A MEMS Based Experimental Colloid Thruster Package for Nano satellites

Barry Kent[1], John Stark[2], Bob Stevens[3], Matt Alexander[2],
Adam Baker[4], Dave Gibbon[4], Douglas Liddle[4]

[1] Space Science and Technology Department, Rutherford Appleton Laboratory,
Didcot, OX110QX, U.K. + 44 1235 446364, **b.j.kent@rl.ac.uk**

[2] Department of Engineering, Queen Mary, University of London,
London E1 4NS U.K., + 44 207 882 5195, **j.p.w.stark@qmul.ac.uk**

[3] Central Microstructures Facility, Rutherford Appleton Laboratory,
Didcot, OX11 0QX, U.K. + 44 1235 446504, **r.stevens@rl.ac.uk**

[4] Surrey Satellite Technology Ltd, University of Surrey,
Guildford GU2 7XH, U.K. + 44 1483 683 474, **a.baker@sstl.co.uk**

ABSTRACT: Colloid propulsion can provide flexible electric propulsion systems capable of delivering both high thrust density and high specific impulse at low power demand. It thus has wide applicability over a range of missions; from end of life de-orbit devices to sophisticated formation flying or disturbance compensation systems. Because of the low system demands colloid propulsion is particularly applicable to small satellites. Scalable micro-fabricated colloid thrusters allow many options: The total thrust available is simply a function of the number of nozzles and, for a given propellant the specific impulse is a function of the acceleration potential.

We outline a nano-satellite micro propulsion system in support of a concept proposed by Surrey Satellite Technology Ltd (SSTL), and demonstrated with their SNAP-1 nano-satellite mission. In the proposed target mission the nano-satellite co-flies with a micro satellite in low Earth orbit.

The micro-fabricated propulsion system has multiple thrust heads which in the present concept gives full 6 degree of freedom control using appropriately positioned thrust heads each capable of delivering 100 micro Newton. The overall system mass is 524g with an orbit average power demand of less than 0.1W.

INTRODUCTION

In previous papers [1,2,3] we have outlined the basic concepts involved in the production of micro fabricated colloid thruster systems and described some of the relevant physical processes driving the electrospray phenomenon. Others have also highlighted the beneficial qualities of colloid thruster systems [4, 5]. It is apparent that a detailed understanding of the electrochemical interactions in all parts of the system is important in preparing a system suitable for space application. It is now equally clear that at low system cost, colloid thruster technology can provide an interesting option to the problem of attitude control and manoeuvrability in small satellites. Our particular interest is in using the techniques of micro fabrication so that the versatility already present in colloids is greatly enhanced such that a full 6 degree of freedom spacecraft becomes possible for a satellite with a total mass of 10kg and size 310x310x210 mm^3.

In order to provide a realistic and yet ambitious target we propose a propulsion system for a nano satellite based on the previously flown SNAP-1 satellite. We have constrained our design to fulfil the requirements of a challenging mission in which the nano-satellite co-flies with a micro satellite in low Earth orbit. The applications for such capability are numerous; ranging from inspection and possible repair or upgrade of existing spacecraft, to inspection, information gathering or surveillance of satellites.

The colloid thruster experimental package we discuss supports a nanosatellite concept originally proposed by Surrey Satellite Technology Limited (SSTL), and demonstrated with the SNAP-1 mission using a 6.5 kg nano-satellite. The SNAP-1 satellite was launched in 2000 by SSTL. At that time the spacecraft contained a cold gas propulsion system weighing 450 grams in total and achieved a measured $2.1 m.s^{-1}$ of delta V in orbit. It was able to use its cold gas system to fly past and image its launch partner [6].

Figure 1: SSTL's SNAP-1 spacecraft

Using the existing spacecraft geometry of an equilateral triangular prism, the colloid propulsion thrust heads would be mounted on brackets from each of the three rectangular sides. The small propellant tanks are in the central well and the high voltage units and control electronics are housed in one of the electronics trays which make up the sides of the satellite.

