A SMALL ION-PROPELLED SPACECRAFT FOR NEAR EARTH EXPERIMENTS

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K A Ryden and N C Wallace, Space Department,
Royal Aerospace Establishment, Farnborough, Hants, UK

ABSTRACT

The suitability of ion propulsion for small spacecraft is studied, with particular emphasis on low altitudes where air drag can be a serious problem. It is shown that the use of the UK-10 ion propulsion system can enhance the value of a small spacecraft as a technology demonstration vehicle, permitting the test of a range of devices under realistic conditions. Results of orbit analysis using a numerical program are given for two spacecraft configurations.

1.0 INTRODUCTION

The application of ion propulsion to small spacecraft is a concept which would enable a greatly extended range of interesting missions to be accomplished, within the constraints of the available masses and volumes. The large velocity increment manoeuvres then possible, with the use of small quantities of propellant, seem ideally suited to small spacecraft. Propellants can be completely chemically inert, representing a minimal added risk to other primary payloads during the integration and launch stages. Thrust levels are low, being at the milli-newton level, producing slow steady manoeuvres not requiring accurate pointing or precise timing, and can be applied whenever necessary as long as power is available. Spacecraft can be made more flexible, capable of achieving a wider range of orbits (and therefore applications), which were previously unattainable because of a lack of launch opportunities, suitable propulsion systems or budgets. This potential was first brought to the attention of many interested in small spacecraft by the JPL proposal of a 'Lunar GAS' mission[1].

Missions such as 'Lunar GAS' or those outlined in this paper could also provide conclusive evidence of the effectivness of ion propulsion and other innovative technologies during a long term in-orbit test. In addition, the use of ion propulsion to slowly raise the altitude of the spacecraft would enable the onboard technologies to experience a wide range of gradually changing radiation conditions. The manoeuvre could also be stopped temporarily at a specific altitude, while interesting operational experience or scientific data are aquired.

2.0 PROPULSION OF SMALL SPACECRAFT

Since cost and complexity are normally the driving factors in the design stage of many small spacecraft, the use of propulsion in general carries with it considerable disadvantages. This is especially true with electric propulsion since unfortunately it cannot be simply plugged into a small spacecraft as another sub-system. The relatively large power requirements are the immediate problem, necessitating the use of large solar arrays. An ion thruster system also requires a power conditioning and propellant storage, flow and monitoring system as well as some form of central control. This results in a relatively bulky system that can occupy a large fraction of a small spacecraft's volume.

The study reported in this paper is based upon the UK-10 ion thruster system currently being developed by a team led by the Royal Aerospace Establishment at Farnborough in the UK. This is a 10cm diameter Kaufman-type thruster, capable of producing 3-70mN of thrust, depending on the power available, and using xenon gas as a propellant (see Fig. 1).

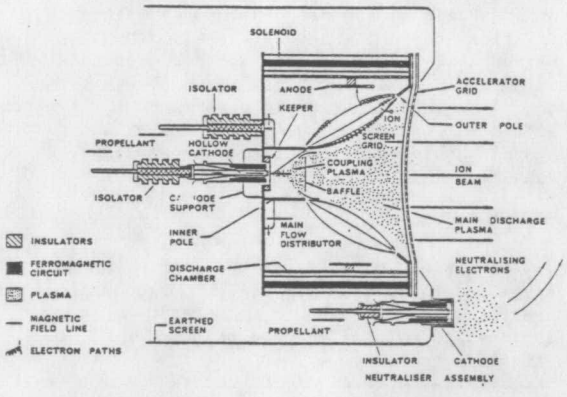

Fig. 1 Schematic of UK-10 Ion Thruster

The thruster consists fundamentally of a cylindrical discharge chamber in which a highly ionised xenon plasma is produced by a dc discharge between a cylindrical anode and an axial hollow cathode. This process is enhanced by the inclusion of a magnetic field generated by solenoids and applied between the inner and front poles inside the discharge chamber. Ions from this plasma are then accelerated and ejected out of the discharge chamber by a strong electric field, set up between two perforated and aligned grids forming the exit to the discharge chamber. The positive space charge of the emerging ion beam is then bulk neutralised by electrons emitted by a second external hollow cathode (see table 1). The thruster requires nine independent power supplies and a schematic of how they interface to the thruster can be seen in Fig. 2

