration

To date, the mask needed to etch the features into the silicon has been fabricated, and the etching of the silicon is in process.

Reactor Tests

Evaluation of the various reactor designs will be accomplished by a series of experiments. The reactors will be integrated with a macro sized propellant manifold and feed system consisting of a precision syringe pump. Once integrated with the propellant feed system, the reactors will undergo several flow tests. An inert reference fluid will be used to initially validate pressure drop and flow predictions. Upstream feed pressure will be varied and measured at discrete intervals while the mass flow through the reactor is measured. Mass flow measurements for these and subsequent tests will be made with a precision thermal mass liquid flow sensor with a range of 0 –0.5 g/hr.

The use of a Pyrex cover plate will allow for visual observation of the reaction process when HTP is introduced into the reactor. Observations will be made and recorded using an inspection microscope equipped with a color video camera and monitor.

To determine the stability of the reaction, reactors with pressure taps as an integrated part of the chip will be utilized. The simultaneous measurement of chamber pressure, mass flow rate and inlet pressure will allow thrust and specific impulse to be estimated. These measurements will be made with the thruster at atmospheric conditions and at vacuum in a bell jar. Flow characteristics of reactors with and without pressure taps will be compared to evaluate possible degradation of performance caused by the taps.

Insight into the thermal response and characteristics of the reactors will be obtained by observing the operation of the reactors with an infrared thermal imaging camera. Existing equipment at GSFC can determine temperatures up to 1500 °C with spatial resolutions as small as several microns. The use of optical coating on the exterior of the reactor chips may be employed to improve the accuracy of the measured temperature profile.

The composition of reactor exhaust gases will be analyzed to determine the effectiveness of the catalyst design. The relative amounts of hydrogen peroxide, water, and gaseous oxygen will determine the percent decomposition of the propellant. Exhaust gases will be collected in sample bottles that are initially evacuated or contain inert gas.

Low Temperature Hydrazine Blends for Miniature Thrusters

The application of low temperature hydrazine propellant blends (HPBs) in miniature nanosatillite attitude control thrusters was investigated [14]. This work was initiated by GSFC in partnership with NASA/Glenn Research Center. and was carried out by Primex Aerospace Company (PAC). The purpose of this effort was to determine the feasibility of using a blended hydrazine propellant which would have a sufficiently low freezing point to enable the elimination of propulsion survival heaters and their associated power while at the same time achieving improvements in specific impulse and mass density over straight hydrazine. At the time this effort was initiated, the MagCon design dictated that the propellant temperature be less than or equal to minus 10 °C to preclude the requirement of survival heaters. The objective of this work was to demonstrate that a HPB could be used in a low thrust engine firing at duty cycles that would be representative of attitude control requirements for a nanosatellite.

The demonstration objectives adapted from GSFC requirements are listed below.

Table 6. Performance Goals

Parameter	Demonstration Objectives
Inlet Pressure	0.896 – 2.62 MPa
Operating Temperature	- 20°C (10 °C margin added)
Minimum cycle life	1000 cycles
Minimum cold starts	10
Minimum Isp	240 sec (to be calculated)
Avg. vacuum thrust	4.445 N @ 2.62 MPa
Minimum mass throughput	11 kg.
Minimum duty cycle	Pulse Mode Operation
	0.020 sec On
	0.4% D.C.
	Steady State Operation
Total Impulse	23470 N sec

Propellant Selection

In the 60's, 70's, and 80's Primex Aerospace Corporation developed a variety of HPB's for defense applications. However, this work with HPB propellants did not focus on small satellite applications where low thrust and low duty cycles are of primary concern. The demonstration of a HPB in a low thrust thruster in this current effort would address such questions as the ignitibility of the propellant, stability of the propellant in the thruster during low duty cycles, and the effect of the propellant on catalyst life. The HPBs under consideration for this effort consisted of a mixture of N2H4, H20, and HN (N2H4N03).

Listed above are the HPB candidates which where previously developed by PAC and considered for this effort. HP-1808 was the propellant chosen for this effort because it has the best combination of Isp and combustion temperatures that would allow existing rocket engine materials to be used. Additionally, it is the HPB farthest outside the shock sensitivity range.

Table 7. HPB 's Considered for Demostation

HPB	Freezing Point (°C)	Composition by Weight %			Density (g/cc)	Est. Isp (sec)
		HN	H2O	N2H4		
N2H 4	1.7	0	0	100	1.004	245.0
1808	-20.0	18	8	74	1.08	243
2012	-34	20	12	68	1.093	236.2
2400	-18	24	0	76	1.11	263.8
2409	-34	24	9	67	1.109	245.8
2517	-54	25	17	58	1.12	229.9

Other HP-1808 physical and safety properties are given below.

Table 8. HPB-1808 Properties

Density	@25C, 1.082 g/cm3
	@71C, 1.046 g/cm3
Kinematic Viscosity	@-21C, 4.9 centi-Stokes
	@ 25C, 1.55 centi-Stokes
	@71C, 0.70 centi-Stokes
Vapor Pressure	@25C, 2.1 kPa
	@71C, 21.0 kPa
Surface Tension	7.3x10-2 N/m @ 25 °C
Flammability, 66.7C	-Flash point, Cleveland open, cup -ASTM-D92-78
Explosion Sensitivity	-Cap sensitivity • No.8 cap • Negative 3/3
	-Card gap sensitivity • TB700-2, Neg@ 0 cards, 3 of 3 • NAVORD 2563 (Neg 1 of 1) - 2.54 ID, 7.62cm long tube - 50g booster (pentolite)
	- Drop weight sensitivity, negative • 10 out of 10 tests • 140 kg cm - negative
	Thermal stability • No ignition after 48hrs @75 °C • DSC - Endotherms @ 54 & 127 °C - Exotherms @227 & 286 °C

Hot Fire Demonstration

PAC performed an ambient pressure hot fire test of HP-1808 in an existing PAC 4.445 N hydrazine thruster (MR-111G). The test plan for the hot firings is given below. Figure 10 shows the PAC MR-111G engine firing with HP-1808.

