A STUDY INTO THE TECHNIQUES FOR MINIATURISED ELECTRIC PROPULSION SYSTEMS, AND MISSION CATEGORIES, FOR SMALL SPACECRAFT.

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INTRODUCTION

Since the beginning of the space age, mainstream scientific and commercial spacecraft have tended to become larger, heavier and more costly as mission requirements have escalated. There has been a parallel trend in the opposite direction, towards the development and operation of smaller spacecraft for a variety of purposes. Although clearly not as capable as the large and heavy spacecraft, these smaller vehicles have provided valuable service in many fields, for relatively modest funding levels.

Up to the present, most small spacecraft have avoided the complexity of on-board propulsion systems, and have remained in the orbits into which they were initially deployed. This has limited their operational flexibility, and has restricted the range of tasks for which they can be used. Similarly, suitable launch opportunities have been difficult to find, since the "piggy-back" principle has often been adopted for economy. The range of applications for which small spacecraft might be suitable can be expanded if adequate propulsion systems can be made available. Conventional chemical systems provide a relatively low specific impulse (SI), with a maximum of about 320 s for storable bi-propellants. As a consequence, a large fraction of the spacecraft launch mass will be devoted to propellant. With electric propulsion, a much higher value of SI is available, with a current maximum exceeding 6000 s, so that the overall propellant mass is reduced substantially, with an improvement by a factor of 10 to 20 being feasible. In this context, a small spacecraft was defined, as a vehicle having a launch mass of less than 200 kg. Various new technologies make this possible, and this paper concentrates on those technologies.

The review of possible missions with which the study commenced showed that they are relevant to a very large number of significant space applications. Advantages include reduced cost and improved flexibility, amongst other factors. Three missions were selected for in-depth consideration. They were:

a. An interplanetary mission to an asteroid requiring a very high velocity increment, ΔV. This might be greater than 10 km/s if an initial orbit-raising phase to Earth escape velocity is needed.
b. A commercial Earth orbital mission involving orbit emplacement at the operational altitude, acquisition of the correct orbital plane, and orbit control during the mission. This application would be most effective if applied to the deployment and control of a constellation of small satellites.
c. A precise station-keeping mission in Earth orbit, involving a moderate cost synthetic aperture (SAR) low altitude spacecraft, with accurate drag compensation by electric thrusters to minimise the propellant needed. This concept is particularly attractive for oceanographic applications.

It was found in each case that the spacecraft power system dominated the design, since electric thrusters require significant levels of power for their operation.

SUMMARY OF SELECTED MISSIONS

The analysis of the interplanetary[1] mission commenced with an examination of a wide range of possible asteroid targets. However, it was concluded that a decision here should be left to the scientific community, so a total value of ΔV of 12 km/s was adopted, which will permit a very wide variety of asteroids to be examined; indeed, this is sufficient for several dual and triple target missions to be contemplated. It was found that this velocity increment would be feasible using two 10 cm beam diameter ion thrusters operating at 25 mN thrust and an SI of 5000 s.

The station-keeping SAR mission was dominated by the mass of the payload, assumed to be optimised for oceanographic applications. For the radar antenna, the mass of was particularly heavy at 50kg. It was decided that the operational altitude should be 300 km, as a compromise between sensor performance and propulsion requirements. The use of high thrust (10-20mN) (field emission EP) technology, combined very high SI with a smooth throttling capability. In establishing a baseline design, a typical space computer with a standard 1 Gbit memory will provide a data storage capability of the sensor over an ocean swath of almost 8000 km length.

Careful consideration of the commercial mission led to the conclusion that the analysis should focus on a multi-mission platform capable of attaining a wide variety of operational orbits after deployment in either LEO or GTO. It should also carry sufficient propellant for orbit maintenance for 10 years. It was shown that the optimum SI for this, is in the range of 1300 to 1800 s. Thus the Hall-effect thruster (HET) was selected for this task.

EP SYSTEMS

The Benefits of Using EP

As already mentioned, if a mission requiring a significant value of ΔV is undertaken using conventional chemical propulsion, the mass of propellant is substantial. The solution is to use EP, which provides a much improved performance in terms of the available SI. The SI can be increased by a factor of about 5 above the best storable chemical systems using HETs and by an order of magnitude or more with gridded ion thrusters and FEEP devices. A comparison of the performance of different systems is presented below in Table 1, in which the effective exhaust velocity of a thruster is denoted by v_{eff}.

