Experimental Validation of a 1 Newton Hydrogen Peroxide Thruster

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The evolution of a 1 Newton monopropellant thruster utilizing 87.5 % concentration hydrogen peroxide is reported. The design of the thruster is described, along with the testing of the catalyst material. Thruster firings were completed at sea-level atmospheric conditions and in vacuum, with a vacuum specific impulse of 160 seconds and a characteristic velocity efficiency of greater than 95 % achieved. The final thruster design demonstrated good overall performance, with initial lifetime testing suggesting that the catalyst offers good robustness. A case study suggests that the thruster would provide a level of performance comparable to that of a hydrazine-based propulsion system for a high velocity-change, volume-constrained Low Earth Orbit spacecraft mission, but at lower cost.

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Nomenclature

A = area (m^2)

$C*$ = characteristic velocity (m/s)

k = average ratio of specific heat of combustion products

$\dot{m}$ = mass flow rate (g/s)

P = pressure (bar)

R = universal gas constant (8.314 J / mol. K)

T = temperature (^{0}C)

W = average molecular weight of combustion products (kg/mol)

η = efficiency

Subscripts

C = thruster chamber

$C*$ = characteristic velocity

t = throat

I. Introduction

The monopropellant of choice for in-space satellite propulsion systems is hydrazine. However, the use of hydrazine carries toxicity concerns, and in 2011 it was added to the European Union's Registration, Evaluation, Authorization and Restriction of Chemicals (REACH) list of substances of very high concern. Although the likelihood of hydrazine being banned would seem reduced, there are still concerns over its carcinogenic nature and the rigorous handling procedures that are required. This disproportionately affects the cost of smaller propulsion systems using hydrazine, where cost and ease of use are paramount concerns. Consequently alternative, low toxicity "green" propellants are being been sought and developed [1]. At present there are various alternatives including ionic liquid based propellants, for example LMP103S [2, 3] and AF-M315E [4]. Another option is high concentration hydrogen peroxide, also known as High Test Peroxide (HTP). HTP has flight heritage [5] and can provide a high performance and low-cost green monopropellant choice, particularly for small-medium sized satellites.

There is currently considerable work being completed on the development of hydrogen peroxide based thrusters [6-10]. Here we describe work undertaken to develop a 1 N monopropellant thruster utilizing 87.5% concentration HTP. Although, the performance of the thruster would be expected to improve with HTP of a higher concentration (e.g. 98%), 87.5% concentration HTP was more readily available. It is envisaged that the thruster would be primarily used for orbit raising, rather than attitude control, on a small-medium sized spacecraft (200-500 kg) platform. This application demands a relatively high velocity-change (ΔV) requirement in which the thruster steady-state performance is considered most important. Hence, in this study, transient operation of the thruster has not been explored.

The ultimate aim of the project is the flight qualification and application of the thruster. Here we describe the evolution of the thruster design from an initial 'Mk 0' design, to Mk 1 and then to Mk 2, and the results of hot firing tests of these thrusters. The development program culminated in the testing and performance evaluation of the Mk 2 thruster under sea-level atmospheric and vacuum conditions, each with a substantial throughput of HTP. A case study considers the potential advantages of the HTP-based propulsion system compared with a system comprising hydrazine thrusters of a similar size.

II. Design Methodology

Three different versions of 1 Newton hydrogen peroxide thrusters were developed. The design methodology for each of the thrusters remained relatively consistent throughout and used a series of input parameters to a design model that utilized thermo-fluid relationships based on ideal, incompressible flow in the propellant liquid phase (e.g. through the injector) and isentropic, choked flow in the gaseous phase (e.g. through the nozzle). Firstly, the propellant mass flow rate was estimated from the required thrust (1 Newton) and the target specific impulse. The latter was based on the theoretical value for 87.5% concentration HTP, predicted by the Chemical Equilibrium with Applications (CEA) program [11], but reduced slightly to account for anticipated heat losses in the catalyst bed.

Once the design mass flow rate has been estimated the basic thruster dimensions (e.g. injector diameter, throat diameter, etc.) were determined. The injector orifice diameter determines the pressure drop between inlet and catalyst bed. The latter is necessary to avoid an oscillatory operational phenomenon where inlet and chamber pressure become linked ("chugging"), which is particularly prone to occur at low propellant inlet pressures and flow rates. The injector hole size is a compromise between having too low a pressure drop at low inlet pressures, possibly leading to this instability, and too high a pressure drop at high inlet pressures which reduces performance. The catalyst bed dimensions were based on our previous experience with hydrogen peroxide monopropellant thrusters, sized appropriately for the estimated propellant mass flow rate.

The design parameters of each of the three thrusters are shown in Table 1. Each of the designs assumes that the HTP concentration was 87.5%, the design chamber (i.e. nozzle plenum) pressure was 17 bar absolute and a single-hole injector was used. The evolution of the thruster design from Mk 0 to Mk1 and then from Mk 1 to Mk 2 is described in later sections.

Table 1. Thruster design parameters

Variable	*Mk 0*	*Mk 1*	*Mk 2*
Injector pressure drop, bar	2	2	3
Catalyst bed length to diameter ratio	2.5	2	1.34
Nozzle expansion ratio	167	3	300
Target vacuum specific impulse, s	165	166	172
Design propellant mass flowrate, g/s	0.62	0.61	0.59
Nozzle throat diameter, mm	0.63	0.64	0.63
Injector hole diameter, mm	0.24	0.22	0.20
Catalyst bed diameter, mm	12	12	13
Catalyst bed length	30	24	17.5

III. Experimental Facilities and Uncertainty

A. Sea-level atmospheric testing

For atmospheric testing at sea-level conditions a HTP propellant delivery rig had been previously developed and used to test both hybrid and monopropellant HTP thrusters in the 20 to 50 Newton range [12]. Although some testing of the Mk 0 and Mk 1 thrusters was initially conducted using this facility, it proved to be not suitable for testing small thrusters in the 0.2 to 1 N range. Hence a new laboratory was commissioned along with all the associated propellant delivery systems, thrust stand, exhaust extraction system and data acquisition and control system, specifically for testing HTP thrusters in this lower thrust range. In particular, two load cells (Tedea-Huntleigh Model 1042 and Model 1041, respectively) allowed for the thrust and also propellant mass to be measured continuously. The nitrogen-pressurized propellant delivery system, incorporating a 3.8 liter 316L stainless steel tank, was redesigned to include components cleaned to Swagelok 'Special Cleaning and Packaging' (SC-11) standards throughout, whilst the propellant tank was passivated using 30 % concentration of nitric acid. A Bronkhorst mini Cori-Flow flowmeter provided measurement of the propellant mass flow rate. A National Instruments LabView data acquisition (DAQ) system provided control of the pressurization bang-bang control system. Data were sampled at rates ranging from 50 – 400 samples/s during the various test campaigns, with then data sets down-sampled to a data rate of 0.05 s (20 Hz) to reduce aliasing errors. Temperatures and pressures were measured at strategic locations on the rig and thruster using TC Direct miniaturized Type K thermocouple probes and Gems Sensors Series 3100 strain gauge pressure transducers.

B. Vacuum testing

The vacuum facility is used to test commercial bipropellant thrusters up to 35 N and monopropellant thrusters up to 15 N, with vacuum generated and maintained using a two stage steam ejector system achieving 0.6 millibar during the 1N HTP thruster firing. Instrumentation available allowed the measurement of the thrust, propellant mass flow rate, thruster chamber pressure and gas temperature, the thruster surface temperature at various locations and the vacuum cell pressure.