Table 9. HP-1808 4.55N Test fire Sequence

Sequence	Feed Pressure (Mpa)	Fuel Temperature, deg C	ON Time, sec	OFF Time, sec.	No. of Pulses	% Duty Cycle	Start Temperature, deg. F	Comments
0	2.62	21.1	0.02	0.98	30	2	Ambient	Warming Pulses
1	2.62	21.1	100	----	1	10	Existing	ATP Start
2	2.62	21.1	0.3	0.9	100	25	Existing	
3	2.62	21.1	1.5	4.5	25	25	Existing	
4	2.62	21.1	0.75	165	10	0.45	Existing	
5	2.28	21.1	100	----	1	10	Existing	Blowdown 1
6	2.28	21.1	0.3	0.9	100	25	Existing	
7	2.28	21.1	1.5	4.5	25	25	Existing	
8	2.28	21.1	0.75	165	10	0.45	Existing	
9	1.86	21.1	100	----	1	10	Existing	Blowdown 2
10	1.86	21.1	0.3	0.9	100	25	Existing	
11	1.86	21.1	1.5	4.5	25	25	Existing	
12	1.86	21.1	0.75	165	10	0.45	Existing	
13	0.896	21.1	100	----	1	10	Existing	Blowdown 3
14	0.896	21.1	0.3	0.9	100	25	Existing	
15	0.896	21.1	1.5	4.5	25	25	Existing	
16	0.896	21.1	0.75	165	10	0.45	Existing	
17	2.62	21.1	100	----	1	100	Existing	ATP Finish
Sequence	**Feed Pressure (Mpa)**	**Fuel Temperature, deg C**	**ON Time, sec**	**OFF Time, sec.**	**No. of Pulses**	**% Duty Cycle**	**Start Temperature, deg. F**	**Comments**
1	2.62	21.1	0.02	0.98	30	2	Ambient	Warming Pulses
1A	2.62	21.1	650	----	1	100	Existing	Start No. 1
2	2.62	21.1	0.02	0.98	30	2	< 37.7	Warming Pulses
2A	2.62	21.1	650	4.5	1	100	Existing	Start No. 2
3	2.28	21.1	0.02	0.98	30	2	< 37.7	Warming Pulses
3A	2.28	21.1	650	----	1	100	Existing	Start No. 3
4	2.28	21.1	0.02	0.98	30	2	< 37.7	Warming Pulses
4A	2.28	21.1	650	4.5	1	100	Existing	Start No. 4
5	2.28	21.1	0.02	0.98	30	2	< 37.7	Warming Pulses
5A	2.28	21.1	650	----	1	100	Existing	Start No. 5
6	1.86	21.1	0.02	0.98	30	2	< 37.7	Warming Pulses
6A	1.86	21.1	650	4.5	1	100	Existing	Start No. 6
7	1.86	21.1	0.02	0.98	30	2	< 37.7	Warming Pulses
7A	1.86	21.1	650	----	1	100	Existing	Start No. 7

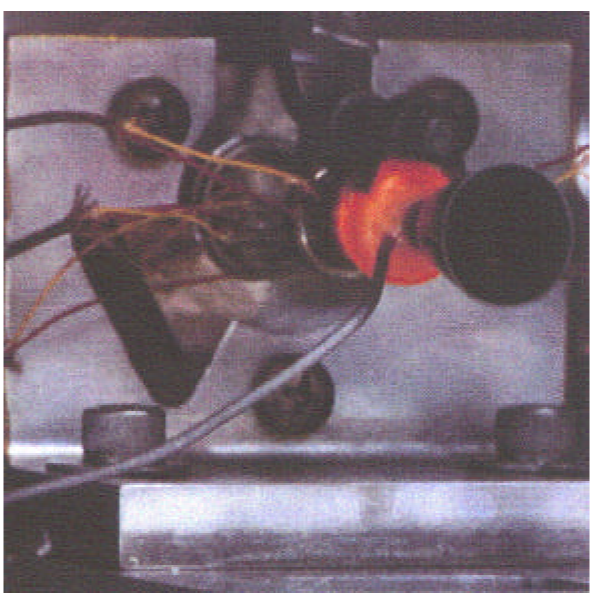


Figure 10. HP-1808 Test fire in MR-111G

The hot fire demonstration was successful; the performance objectives were demonstrated with the HPB firing in the MR-111G. Smooth combustion was observed for all runs. The catalyst life degradation was minimal (approx. 5%) and only twice that of the degradation rate for straight hydrazine. The test sequence demonstrated a HP-1808 throughput of 16.1 (40% greater that objectives). The averaged delivered Isp was calculated to be between 216-235 sec with an average C* of 1372 m/sec. The test demonstrated a range of duty cycles between 0.4% and 100 % and a feed system blow down ratio of 3:1.

Follow on development effort of HPB's for nanosatillite missions will include vacuum hot fire tests utilizing a milli-Newton thrust engines, characterization of engines performance at low temperatures, and additional materials compatibility tests for ground processing.

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