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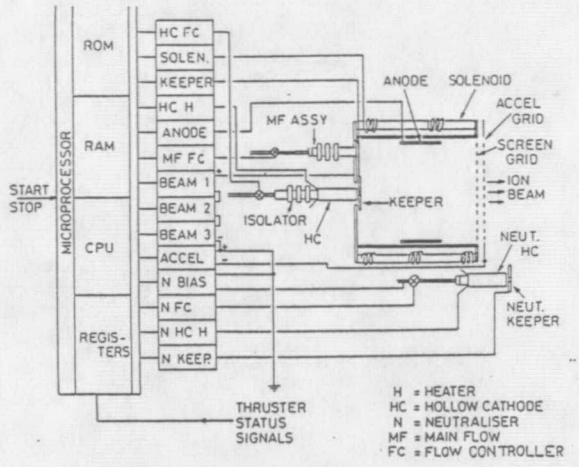

Fig. 2 Interface of Power Conditioning and Control System (PCCS) to Thruster.

Table 1 T4A THRUSTER PERFORMANCE DATA AT NOMINAL 11mN THRUST

PROPELLANT XENON GAS

Beam energy (V)	940
Thrust (mN)	11.4 (11.2)
Demonstrated thrust range (mN)	3.5 - 70
Exhaust velocity, singly charged ions (km/sec)	37
Specific impulse (sec)	3790 (3171)
Total input power (W)	275
Power/Thrust ratio (W/mN)	24.1 (24.6)
Total mass flow rate (mg/sec)	0.341 (0.36)
Electrical efficiency (%)	77
Mass utilisation efficiency(%)	89.8 (82.7)

Note: Figures in brackets have been corrected for the presence of doubly-charged ions, neutraliser power and flow rate, and keeper discharge power, as appropriate.

Table 2 ESTIMATED MASSES OF THRUSTER SYSTEM (kg)

	Flight test	Operational applications
Thruster	1.0	1.0
PCCS	8.0	3.9 - 5.1
Cabling	0.5	0.5
Tank (5Kg capacity)	1.4	0.5
Flow control	4.3	2.8
Totals	15.2	8.7 - 9.9

Propellant storage can be at any pressure, although high values would require expensive qualified tanks. In addition thermal control of propellant tanks may be required to prevent large variations in pressure. Storage at lower pressures would be more acceptable for safety reasons and could reduce costs significantly, since the use of non-space qualified tanks might be feasible. An example of this is the proposed flight of a RIT-10 thruster on EURECA[2], where a diver's air tank is to be used for xenon storage. Storage at approximately 60 Bar would enable the gas to be kept at its critical point, so reducing the required storage volume[3]. At this presure the density of xenon is approximately. 0.5gm/cc, half that of water.

3.0 ION PROPULSION IN NEAR EARTH APPLICATIONS

Designing a small spacecraft around an electric propulsion system can severely restrict the choice of launches. This results from the need for relatively large solar arrays to operate the thruster, which can lead to drag forces comparable to thrust levels in the initial stages of a low orbit mission. In many cases examined during this study, these drag forces could become dominant, resulting in the re-entry of the vehicle. The relationship between the minimum altitude at which a purely ion propelled spacecraft can just maintain its orbit is determined by a number of factors, which are outlined below.

A) Solar array and spacecraft body area:- this determines drag and power available; larger arrays provide more thrust but in turn cause more drag.

B) Eclipse time:- no power is available in shadow, hence no thrust. In an equatorial Earth orbit of 300Km altitude, the spacecraft can be in shadow for approximately 40% of the time.

C) Efficiency of solar cells:- this determines the power to area ratio.

D) Electrical efficiency of thruster:- this determines the power usefully employed in drag make-up.

E) Thrusting strategy:- the thrust vector may not be optimally directed for drag make-up at all times.

F) Solar cycle:- at a constant altitude, the residual air density can vary by an order of magnitude over the 11 year cycle (see Fig.3).

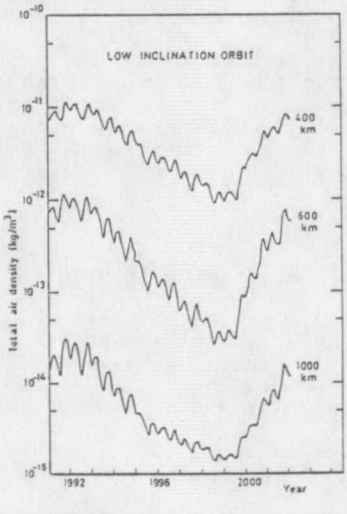

Fig. 3 Variation of Atmospheric Density with 11 Year Solar Cycle.