The propellant delivery system used for testing was designed and built specifically for use with ‘green’ propellants. This system utilized a 30-liter 316L stainless steel tank. The whole system was passivated to maintain compatibility with up to 98% concentration hydrogen peroxide. A regulated system was used to pressurize the propellant tank and maintain the required test pressure (up to 30 bar) to an accuracy of ±0.01 bar. Remote operation was achieved through the use of pneumatically actuated solenoid valves on both the tank stop and emergency dump valve. An Emerson Micro Motion Coriolis CMFS010 flow meter was used to monitor the propellant mass flow rate.

C. Experimental uncertainties

Although the two test facilities had different configurations and capabilities, the instrumentation used for the tests reported here had similar characteristics. Hence the experimental uncertainties in the primary measurements and derived performance parameters are expected to be similar. The accuracies of the instruments used are claimed by the manufacturers to be ± 0.2% (reading) for mass flow rate, ±0.25% (full-scale, 25 bar) for pressure, ±0.8% (reading) for temperature and 0.04% (full-scale, 50 N) for thrust. It is assumed that the 16-bit data acquisition system introduced negligible error or uncertainty during the analogue-to-digital conversion process.

Using these values, the uncertainties in the primary measurements and derived performance parameters have been estimated using a propagation of errors analysis. The estimation has been carried out over the range of pressures at which the thrusters were tested (approximately 5 bar absolute to 25 bar absolute). The results are summarized in Table 2 and indicate that the uncertainties are of the order of 1% or less in all of the parameters. Note that the characteristic velocities (C^*) have been estimated using both the measured chamber pressure and mass flowrate (in combination with the design nozzle throat area) and also in terms of the measured chamber temperature (see Section IV). In the latter case the values for k (ratio of specific heats) and W (molar mass of decomposition products) predicted by CEA [11] have been assumed.

Table 2. Estimated experimental uncertainties

	Error at Inlet Pressure,± %				
Parameter	*5 bar*	*10 bar*	15 bar	20 bar	25 bar
Inlet Pressure, bar	1.2	0.6	0.4	0.3	0.3
Chamber Pressure, bar	1.3	0.7	0.5	0.4	0.3
Thrust, N	0.4	0.1	0.1	0.1	0.0
Mass Flow Rate, g/s	0.2	0.2	0.2	0.2	0.2
Chamber Temperature, ^{0}C	0.8	0.8	0.8	0.8	0.8
Specific Impulse, seconds	0.5	0.2	0.2	0.2	0.2
C* (Pressure), m/s	1.3	0.7	0.5	0.4	0.3
C* (Temperature), m/s	0.2	0.3	0.3	0.3	0.3

IV. Mk 0 Thruster Design and Testing

The design and results of initial testing of the breadboard prototype thruster, designated Mk 0, are described fully in [13] with an overview given here. The overall design was based on an existing and in-service 1 N resistojet thruster architecture. Modifications for operation with HTP included the addition of a single-hole injector with a wire-mesh distribution plate, sized to provide the required pressure drop at the design mass flowrate, and a wire mesh at the entrance to the nozzle to act as a catalyst retainer. An image of the assembled thruster is shown in Fig. 1, with standpipes attached to the catalyst bed and nozzle plenum chamber to allow pressure and temperature measurements to be made. No attempt was made to minimize either the thermal mass of, or heat losses from, this prototype thruster.

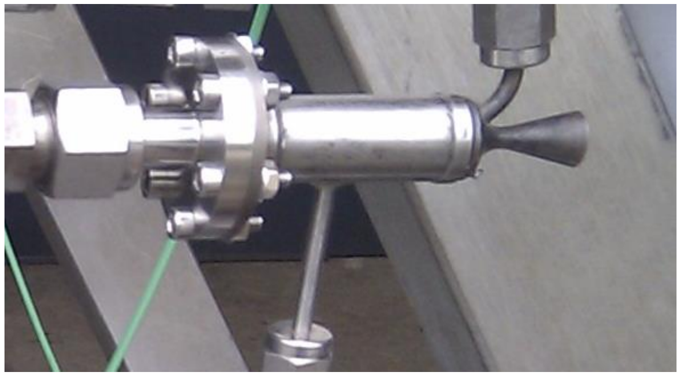

Fig. 1 Assembled Mk 0 thruster.

The catalyst used for this initial study comprised cylindrical pellets of ceria, approximately 1.4 mm in diameter and 6 mm in length, coated with an active phase of Manganese oxides (M_nO_x). This type of catalyst had been developed previously and successfully tested in a 20 N monopropellant thruster with 87.5% HTP [12].

Tests were conducted on the Mk 0 thruster under sea-level, atmospheric conditions and consequently thrust and specific impulse measurements are not discussed given the large nozzle area ratio. The thruster was operated from inlet pressures (P_{in}) in the range of 5.5 to 25 bar, as would be experienced by the device operating in blowdown mode in service on a satellite platform. The run duration was 180 seconds in each case, sufficient for steady-state conditions to be achieved.

Fig. 2 shows typical variations with time of pressure and temperature for a propellant feed pressure of approximately 15 bar absolute. Pre-heating of the catalyst bed was considered desirable to minimize thermal shock damage to the catalyst and, since no electrical pre-heating element was available, a short pulse of HTP was introduced into the thruster at around t = 10 s. A similar pre-heating strategy for this catalyst had previously been carried out successfully on a larger thruster [13].

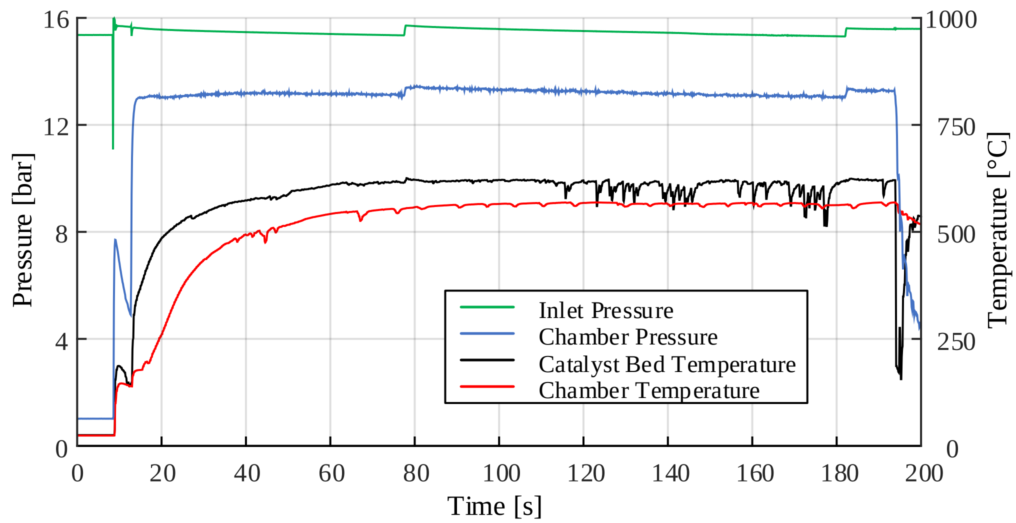

Fig. 2 Variation of pressure and temperature with time for Mk 0 thruster firing at nominal 15 bar inlet pressure.

