Low Cost Propulsion Development for Small Satellites at The Surrey Space Centre

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ABSTRACT. The Surrey Space Centre (SSC) has led the way in demonstrating the utility of microsatellite size spacecraft for research, humanitarian, commercial, and military applications. SSC recognises that cost effective propulsion technology for small spacecraft is an enabling technology for expanding the utility of these assets and has been actively researching this field since 1993. This paper provides an overview of propulsion research and development at the Surrey Space Centre.

The paper will summarise SSC goals for small spacecraft propulsion technology and link them to areas of propulsion research past, present and future. A review of Surrey's propulsion history to include hybrid, monopropellant, cold gas and resistojet technology is presented. Design and integration of SSC cold gas and resistojet technologies on flight spacecraft will also be covered with an emphasis on the SSC low cost approach to qualification, integration and operation of these systems. These topics will be followed by a discussion of areas that are currently being investigated for near term research, specifically, H2O2 long term storage, expulsion, catalysis, "Green" monopropellant and hybrid technology utilising both N2O and H2O2.

One topic covered in detail is a novel alternative geometry hybrid rocket motor. This motor is currently under development to provide a low-cost, intrinsically-safe and easy to integrate orbital upper-stage for small spacecraft. A prototype motor has been constructed and test results are presented.

SSC Propulsion Philosophy

Propulsion research at the Surrey Space Centre (SSC) has been ongoing since 1993. The result of these research efforts culminated with the flight of two new propulsion systems onboard the 350kg SSC Minisatellite UoSat-12 in April of this year.

SSC has enjoyed enormous success by taking a measured approach to every aspect of small spacecraft engineering, spanning the spectrum from basic research to flight operations. Propulsion research, like every other spacecraft related discipline at SSC, works towards achieving the common goal of "affordable access to space".

The SSC propulsion goals are to develop technologies that provide viable, cost effective propulsion solutions, directly applicable to real-world, small spacecraft applications. In addition, there is an emphasis placed on the low cost nature of SSC missions, life-cycle environmental and safety issues, and the optimisation of systems only when necessary. Of course optimisation has mass, volumetric and performance benefits, but in the secondary spacecraft arena, these aspirations can be at odds with the SSC goal of keeping space access affordable. Environmental and safety aspects of SSC propulsion receive priority because they are recognised as cost drivers and not just industry "buzz" phrases. Propulsion research at SSC is conducted in a university environment where the infrastructure costs for dangerous and environmentally damaging fuels and oxidisers is prohibitive. In addition, export, shipping and launch preparation costs are all reduced by using propellants that are of lower danger to the environment and personnel.

SSC Propulsion History

SSC began it's first research project devoted to propulsion in 1993.[1] This research focused on quantifying all elements of propulsion system cost. The research program provided an in-depth analysis of how to calculate true propulsion system cost spanning multiple disciplines from spacecraft engineering to program management and on-orbit operations; it developed a cost model utilising nine separate components - each of which must be considered to accurately determine propulsion system cost. The nine components are defined as:

Propellant/system mass
Propellant/system volume
Total elapsed thrust time
Power required (electric propulsion)
System price
Technical risk (to the program)
Safety (personnel)
Integration
Logistics

During the course of this research, a 400N (sea level) high test hydrogen peroxide (HTP) and polyethylene (PE) hybrid rocket motor was built, tested and characterised. The HTP/PE hybrid demonstrated attractive characteristics for small spacecraft applications, among the strongest were safety, environmental friendliness, reasonable performance and the potential for these characteristics to add up to large cost savings over current state-of-the-art propulsion systems.

An interesting by-product of this hybrid program was the research devoted to the catalytic decomposition of HTP. By employing catalytic decomposition of HTP, auto-ignition of the solid fuel in a hybrid rocket is possible. When HTP is properly decomposed, the decomposition products alone release enough heat (>600°C) and gas to provide efficient thrust. A HTP monopropellant capability would be useful as a cost effective stand-alone system or be easily added to the hybrid infrastructure providing a bimodal system at minimal additional cost. Thus the hybrid catalyst research effectively kicked-off SSC's first research with relevance to HTP monopropellant propulsion as well. HTP catalysis research focused on 5 separate catalysts:

1. Maximum surface area silver-plated nickel gauze Type-2
2. Type-2 catalyst, sintered for 30 min at 800°C Type-2A
3. LCH 212 Type-4
4. Very rough silver plating of nickel gauze S-2 Type 7
5. Pure silver gauze treated with samarium nitrate Type 8

Eventually settling on the pure silver gauze with samarium nitrate treatment due to its superior heat producing (~decomposition) performance and lifetime characteristics (fig.1).[1]

SSC's first propulsion research program provided a valuable tool for identifying all parameters that effected the price of propulsion for small spacecraft. In addition, the practical hybrid research convinced SSC that spacecraft propulsion doesn't necessarily have to be extraordinarily expensive.

