Results of Cold Gas and Resistojet Research for Small Satellite Application

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Abstract

The paper summarises on-going research into low-cost propulsion system options for small satellite missions at the University of Surrey. Recent results of the engine flight qualification tests of a 0.01 N water resistojet for the UoSAT-12 minisatellite are discussed. The research programme (done in collaboration with the United States Air Force Academy (USAF)) investigating nitrous oxide hybrids for small satellite upper stage application is described. A study (also done in collaboration with USAF) studying the use of cold gas thrusters for microsatellite stationkeeping and constellation maintenance is presented. Future propulsion research work is summarised.

Introduction

The objective of the research described in this paper is to investigate cost-effective propulsion system options for small satellite applications. The research first investigates the propulsion options to provide stationkeeping and constellation maintenance to the existing SSTL microsatellite platform (~50 kg). On-going resistojet work for stationkeeping of the UoSAT-12 minisatellite (~300 kg) and nitrous oxide hybrid rocket upper stage application is discussed. More background information on the history and early developments of these programmes can be found in [Lawrence, 1997].

Some of the work presented in this paper was done under a collaborative agreement between the University of Surrey, the US Air Force Academy, and the European Office of Aerospace Research and Development (EOARD).

The authors would also like to thank Royal Ordnance, Rocket Motor Division, Wescott, UK for use and support of their vacuum facility and Allied Signal, for providing data on the design of a microsatellite cold gas system.
Discussion

**Microsatellite Cold Gas System**

*Introduction*

Surrey Satellite Technology Ltd (SSTL) at the Centre for Satellite Engineering Research at the University of Surrey (UK) has been designing small, inexpensive, and sophisticated micro-satellites for 18 years. Traditionally, each micro-satellite has had a mass of approximately 50 kg. These very successful spacecraft have had to use whatever orbit the launcher could provide. Future mission requirements for the micro-satellite require propulsion systems to provide stationkeeping and constellation maintenance in low Earth orbit (LEO). The proposed satellite which will contain a propulsion system will have a mass of 65 kg. This will allow the extra mass (15 kg) to be used for the propulsion system components and propellant.

The volume in which the propulsion system must fit is going to be much more restrictive than the mass limitation. The micro-satellite is system of module boxes stacked to form a rectangular satellite. The propulsion system has been allotted a 100mm high module box in which all of the system must fit. This box has an approximate volume of 7 litres. Figure 1 shows the satellite configuration and where the propulsion system must fit. It also shows the suggested location of the one thruster.

This thruster location was also chosen so that the thrust vector would be along the centre of gravity (CG). This configuration is important because the attitude control for the satellite will be done using only reaction wheels and magnet-torquers. There will be no gravity gradient boom like most traditional SSTL satellites. For this reason the thruster for which the system was designed was a 0.01 N force. This small force was chosen so that if there was any CG off-set the satellite would not immediately go unstable. Figure 2 shows the attitude deviations in degrees during a 24 hour continuous firing (reasonable expected burn time).

![Figure 2: Attitude deviations for a 0.01 N continuous firing for thruster location shown in Figure 1.](image)

*The Mission*

The proposed missions for which the propulsion system would be required all include a constellation of satellites. The satellites could be used for remote sensing so that a portion of the world could have 24 hour satellite coverage. This is ideal to use for disaster relief and warning for many small countries. The satellites could also be used for near real time communications such as e-mail. The satellites could provide a way to send messages around the world through inter-satellite links. Figure 3 shows a possible configuration for an equatorial constellation.

For the study presented in this paper a constellation of 8 satellites in a final orbit of 650 km was chosen. All eight satellites will be launched on one launch platform into a parking orbit of 625 km and phased into the final 650 km orbit. Figure 4 shows a possible launch configuration.

![Figure 3: Equatorial Orbit Constellation](image)

T. Lawrence, M. Sweeting, M. Paul, R. Humble, J. Drum 2
When conducting the mission analysis for this system two major factors must be considered when determining the required $\Delta V$. The initial phasing of the satellites from their parking orbit into the final orbit is the first consideration. The second consideration is station keeping for the constellation.

The initial phasing from the 625 km orbit into the final 650 km orbit will require a $\Delta V$ of 13.43 m/s for each satellite. This will allow the constellation to be complete for all 8 satellites in 88 days. Figure 5 shows the phase time and $\Delta V$ trades.

The phase time can be shortened but this requires a significant increase in the required amount of $\Delta V$ that will be needed. Likewise, the $\Delta V$ required can be decreased at a significant increase in the phase time needed to complete the constellation. The chosen phase time of 3 months is acceptable since SSTL has used in the past the first 2 months of the satellite's lifetime for commissioning.

Because the design thrust is so low (0.01 N) the thrust time to achieve this phase will be a 24 hour thrust. Figure 6 shows how the satellite will spiral its way into the final orbit.