G) Orientation of orbit plane to sun line:- this is an additional factor to be taken into account when calculating the instantaneous area of the solar array normal to the velocity vector of the spacecraft.

An approximate value of the minimum altitude of a certain spacecraft configuration can be estimated as follows:

$$F_D < f.T \qquad (1)$$

where:

F_D = the average drag on the satellite

T = the thrust provided by each ion thruster

f = the 'thrusting factor' which accounts for the proportion of the orbit in which thrusting occurs ($0 \leq f \leq 1$).

f also accounts for any inefficiency in thrusting due to out of plane or non-tangential components.

Now consider a sun-pointing spacecraft of the general type shown below (see Fig.4).

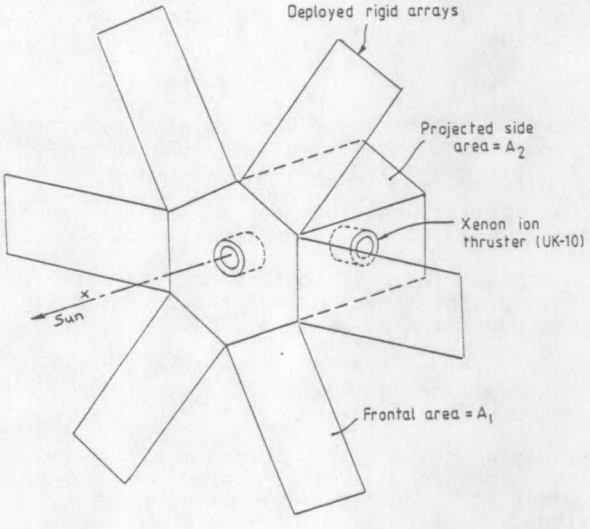

Fig. 4 Proposed Spacecraft Configuration.

Drag on the vehicle around the orbit will vary due to the changes in frontal area (A_p) projected in the direction of motion, if the x-axis is fixed in inertial space. Consider the case where the x-axis is within the orbital plane (see Fig. 5).

Then:

$$A_p = A_1 |\cos\theta| + A_2 |\sin\theta| \qquad (2)$$

where: A_p = projected area
A_1 = area of sun-pointing side which is covered in solar cells
A_2 = side area (see Fig.4).

The mean value of A_p is:

$$A_p' = 2(A_1 + A_2)/\pi \qquad (3)$$

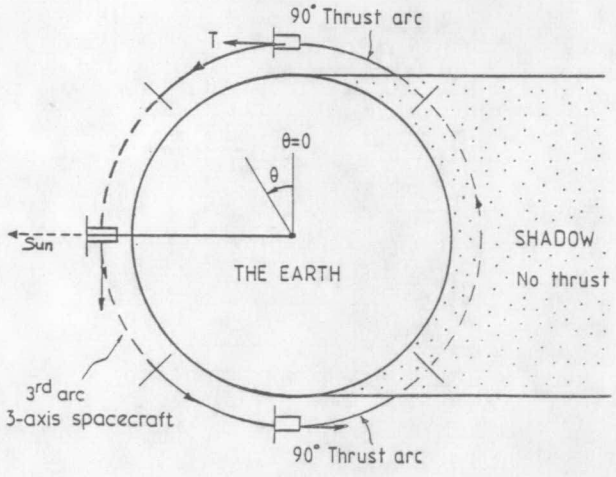

Fig. 5 Ecliptic Plane Orbit

Now the mean drag is given by:

$$F_D = \rho(r).C_D v^2 A_p' /2 \qquad (4)$$

where: r = the orbital radius
C_D = the drag coefficient
v = the orbital velocity
$\rho(r)$ = the air density at orbital radius r.

For a circular orbit we have:

$$v^2 = G.M/r \qquad (5)$$

where: G = universal gravitational constant
M = mass of the Earth

and so the minimum radius of the orbit (r_{min}) for successful orbit-raising is found from substituting Eqs. (4) and (5) into Eq. (3). Thus:

$$f.T = \frac{\rho(r).C_D.GM.(A_1 + A_2)}{r_{min} \cdot \pi} \qquad (6)$$

Now the area A_1 depends upon the thrust T:

$$T = W.A_1.\alpha.\eta \qquad (7)$$

where: α = the thrust to power coefficient for the ion thruster

$\eta = \eta_a.\eta_p.\eta_s$ where:

η_a = the solar cell efficiency

η_p = the packing efficiency of solar cells onto the surface A.