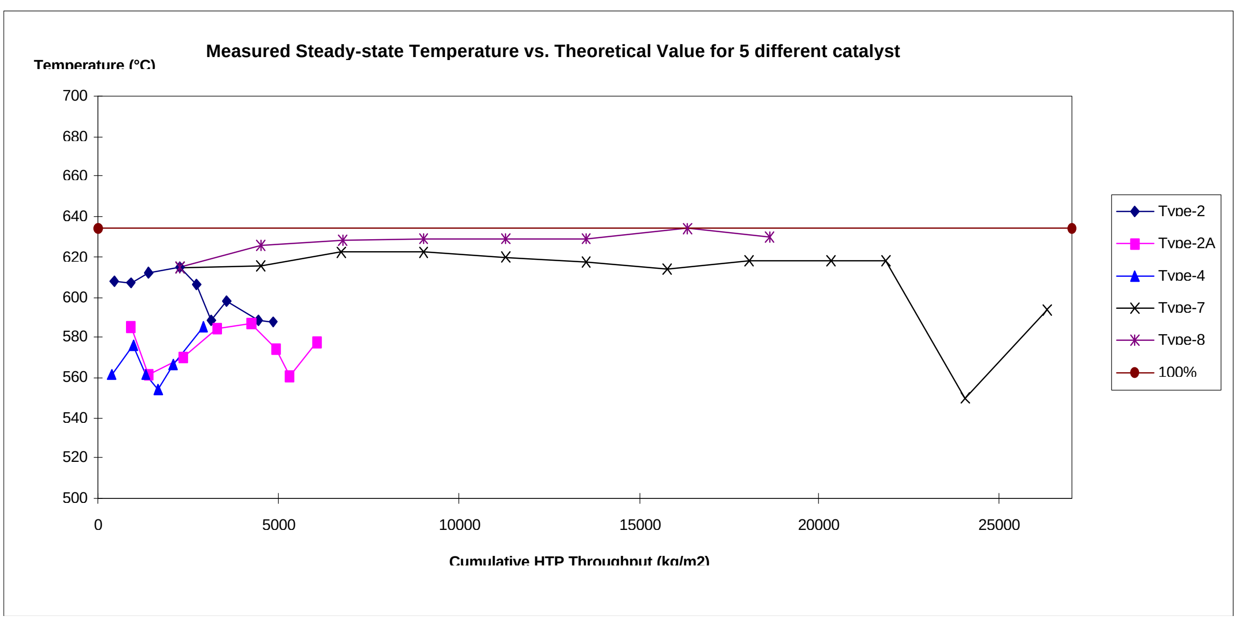

Fig. 1 Performance summary of 5 catalyst pack materials.

In 1995, another propulsion research program was initiated at SSC.[2] This program focused on resistojet technology. Initially, the resistojet research focused on finding the right combination of power, working fluid, and heat transfer medium to produce a cost effective thruster for small spacecraft applications. As the resistojet research progressed, the design converged on two systems, both utilising the same heating element and silicon carbide heat transfer bed but having different working fluids: water and nitrous oxide (N2O). By the time the resistojet research was completed, both thrusters (fig. 2) were manifested on two separate space missions. The water and N2O resistojet designs each consume 100 watts of electrical power providing an Isp of 127s (N2O) and 152s (H2O).

Fig. 2 SSC Flight Resistojets

One particularly significant accomplishment of this research program was SSC's first observation of a self-sustained decomposition of nitrous oxide. The self sustained N2O decomposition was significant because it indicated that N2O resistojet performance could be attained at much lower power levels. In addition, the research provided proof that there was significant savings to be had in the propulsion arena, demonstrated by actually lowering the threshold price of a space qualified resistojet thruster by a factor of 18[2].

UoSat-12

UoSat-12 incorporated two separate propulsion systems, a compressed nitrogen cold gas system and a nitrous oxide resistojet system (fig. 3). The combined delta V afforded to the UoSat-12 mission is 26.8m/s (16.4m/s from the N2 cold gas, 10.4m/s from the N2O resistojet)[2]. The N2 system is used for attitude control and velocity change whereas the resistojet is a technology demonstrator primarily used for velocity change. After initial checkout and experimentation, the N2 propulsion system is being configured to operate in an experimental autonomous mode via software developed by the Microcosm corporation.