Propulsion System	Typical v_{eff} (km/s)
Cold gas	0.3-0.65
Solid propellants	2.1 – 3.2
Liquid propellants	2.9 – 4.5
Storable bi-propellants	2.0-3.2
Exotic bi-propellants	4 – 6
Liquid hydrogen/liquid oxygen	< 4.7
Hydrazine monopropellant	1.7 – 2.9
Power augmented hydrazine	2.8 – 3.2

Propulsion System	Typical v_{eff} (km/s)
Electric propulsion	
Arcjets	6 – 10
Hall-effect thrusters	14 – 20
Gridded ion thrusters	22 – 60
Advanced gridded ion thrusters	> 60

Table 1. Typical values of effective exhaust velocity.

The advantage of using a high value of SI can be seen immediately by consideration of the rocket equation, which is derived from Newton's Laws of motion and can be expressed as

$$\Delta V = v_{eff} \log_e \left(M_o / M_f \right) = v_{eff} \log_e \frac{M_o}{M_o - \Delta M} \tag{1}$$

where M_o and M_f are the masses of the spacecraft at the beginning and end of the manoeuvre, and $\Delta M = M_o - M_f$ is the mass of propellant consumed.

The SI, denoted by I_{sp}, is related to v_{eff} by $I_{sp} = v_{eff}/g_o$, where g_o is acceleration due to gravity at sea level, so

$$\Delta V = I_{sp} g_o \log_e \frac{M_o}{M_o - \Delta M} \tag{2}$$

It is therefore clear that an increased value of I_{sp} enables ΔM to be reduced considerably for a given ΔV.

PROPULSION REQUIREMENTS

Primary Propulsion

Gridded Ion Thrusters: Interplanetary Mission

A cursory examination of Equ 2 then shows clearly that reasonable values of ΔM can be attained only if the SI is large; within limits, the higher the value achieved the better. This conclusion is illustrated by the data in Table 2, which assume an initial deployed spacecraft mass of 200 kg and the use of two T5 ion thrusters[3,4] operating at 25 mN, using xenon propellant.

SI (s)	ΔM (kg)	Thrusting time (hr)
3000	67.0	11,445
4000	52.7	11,491
5000	43.4	11,831
6000	36.9	12,059

Table 2 Propellant mass and thrusting time for a range of values of SI, using two T5 ion thrusters.

Again for reference purposes, typical thruster parameters are given in Table 3 at an SI of 3500 s. The estimated thruster lifetime[5] is in excess of 10,000 hours with molybdenum grids, so all values of SI listed are probably achievable employing this technology. However, an improvement by a factor of 3 to 5 is immediately available using carbon.

THRUSTER PARAMETER	VALUE
Thrust	25 mN
Ion beam current	457 mA
Beam accelerating potential	1100 V
Beam power	503 W
Ion velocity	40.2 km/s
Total mass flow rate	0.725 mg/s
Neutraliser mass flow rate	0.04 mg/s
Specific impulse	3515 s
Propellant utilisation efficiency	85.7%

THRUSTER PARAMETER	VALUE
Anode current	3 A
Solenoid power	< 3 W
Keeper discharge power	< 15 W
Maximum discharge power	138 W
Neutraliser discharge power	< 16 W
Maximum total power	657 W
Electrical efficiency	76.5%
Total efficiency	65.6%
Power to thrust ratio	26.3 W/mN

Table 3 Typical operating parameters of a T5 ion engine at 25 mN thrust and an SI of 3500 s.

To summarise the primary propulsion requirements, assuming that the total velocity increment needed is 12 km/s and that two T5 thrusters are used in parallel operating at an SI of 5000 s, the propellant mass needed is 43.4 kg, (47.7 kg 10% safety margin added).

HET: Commercial Mission

This mission also demands high SI, but time and power are limited. This causes the optimum SI to be 1500 to 2000. For GEO missions, orbit raising from GTO by use of EP will need a ΔV of 2200 m/s, orbit maintenance 50 m/s per year, and attitude control 100 m/s.