In Fig. 2 it is observed that the catalyst mid-bed temperature rapidly reached approximately 150 °C after the warming pulse and then more slowly increased to a temperature of approximately 620 °C, taking around 70 s to reach steady-state. It should be noted that the complete and adiabatic decomposition temperature for 87.5% HTP is 690 °C, as given by Chemical Equilibrium with Applications (CEA) simulation [11]. Towards the end of the run the mid-bed temperature became unsteady, with peak-to-peak fluctuations of around 30-40 °C observed. The cause of the fluctuations is uncertain but may be indicative of incomplete decomposition in the catalyst bed. Meanwhile the nozzle plenum chamber was steadier, with fluctuations of only around 10 °C observed, although the mean temperature was lower by approximately 50 °C than that at the mid-bed location. Given that the catalyst bed geometry had not been optimized for HTP operation in this prototype thruster, none of these observations were unexpected.

The pressure signals shown in Fig. 2 indicate that these remained steady throughout the run (the slight saw-tooth variations being due to the operation of the bang-bang pressure regulation system). The average pressure drop across the injector and catalyst bed was approximately 2.3 bar. The roughness in the chamber pressure, estimated as one standard deviation of the signal, was approximately 0.11 bar or 0.8% of the mean value.

A summary of the Mk 0 thruster steady-state performance tests is given in Table 3. The characteristic velocity (C^*) is used a performance metric, providing a measure of how effectively the HTP decomposes in the catalyst bed, although the value is also affected by heat losses. C^* values were estimated based either on the chamber pressure (P_c) and propellant mass flow rate ($\dot{m}$) or the measured chamber temperature (T_c). The equation for C* in terms of pressure and mass flow rate is given by the standard formula:

$$C^*\left(P_C\right) = \frac{P_C A_t}{\dot{m}}\,, \tag{1}$$

whilst C^* in terms of temperature is given by;

$$C^*\left(T_C\right) = \frac{\sqrt{kRT_c/W}}{k\sqrt{\left[2/\left(k+1\right)\right]^{(k+1)/(k-1)}}}\,. \tag{2}$$

In the ideal (isentropic) case these two values would be identical. The corresponding C^* efficiencies (η_{C*}) were calculated by dividing these values by the theoretical value for 87.5% HTP, given as 913 m/s using the CEA program [11].

Table 3. Summary of Mk 0 thruster tests.

P_{in}, bar	P_C, bar	T_C, °C	$\dot{m}$, g/s	η_{C*} (P_C), %	η_{C*} (T_C), %
5.2	4.4	433	0.32	47.4	85.6
10.3	7.9	519	0.40	67.7	90.6
15.6	13.3	565	0.59	76.2	93.2
20.6	16.9	585	0.74	78.0	94.3
25.6	20.8	600	0.89	79.8	95.2

In Table 3 it is observed that the C^* efficiencies based on the measured chamber temperatures were significantly higher than those estimated from chamber pressures and mass flowrates, which would be expected if the temperature profile in the nozzle plenum chamber was non-uniform (the temperature being measured near the plenum axis). The effective (mean) plenum chamber temperature would be lower than that measured at the center of the chamber, hence the C^* (and C^* efficiency) based on temperature may be over-estimated. Both values increased as the propellant inlet pressure was increased and the difference between the two values decreased. Both trends are likely due to the heat losses becoming proportionately less (compared with the rate of chemical energy input) as the inlet pressure increased. The main conclusion from these tests is that the overall performance of the Mk 0 thruster at high inlet pressures is reasonable, but at low inlet pressures it is poor, due to a combination of non-optimized catalyst bed design and excessive heat losses, both of which were addressed in the design of the Mk 1 thruster described below.

V. Mk 1 Thruster Design and Testing

The Mk1 thruster was designed specifically for testing with 87.5% HTP in atmospheric conditions and was based on the experience gained in testing the Mk 0 breadboard thruster. Its overall characteristics are also indicated in Table 1. It included a shortened catalyst bed to reduce heat losses, an “anti-channeling” baffle in the catalyst bed to prevent the preferential flow of undecomposed HTP along the wall (and hence improve the temperature uniformity in the nozzle plenum chamber), a catalyst bed compression mechanism and the inclusion of an electrical heating element to enable more controlled pre-heating of the catalyst material prior to test firing. A single-hole geometry was selected, as on the Mk 0 thruster, as this provided repeatable pressure-drop characteristics. A low expansion ratio nozzle was utilized to provide perfect expansion at sea-level conditions.

For the initial tests with the Mk 1 thruster a new batch of catalyst material was manufactured with M_nO_x-coated ceria pellets that were in the 2 – 4 mm length range (compared to 6 mm length in Mk 0 thruster), as it was expected that this would improve the packing density of catalyst in the bed compared with the longer pellets used previously in the Mk 0 thruster.

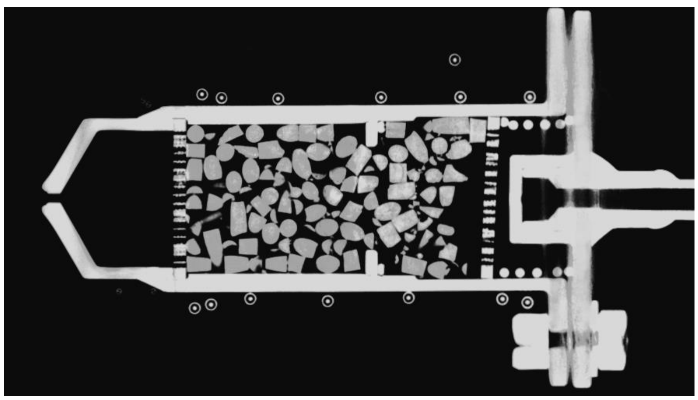

Fig. 3 Post test CT scan of Mk 1 thruster.

Fig. 3 shows a Computerized Tomography (CT) scan of the Mk 1 thruster, showing the thruster features described above. Analysis of the CT scan data indicated that the average void fraction in the catalyst bed was approximately 0.47. As on the Mk 0 thruster standpipes enabled access to the catalyst bed and nozzle plenum chamber to enable pressure and temperature measurements to be made. Other measurements included the temperature and pressure of the HTP upstream of the injector and also the propellant mass flowrate.

As with the Mk 0 thruster, the Mk1 thruster was operated over an inlet pressure range of 5.5 to 25 bar. The run duration was increased to 300 seconds to increase the overall propellant throughput, whilst the catalyst bed pre-heat temperature was set at 200 °C. Although this was a relatively high pre-heat temperature, for the thruster's intended application for beginning of life orbital maneuvers it was considered that there would be power available on board the spacecraft to enable this level of heating. It should be noted that, although hydrogen peroxide has some ability to be operated without pre-heating, others have utilized pre-heating on very similar thrusters [14].