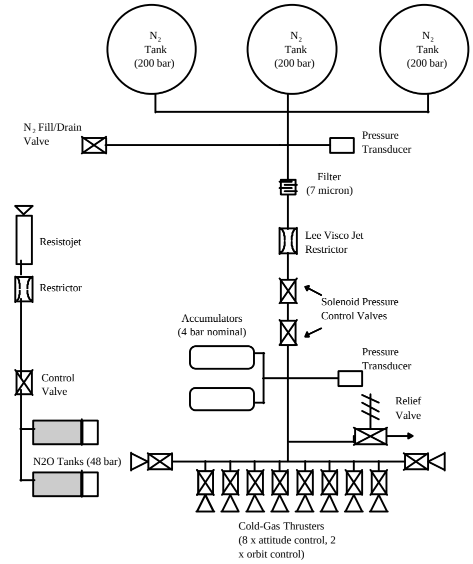

Fig. 3 UoSat-12 Cold Gas and Resistojet Diagram

The Uosat-12 cold gas design is composed largely of off-the-shelf components assembled into a seemingly ordinary cold gas propulsion system. One significant departure from an ordinary system design is the inclusion of a bang-bang (BB) pressure regulation system (fig. 4). Space qualified pressure regulators have historically been high cost items; The BB regulator replaces the conventional regulator with two valves, two pressure transducers and two small accumulators. The BB system effectively steps down the stored N2 pressure to the operating pressure of the cold gas thrusters. The BB system does require more volume and mass than it's costly regulator alternative, but for this mission (as well as others currently under consideration) it provides at viable trade to save a significant amount of mission funds.

The UoSat-12 N2O resistojet demonstrator is the first of it's kind ever to be flown in space. The resistojet produces 125mN of thrust at 100W. One particularly favourable characteristic of the N2O propellant is a storage vapour pressure of approximately 48 Bar. The high vapour pressure effectively negates the need for the mass, volume, and cost of a separate pressurant system. This propulsion system utilises a cost effective BB pressure regulation scheme as well.

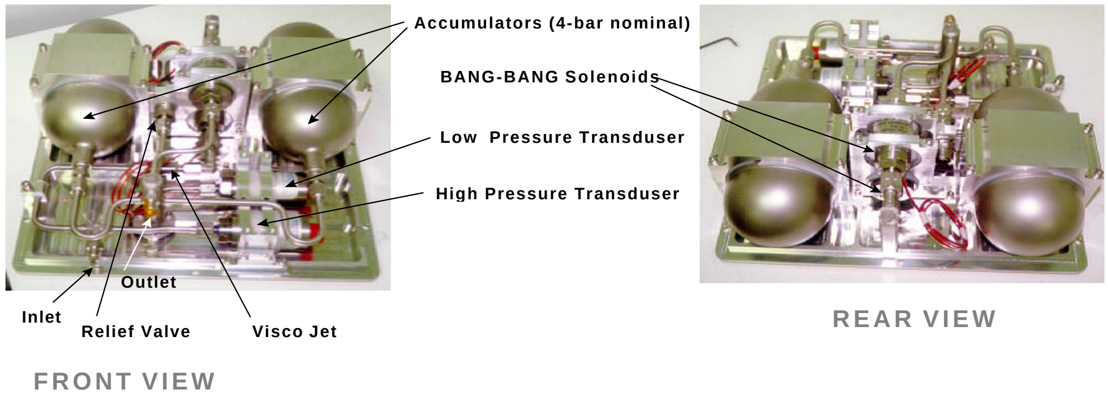

Fig.4 UoSat-12 "Bang-Bang" pressure regulator

Current research areas

At the time of this writing SSC has no less than four propulsion research areas under active investigation:

1. HTP as a propellant (storage, expulsion, handling)
2. HTP mono-propellant thruster research and design
3. N2O catalysis/mono-propellant research
4. Hybrid research covering both:
   - Conventional
   - Alternative geometry hybrid motors

All of these research areas focus on lowering the cost for propulsion systems on small spacecraft by finding the right mix of performance and cost reduction strategies. Each of these topics will be discussed in detail in the following sections.

HTP as a propellant

HTP (for the purposes of this paper ~ 85-100% H2O2 by weight) is an attractive propellant/oxidiser because it has favourable storage density and energy characteristics. HTP decomposes into super heated steam and oxygen according to:

H2O2 => H2O + ½ O2 + Heat

Finding a more environmentally benign oxidiser would be difficult indeed. Although HTP is considered a toxic substance, and under certain conditions presents an explosion hazard, HTP safety is easily managed in comparison with other high performance liquid rocket propellants. HTP safety is attributed to the fact that it can be rendered harmless with the addition of water and because it only presents an explosion danger in specific and avoidable regimes (fig. 5).[3] In addition to these favourable characteristics, HTP is extremely versatile, able to fulfil roles of monopropellant thruster propellant, warm gas generation, and oxidiser in hybrid and bi-propellant rocket systems.