State of the art performance and trends for HET thrusters are summarized in Fig 1. Based on this extensive heritage and on current understanding of the fundamental physical processes, which drive Hall thruster operation, several scaling laws may be derived in order to size, optimize and predict the performance.

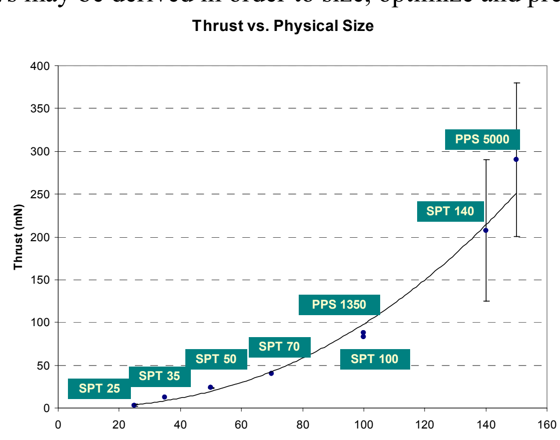

Figure 1 Thrust as a function of physical size for selected SPTs.

Miniaturised HETs require a stronger magnetic induction in the exit plane than conventionally sized thrusters. However, because it is easier to maintain a strong magnetic field across a magnetic gap of reduced dimensions, this condition does not imply dramatically increased magnet current needs but the difficulty then resides in avoiding saturation of the magnetic circuit. Total efficiencies as high as 60% that characterise larger-dimension HETs become difficult to achieve as the surface-to-volume ratio in the plasma discharge increases. As a conclusion, the requirements of the commercial missions envisioned in this study can be best met with a HET having a nominal power level of 700 W, a specific impulse of 1600 s and a thrust of 40 mN. These characteristics can easily be obtained with a thruster of 7 cm in diameter.

FEEP: SAR Station Keeping Mission
Although the SAR antenna to be flown on the oceanographic mission is relatively large, it is assumed that its orientation parallel to the direction of flight does not add appreciably to the atmospheric drag. For a cross section area of $2m^2$ and a drag coefficient of 2.6, the maximum thrust requirements for the EP system are in the range 10 to 15 mN. Assuming that the FEEP power input is 65 W per mN[6] results in a mean power requirement of 455 W and a peak of 975 W. Given a specific impulse of 6800 s, the propellant flow rate is approximately 0.1 mg/sec at 7 mN. The annual impulse for drag compensation will be of the order of 190,000 Ns and the propellant needed per year will be approximately 3 kg, or 15 kg over a mission duration of 5 years.

It should finally be mentioned that there is a theoretical possibility that the Cs propellant used in the main and any auxiliary FEEP thrusters may be oxidised by the residual oxygen encountered at an altitude of 300 km.

POWER TECHNOLOGIES

Secondary Propulsion

Attitude Control System (ACS)
Although the three missions are, by definition, very different there is a significant degree of commonality between their attitude determination and control requirements. All three spacecraft require 3-axis control throughout their respective missions, and they will all need appropriate sun sensors and perhaps GPS systems for attitude and orbit determination. Another common factor is the use of thrust vectoring to control attitude, or to de-saturate momentum wheels.

Attitude Sensors
For all three missions, sun sensors are necessary for attitude determination, with a suitable gyro pack. Earth sensors may also be required. Orbit determination whilst in Earth orbit can utilise standard spacecraft ranging through the transponder, the possibilities offered by GPS and, later, Galileo, must be considered.

Attitude control actuators
It is likely that the optimum method of attitude control actuation will involve a combination of thrust vectoring of the EP system and momentum wheels. The latter will be mainly exercised when the EP system is not operating, but will also be required continuously for control about the roll axis of the spacecraft. As well as using the main thrusters for real-time attitude control, they will also be suitable for momentum dumping about two axes after periods when no thrust is required. It is recommended that this function, be carried out by hollow cathode arcjet's (HCA'S), small FEEP devices, or miniature colloid thrusters.