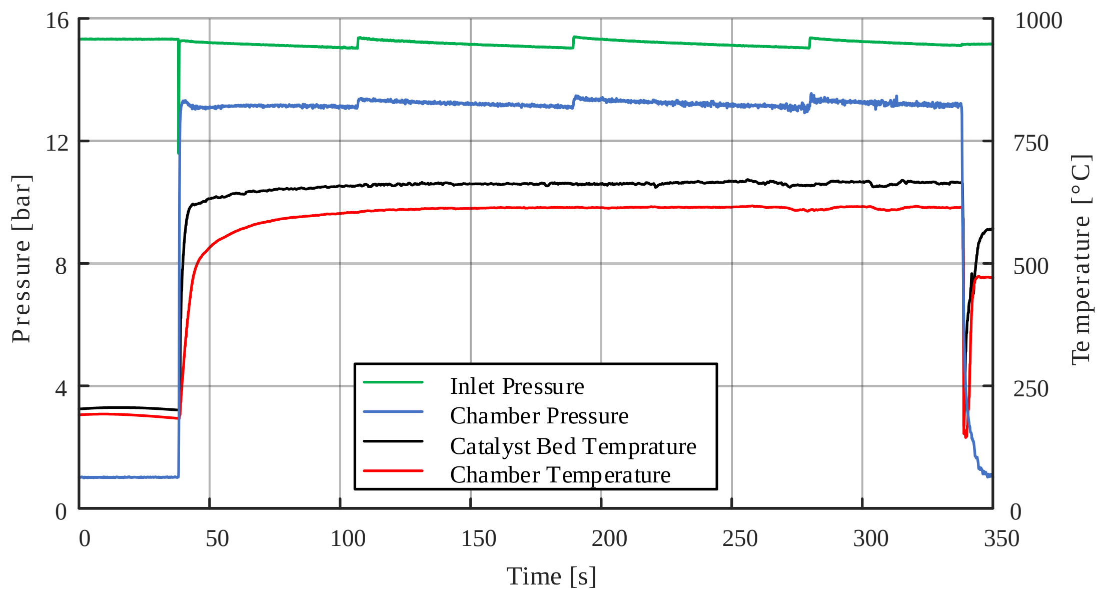

Fig. 4 Variation of pressure and temperature with time for Mk 1 thruster firing at nominal 15 bar inlet pressure.

Typical results for pressure and temperature are shown in Fig. 4, in this case for a run at a nominal 15 bar propellant inlet pressure. On commencement of the run at 40 s the catalyst mid-bed and nozzle plenum temperatures took approximately 50 s to achieve their steady-state values, slightly quicker than observed in the Mk 0 thruster tests. This thermal equilibration time is fairly typical for hydrogen peroxide thrusters of this size [15, 16].

The average steady-state temperatures achieved at the catalyst mid-bed location and in the nozzle plenum chamber were approximately 670 °C and 614 °C. These are 50 °C higher than those achieved in the Mk 0 thruster at the same run condition, but once again a substantial temperature drop, similar to that observed in the Mk 0 thruster test data, was observed between the mid-bed and nozzle plenum locations, indicative of heat losses. Fluctuations (of around 5-10 °C) were observed in both temperature signals towards the end of the run but these were much less than observed in the Mk 0 thruster.

As with the Mk 0 thruster, the pressure signals shown in Fig. 4 for the Mk 1 thruster indicate that these remained steady throughout the run (with, again, the saw-tooth variations being due to the operation of the bang-bang pressure regulation system). The average combined pressure drop across the injector and catalyst bed was approximately 1.9 bar, slightly lower than observed with the Mk 0 thruster at the same run condition. At this condition the roughness in the chamber pressure was approximately 1% of the mean value, slightly greater than that observed in the Mk 0 thruster. Some evidence of the onset of oscillatory behavior (chugging) at low inlet pressures was noted, indicating too low an injector pressure drop at low propellant flow rates. Overall however the smaller temperature fluctuations suggested steadier operation than the Mk 0 thruster and that the HTP decomposition was complete at the mid-bed location.

A summary of the Mk1 thruster steady-state performance tests is given in Table 4. As with the Mk 0 thruster, the C^* efficiencies increased as the inlet pressure was increased but are higher than those achieved in the Mk 0 thruster, achieving values between about 95-98%. However, the most striking observation is that the differences in the C^* efficiencies based on pressure and mass flowrate and those based on nozzle plenum temperature are much smaller than with the Mk 0 thruster. This suggests that the temperature profile at the exit from the catalyst bed (and entry to the nozzle plenum chamber) was much more uniform in the Mk 1 thruster.

Table 4, Summary of Mk 1 thruster tests with MnOx-ceria catalyst.

P_{in}, bar	P_C, bar	T_C, °C	$\dot{m}$, g/s	η_{C^*} (P_C), %	η_{C^*} (T_C), %
5.6	5.1	593	0.22	92.9	94.8
10.1	8.9	628	0.37	93.5	96.7
15.2	13.3	614	0.56	92.3	95.9
20.2	16.6	649	0.63	93.7	97.8
25.1	19.6	661	0.81	94.3	98.4

Figure 5 compares CT scan images of the catalyst bed before and after the above series of tests were conducted. Overall the void fraction after testing was slightly higher (by 1%) than before testing, but there is a significant increase in void volume in the upper catalyst bed resulting from catalyst bed fragmentation during testing. The damage could have been caused by the impingement of the high velocity jet of undecomposed HTP from the injector or due to thermal shock as the HTP started to decompose in the bed. This suggests that the catalyst lifetime may be limited, with less than 1 kg of propellant had been put through the thruster.

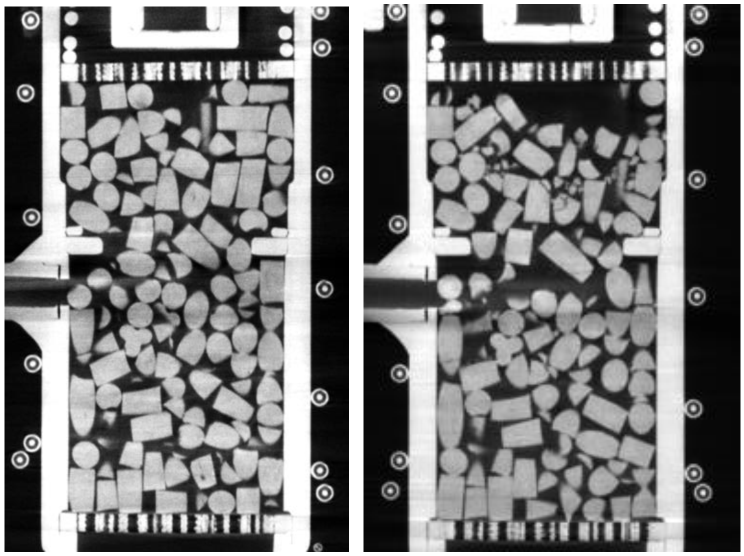

Fig. 5 CT scan of ceria-based catalyst before (left) and after (right) testing.

A. Testing of an Alternative Catalyst

An alternative, commercially-available catalyst material, comprising Platinum-coated gamma alumina particles, was also tested. The particles were irregular in shape but had an average diameter of 1.2 mm, significantly smaller than the cylindrical ceria pellets previously used. It was expected that this would improve the packing density in the bed. This was confirmed in the CT scans shown in Fig. 6; analysis of these images has indicated that the void fraction with the smaller, gamma alumina-based catalyst was 0.42, a 10% improvement compared with the ceria-based catalyst. Besides the greater surface are achieved with these smaller particles, the higher packing density is expected to inhibit the motion of the particles within the bed which may result in less mechanical damage to the catalyst pellets.