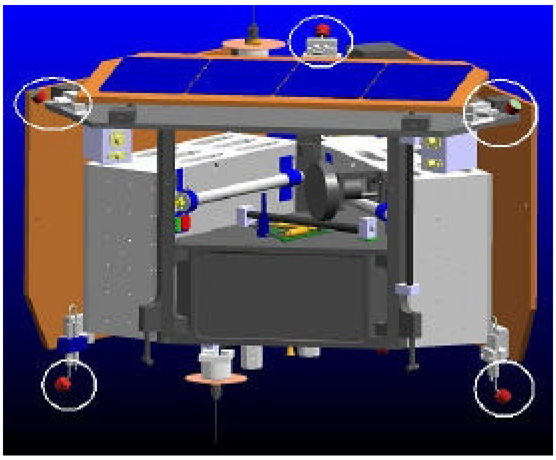

Figure 2: Solid model (panel cutaway) of SNAP-2 platform, incorporating colloid thruster heads as indicated.

The primary driver for this type of platform, which will apply on any electrical propulsion system, is the power available. The spacecraft has body mounted solar panels to avoid the cost and complexity of deployable panels. However, as the spacecraft is small, the panel area is also small, limiting the power available. On SNAP-1 the power available to the whole spacecraft was around 6 watts orbit average power, for SNAP-2, 15W will be available during illumination of the arrays.

BASICS OF COLLOID THRUSTERS

In essence a colloid thruster, uses the principle of electrospraying of a conductive liquid to generate thrust. A fine droplet spray is created from a fluid jet that derives from an array of micro-fabricated emitters arranged in a structured pattern. An intense electric field, of order 10^7 V/m is applied using a micro-fabricated grid, which under appropriate conditions forms a stable Taylor cone-jet, as seen in the figure 3.

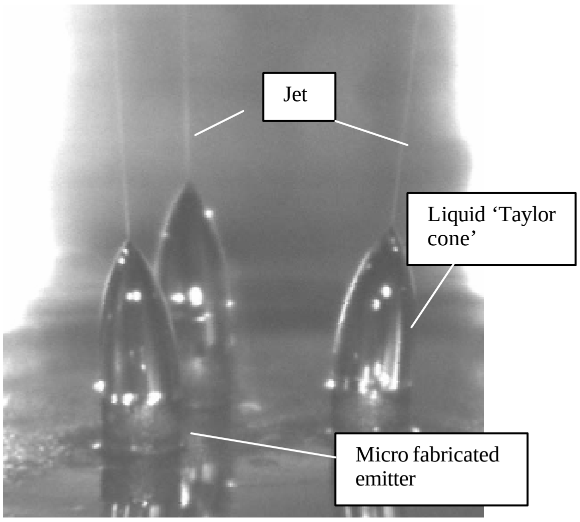

Figure 3: Electrospray from 3 nano-emitters

The jet itself breaks up into a stream of mono-disperse charged droplets. After passing through a first extraction grid the charged droplets are accelerated by a second accelerator grid to become a collimated charged beam. In this the method of acceleration is similar to an electrostatic ion thruster; the system is maintained at electrical neutrality by means of a micro fabricated field effect neutraliser[7]. The fluid droplet velocity is determined by the overall accelerating potential and the selected high conductivity fluid, with typical Isp values in the 500-600s range. The thrust level can be selected over a wide range by varying the number of emitters in the array. Typical thrust is in the range from 20 to 200 μN / cm^2.

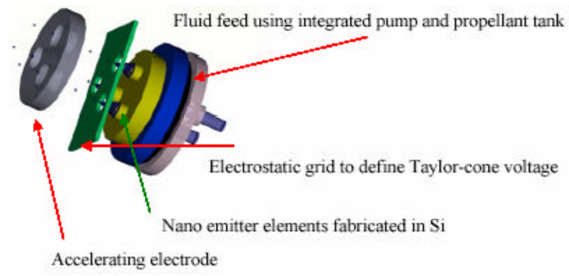

Figure 4: Key elements of a colloid thruster

In a micro fabricated system a large number of emitters (figure 5) and their integrated field and acceleration grids can be produced as an array. As shown in figure 4 the integration can be extended to include an integrated micro fabricated pump and propellant storage, but in this paper a hybrid system is considered in which only the thruster head components are micro fabricated and all other components are macro scale.