Hollow Cathode Arcjet Thrusters
Initially these devices were intended for application to Kaufman gridded ion engines[3], in which a discharge cathode and a neutraliser cathode are employed, and recently found applications in high power HETs. These developments have culminated in a highly efficient, scaleable cathode design, which can be modified as appropriate to suit the performance and interface requirements of any new application. As an example of availability, QinetiQ currently produces several sizes of hollow cathode[2], two of which are pictured in Fig 2. The T6 design has been extensively tested to 35 A, and an earlier version was used successfully in the UK-25 300 mN ion thruster[7], where currents of 30 A were routinely employed.

Figure 2 Photographs of QinetiQ hollow cathodes.

An investigation of the feasibility of adapting hollow cathode (HC) technology[8] for use in medium SI (> 500 s) thrusters is under way at both QinetiQ and the University of Southampton[9]. Research performed at Aerospace Corporation using a T5 thruster cathode [10] has shown that the propellant gas flowing through the HC is highly ionised (60 to 100%). Axially directed high energy ions are also generated within the orifice, with energies well in excess of the voltages applied to the external electrodes[11,13]. As an example, 60 to 100 eV ions have been observed from cathodes operating with only 10 V on the anode[11].

Micro-Machined Colloid Thrusters

A new generation of colloid thrusters manufactured using micro-machining techniques offer promise in this application. They differ from standard colloid thrusters[14,15] in that they utilise well-defined and controlled single Taylor cones for the generation of thrust by each individual emitter. Apart from using a variety of propellants with different electrical conductivities to vary the operating characteristics, particularly the SI, a dual electrode droplet accelerating electrode system is employed, which permits much better control of the emitted beam than is normally the case. Although each emitter, which has a typical diameter of 50 *u*m, can produce a thrust which is limited to the sub-*u*N range, any number of such sources can be operated in parallel to produce a fully controlled aggregate thrust of interest for attitude and fine orbit control.

SOLAR CELL AND ARRAY TECHNOLOGIES

The solar array is a dominant feature of all three spacecraft. Its mass is the most critical parameter, but this is very sensitive to the technologies assumed for the solar cells as well as for the array itself.

High-Eta Silicon Cells

High-eta Si cells include several different types with initial efficiencies ranging from 15.5 to 17.3%, depending on whether the cell is optimised for beginning-of-life (BOL) efficiency, radiation resistance or cell mass. The main advantages of these cells over the other candidates are that they are lighter (approximately one third of the mass) and less than 50% of the cost.

GaAs/Ge Cells

These cells are available from commercial sources. Typical BOL efficiency is 19% with a superior radiation resistance over the silicon cells. The temperature coefficients are also better and therefore the loss of efficiency with increasing temperature is less.

Dual/Triple Junction Cells

Dual and triple junction cells are available commercially and are the current state-of-the-art solar devices, although the dual junction types, are already being superseded by the triple junction ones. Dual junction cells are typically 24 to 25% efficient and the triple variants reach 27 to 28%. These cells have a superior radiation resistance compared to GaAs/Ge technology. The performance of the TJdevice is forecast to be close to 30% in the next few years. These cells have the advantage of a high operating voltage which allows high voltage arrays to be built more easily. 2.2 V compared to 0.85 V and 0.45 V GaAs/Ge and Silicon.

Rigid Panel Arrays

Rigid panel arrays are the most common type in use on spacecraft. Carbon-fibre skinned aluminium panels are hinged together and deployed out from the side of the spacecraft. Although most applications use two wings, some employ one wing (eg on various remote sensing spacecraft).

Flexible Blanket Arrays

A fold-up blanket array consists of a reinforced Kapton blanket supporting the solar cells and their interconnection, which is folded up, concertina fashion, into a box and then deployed using an appropriate boom or mast. In between each fold is a "leaf" of Kapton to protect the cells. An early example of this type of array was the RAE (Royal Aircraft/Aerospace Establishment) version[16] used on the Miranda X4 spacecraft flown in 1974 .

The Miranda array produced 310 W from 7440 2 x 2 cm silicon cells with an efficiency of 7.7% for a mass of 6.25 kg, ie about 50 W/kg. If modern cells are transposed onto this array, the resulting performance figures are shown in Table 5.