Drag compensation will be a major concern. This is due to the fact that the solar activity will be at a maximum in the years of 1999-2000. This means that the $\Delta V$ required to overcome the effects of drag are at a maximum of 3.61 m/s/year as well. This will result in the need for a 30 minute thrust every month or approximately a 1 minute burn per day. For a 3 year lifetime a total $\Delta V$ of 10.83 m/s is required. This makes the total $\Delta V$ required approximately 25 m/s.

System Options

When deciding what type of system is best for the microsatellite, all of the traditional options were considered. The options studied are listed below:

1. Cold-gas
2. Chemical

T. Lawrence, M. Sweeting, M. Paul, R. Humble, J. Drum
Looking at the design criteria established by Sellers in [Sellers, 1996], cold gas propulsion systems emerge as the best candidate. This is due to several factors:

- low \( \Delta V \) required
- low toxicity
- low power requirements (only have 30 W on orbit average)
- low thrust
- low cost (experience from UoSAT-12 system purchased from Arde).

Cold gas propulsion is a proven technology. It has flown on many different space-craft but never on anything as small as a micro-satellite. The initial design choice which needed to be made was the selection of a propellant. Figure 7 shows how the tank volume compares to the tank pressure.

![Figure 7: Tank volume versus pressure for various working fluids. The liquid systems have a very low storage volume, but must enter the chamber as a gas hence requiring some kind of propellant management system.](image)

Because the system volume is the most restrictive specification it is important to choose a propellant which does not take up much room. For this reason a 600 bar Nitrogen (\( \text{N}_2 \)) was investigated.

The design of the system is also important. Two possible configurations for the system were investigated, the bang-bang and regulated system (see Figure 8).

![Figure 8: Bang-bang and regulated cold gas propulsion system drawings](image)

The regulated system uses much less volume than the bang-bang system. However, the bang-bang system has also been thoroughly tested for the SSTL mini-satellite. The price of each system quoted by Allied Signal is not too different (ROM cost = \( \sim $50,000 \)).

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Feed System</th>
<th>Propellant mass, kg</th>
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<td>Storage pressure, bar</td>
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<tr>
<td>Tank volume, L</td>
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<td></td>
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<tr>
<td>Feed pressure, bar</td>
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<tr>
<td>Flow rate, kg/sec</td>
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<td></td>
</tr>
<tr>
<td>Thruster throat diameter, cm</td>
<td>0.0133</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

A possible system configuration is shown in Figure 9. It provides storage for a maximum of 3 litres of propellant and leaves room for the system components.
Since the provided volume will only allow for 3 litres of propellant it is necessary to have a 600 bar storage tank for the N₂ system. This pressure will provide the needed mass of propellant to achieve the desired ΔV of 25 m/s.

**Mark-II Water Resistojet Test Results**

In January 1996, the water resistojet programme was started at the University of Surrey. The first thruster (*Mark-I*) test results were presented at the 10th AIAA/USU Small Satellite Conference and are described in more detail in [Lawrence, 1996]. The improvements needed for the next phase were:

- heater (higher sheath temperature and lifetime)
- efficiency (high energy losses)
- start-up (took at least 30 minutes to reach steady state)

These improvements were incorporated into the design of the *Mark-II*.

The *Mark-II* thruster (shown in Figure 10) has the following design features:

- new 200 W (12.5 mm by 55 mm) cartridge heater (nickel chromium alloy filament surrounded by magnesium oxide insulation and an Inconel sheath) provided by ISE Inc. of Cleveland Ohio USA
- 30 mm by 59 mm 316 stainless steel thrust chamber (3mm in thickness) surrounded by 25 mm thick Micropore insulation
- 350 μm particle diameter silicon carbide bed heat transfer medium (packed in the thrust chamber surrounding the heater at a porosity of 35 %)
- thrust ~ 0.01 N
- inlet pressure ~ 10 bar
- welded connections
- 2 pressure transducers (inlet and chamber) and 5 thermocouples (located in the middle of the heater, at exit right before nozzle, outer wall, inlet to the injector, and ambient air/vacuum)
- conical nozzle (half angle = 15 degrees), 0.1mm throat diameter, expansion ratio = 100:1

Testing started in April 1997 and is ongoing. Results were obtained at seal level and in the RO Wescott vacuum facility. A total of 54 hours of test data has been collected (20 hours @ sea level and 34 in vacuum). The sea level results (Figure 11) gave:

- improved efficiency (almost 2 x at half of the power input)
- no vapour in exhaust at start-up
- lower power required from heater to reach temperature
- no leaks in welded connections
- insulation wall temperature remains cool (20 C)

T. Lawrence, M. Sweeting, M. Paul, R. Humble, J. Drum 5
Figure 11: Comparison of Mark I and Mark-II sea level test results. Notice almost the 2 x increase in Chamber temperature at a lower power input for the Mark-II. The Mark-II was also run on a mixture of 60% water and 40% IPA. This was done to lower the freezing point of the water to -20 C (for spacecraft thermal integration concerns). This mixture lowers the performance compared to water for the same energy input.