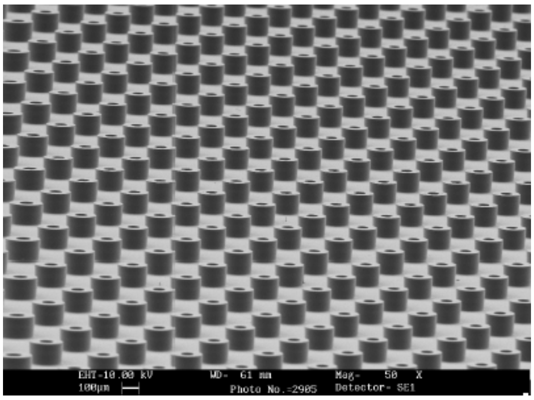

Figure 5: Section of micromachined nozzle wafer (total 20000 nozzles in 75mm diameter)

OUTLINE SYSTEM DESIGN

A concept design for a colloid thruster based 6DOF manoeuvring system has been created which could be manufactured and integrated with the SNAP-2 platform.

With a propulsion system as versatile and scalable as micro-fabricated colloid thrusters many options become possible. The total thrust available is simply a function of the number of nozzles [1] and, for a given propellant the specific impulse is a function of the acceleration potential used in the final grid [3]. A wide range of different configurations is possible and a large number of scenarios and experiments may be investigated. In order to provide a clear expression of capabilities we provide detailed system parameters for one configuration only.

For this mission our hybrid micro/macro-fabricated design concept consists of a micro propulsion system having nine separate thruster heads, with each head capable of delivering 100 µN. These are supplied from tubular propellant reservoirs. In other designs [8] we have provided variation in thrust by implementing fluid feed to separate areas (sub-divisions) within the nozzle arrays. For example we could select arrays areas of at 10, 40 and 50 micro Newton on each 100µN thrust head, to give a range of selectable thrust. However in a 9 thruster design the complexity could become unwieldy and although the final selection of sub division of thrust level availability is subject to further evaluation, in this paper the thrust heads are not sub divided. To keep system complexity low, the high voltage is applied to all thrust heads simultaneously and a valve switches on the fuel supply to each thrust head as required.

Thus the outline colloid thruster subsystem consists of a fluid feed system, which supplies the working fluid to the thrust heads via control valves. High voltage units provide the extraction and acceleration fields. A neutraliser and system monitors (pressure and temperature) together with an isolated power supply completes the system. The overall system mass is 524g with an orbit average power demand requirement of less that 0.1W. This highly versatile approach is made possible by the use of lightweight low power control valves many of which are available as COTS items. Design iteration is expected to reduce both mass and power values further.

The experimental propulsion system using nine electrospray 'colloid' thrusters will provide full agility in 6 Degrees of Freedom, adding pitch, yaw and roll to x, y and z translation. This is shown schematically below in figure 6. The colloid thruster system envisaged will allow rotational control and hence precise attitude control during tracking of the micro satellite. DeltaV requirement for pitch / attitude manoeuvres has been determined to be 0.866m/s.

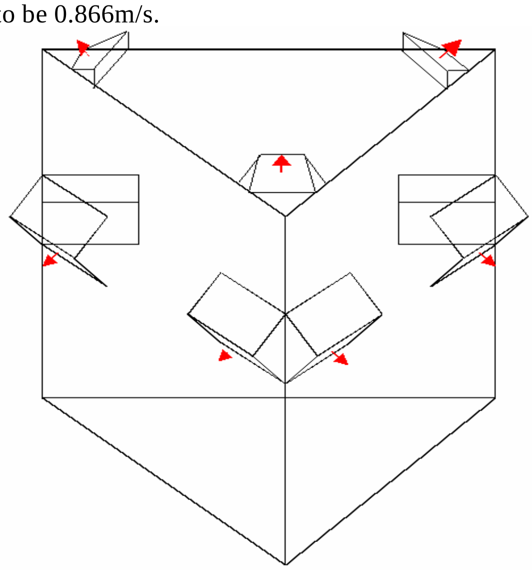

Figure 6: SNAP-2 6DOF thruster arrangement

PROPULSION SYSTEM DETAILS

Propellant / tank

Depending on the mission requirements the propellant is selected from solutions of ionic salts in organic solvents, e.g. sodium iodide in tri ethylene glycol, or one of the high conductivity ionic liquids. These are essentially liquid salts. A possible propellant is the ionic liquid based on an imidazolium salt (e.g. 1-Butyl-3-methylimidazolium tetrafluoroborate ($BmimBF_4$) with a density of 1150kg/m^3. This is non-toxic and has negligible vapour pressure, and is liquid above a temperature of -80ºC.