As with any propellant, HTP has some unfavourable characteristics. Pure HTP continuously decomposes into oxygen and water and builds up pressure in a closed system. In order to minimise the decomposition of stored HTP, most manufacturers add small quantities of stabilisers (mostly Sn, and Phosphate but others as well) that desensitise the solution to contamination. The HTP concentration, level of stabilisation, temperature, and storage vessel characteristics all play a role in the observed decomposition rate. Although a military specification (milspec) for propellant grade H2O2 (and associated material compatibility data) does exist (MIL-P-16005D, 18 Mar 1965), it is difficult to find H2O2 stock that exactly matches the specification in concentration and/or particular levels of stabilisers and allowable contaminants; in the event one could find the exact (milspec) material it would still be difficult to convince any potential launch provider of the safety/risk issues involved without presenting some modern empirical data. Therefore, it becomes necessary to determine the decomposition characteristics of the particular HTP stock and storage/expulsion system combination. Previously, HTP decomposition has been controlled to a rate of less than 1% per year.[4] This decomposition rate was achieved through careful material selection, meticulous contamination avoidance procedures and storage vessel passivation. SSC's HTP research aspires to achieve similar performance (less than 1%/annum) for the specific HTP stock used, stored within a low cost tank/expulsion system.

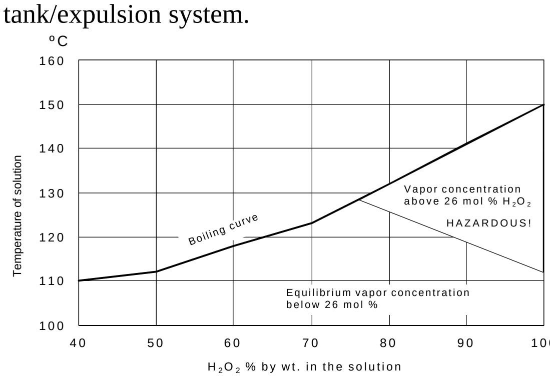

Fig. 5 Phase Diagram Showing the Explosive Range of Hydrogen Peroxide

HTP Monopropellant

SSC's HTP monopropellant research first started with the investigation of suitable catalyst materials for a 400N HTP/PE hybrid rocket. SSC interest in an HTP monopropellant thruster stems from the performance available in such a system (vs. cold gas) combined with the relative safety and environmental friendliness of HTP (vs. hydrazine monoprops). In addition, the flexibility and performance of a dual mode system comes at a low price when combined with a HTP hybrid propulsion system.

At the time of this writing, SSC research has built and is preparing to test a 10N HTP monopropellant thruster using pure silver gauze as the catalyst material. This particular thruster is expected to achieve an Isp of approximately 150s and a minimum impulse bit of 0.1Ns.[5]

N2O Monopropellant

During the course of the N2O resistojet research, long duration vacuum firings were conducted. On one particular firing, power was cut during the test and the resistojet continued to fire for 18+ hours. This was the first self-sustained thermal decomposition of N2O witnessed by SSC for spacecraft propulsion applications. The realisation that N2O could be thermally decomposed in a self sustained manner led SSC research to investigate a N2O monoprop thruster utilising catalytic decomposition.

Catalytic decomposition of N2O is a very promising technology for small spacecraft propulsion. Although N2O does not store as densely as HTP or Hydrazine, it has reasonable performance as a monopropellant and the property of self pressurisation to approximately 48 Bar. In addition, N2O is cheap, safe to work around and poses little danger to the environment (compared with other rocket fuels) when it escapes into the atmosphere. Under normal operating procedures, N2O is decomposed into environmentally friendly products according to:

$$N_2O \Rightarrow N_2 + \tfrac{1}{2} O_2 + \text{Heat}$$

SSC has determined that by using the right catalyst, N2O decomposition can be achieved with power levels much lower than the resistojet. Current analysis indicates a self sustaining N2O decomposition can produce temperatures in the 1200-1600°C range.[5]

Currently, the N2O monopropellant research is concerned with refining the catalyst material and support substrate combination, accurately determining the initial activation energy threshold, determining N2O flow rates conducive to self sustained decomposition, and performance measurement. Once the concept is fully proven, the research will focus on thermal efficiency, long duration/lifecycle testing and packaging the technology into a flight system.

In addition to having good monopropellant characteristics, N2O provides a hot, oxygen rich exhaust. Once catalytic decomposition of N2O is realised, the exhaust products can be fed into a solid fuel to achieve much higher performance. Fig. 6 illustrates potential performance for decomposed N2O with PE and Hydroxyl Terminated Polybutadiene [5,6].