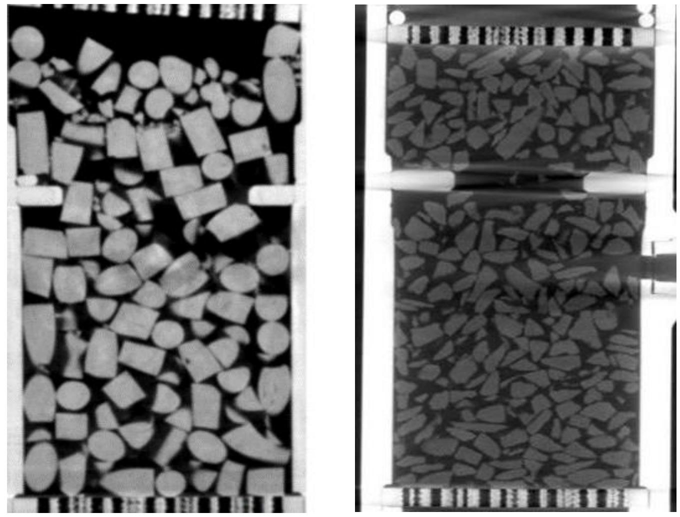

Fig. 6 Comparison of the packing density of ceria catalyst (left) and gamma alumina catalyst (right).

A set of steady state tests similar to that conducted with the Mk 1 thruster containing MnOx-ceria catalyst was carried out with the thruster containing the Pt-alumina catalyst. The smaller catalyst particles and increased packing density did not produce any significant increase in overall injector-nozzle pressure drop, likely due to the small catalyst pressure drop in comparison to the injector. Fig. 7 shows the variation in C^* efficiency (based on chamber temperature) with inlet pressure for both sets of catalysts. Overall the performance of the two catalysts appears to be similar, with perhaps the platinum catalyst slightly better. Note that there was quite a large spread of data for the MnOx-ceria catalyst when repeat tests were carried out at 10 bar inlet pressure, with two runs not achieving such high temperatures. The reason for this is unclear, but for full disclosure the data has been included.

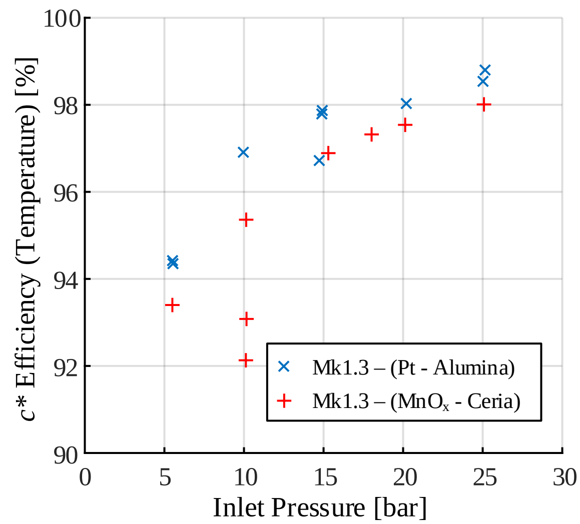

Fig. 7 Variation of C* efficiency (based on temperature) with inlet pressure for both the MnOx-ceria and Pt-alumina catalysts.

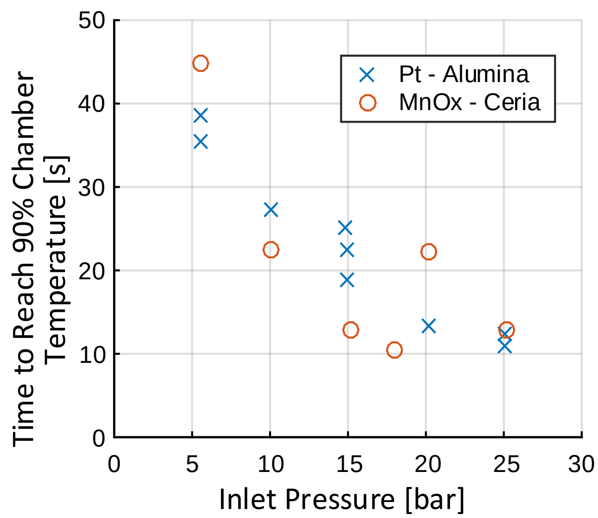

Fig. 8 Time to reach 90 % chamber temperature plotted against inlet pressure for MnOx-ceria and Pt-alumina catalysts.

Fig. 8 indicates the time taken for the chamber temperature to reach 90% of its steady-state value against inlet pressure. As expected for both catalysts the time taken is longer at the lower inlet pressures since the rate of chemical energy input is lower, however there seems to be no difference between the performances of the catalysts based on this criterion.

Unlike the MnOx-ceria based catalyst no degradation of the Pt-alumina catalyst was observed in post-test CT scans. I was decided to develop the thruster further using the Pt-alumina catalyst.

VI. Mk 2 Thruster Design and Testing

Table 1 indicates the main changes that were made to the Mk 2 thruster design. Compared with the Mk 1 thruster various modifications were made, including the use of a vacuum conical nozzle with an expansion ratio of 300. The catalyst bed was shortened (to an L/D = 1.34), as the data obtained from testing the Mk 1 thruster suggested that decomposition was complete at the mid-bed location. At the same time the internal diameter of the catalyst bed was increased slightly to 13 mm to accommodate a revised anti-channeling baffle arrangement, which allowed it to be inserted and removed more easily during loading of the bed with catalyst material. The (single-hole) injector pressure drop at design conditions was increased slightly to avoid the oscillatory behavior which had been observed in some tests of the Mk 1 thruster at low inlet pressures. A thermal stand-off was incorporated to reduce heat losses from the thruster body by conduction to the support structure. Unlike the Mk 0 and Mk 1 thrusters there were no bolted flanges, the catalyst bed being welded once the catalyst had been loaded. Also, only a single standpipe was included to enable the nozzle plenum chamber temperature and pressure to be measured. The plenum chamber was shorter than on the Mk 1 thruster, again to reduce the thermal mass of the thruster.

Fig. 9 illustrates a cut through of the design of the Mk 2 thruster. A spring situated upstream of the catalyst bed compresses the catalyst. Figure 10 illustrates the fully assembled Mk2 thruster, with the exception of the external electrical heater which was later wound around the outer wall of the catalyst bed.

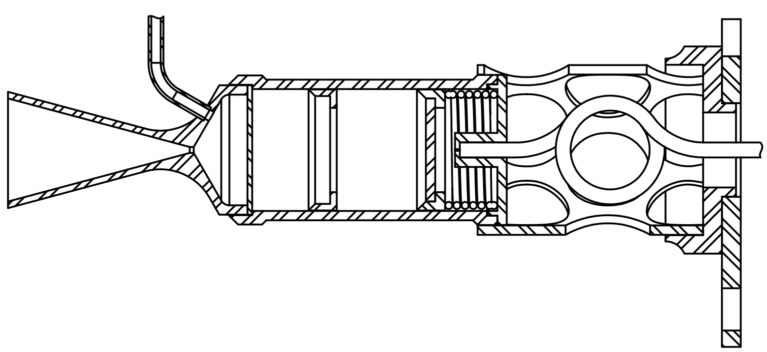

Fig. 9 Cut through of the designed Mk 2 thruster.

Fig. 10 Fully assembled Mk 2 thruster.

Two thrusters were manufactured and assembled, designated Mk 2.0A and Mk 2.0B, for sea-level and vacuum testing respectively. The injector hole sizes differed slightly, from 0.2 to 0.19 mm. The heater element on the Mk 2.0A thruster was wound around the outside of the catalyst bed whereas on the Mk 2.0B thruster it was vacuum-brazed onto the outside of the catalyst bed, as it would be for a flight-rated model.

A. Sea-level testing

A test program at sea-level conditions, similar to that conducted with the Mk 1 thruster, was performed on the Mk 2.0A thruster, with again the primary aim to determine the steady-state performance. A secondary aim was to perform high-throughput tests to investigate the lifetime of the catalyst and to bring the thruster closer to flight-readiness.