The propellant tank will require an expulsion device, for propellant management to ensure separation of liquid and gas in microgravity. For the high aspect ratio cylindrical propellant tanks preferred for SNAP-2 a spring loaded piston expulsion system sealed using O-rings offers a simple solution to pressurising the propellant to up to a few Bar, i.e ~0.3MPa above atmospheric pressure. Gas pressurisation of the piston is also feasible however this will complicate propellant loading. Propellant will be loaded before shipping to launch site, and before final AIT.
Ti-3Al-2.5V tube qualified on the SNAP-1 mission had an OD of 3/8" (9.525mm) with 0.019" thick wall, giving an ID of 8.560mm. Propellant volume is determined by the length of tube, which is constrained due to space within the SNAP-2 stack. The pressure rating of this Ti alloy tube far exceeds the requirement of around 3Bar.

An interim set of specifications suggested a volume of 17.3cc propellant. Use of the SNAP Ti tube would require 295mm total length. However the maximum internal dimension, dictated by the electronics trays and the butane triangular prism tank is ~167mm. The preferred solution is to split the propellant tank into three straight sections each with ~100mm length for propellant. Splitting the propellant into 3 tanks enables each tank to supply 3 thrusters and considerably reduces length and complexity of propellant feed tubing, as well as the concomitant pressure drop.

A propellant tank pressure of 0.3MPa, on a plunger (sliding bulkhead) of 8.55 diameter (slide fit inside of Ti tube), requires a spring force of 17.3N. If a spring has a compressed length of ~50mm, and an extended length of 150mm, ignoring plunger thickness this gives a tank length of ~150mm. Careful examination of the SSTL solid model for SNAP-2 shows that the cylinder length which can easily be accommodated and mounted on existing electronics trays, accounting for end caps is 134mm. 3 such tanks each with an internal spring compressed to 50mm length, but ignoring piston length gives a propellant volume of 14.7cc, or a mass of 17g at a density of 1150kg/m^3.

Each tank has an estimated mass, excluding spring and piston, of 35g. A single tank clamped using insulating spacers to the side of one of the 3 electronics box stacks in SNAP-2 is shown in figure 7 below:

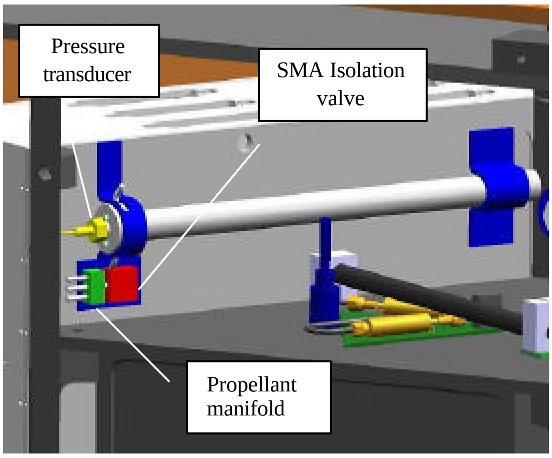

Figure7 : Solid model of the Propellant tank

Feed system

Propellant feed and control has been assumed using ultra fine, 1.6mm, stainless steel tube and small solenoid valves available from the Lee company (EPSV Extended Performance Solenoid Valve, model # INKX0514300A). Each valve has a mass of 6g including fittings and draws ~0.75W average power. To minimise the risk of propellant leakage twin series redundant valves will be connected to each thrust head hence a total of 18 valves (and flight spares will be required). Valves are currently rated to 0.8MPa. The required propellant feed pressure of ~0.3MPa is well within this specification. A single additional isolation valve based on a MEMS valve using a Shape Memory Actuator (SMA), per propellant tank can be optionally fitted [9].

Pipe runs between individual thrust heads (3 per propellant tank) and propellant tanks have not yet been designed, although this is not expected to be problematic, careful thought will have to be given to electrical isolation relative to the SNAP-2 structure. A simple and adequate solution involves insulating each pipe with an electrical insulating coating, increasing OD from 1.6 to ~6mm.