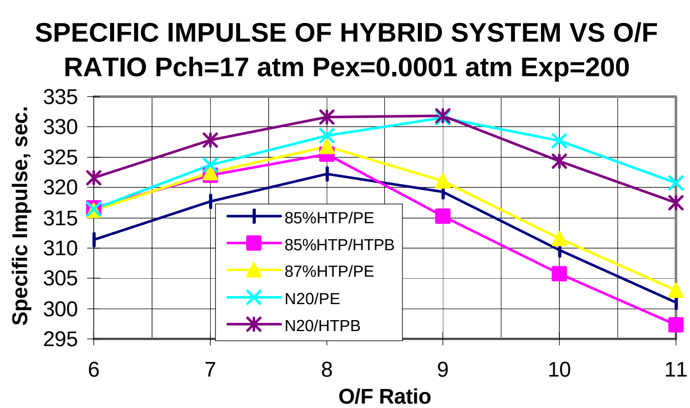

Fig. 6 N2O Theoretical Hybrid Performance

Hybrid Propulsion

High test peroxide (HTP) and polyethylene (PE) hybrid propulsion have been a subject of study by SSC since 1993. These systems have the following advantages:

1. The inherent safety afforded by the technology
   - --Fuel and oxidiser are never mixed to produce an explosive condition
   - --Fuel is not sensitive to cracks/voids
   - --Fuel is harmless (plastic)
   - --Oxidiser has a low vapour pressure (low fume dangers)
   - --Oxidiser is rendered harmless with the addition of water
2. The environmentally friendly products of combustion
   - --H2O2/PE combustion produces (primarily) O2, H2O and CO2
3. The potential for cost savings
   - --Favourable aspects of safety and environmental impact keep cost low
   - --Single liquid - fewer costly components (regulators, valves, tanks, etc.)
   - --Catalyst induced auto-ignition - complex ignition devices not necessary
4. The flexibility afforded by the technology over solid propulsion options
   - --Re-startability / multiple firings

For the purposes of this paper, conventional hybrid propulsion is defined as an axial flow hybrid motor with oxidiser being injected in one end and products of combustion exiting out the other. The conventional hybrid typically employs one or more combustion ports whose diameter continuously grows as the burn progresses. This changing geometry effects two critical performance parameters; the available burn surface and the combustion port cross-sectional area. As a result of these varying parameters, the oxygen to fuel ratio (O/F) defined by:

$$O/F = Gox/Gfuel$$

Where:
Gox - Oxygen mass flux (oxygen mass flow rate/port cross-sectional area)
Gfuel - Fuel mass flux (fuel mass flow rate/port cross-sectional area)

continues to increase during the duration of the burn. The regression rate or surface burn rate of solid fuel is affected by Gox according to:

$$Rdot = a\, Gox^n$$

Where:
Rdot - Solid fuel regression rate (m/s)
a - regression rate coefficient with grain length term
Gox - Oxygen mass flux (kg/m^2s)
n - regression rate exponent

A varying O/F has an operational impact in that the hybrid motor will only operate at maximum performance at a specific O/F ratio. Therefore, hybrid motors are typically designed to operate through their optimal O/F, accepting performance losses on either side.

Taking a closer look at this phenomena, one can see that the driving mechanism for an increasing O/F is a decreasing fuel mass flow rate. Since the denominator of both Gox and Gfuel increase at the same rate and the mass flow rate of the oxidiser is held constant, Gfuel must decrease in order to have an increasing O/F. In essence, both the oxygen mass flux and fuel mass flux are decreasing but the fuel mass flux decreases faster producing an increasing O/F shift. The increase of available fuel burn area while less fuel is being vaporised off the solid fuel wall can only be accounted for by a decrease in heat transfer to the solid fuel (as the port area increases) and thus less fuel being liberated from the solid fuel surface.

The initial port diameter and length of the hybrid motor are critical parameters because they determine the amount of fuel available at a given fuel regression rate. More basically, if the motor is too short and the port too small, O/F will start high and continue to climb higher. If the port area is made large to compensate for a short fuel grain, fuel liberation (due to poor heat transfer) and volumetric efficiency decrease. Hence, hybrid fuel grains tend to be designed long and slender. There are other alternatives to increasing fuel surface area (multiple ports, other geometry's) but these must be weighed with the subsequent effect on cross-sectional area (increasing complexity) and the risk of other adverse effects (i.e. fuel slivering, reduced volumetric efficiency, etc.). In addition, conventional hybrid motors can suffer inefficiencies due to a lack of complete mixing between fuel and oxidiser. These inefficiencies can result in a loss of 1-2% of impulse efficiency vs. liquids or solids systems.[7]