η_s = the power processing efficiency from solar array output to ion thruster PCCS input.

W = the solar flux constant at 1 A.U.

Re-arranging Eq.(6) and substituting Eq. (7) gives:

$$\frac{\rho(r_{min})}{r_{min}} = \frac{W.f.T.\alpha.\eta.\pi}{C_D.G.M.(T+T')} \qquad (8)$$

where:

$$T' = A_2.W.\alpha.\eta$$

The right hand side is a function of orbital radius and $\rho(r)$ may be expressed in an approximate analytical form:

$$\rho(r) = 4.165\times10^{-8} \exp\left[-(0.023h' - 5\times10^{-6} h'^{7/4})\right] \text{ Kg/m}^3$$
for $h' < 300$km

$$\rho(r) = 1.31\times10^{-7} \exp\left[-(0.0115h' - 5\times10^{-6} h'^{7/4})\right] \text{ Kg/m}^3$$
for $h' > 300$km

where: R = radius of the Earth
h' = $r-R$

These values of $\rho(r)$ correspond well with those recorded for the 1954 sunspot maximum[4]. Eq. (8) can be solved most easily on a computer. To do this, the following parameter values were assumed:

α = 3 x 10 N W (UK-10)
W = 1.3 K W m
C_D = 2.6
η_o = 0.14 (silicon) η_p=0.9 η_s=0.9
f = 0.3 for two arc thrusting (eclipse reduces 90 degree thrusting arcs)
= 0.15 for single arc thrusting
A_2 = 0.5 m

For simplicity, an orbit in the ecliptic plane was considered; other orbits such as polar must be considered individually and the equations adjusted accordingly. The graph below (see Fig. 6) shows the resulting minimum launch altitude against thrust for one and two arc thrusting strategies ie. one or two ion thrusters. Clearly, as the thrust increases so that $T >> T'$ (ie $A_1 >> A_2$) the minimum altitude becomes independent of thrust.

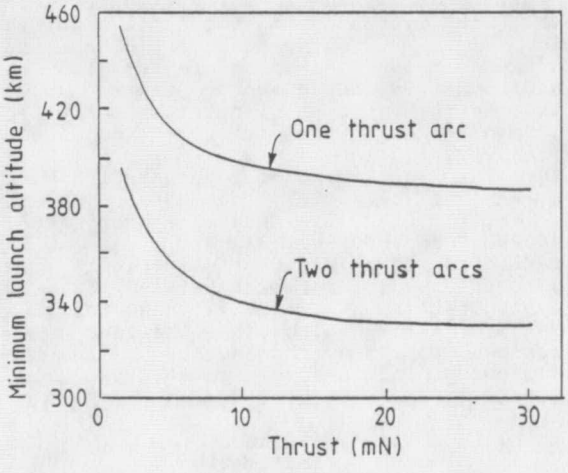

Fig. 6 Minimum Altitude Comparison between 1 and 2 Arc Thrusting.

This analysis indicates that most currently available expendable launchers would be suitable for a purely ion propelled spacecarft. The drag problem could become serious, however, with low altitude Shuttle missions, or low altitude expendables at solar maximum.

4.0 SPACECRAFT DESIGN APPROACH

The development of an initial design followed four basic criteria; these were

1) Flexibility of launch requirements and potential applications
2) Simplicity of design
3) Safety
4) Cost and availability

Since low cost launch opportunities are difficult to find, it was decided to make flexibilty of launch a driving factor, to enable the spacecraft to be launched at short notice, whenever an opportunity became available. From the analysis shown in the previous section, it is clear that a purely ion propelled spacecraft would not always satisfy this prerequisite and so provision should be made for a small, scaleable chemical propulsion system for the initial low altitude stages. This would be used to increase the altitude soon after launch, but it would only be adopted if made necessary by the lack of suitable launches.

4.1 Chemical Propulsion System

An investigation of the potential systems showed that the most favourable approach is to use a small hydrazine system. These appear to be the most developed and reliable liquid systems available. The thrusters proposed are small 400N multiple burn devices requiring only three valves each. The entire system can also be easily scaled by the addition of more propellant.

A simple analysis of the order of propellant masses required for certain missions is possible using the rocket equation, each manoeuvre being characterised by the velocity increment ΔV necessary to achieve it.

$$\Delta V = V_e \ln\left[M_o/M\right]$$

where: V_e = exhaust velocity of propellant
M_o = initial spacecraft mass
M = final spacecraft mass

It can be shown that to raise a 150 kg spacecraft from a standard 290 km Shuttle orbit to a 500 km orbit where air drag is negligible, would require approximately 6kg of hydrazine propellant.