Thruster Head Mounting

The current assembly combines twin Lee solenoid valves held in a screwed Al alloy clamp arrangement, mounted on the side faces of the electronics trays (thrusters 1-6) and the +Z solar panel (thrusters 7-9) and isolated using a Teflon based standoff. The stainless steel propellant feed tube will be brazed directly onto the valve inlets/outlets. The thruster head is cylindrical with a diameter of ~ 2.5 cm and a thickness of ~ 8mm, and is shown schematically in the mounting arrangement figures. The detailed engineering design of the thrust head mounting has not been completed at the time of writing, and the two possible schematics are shown for illustrative purpose only.

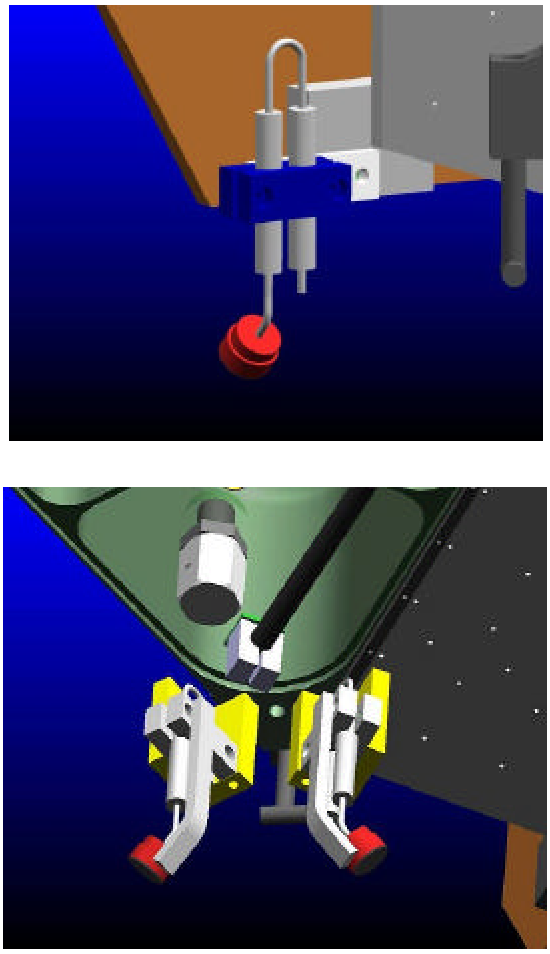

Figure 8: Two solid model views of possible thruster head mounting

The complete system mass and power breakdown is tabulated in table 1.

Note that the figures given are upper limits, and based in part on the extrapolation of COTS data or limited mission analyses. For example removing the serially redundant solenoid valves, employing only 1 neutraliser and reducing the thrust head mounting bracket size would save ~100g, and replacing the COTS power supply with a custom high voltage unit could save ~1.25W. Design iteration is expected to reduce both mass and power values further. The orbit average power is also expected to be only ~0.1W.

All components in direct contact with the propellant will reach a high voltage. Mechanical items, the power supply and monitors which interface with the spacecraft power supply and data bus must be isolated to ~10kV. Insulating will add mass and bulk and needs to be analysed in further detail for the effects on the system

Table 1: Power and mass estimates

Subsystem	Number	Mass (g)	Power (W)
Thrust head assembly inc. valves	9	217.8	Up to 4.5 *(3 x 2valves simultaneously.)*
Propellant	N/A	17	0
Neutraliser	Up to 3	Up to 45.9	Up to 0.75
Propellant tanks & mounts	3	105.0	0
High Voltage supply	1	56.6	2.5
Interface electronics	1	21.0	0.6
Power conditioning	1	20.0	2.5
Fluid feed pipes, misc. attachments	N/A	~20	0
Electrical isolation	N/A	~20	0
Estimated TOTAL (upper limit)		**523.3**	**10.85**

Performance

Up to 17g of an ionic liquid propellant (similar to water but with a density of 1150kg/m3) can be stored in the three cylindrical tanks described above. At the predicted Isp of 500s, for a SNAP-2 platform mass of 10kg, a DeltaV of 8.34m/s is achievable. Although the triple tank design can store considerably more propellant than is required for the required experiment, the design allows the flexibility to either reduce propellant filling or to include additional manoeuvres.

Further system benefits include:

- Small envelope of system components, adding multiple thrusters with minimal impact on overall SNAP-2 platform envelope.
- 6DOF control with rapid response.
- No 5-10 minute warm-up time as required by the SNAP-1 butane thrusters.
- Low power orbit average power, with peak loads able to be met by SNAP-2 battery.
- Low thrust (μN level) enabling high fidelity formation flying.