Alternative geometry hybrid propulsion

Alternative geometry hybrid propulsion is being pursued at SSC because it offers the benefits of hybrid rocket propulsion without having the hybrid rocket motor dominating the small spacecraft design. It can be argued that the conventional hybrid's most obvious negative characteristic for small spacecraft applications is the physical geometry required to obtain efficient performance (long and slender). A long and slender rocket motor design can present integration difficulties for small spacecraft. The conventional hybrid's long, slender geometry must be accommodated within a spacecraft and inline with the s/c centre of gravity (or employ multiple rocket schemes). Conventional hybrid geometry can drive the rest of the s/c bus design. Alternative geometry hybrid propulsion is aimed at reducing the impact of the conventional hybrid on small spacecraft design whilst simultaneously increasing performance over conventional hybrid designs.

The conventional hybrid can also present thermal difficulties for small spacecraft if catalytic decomposition of the oxidiser is employed (H2O2 or N2O for example). A H2O2 catalyst pack can generate greater than 600°C temperatures, while the N2O catalyst pack can theoretically double this value. Conventional hybrid geometry combined with limited integration options on small spacecraft would typically place the hot catalyst pack deep within the spacecraft generating unwelcome heat in close proximity to payloads or other spacecraft subsystems. One particular SSC alternative geometry hybrid configuration, the Vortex Flow Pancake (VFP), would place the motor external to the spacecraft with the catalyst pack exposed to space rather than confined within the spacecraft (fig 7).

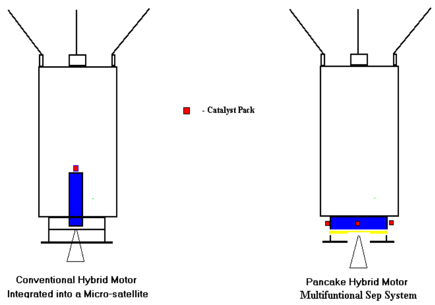

Fig. 7 Conventional vs. Alternative geometry hybrid motor placement

Currently, SSC has produced and fired an engineering model of a 15 cm diameter VFP Gox/Plexiglas hybrid. This particular design injects gaseous oxygen between two "pancake" shaped fuel disks with the products of combustion leaving through a nozzle situated in the centre of one fuel disk (see fig. 8).

In a conventional hybrid, the fuel burn area steadily increases as the burn progresses. In the pancake design, the fuel area remains relatively constant during the burn. For illustrative purposes, table 1 compares a conventional (single port) and pancake hybrid combustion chamber with the same initial fuel surface (burn) area and the same fuel mass.

Getting enough fuel surface in a short conventional hybrid (for a particular thrust level) to approach high performance mixing ratios can be difficult without resorting to the complexity, inefficiencies and risks (i.e. fuel slivering, lower mass fraction) of multiple port schemes. The pancake design provides sufficient fuel surface area for the duration of the burn.

In an earlier pancake design, the plan was to inject O2 from the outside of the motor toward the centre (along a radius), but it appeared that the overall motor diameter would have to be bigger in order to increase the mean path of the oxidiser over the fuel surface to accommodate sufficient fuel vaporisation, mixing and combustion. In addition, this design required a large number of oxidiser injectors in order to ensure all fuel surface areas were adequately "wetted" with O2 and that fuel regression would be even. After conferring with another researcher in the field, it was decided to employ a tangential or "vortex" injection scheme[8]. Tangential injection has many advantages. First, tangential injection would increase the mean path length of the oxidiser over the fuel surface whilst simultaneously increasing the fuel and oxidiser mixing within the combustion chamber, potentially gaining back some of the 1.5-2% impulse efficiency lost from poor mixing in conventional hybrid designs[7]. The tangential injection would theoretically increase performance by forcing the cooler, denser, un-combusted gases toward the chamber wall while forcing the lighter, hotter, exhaust products to the nozzle situated in the centre. This configuration would also reduce potential channelling effects anticipated with injecting straight across the fuel surface to the rocket nozzle.