Cell Type	Cell Efficiency	Mass (kg)	Watts (W)	Watts/kg
Original X4	7.7	6.25	310	49.6
High Eta	16	6.85	644	94.0
GaAs/Ge	19	8.04	724	90.0
Triple Junction	28	8.26	1127	136.4

Table 5 *Performance of the X4 solar array with modern solar cells.*

Concentrator Arrays

One technique for increasing the power from an array is to increase the light intensity on the cells. Several techniques have been proposed, including cassegrainian reflectors, reflective troughs and fresnel lenses. The simplest technique is the trough concentrator. This gives typically a 50% increase in power for 33% less solar cells, compared to the equivalent planar solar array. All concentrators require more stringent pointing accuracy in one axis compared to planar arrays. This leads to important (ACS) requirements.

Hybrid Arrays

Hybrid arrays are usually blankets or reinforced skins in a carbon fibre frame. This scheme allows the simpler deployment methods, while minimising the amount of structure required and the mass. QinetiQ have developed a hybrid solar array, specifically aimed at small satellite applications. The current design is for a 1-2 kW array which will fit on a 120 kg minisatellit, and is depicted in Fig 4. The frame is constructed from carbon fibre reinforced tubing joined with moulded corner connectors. The cells are supported on a stretched membrane, with the panel interconnects attached to the rear of the membrane.

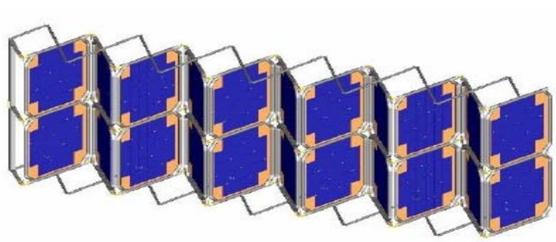

Figure 4 *Partially deployed QinetiQ hybrid solar array.*

The initial performance figures for the QinetiQ array include power to mass and power to area ratios of 77 W/kg and 230 W/m2, respectively.

Mission Consideration

For LEO missions the radiation environment is very benign. For orbit raising through the earth's radiation belt, the radiation total dose received by the array is the main design driver. This dose will depend upon the initial orbit and how quickly the radiation belts can be transited. Ideally, this orbit transfer requires radiation hard technology and a thick coverglass to shield the cell, thus a TJ type is preferable. As an example the degradation of GaAs/Ge cells with a 500µm coverglass as flown on the QinetiQ STRV1-b satellite [17,18] in GTO is illustrated in Fig 5.

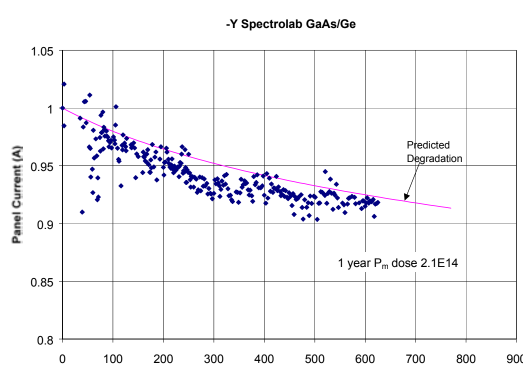

Figure 5 Degradation of STRV-1b GaAs/Ge solar panel in GTO.

The long-term performance of solar arrays in GEO is dominated by the radiation dose to which the solar cells are subjected. To differentiate between the various technologies, Table 5 shows the results of an analysis of the relative merits of the different cell types during a 15 year GEO mission, where EOL is the end of life. This shows, for example, that a high-eta array would be 75% larger than the TJ equivalent.

Cell Type	Normalised EOL Power
High eta Silicon	0.66
GaAs/Ge	0.80
Dual Junction	1.00
Triple Junction	1.16

Table 5 Comparison between solar cell technologies over 15 years in GEO.

Time in GEO (YRS)	Degredation (%)
1	4
2	6
3	7.5
5	9

Table 6 Radiation degredation of TJ cells in GEO

The annual equivalent radiation dose on a triple junction cell is 5 x 10^{13} 1 MeV electrons. The estimated degradation is shown in Table 6. These results are based upon the use of a 80 µm thick coverglass.

Recommendations

The above discussion leads to the various possible array values summarised in Table 8. From this it is clear that TJ cells provide major advantages and that they should be utilised unless strong reasons are available to dictate otherwise.