Solid rockets were also considered for this application but can only be obtained in set sizes, special designs being prohibitively expensive. They are single shot devices and the extremely short burn times (of the order of a second for a 10 kg booster) result in high acceleration rates which can have a serious impact on the design of the spacecraft structure and AOCS. The simplicity of many solids was very attractive though, with often two wires forming the only electrical interface, so their use in future missions should not be ruled out.

4.2 Power Generation and Electric Propulsion System

The thrust delivered by the UK-10 ion propulsion system is in the range 3 to 70 mN and depends entirely on the power available. In turn, the power depends on the solar array size and design, which is governed by the volume and shape of the envelope in the launch vehicle. The design of the solar array also has significant impact on the volume of the spacecraft available for payload. As an example, an extended Shuttle GAS cannister, being rather small, makes flexible wrap-around arrays, as proposed by JPL[1] essential for high power levels. Rigid panels would occupy too much volume in an inefficient way. If however lower power levels are acceptable, then rigid panels become feasible. Small rigid panels have a number of advantages for the small spacecraft builder, since they are easier to construct and have simpler deployment mechanisms. Also, the packing density is greater, they are more robust and if one of a number of small panels fails to deploy or stops operating, there is still sufficient power for the mission to continue, albeit at reduced levels.

These advantages led the design towards a spacecraft configuration in which a number of small deployable arrays are maintained pointing towards the sun, so maximising power output. The power level would ideally be about 400 Watts: 350 watts for the ion propulsion system and 50 watts for spacecraft housekeeping. This power level would require about 2 square metres of array and would allow the thrusters to produce a minimum of 10 mN of thrust after 2 years operation. Propellant quantities would, as with the chemical system, be scaled to suit.

Silicon arrays have been assumed for much of the mission analysis, but it should be noted that the use of Gallium Arsenide (GaAs) solar cells would have considerable advantages, especially if they could be exposed to high radiation doses as a technology demonstration. They have an improved efficiency of 18% as compared to 14% for silicon, but they are considerably more expensive and at present more fragile.

4.3 Spacecraft Configuration

The basic configuration of the spacecraft depends largely on the array design and the thrusting strategy employed by the electric propulsion. For simplicity, a spinner is appropriate but this limits the thrusting strategy to the use of 2 ion thrusters (see Fig. 4). The thrusters are placed on the spin axis of the spacecraft, one at each end. By adopting a 3-axis stabilised spacecraft, 3 or 4 thrusters could be used, enabling easy plane changing and relatively rapid orbit raising, although the basic spacecraft design is the same in both cases. The spinner design might be used in initial missions, with the 3-axis system possibly being adopted when the basic design has been proven.

5.0 MISSION ANALYSIS

Orbit analysis was carried out with a numerical computer program[4,5] which takes into account the following factors.

1) Thrust vector:- components of thrust along and normal to the velocity vector of the spacecraft.
2) Air drag
3) Shadowing by Earth eclipse:- in shadow no thrust is applied.
4) Time of year:- relative inclination of orbital plane to sun line.
5) Orbit inclination
6) Orbital eccentricity
7) Perturbations due to Earth asperity

In all cases the effects of the components of thrust normal to the orbital plane are ignored and the thrust level set at 10mN. Two general configurations were examined and are:

Configuration 1:
Spinner with 2 ion thrusters. In this configuration the thrust vector is always applied along the spacecraft sun-line, and two thrusting strategies are examined.

1) Thrust is applied in 2 x 90 degree sectors centred about the line perpendicular to the Earth-Sun line, (see Fig. 5)
2) Thrust is applied at all times power is available.

Configuration 2:
3-axis stabilised with 3 ion thrusters. In this configuration the thrust is applied in 2 x 90 degree sectors along the spacecraft-sun line as before but with an additional 90 degree sector centred about the Earth-Sun line with thrust directed perpendicular to this line, as indicated, (see Fig. 5).
The main altitudes of interest in many areas of technology demonstration and scientific study lie between approximately 2000km and 6000km, where the major radiation doses would be experienced. A comparison was first made between the two configurations to examine the relative rates of altitude increase. The launch was assumed to be by the Shuttle, with the following initial orbit parameters being used.