MISSION ANALYSIS

The use of an experimental low thrust capability on a nano-satellite co-flying with a larger micro-satellite provides the opportunity to investigate a variety of formation flying/station keeping options not available at higher thrust. We note that in adopting a micro-fabricated electric propulsion system for a nano satellite mission it is to be expected that the greatest demand on system resources will be from the power requirements rather than the mass associated with the propulsion system.

The principal issue to address for this mission is the control of the differential acceleration of the nano-satellite relative to the micro-satellite. We have based our calculations on geometric and mass data in which the micro-satellite has an area of 600 x 800 mm, and mass 100 kg. The reference projected area for the atmospheric drag on the nano-satellite is the SNAP-1 value of 465mm x 210 mm; the reference mass for the nano-satellite is 10 kg. We have assumed launched will be near solar minimum to make it compatible with our aspirations of a demonstration in the 2006 time frame; calculations have been performed, using Jacchia's reference atmosphere[10]. In this model density predictions are sensitive to the assumed level of solar activity through a value assumed for the exospheric temperature. 'Low' and 'medium' temperatures have been adopted yielding a density at 700km of 1.5×10^{-14} and 3.1×10^{-14} kg/m^3 respectively. The ballistic coefficient adopted in our model calculations for the satellite assumes a value for drag coefficient of 2.2.

As a first step it is necessary to identify the level of drag force expected on the satellite. Evidently the minimum value of thrust that could be used to control the satellite against drag perturbation is simply the drag force experienced. In circular orbit for the environmental conditions under consideration the maximum for this drag force on the micro-satellite is approximately 1 μN, and that on the nano satellite is less than 0.3μN. Thus if the thrust available from the colloid system is greater than these values it will always be possible to match the orbit decay rates of the 2 satellites by thrusting appropriately with the nano-satellite, albeit with the thruster operating throughout the orbit. The altitude of a satellite in a circular orbit is also subject to short-term periodic fluctuations arising from the Earth gravitational harmonics. The dominant harmonic in low Earth orbit is that of J_2, and at an altitude of 700km the consequent variation in altitude is of order 5 km; at this level of altitude variation there is negligible effect on the satellite drag experienced. In the analysis we have performed, we have therefore neglected the effects of Earth harmonic terms on orbit properties.

Clearly the minimum thrust to overcome drag make-up is not necessarily the optimum thrust, even in a system which is power limited. Thus an obvious feature of operating at minimum thrust would be the requirement for continuous operation of the thruster, most probably an undesirable feature operationally. We note one attractive feature of a colloid thruster is that the thrust level available may be varied in a number of ways including by variation of the mass flow rate, or by the applied acceleration voltage. The adoption of throttlable thrusters provides the opportunity to realize a thrust solution that is closer to an optimum solution and gives the added flexibility of examining different solutions and approaches to formation flying in space. In fact even greater variation in thrust level from a micro-fabricated colloid propulsion system is available. Direct throttling by voltage or flow rate provides only a limited range of thrust variation, more substantial variation can however be achieved at discrete thrust levels by controlling the number of the emitters used at any one time in the colloid emitter array; this approach is the one we have adopted in other missions, and is described in more detail in reference [9].

In order to assess analytically the potential for optimisation we have investigated a simplified concept, wherein it is assumed the thrust is applied purely tangentially to the orbit arc. The analysis we have undertaken is based on a linearised approach to the equations of motion as outlined by Cornellisse [11]. In this formulation it is possible to write, for the case of a low magnitude tangential thrust *T*, the rate at which an initially circular orbit is raised through a height *h* by:

$$h = 2\times\frac{m}{r_0^2}\times\frac{T}{m}\times\left(\frac{V_{c0}\times t}{r_0} - \sin\left(\frac{V_{c0}}{r_0}\times t\right)\right)$$

where *m* is the gravitational parameter for the Earth, r_0 is the initial circular orbit radius, with an associated satellite velocity of V_{c0}, *T* is the thrust from the colloid system, *m* is the satellite mass and *t* is the time since start of thrust application.