Array Type	Cell Type	Efficiency	Mass/Area (kg/m2)	Power/Mass (W/kg)	Power/Area (W/m2)
Muses-C (EOL data)	TJ	-	4.45	53	236
Upgraded X4/X5 flexible type	GaAs/Ge	19	2.01	90	181
	TJ	28	1.96	136	267
QinetiQ hybrid	TJ	25	3.00	77	230
QinetiQ X4 type	TJ	25	1.64	100	164
QinetiQ hybrid; prediction	TJ	25	6.7-5.0	45-60	300
Recommended - hybrid	TJ	28	3.00	83	250
Recommended - flexible	TJ	28	2.00	125	250

Table 8 Summary of solar array characteristics and recommendations.

ADVANCED ELECTRONICS

The problem of reducing the mass and volume for power electronics systems can be distilled to the single issue of heat rejection. If a system could be produced where the dissipation was lower, the operating temperature was higher and the thermal conductance to the spacecraft was better, major reductions in the mass and volume of the system could be made. The temperature of any electronic system is limited to that of the maximum safe operating temperature of the semiconductor components. The upper junction operating temperature for typical Si-based semiconductors is 110ºC. In analysing this situation, it is assumed that increasing the operating temperature of the EP PCU would allow the desired reduction in the physical size of the unit, if all other aspects are kept constant. As the power is dissipated to the junction doubling from ~100°C to 200°C would double the power density, leading to either half the mass, half the volume or twice the power capability for the same power conversion efficiency and conductive thermal resistance.

The Use of SiC

The recent availability of SiC diodes has led to the advantages of SiC over Si for power converter applications [19,20]. The application of mature SiC technology, in the form of diodes and switching transistors, yields savings of 50% and more in the losses in power converter systems [21]. A further advantage accrues from the ability of SiC to operate at high switching frequencies without performance degradation. It is this ability of SiC devices to operate at high temperatures, as well as its very good radiation tolerance, that provides the greatest impetus for its use in spacecraft power electronics systems. The technologies for this exist already, in the form of high temperature Si and SiC devices. The near-term realisation of the advantages of SiC is now very likely, because manufacturers are developing and marketing many relevant components and devices such as, Schottky diodes with operating and storage temperatures of-55ºC to +175ºC, with current capabilities of 4, 6 and 12 A, with continuous forward potential of 600 V. Also available 10 volt linear products, 5 volt high performance digital devices, operating from -55 to 300°C.

EP System Power Processing

The electronics systems required for an EP installation in a spacecraft include more than the PCU. A computer is often specified to control the complete installation, interfaces are needed with the spacecraft's data bus, and the propellant supply and monitoring equipment (PSME) requires electronic support. When combined together, these units are sometimes termed the power conditioning and control equipment (PCCE). The mass of a typical PCCE is very significant. For example, the UK-10 IPS[30] flown on the Artemis geostationary communications satellite[23] had a mass of about 15.5 kg, of which 11.7 kg (75%) was contributed by the PCCE. An important feature of the PCCE was the need to remove waste heat without exceeding maximum device temperatures.

Batteries

Although Ni-H2 batteries[24] were, until recently, regarded as the best available technology, high performance Li-ion batteries are now much superior[25], and their use was assumed in the present study. Bearing in mind that many discharge cycles may be needed in the missions of interest, it might be assumed on the basis of the current status of this new technology[22] that the depth of discharge (DoD) will be limited to 30%, giving a performance of about 100 Wh/kg. This limitation to a 30% DoD applies to applications in which more than 100,000 cycles are required. For a lower number of cycles the acceptable DoD increases with 150 Wh/kg being available with 80% DoD at 30,000 cycles.

DATA HANDLING AND STORAGE

Ideally, a single data handling unit (DHU) is required to perform all of the spacecraft and payload functions, i.e. housekeeping, autonomous navigation and control, and operation of the EP system and of the payload. Each mission has a different set of requirements to be met by the OBC, although it is anticipated that a common design can satisfy them all to minimise the need for the development of more than a single unit. In general, the ERC-32 (Sparc) processor is likely to be the best candidate to meet these requirements. It is commercially available in a radiation-tolerant form. An additional digital signal processor (DSP) could be accommodated for the data processing of the payload. The main design driver is the radiation dose expected during the mission. The aggregate masses are given in Table 9 for a variety of memory capacities.