Figure 12 shows the Mark-II vacuum temperature results. These results were better than the sea level results (expected due to the nozzle design):

- improvement in chamber temperature by 200 K at a lower input power compared to sea level results
- no ice in exhaust plume at 0.4 millibar of vacuum pressure
- no variations in performance with respect to gravity
- Calculated Isp's from 150 - 224 sec (see Figure 13)
- Calculated Thrust levels ranged from 0.012 - 0.018 N (see Figure 14).

Some new problems developed while testing the Mark-II. As shown in Figure 14, there were some flow oscillations observed at start-up as the thruster was reaching steady state. This was due to the coupling of the inlet pressure to the chamber pressure as the water evaporated. A stainless steel sintered disk was added just aft of the injector to give a pressure drop to prevent the flow oscillations regulating the inlet flow. The disk did not present enough of a pressure drop, so a Lee Visco Jet flow restrictor was added to the inlet. This worked very well in preventing the oscillations and a smoother start-up was observed.

The next problem encountered was clogging of the nozzle. Figure 15 shows the thrust versus time for two of the thrusters used in the Mark-II programme (same design). The clogging is caused by oxidation of the stainless steel. Figure 16 shows the oxidation rate of steel as a function of temperature. The Mark-II operated at temperatures close to 1000 C for long duration during the test programme. Since the nozzle was only 0.1mm in diameter at the throat, the oxidation of the stainless caused flaking which gradually clogged the nozzle over time. Figure 17 shows the results of an electron microscope analysis of the nozzle. The picture shows the flakes with the nozzle exit completely clogged. The Scanning Electron Microscope also analysed the surface material composition. The results showed high concentrations of oxygen in the clogged area.
The test apparatus is shown in Figure 18 below. All of the tests were conducted at RO Wescott at the J-4 test site (sea level).

Figure 18: Pictures of the N2O Test Facility. The Mark-I thrust chamber was used to get the first phase results.

It was decided to use the Mark-I chamber to get the initial results (programme was started concurrently with the Mark-II programme). Gaseous N2O was stored at 48 bar and regulated down to 10 bar at the chamber inlet. Mass flow rate varied from 0.00025 - 0.0005 kg/sec at powers ranging from 200 - 560 W.

Figure 19 shows the results of the programme to date. Notice in the last run the verification of decomposition since more energy was coming out than was being put into the system.

Figure 19: Initial N2O test results conducted at sea level. The vacuum isp was calculated based upon the sea level results. The last chart showed dissociation occurred due to energy balance calculations and C*.

The proposed future work will be to build a bigger resistojet at 1200 W input power and test at flow rates more prototypic of hybrid start-up ~0.1 kg/s. This work will be done by using the Mark-III resistojet described above. A literature search has also revealed that there are some materials that can cause catalytic decomposition at lower temperatures (400 C). Suitable bed type materials will replace the SiC to investigate this option.

Conclusions

For very low-cost, logistically constrained missions, unconventional options such as water resistojets and cold gas systems offer many unique advantages over current off-the-shelf options. Future research will focus on demonstrating these technologies in orbit.

T. Lawrence, M. Sweeting, M. Paul, R. Humble, J. Drum 8
Future work will focus on more endurance tests of the Mark-II with the new thrust chamber and nozzle materials. The heater and bed lifetime have been very good to date. Post experiment inspection and material analysis has revealed no degradation or material change in the bed and heater material.

A Mark-III is currently under development and will be tested in parallel with the Mark-II starting in October 1997. The Mark-III is a 1 kW heater that will run at approximately 0.1 N thrust. It will use water, biowaste, nitrous oxide, nitrogen, and a mixture of water and IPA as the working fluids. The design has been slightly scaled up for possible future mission opportunities and ease of integration at the Edward’s AFB AFRL Rocket Propulsion Directorate thrust stand. The Mark-II and Mark-III will be tested at Edward’s AFB.

N₂O Resistojet Research

A hybrid rocket engine using the propellant combination of Nitrous Oxide and HTPB represents the least hazardous approach to building and testing of hybrid rocket engines.

The usual method of initiating combustion is to supply thermal energy, typically with some form of pyrotechnic, at the start of N₂O injection. Once N₂O /HTPB combustion has started the process is self-sustaining. However, pyrotechnic ignition is acceptable only for a single shot device, as multiple starts require the complication of a breech mechanism.

An alternative approach to ignition would be to supply the thermal energy directly to the N₂O until such time as steady state combustion is achieved.

It is suggested that passing some of the incoming N₂O through an electrically heated bed would raise it’s temperature sufficiently to sublime the surface of the HTPB and thus start the combustion process.

The resistojet currently under development at SSTL is an ideal candidate device for an experimental programme to investigate the suitability of electrical heating for hybrid engines.

Thus an effort (in collaboration with the USAF EOARD and USAFA) was started to investigate this technology. The key test results in the programme will be to empirically determine the rate of decomposition of the nitrous oxide as a function of time and mass flow rate.

T. Lawrence, M. Sweeting, M. Paul, R. Humble, J. Drum
References