The initial flight angle, the angle the velocity vector makes with the radius vector will for a circular orbit be $\pi/2$. Following application of the tangential thrust at some later time *t* this angle will have changed to an angle of $\gamma+\pi/2$ where the angle γ, given by:

$$g = 2\times\frac{m}{r_0^2}\times\frac{T}{m}\times\left(1-\cos\left(\frac{V_{c0}}{r_0}\times t\right)\right)$$

Using these equations, representative data is presented for three levels of thrust in Figure 9. In these the increase in altitude during a period of 2000seconds is plotted. The initial orbit is at 700km altitude. The three thrust levels shown are 100μN, 333 μN and 1000μN. We need now to consider how this approach can be used to identify a control strategy for the satellite due to aerodynamic drag.

Using the data for satellite properties as identified, at an altitude of 700km, the reduction in altitude due to aerodynamic drag is approximately 10cm per orbit at the assumed value for a low level of solar activity. As an example of how optimisation may be achieved we have taken the case where we have assumed that one orbit burn is provided on a daily basis to provide drag make-up. Since we need to minimize the amount of propellant, for a given specific impulse *Isp*, we have calculated how long a thrust arc is required, from the data presented in figure 9 to regain the altitude lost due to drag. These results are given in Table 2. In this table, the second column identifies the daily reduction in altitude in metres for an orbit, uncorrected for atmospheric drag. The thrust application time required for the three levels of thrust, at two different altitudes and two solar activity conditions are given in the other columns both for the case of drag make-up for the nano-satellite alone, and for the case where the nano-satellite is required to match the decay rate for an uncontrolled micro satellite.

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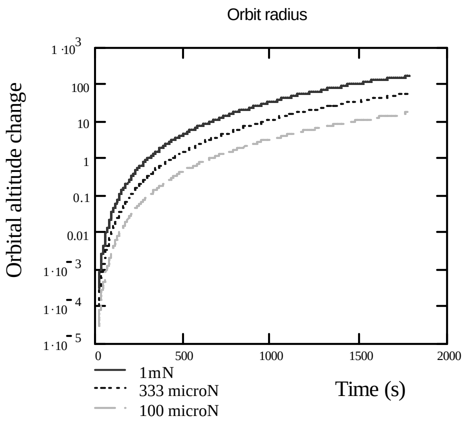

Figure 9: The Micro satellite change in altitude plotted for thrust levels of 1mN, 333 mN and 100mN.

Table 2: Orbit Maintenance requirements.

	Decay Rate (m/day)	Nano Sat Maintenance			Nano Sat Match Micro Maintenance		
Altitude (km)	Solar Activity	Burn time (s) per day			Burn time (s) per day		
	Low	100µN	333µN	1mN	100µN	333µN	1mN
700	1.5	755	503	347	598	399	276
660	2.5	898	596	412	710	473	327
	Medium						
700	3.0	968	642	444	765	509	352
660	4.9	1140	753	520	900	597	413

Assuming that the specific impulse of the colloid thruster is independent of the level of thrust, the amount of propellant consumption will be proportional to the product of thrust duration and thrust level. This can then be used to evaluate the relative mass of propellant for the various cases examined. This data is presented in Table 3, for the case of drag make-up for the nano-satellite. The propellant mass shown in columns 3 to 5 are normalized to that for the 100µN thrust, low solar activity and 700km altitude.

As anticipated lower propellant usage will occur for higher altitude and lower solar activity. However the results presented in this table demonstrate the added interesting feature that at the lowest level of thrust examined, *less* propellant will be required, relative to that required for the highest level of thrust. This identifies that there is indeed an opportunity to optimise the thrust arc length for given mission constraints. It is clear that there is not linear relationship between thrust arc length and thrust level, implied by the sinusoidal term in eqation.1.

Table 3: Relative Propellant Mass Requirements

Altitude (km)	Decay Rate (m/day) / Solar Activity	Nano-satellite orbit control relative mass of propellant		
	Low	100µN	333µN	1mN
700	1.5	1.00	2.20	4.60
660	2.5	1.19	2.61	5.46
	Medium			
700	3.0	1.28	2.81	5.88
660	4.9	1.51	3.29	6.89