Typical results from these tests are shown in Fig. 11 for pressure and temperature, in this case for a propellant feed (inlet) pressure of 15 bar. The catalyst bed was electrically pre-heated until the nozzle plenum temperature reached 100 °C. The run duration was 180 seconds, sufficient to attain steady-state conditions. Compared with the Mk 1 results, the overall pressure drop from inlet to plenum chamber (approximately 3.5 bar) is slightly higher, as expected due to the smaller injector orifice. There was no indication of chugging at any of the inlet pressures, however some initial oscillatory behavior was observed in the plenum pressure when compared to the Mk 1 thruster (Fig. 4).

Similar behavior was noted in the plenum temperature signals where the overall steady-state temperature was approximately 100 °C lower than in the Mk 1 thruster. Consequently the C^* efficiencies derived from the data obtained in these tests were also lower than for the Mk 1 thruster, although still in the 90 – 96% range (Fig. 12).

Overall, the thruster performed reasonably well, achieving a characteristic velocity efficiency greater than 90 %. However the results were slightly below expectations, given that the design changes from Mk 1 to Mk 2 thrusters had aimed to reduce the thermal mass and heat losses, which should have resulted in an increased *C** efficiency, whilst consistently a 2-4 % drop was observed. A possible explanation is that the catalyst bed length had been overly reduced and HTP was not fully decomposed, causing increased roughness in the signals as well as the reduced plenum temperature.

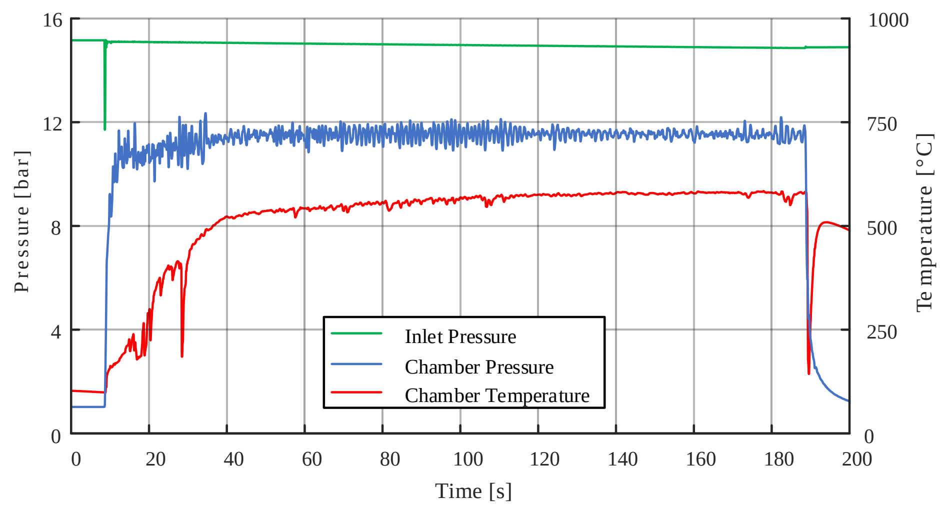

Fig. 11 Variation of pressure and temperature with time for Mk 2.0A thruster firing at nominal 15 bar inlet pressure.

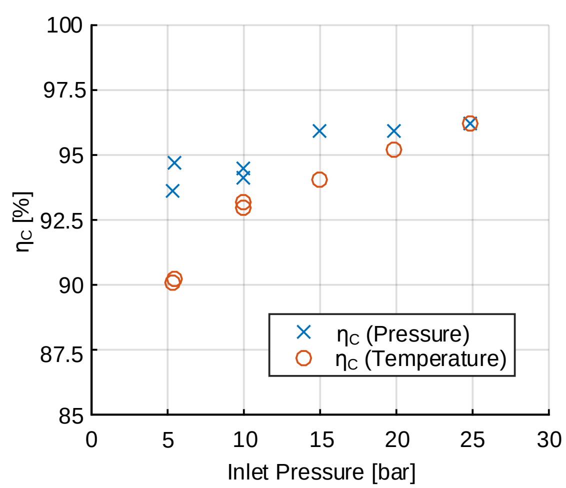

Fig. 12 Variation of C* efficiency with inlet pressure for Mk 2.0A thruster at sea-level conditions.

Following these relatively short-duration steady-state tests a series of longer duration (1200 second) steady-state tests were carried out, followed by a long duration blowdown firing. However, after an hour of the blowdown test an injector blockage occurred, halting testing prematurely. By this point a cumulative HTP throughput of 4.1 kg and a corresponding cumulative run-time of 3.1 hours had been achieved. X-ray microscope images taken after this test confirmed that the injector had become blocked, probably by a stainless steel swarf particle that had been generated during the thruster manufacturing.

Figure 13 illustrates a CT scan of the catalyst bed, obtained following the long duration tests. It confirms that no discernible damage to the Pt-alumina catalyst had occurred, with no breakup of the pellets visible. The void fraction in this case was 0.39, lower than observed in the Mk 1 thruster. It is thought that this improvement in packing density was a result of a revised catalyst bed filling procedure, facilitated by the revised anti-channeling baffle arrangement.

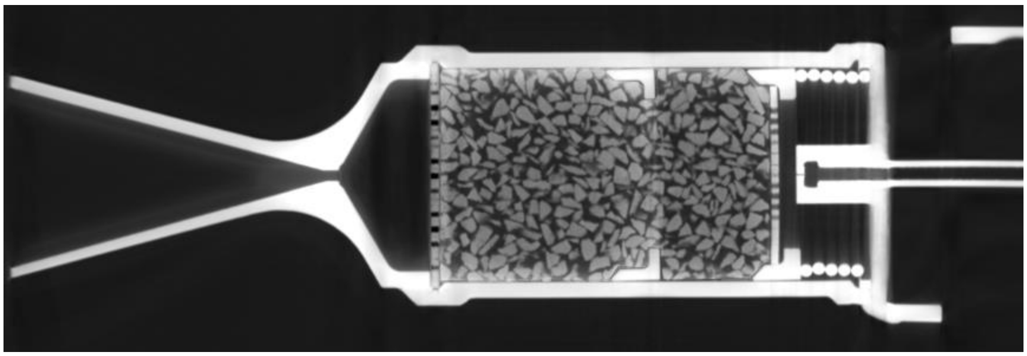

Fig. 13 CT Image of the Mk2.0A thruster post long-duration tests.

B. Vacuum testing

A similar test program to that described above was carried out under vacuum conditions using the Mk2.0B thruster. The instrumentation on the thruster was similar to that on the Mk 2.0A thruster but with additional thermocouple channels included, located as illustrated in Fig. 14.

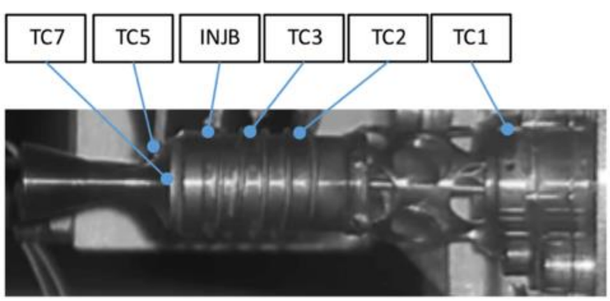

Fig. 14 Mk2.0B thruster illustrating placement of extra thermocouples along the thruster body.