Initial Orbit:-		
	Altitude	350km
	Inclination	28.5°
	Year angle of sun	90°
	R.A ascending node	180°
	Initial eccentricity	0
	Solar cycle	Max.

In the graph shown overleaf (see Fig. 7) altitude is plotted against time for the two configurations (2 or 3 arc thrusting). Transfer times to geostationary altitude are given as an indication of the time spans involved with these levels of thrust.

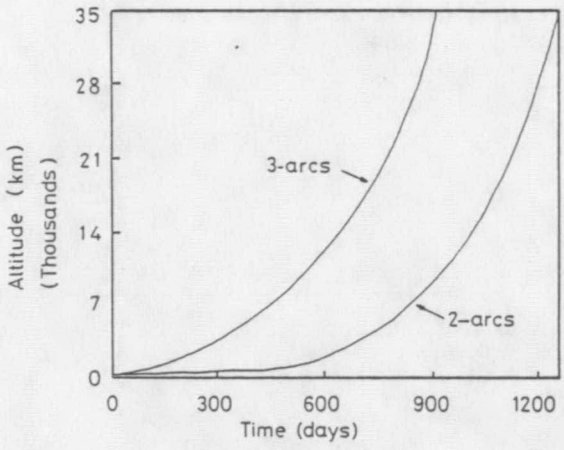

Fig. 7 Altitude vs. Time Comparison between configuration 1 and 2 Spacecraft.

Propellant quantities and transfer times between 2000km and 6000km are given in Table 3.

Table 3. Propellant and Time Required to 2000km and 6000km

	Config.1	Config.2
Time taken to 2000 km (days)	360	240
Propellant to 2000 km (kg)	4	4
Time taken to 6000 km (days)	800	540
Propellant to 6000 km (kg)	5.8	5.8

These figures suggest that for low Earth applications, such as the mapping of the radiation belts, the use of 3 ion thrusters and hence the adoption of 3-axis stabilisation, although faster, is not really warranted. The use of 3 thrusters does reduce the initial minimum altitude of the configuration, however.

The second case studied was for a configuration 1 spacecraft in which thrust is applied at all times power is available. As before, the launch was assumed to be by the Shuttle and the same initial orbital parameters were used as in the previous case.The results of this example can be seen below (see Fig. 8), and in Table 4.

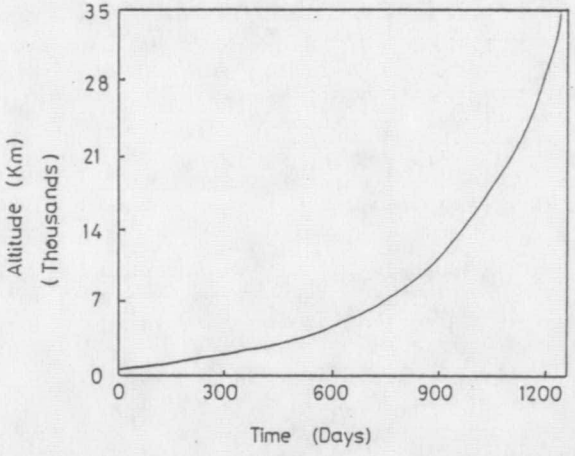

Fig. 8 Altitude vs.Time for Configuration 1 (2-arc thrusting), thrusting at all times power is available.

Table 4. Propellant and Time to 2000km and 6000km for configuration 1 spacecraft thrusting at all times power is available.

Time taken to 2000 km (days)	300
Propellant to 2000 km (kg)	6
Time taken to 6000 km (days)	660
Propellant to 6000 km (kg)	14

Clearly this is not as efficient a strategy for achieving the same orbit raising manoeuvre as just thrusting in the 2 x 90 degree sectors, but it is a quicker way of reaching any altitude with a configuration 1 spacecraft, and it also provides a more rigorous test for the ion propulsion system.

6.0 CONCLUSION

Ion propulsion can provide considerable benefits to small spacecraft, primarily because of the small mass of propellant required to perform significant orbit changes. However, its use represents a not insignificant increase in complexity, particularly in the power generation and conditioning system. Furthermore, the need to consider thrust vectors and some aspect of sun-pointing to ensure illumination of solar arrays means that ion propulsion must be designed-in from the start of the project. The technology required for the kind of applications considered in this paper is at a mature stage of development, and currently available. Nevertheless, the flight of ion thruster systems on small spacecraft would be a valuable technology demonstration in itself.

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