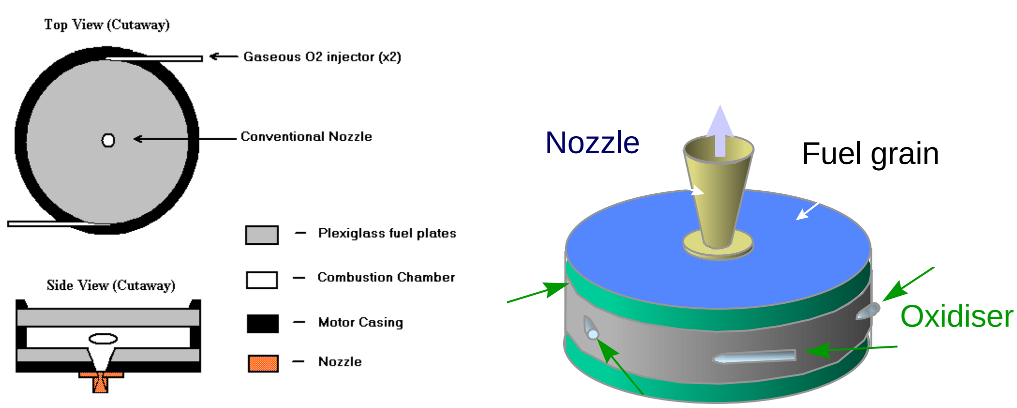

Fig. 8 Pancake hybrid engineering model diagram

Table 1. Conventional vs. Pancake combustion chamber efficiency

(assumes smooth/even regression for both configurations)					
	Fuel Mass Initial (kg)	**Initial Burn Area (cm^2)**	**Final Burn Area (cm^2)**	**Combustion Chamber Density (g/cc)**	**Combustion Chamber Volume (cc)**
Conventional	0.83	344	538	0.70	1185
Pancake	0.83	344	344	0.95	877

Compared with conventional hybrids, the pancake hybrid offers a few concrete advantages for small spacecraft. First, the pancake is short and squat. This feature keeps the motor from extending deep into the spacecraft; or the motor can be mounted external to the spacecraft, possibly as part as a multifunctional separation system, saving mass as well. The overall motor height and to some degree motor diameter can be adjusted to accommodate fuel requirements. As mentioned earlier, an externally mounted pancake motor utilising an annular mounted catalyst pack can radiate heat to space rather than other spacecraft subsystems and components. Fuel slivering within a conventional hybrid presents a nozzle blockage problem, especially when attempting to burn the solid fuel toward completion. Preliminary observations with the VFP hybrid suggests that the centrifugal force of the rotating gas is too strong to allow small amounts of slivered fuel to approach the nozzle. Slivered fuel would continue to rotate around the combustion chamber until it is completely consumed. In this particular design, there is a large fuel area exposed at ignition and very little variation in solid fuel burn area during operation, the primary variable parameter is combustion chamber height and the subsequent effect on combustion chamber volume.

The VFP hybrid has significant differences when compared to the conventional hybrid. First, the VFP is very much a three dimensional device. Once the oxidiser flows into the VFP it's path is continuously bent toward the centrally located nozzle (motion in the x, y plane) (fig. 9). Upon arriving in the vicinity of the nozzle, the flow is once again bent (90 degrees - down the z axis) to enter the rocket nozzle. The nature of the flow makes applying conventional hybrid wisdom more difficult because determining a value for Gox (Gfuel, and Gtotal as well) becomes very tedious. As mentioned earlier, Gox is the mass flow rate of oxygen divided by the cross sectional area. This parameter is easily defined in conventional hybrid analysis because local flow velocity is perpendicular to the port cross-sectional area. In the VFP, the flow direction constantly changes making analysis of the cross sectional area (perpendicular to the flow) very difficult. In order to accurately link the effect of oxidiser flow on fuel regression one needs to consider instantaneous velocity at a given position combined with the combustion chamber height and the subsequent effect on the boundary layer and heat transfer efficiency to the fuel grain.

Fig. 9 VFP combustion seen through a Plexiglas fuel grain (with a UV filter)

Performance:

All VFP firings conducted to date had a 2 inlet, tangential O2 injection scheme, the inlets being separated by 180 degrees. All firings demonstrated smooth combustion with a steady state combustion chamber pressure variation of less than 5% (fig. 10). Although the fuel plates exhibited a uniform regression pattern, there was evidence of fuel erosion where the high pressure/velocity O2 was forced off the combustion chamber wall and impinged on the fuel plate surface. This erosive effect tends to get more severe as the fuel regresses further away from the oxidiser inlets and is potentially aggravated by heat transfer from the steel combustion chamber wall softening the solid fuel. Currently, the plan is to reduce this erosive effect by increasing the number of O2 inlets (lower local velocity) and preventing (redirecting) the O2 from impinging on the fuel surface. Future designs will compensate for the heat transfer effects. Most firings employed a small nozzle entry port "carved" from the fuel grain which regressed in a fashion similar to a circular port in a conventional hybrid (fig 11). This variation of geometry is not desired and a experimental modification has been employed that incorporates a "submerged" nozzle that prevents this from occurring. Previous SSC experience with conventional hybrids had indicated that their was a significant amount of film cooling occurring during