The testing completed consisted of 600, 2400, and 300 second runs over inlet pressures ranging from 5.5 to 25 bar absolute. All tests were sufficiently long to achieve steady-state conditions.

Figure 15 illustrates the temperature and pressure-time variation for the first run completed under vacuum conditions. The run was 600 seconds long, at an inlet pressure of 15 bar. The thruster achieved temperatures above 650 °C within the nozzle plenum chamber, compared to approximately 600 °C with the Mk2.0A thruster tested at the same inlet pressure in atmospheric conditions (Fig. 11). The chamber pressure was more stable for this thruster operating under vacuum conditions, as can be seen through comparing Fig. 15 to Fig. 11.

The thermocouples positioned at other points along the thruster outer wall indicate that the greatest temperature rise occurred at the end of the catalyst bed. This suggests that the decomposition of the peroxide may not have been fully completed within the catalyst bed. It should also be noted that the thermal standoff would seem to be operating reasonably well, its temperature increasing to approximately 50 °C.

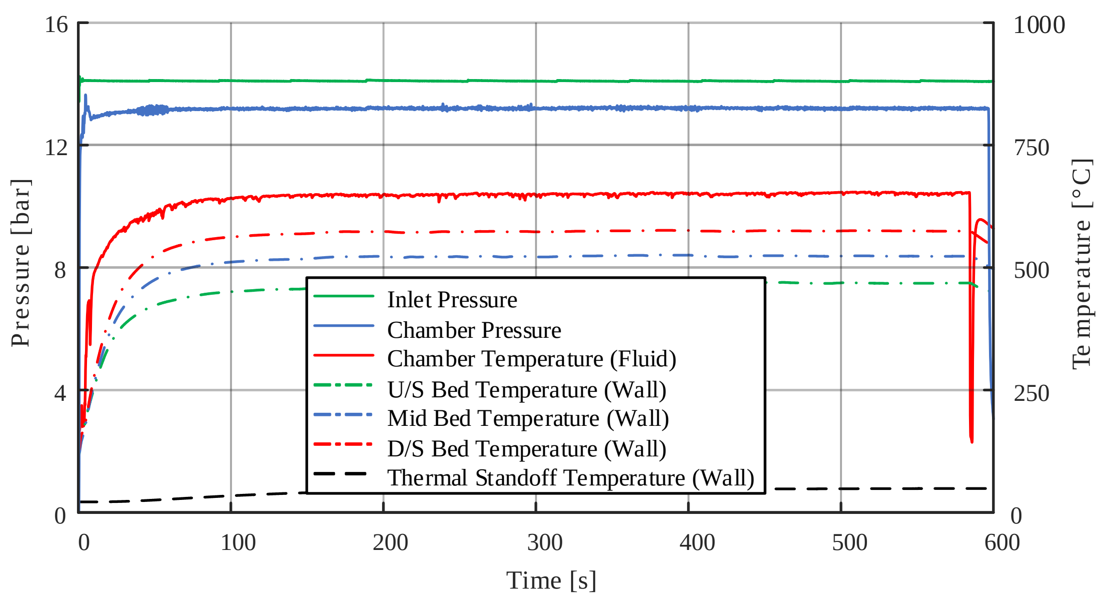

Fig. 15 Variation of pressure and temperature with time for a 600 second vacuum run at an inlet pressure of 15 bar.

The variations of steady-state chamber temperature and corresponding *C** efficiency with inlet pressure are illustrated in Figs. 16 and 17, respectively. For comparison, steady-state data are also shown for the Mk 2.0A thruster operating at sea-level atmospheric conditions.

The Mk 2.0B thruster operated better under vacuum, achieving chamber temperatures between 50°C to 100°C higher than the Mk 2.0A thruster that was tested under atmospheric conditions, probably due to the lack of convective heat losses. This results in C^* efficiencies, calculated using the chamber temperature, approaching the theoretical maximum, with values between 96% and 99% (Fig. 17).

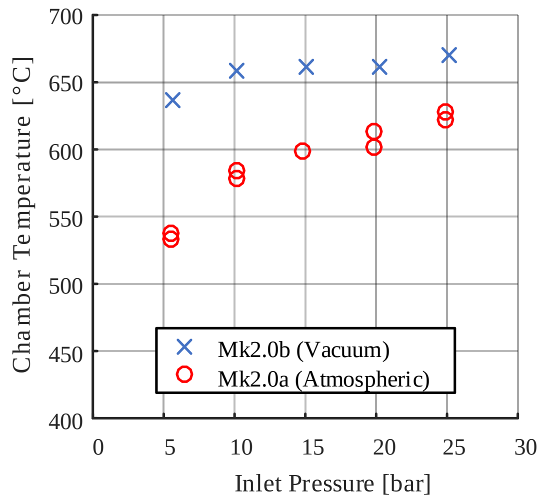

Fig. 16 Variation of chamber temperature with inlet pressure for the Mk 2 thrusters operating in atmospheric and vacuum conditions.

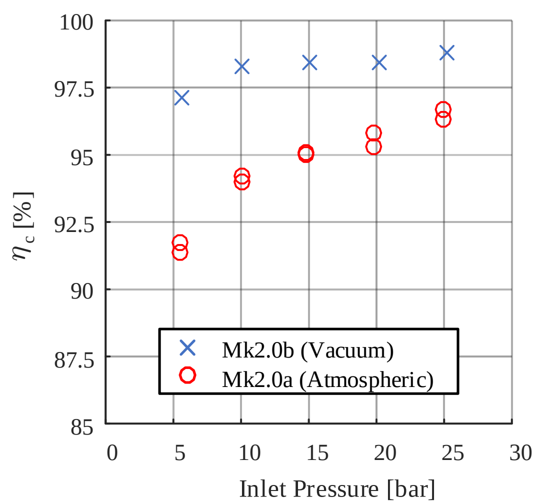

Fig. 17 Variation of C* efficiency based on temperature with inlet pressure for the Mk 2 thrusters operating in atmospheric and vacuum conditions.

Fig. 18 illustrates the variation of thrust and specific impulse (I_{sp}) with inlet pressure for the Mk 2.0B thruster. Data are shown for two different test durations, one of 300 seconds and one of 2400 seconds. There is little difference in performance between the two data sets.

Under vacuum conditions the thrust increased up to the desired 1 Newton at just less than 25 bar inlet pressure, with the I_{sp} in the range 150-160 s. This is only 7-13 % lower than the design target of 172 s. Given that the C^* efficiency achieved was close to or exceeded the design target value of 98%, with the 2 % below maximum performance corresponding to a difference in specific impulse of approximately 4 seconds, it is likely that the slight loss in I_{sp} is due to a non-optimum nozzle design. It should also be noted that the vacuum specific impulse decreases slightly with inlet pressure, which is in contrast to the trend exhibited by the characteristic velocity data. We believe this slight loss of performance can be explained through the nozzle flow transitioning from being slightly over-expanded at 5 bar inlet pressure (exit pressure of approximately 55 Pa compared to the ambient pressure in the test chamber of 60 Pa) to under-expanded at 25 bar inlet pressure (exit pressure of 340 Pa).