Shielding Thickness (mm)	Mass Memory (Mbits)	Computer Mass (g)	Mass Excluding Box (g)	Total Mass (g)
4	500	477	645	1552
4	1000	477	674	1609
7	500	477	645	2321
7	1000	477	674	2400

Table 9 Masses of computer and total OBC system as functions of shielding thickness and memory capacity.

MEMS TECHNOLOGIES

Provided that the commercial problems, which face MEMS systems in new applications, can be successfully overcome, there is no reason why selected components could not be integrated into various EP systems in the near future. The limiting factor is the technical performance achievable when compared to operational requirements. Table 10 summarises some technology areas which are likely to be of benefit to miniaturised EP systems within the next three to five years.

The real advantages is the degree of component integration that can be achieved in a single device, allowing production of a complete system 'on a chip' or the inclusion of multiple redundancy with little extra mass or volume penalty. Two possible examples at opposite ends of the integration spectrum are:

1. A single microvalve chip, containing ten or more identical valves.
2. A complete xenon feed system integrated at wafer level into a single unit.

Component Description	Possible Application
Pressure transducers	High and low pressure devices in a propellant feed system.
Flow rate sensors	Used in the propellant feed system for throttling and telemetry.
Temperature transducers	Used in the propellant feed system for telemetry and possibly as part of an active thermal control system for temperature maintenance.
Microheaters	Used in the propellant feed system as part of an active thermal control.
Microvalves	Used throughout a propellant feed system.
Filters	Used throughout a propellant feed system.

Table 10 Summary of MEMS technology areas which are likely to be of benefit to miniature EP systems within the next three to five years.

In the longer term, it is almost certain that ten years will see the widespread introduction of MEMS components and systems into spacecraft propulsion, including EP, at a cost effective level. Operational systems may be based on hybrid technology, rather than complete MEMS solutions.

OTHER TECHNOLOGIES

Communications for Command, Control and Data Transfer

The communications requirements for the three missions are very different, owing to the great variations between expected data rates, distances from the ground station and degree of spacecraft autonomy. Platform housekeeping data transmission requirements are likely to be similar. In order to maintain a low cost approach, it must be assumed that large ground antennas cannot be used. Where possible, smaller facilities should be employed, such as the 12m antennas available at QinetiQ and RAL in the UK.

As the most demanding communications task will be to transmit imagery, either from the interplanetary or SAR missions, a Ka-band system, which would probably satisfy such requirements. The design is aimed specifically at the LEO application, only minor modifications are likely to be needed to meet the objectives of the interplanetary mission. Having a large downlink capacity also has the obvious advantages of increasing the amount of data that can be collected and transferred to the ground, without the need for on-board processing.

Spacecraft Autonomy

To minimise ground segment costs, the omni-directional low gain antenna on the spacecraft radiates continuously one of four frequencies. These indicate the health of the spacecraft, as determined by the OBC, and the need for intervention from the ground, and the urgency with which action is required. This is very effective in reducing ground segment costs, and it is recommended that a similar approach to autonomy is considered for most future missions.

Structure, Harness and Propellant Tanks

It is not possible, without designing the spacecraft, to derive an accurate mass for the structure and wiring harness. Published data from similar missions, such as the many SSTL spacecraft [26,27,28], the Muses-C design[29], and the four STRV satellites[17,18], can reflect these figures. These suggest that a carbon fibre reinforced plastic (CFRP) structure will have a mass of about 12 - 15% of the launch mass and the harness around 1% of the total. The solution assumed in deriving the above figures employs both CFRP struts and honeycomb panels. A single toroidal filament-wound propellant tank is situated around a central thrust/torsion cylinder, which might also house the main EP thruster. Subsystems and payload units are attached to floors bonded above the tank, and external shear walls carry the loads through to the launcher interface ring.

Thermal Control

The thermal control requirements of each spacecraft will depend critically on its detailed design and on its environment. In the case of the SAR mission, the method of removing the heat from both the transmitter and the PCU needed by the electric thruster must be considered. This should be accomplished within a design, which must also allow for the situation in which neither unit is operational. By comparison of documented missions, it is thought likely that 3 to 5% of the launch mass should be provided for thermal control.

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