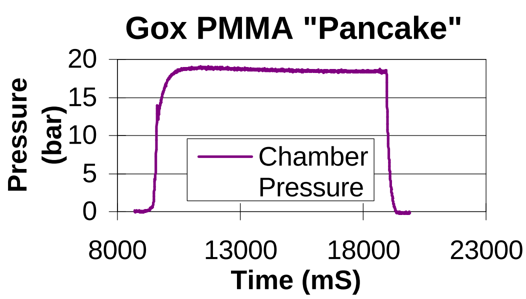

Fig. 10 Combustion chamber pressure trace for a 10 second burn

hybrid operations.[9] It was hoped that the VFP design would also share this film cooling benefit, but initial indications demonstrated otherwise. The first two attempts at achieving high performance mixing ratios (O/F of approx. 1.5) had resulted in the failure of two stainless steel nozzles within two seconds of ignition. Subsequently, all burns conducted since that time have been conducted fuel rich to keep combustion temperatures down and enhance any film cooling effect that may be present. Currently, modifications are underway to incorporate pyrolytically coated graphite nozzles and thus higher performance testing will soon begin.

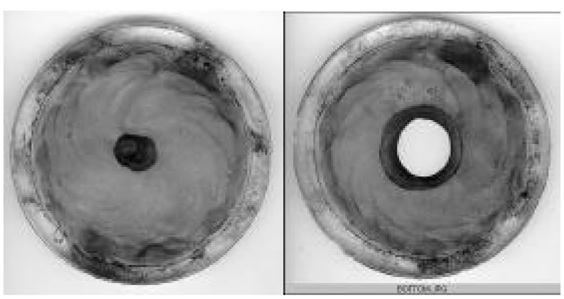

Fig.11 VFP fired fuel grains

Even though the tests were conducted at low O/F ratios, good performance was achieved. By measuring chamber pressure, propellant mass flow and nozzle throat area one can determine the characteristic exhaust velocity or C*.

$$C^* = (Pc * At)/Mdot$$

Where:
Pc = Chamber Pressure (Pa)
At = Area of the throat (M^2)
Mdot = Mass flow rate of the propellants (kg/s)

Measured C* is then compared with theoretical C* performance of the fuel and oxidiser combination. C* theoretical is determined from the thermo-chemical characteristics of the propellants. By using the USAF Isp program[6], theoretical C*s were produced for the applicable oxidiser to fuel mixture ratios and then compared with the actual measured values to determine combustion efficiencies. Combustion efficiencies of 88-92% (+/-1.8) have been demonstrated by the VFP. Other preliminary performance observations include a slightly negative O/F shift. This tendency is currently believed to be associated with a gradual increase in fuel mass flow due to the slow increase of erosive fuel burning as the burn progresses.

Although it is very early in the research and development stage for the VFP hybrid, some very positive results have been witnessed. First, high combustion efficiency has been demonstrated in a very "short" hybrid rocket motor. The VFP demonstrated high fuel availability and combustion chamber volumetric efficiency at an extremely low overall length vs. diameter for a hybrid rocket motor (fig.12). In addition, very smooth combustion (~3%) has been achieved on every firing to date. In the immediate future, the VFP research is concerned with establishing performance at high performance mixture ratios, gaining control of the fuel erosion at the oxidiser injectors and determining the effect of increasing combustion chamber height (i.e. fuel regression) on the rocket performance. In the long term, the research goals are to determine all the performance figures of merit for this configuration, switch to more flight like propellants (such as HTP/PE or N2O/PE), and converge on a flight design.

Conclusions

The propulsion systems currently under development at SSC promise to set new standards for high performance, yet low cost for small spacecraft. SSC is currently in the planning phase for an experimental propulsion mission in mid 2001. Although the details of the particular propulsion technologies to be flown have not been determined, a hybrid (conventional or VFP) rocket is very likely. The N2O and VFP research are new examples of SSC research increasing the capabilities of small satellites while simultaneously chipping away at the threshold cost of access to space.

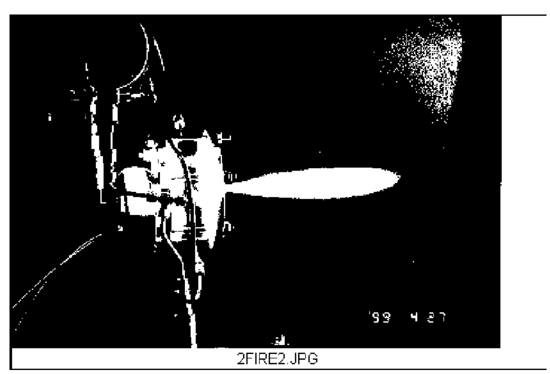

Fig.12 VFP firing

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