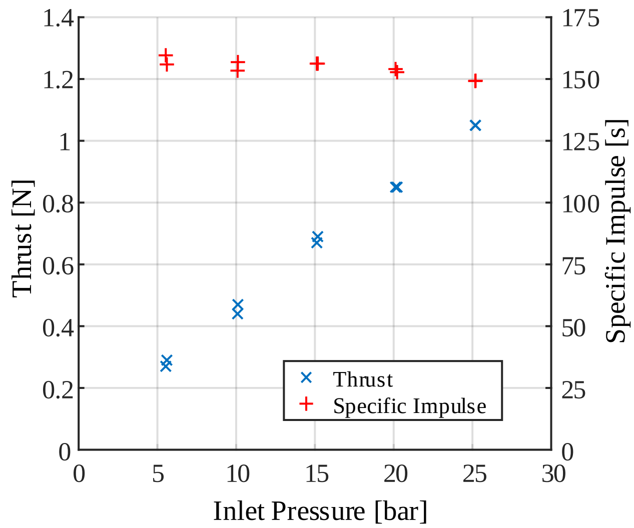

Fig. 18 Variation of specific impulse and thrust with inlet pressure for the Mk2 thruster operating in vacuum conditions, for 'long' tests of duration 2400 seconds, and 'short' tests of duration 300 seconds.

At the point where the test program was terminated, the Mk 2.0B thruster had experienced a total HTP throughput of 7.2 kg under vacuum conditions without any indication of catalyst degradation. Although this total was lower than the target throughput of 20 kg, given that there was no apparent catalyst degradation (in terms of reactivity at the beginning and end of the test and no obvious catalyst fragmentation) it suggests promising catalyst lifetime. However, further testing with greater propellant throughput is required, supported by more extensive catalyst chemical and physical analysis before and after testing.

VII. HTP One Newton Thruster Case Study

A possible in-orbit application for the HTP system using the 1 N thruster is to replace existing hydrazine systems used for orbit acquisition and control on small-medium sized satellites, such as the Surrey Satellite Technology Limited SSTL-42 platform which has a mass of 200 to 500 kg. The platform is tailorable to mission specific requirements, one of which is a high velocity-change (ΔV) option. The high ΔV propulsion option uses up to 37.6kg of hydrazine, stored in a 49.1 liter propellant tank. It is used in blowdown mode with two redundant 1 N thrusters, aligned through the spacecraft center of gravity. On the SSTL-42 platform the hydrazine system is designed such that it just fits into the volume available inside the spacecraft, hence to compare a HTP system with the hydrazine system the volume must be the same.

A typical mission scenario for the SSTL-42 platform could be a communications payload operating at high LEO altitude. To achieve a low cost launch, it could be launched as a secondary payload into a low LEO altitude. Hence it needs its on-board propulsion system to increase the altitude (typically by approximately 500 km) at the beginning of life and to reduce the perigee at end of life such that the spacecraft will re-enter Earth's atmosphere within 25 years. Such changes in altitude have been demonstrated with hydrazine thrusters. This type of application is particularly suitable for an HTP system as more than two thirds of the propellant is used early in the mission, which results in a greater gas volume within the tank, therefore reducing the likelihood of significant tank pressure increase from any HTP decomposition that may take place.

As an example, the specific mission mentioned above had a launch mass of 169 kg. With 37.6 kg of hydrazine in blowdown mode and an I_{sp} of 210 s a total impulse capacity of 77.4 kNs is achieved. On a dry mass of 131.4 kg, this is equivalent to a ΔV of 518 m/s.

In comparing the mission capabilities with HTP, consider the same volume of propellant used in a 4:1 blowdown. Due to the higher density of HTP (approximately 1400 kg/m^3 assuming 87.5 % concentration is used) extra mass of propellant is loaded, giving 52.6 kg of HTP. At an I_{sp} of 160 s (the level of performance demonstrated by the Mk 2 thruster under vacuum conditions) results in a total impulse of 82.6 kNs. In terms of total ΔV on 131.4 kg dry mass this is equal to a ΔV of 528 m/s. It should though be noted that hydrogen peroxide will require an aluminum tank rather than the titanium. The density of aluminum is 2700 kg/m^3 compared 4500 kg/m^3 for titanium but has about half the strength. For the hydrazine tank of dry mass of approximately 8 kg, we estimate given a doubling of the wall thickness the aluminum tank mass would be comparable.

Hence it can be seen that, for systems that are equivalent from an overall spacecraft volume point of view the 87.5% HTP system can deliver a slightly higher ΔV at the expense of 15 kg extra propellant mass compared with hydrazine. The HTP system architecture would be broadly similar to that of a hydrazine system and hence it is expected that most of the valves, filters, and transducers will be the approximately same cost, although HTP compatibility would need to be verified. The cost of an HTP system may also be lower in various other aspects. For example, the thruster developmental cost is lower, as the HTP testing can be performed in-house. Space standard titanium propellant tanks that are compatible with hydrazine are costly, typically around $180,000. The cost of hydrazine can be up to $1,300 per kg, compared to the cost of HTP which can be as little as $50 - $100 per kg. A hydrazine launch campaign is very expensive as the cost of safety is high, with a typical launch campaign for a hydrazine system costing $200,000 - $400,000, depending on the launch site. As HTP is non-toxic, it can be handled safely at significantly lower cost.

It should be noted that for other applications with smaller systems, the material costs will reduce but the launch site costs will not. Hence for smaller systems the launch site costs for hydrazine will be disproportionally higher.

It should also be noted that there is increasing interest and use of hydrogen peroxide at concentrations greater than 87.5 %, in particular at a concentration of 98 % [8]. This is partly due to the greater availability of high concentration hydrogen peroxide. If 98 % peroxide is used within the 1 N thruster, and an I_{sp} of 188 s is taken as the demonstrated performance (95 % of the theoretical maximum that can be achieved), then within the system as described above a total impulse of 105 kNs and a ΔV of 636 m/s can be achieved. This is very competitive with a hydrazine based system providing increased justification in the further research and development of a small satellite HTP-based propulsion system. The next steps in this thruster system development are towards the development of a full system (tanks, valves, etc.), targeting a flight opportunity in the early 2020's.

VIII. Conclusions

The design evolution of a 1 N HTP monopropellant thruster, and testing at various stages of its evolution, has been described. This work culminated in a thruster design approaching flight readiness, which was tested at sea-level atmospheric and vacuum conditions. During this work a new commercially-available catalyst was identified and tested, offering good performance.

Testing of the final thruster design under vacuum conditions indicated that the catalyst bed was performing well, with a C^* efficiency in the range of 96% - 99%. Even so, some further optimization of the catalyst bed may be possible, as testing under sea-level atmospheric conditions suggested that the C^* efficiency was lower than had been achieved in an earlier design. It is suggested that this could be resolved by increasing the catalyst bed length slightly.

Vacuum testing also demonstrated that a specific impulse in the range 150-160 s was achieved with 87.5% HTP, 7-13 % lower than the design target of 172 s. Given that the catalyst bed appeared to be performing to design expectations during the vacuum tests, it seems likely that the loss of I_{sp} can be attributed to a non-optimum nozzle design.

The testing programs conducted at atmospheric and vacuum conditions achieved HTP throughputs of approximately 4.2 kg and 7.2 kg, respectively. At the end of the test campaigns the catalyst beds in both versions of the Mk 2 thruster appeared to be performing well with no visible signs of physical damage to the catalyst material or loss of chemical activity being apparent.

From a systems perspective, a one newton HTP thruster system for a small to medium sized satellite would perform well compared to a hydrazine-based system. Although the I_{sp} is significantly lower, on a volume-limited platform the overall change in velocity and total impulse offered is comparable.

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