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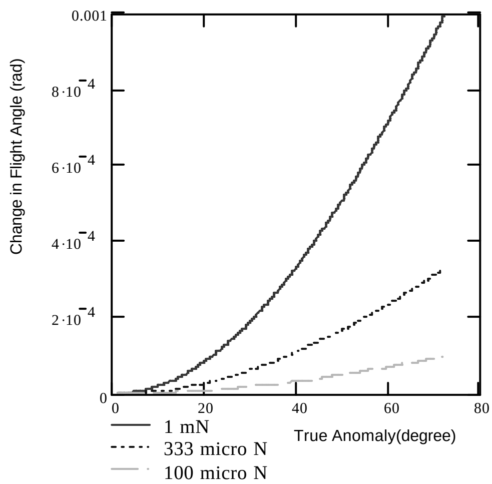

Figure 10: Change in Flight Angle, radians, as a function of True Anomaly

The reason behind these observations may also be understood by considering the way in which the flight angle varies whilst thrust is applied. The results for this calculation are presented in figure 10, again for the three levels of thrust used previously.

From these it is apparent that as the thrust is increased, the rate at which the flight angle varies also increases. That is to say the rate of increase in the orbit's eccentricity is also related to the thrust level. Since our assumption is that the orbit is required to be maintained in a near circular orbit, this increase in flight angle is undesirable.

We emphasize that this analysis is simplified and specific, in that the calculations performed have assumed that the drag make-up manoeuvre is made once per day, and the operational requirement is simply that the thrust level and duration is used to achieve an increase in altitude, that is just sufficient to compensate for the reduction in altitude.

Our baseline configuration investigated, provides for 15g of propellant. With this we are able to provide station-keeping control for the periods identified in Table 4.

Table 4. Days of orbit maintenance available with 15g of fuel

	Nano Sat Maintenance (day)			Nano Sat matches Micro Sat Maintenance (day)		
Altitude	100µN	333µN	1mN	100µN	333µN	1mN
700 (low)	591	266	129	747	336	162
660 (medium)	497	225	108	629	283	137
700 (high)	461	209	101	584	263	127
660 (medium)	392	178	86	496	224	108

A variable thrust system would be able to undertake different control strategies, which are not available to propulsion such systems having a fixed thrust. For example providing thrust at 30 µN, long duration thrust arcs can be achieved, providing more closely simulation of operational conditions for missions such as LISA, which require proportional control for disturbance compensation.

A simple fixed thrust system of multiple thrust heads addresses a different the set of experiments dealing with precision formation flying which are also relevant to LISA and a host of nano satellite constellation proposals. We can explore the thrust efficiency of low thrust level formation flying and further work can identify what range of thrust level is most appropriate for particular missions.

REFERENCES

1. Stark J. et al, Colloid Propulsion – A Re-evaluation with an integrated design. AIAA-2003-4851. 39th Joint Propulsion Conference, Jul 2003.

2. Stark J., Stevens R., Alexander M., AIAA-2003-4852. Fabrication and Operation of Micro Fabricated Colloid Thruster Arrays. 39th Joint Propulsion Conference, Jul 2003.

3. Stark J., Smith K, Robertson S., High Accuracy Measurements in Electrospray Relevant to Colloid Thrusters. AIAA-2003-4847. 39th Joint Propulsion Conference, Jul 2003

4. Hruby V., Camero-Castano M., Falkos P.,Shenoy S, IEPC-01-281 (2001)

5. Velasquez L.F., Martinez-Sachez M., A micro fabricated colloid thrusters array. AIAA-2002-3810, 38th Joint Propulsion Conference, 2002

6. Da Silva Curiel, A. Davies, P. Baker, A. Underwood, C. Towards a Spacecraft on a Chip; 4th Round Table on Micro/Nano Technologies for Space ESTEC Conference Centre, Noordwijk, The Netherlands May, 2003.

7. Kent B. et al, A field effect neutraliser for fundamental physics missions, 4th International Spacecraft Propulsion Conference, Chia Laguna, Italy June 2004

8. Stark et al To be presented at the AIAA 40th Joint Propulsion Conference, Fort Lauderdale, Florida, July 2004

9. Edeson R., Whalley M., Kent B., Canfer S., Sawyer E.. An SMA Gas Release Mechanism for the Rosetta Lander's Ptolemy Instrument. To be presented at the 55th IAC. Vancouver, Canada, October 2004.

10. Jacchia LG, Revised static models of the thermosphere and exosphere with empirical temperature profiles, Special Report 332, Smithsonian Astropysics Observatory, Cambridge Mass., 1971

11. Cornelisse JW HFR Schoyer & KF Wakker, Rocket Propulsion and Spacecraft Dynamics, pb Pitman, 1979