THE POTENTIAL APPLICATION OF SMALL ELECTRICALLY-PROPELLED SPACECRAFT TO LOW-COST INTERPLANETARY MISSIONS

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INTRODUCTION

This paper presents an assessment of the application of small electrically-propelled spacecraft of 200 kg launch mass to low-cost interplanetary missions, for which very high velocity increments are required. Here, the term "low cost" eliminates the utilisation of a conventional launch vehicle for the attainment of escape velocity, thereby implying the use of electric propulsion (EP) for orbit-raising from an initial low Earth orbit (LEO) or from the geostationary transfer orbit (GTO). Thus the missions considered require velocity increments (ΔV) of the order of 10 km/s or greater. In such a demanding mission, although the performance of the solar array is critical, recent advances in solar cell technology have enabled considerable progress to be made in this area[1]. Other relevant advances include improved batteries, attitude and orbit control sensors with reduced mass, and communications systems with enhanced performance but lower mass. Gridded ion thrusters are essential for such missions owing to the need for very high values of specific impulse (SI).

This progress recently led to the conclusion[2] that a 10 km/s mission could be launched as an Ariane 5 auxiliary payload into GTO, which has a mass limit of 120 kg. This mission would commence with an orbit-raising manoeuvre to escape from the Earth's gravitational field, but could not accommodate redundancy within the propulsion system or elsewhere. This earlier work has thus been extended to examine the impact of increasing the launch mass to about 200 kg, with the aim of carrying redundant thrusters and a larger payload, whilst simultaneously raising the total velocity increment available.

Relevant flight experience comes from NASA's Deep Space 1 (DS 1) mission[3], in which an ion-propelled spacecraft undertook a fly-by of the asteroid 1992KD on 29 July 1999 and of the nucleus of comet Borrelly during September 2001. The thruster accumulated more than 16,000 hours and was working perfectly at the end of the mission. Also relevant is the Muses-C mission[4], in which an ion propulsion system (IPS) will be used to rendezvous with the asteroid 1989 ML, land, take a sample of the surface, and return this to Earth.

DEEP SPACE MISSION TO AN ASTEROID

There has, for many years, been great interest in deep space missions, heightened by the need to understand better the origin and evolution of the solar system, and also by the possibility of finding evidence of past or present life elsewhere. This interest extends to the asteroids and comets orbiting the Sun, mainly because they are considered to be remnants of the material from which the solar system was formed. However, a major problem associated with missions to these bodies is the substantial cost involved. This is due largely to the need to provide a very high ΔV to the spacecraft to escape from the Earth's gravitational field and then to reach the target in a reasonable time. A rendezvous with the target, perhaps to go into orbit around it, adds further to this requirement, which totals many km/s.

The initial objective was to select suitable targets, taking into account the need to minimise launch cost. Thus a very large ΔV was deemed mandatory, and preliminary calculations suggested that 12 to 14 km/s might be achieved by a 200 kg spacecraft carrying a reasonable payload. A list of accessible asteroids was examined, together with their orbital parameters and the values of ΔV required to rendezvous with them following Earth escape. These values were between 4.5 and 8 km/s, so are feasible using the spacecraft envisaged in this study. Indeed, it was found to be possible to target up to three such bodies in a single mission. Three asteroids can be visited for a total ΔV of below 10 km/s in at least 14 combinations and all multiple-targets studied require less than 12 km/s.

From this examination of possible targets it was clear that most represent desirable scientific objectives. It was also evident that special expertise and extensive discussion within the science community will be required to select the asteroids giving the best scientific return. Since none of them require a total value of ΔV exceeding 12 km/s, they are all accessible and there was thus no need to be specific at this stage.

LAUNCH OPTIONS

To achieve minimum cost, an attractive option is a launch to a low altitude circular orbit using a relatively inexpensive vehicle, such as the Pegasus-XL, Taurus, Rockot and Vega. Several other low-cost Russian options also exist, together with a variety of US piggy-back alternatives. In addition, a viable possibility is a launch to GTO as an Ariane 5 auxiliary payload within the main thrust cone, for which the mass limit is 350 kg. However, many of these vehicles could launch 200 kg to a much higher altitude. For example, the Rockot could provide an initial circular altitude of several 10,00 km and probably an escape capability.

Thus a very wide spectrum of performance is available, ranging from a low altitude circular orbit to attaining more than escape velocity. So the total requirements for the EP system vary from more than 10 km/s to a few km/s, depending upon the target. As one objective of this mission is to demonstrate the capabilities of ion propulsion, it is appropriate to choose a modest launcher, thereby also minimising cost. If European technology is to be used and Vega[5] is disregarded owing to its relatively high performance, the remaining option is an Ariane 5 auxiliary payload launch into GTO, which requires 4 km/s to achieve escape velocity.

PAYLOAD

An idea of the type of instrument which would be suitable for an asteroid mission can be obtained from the payload carried by the NEAR spacecraft[6], which acquired extensive data from and eventually landed on Eros. Relevant instruments were also carried by DS-1[3] and the Clementine mission to the moon[7]. In the latter case the payload consisted of two star tracker cameras, other cameras operating in the ultra-violet (UV)/visible, near infra-red (IR) and long-wave IR regions, a high resolution camera, and a laser transmitter. The total mass was 7.37 kg and the maximum power consumption 97.3 W. Relevant future missions include ESA's SMART-1 lunar spacecraft[8], which includes the use of a Hall-effect thruster for orbit raising from an Ariane 5 launch to GTO. More ambitious is the Muses-C asteroid sample return mission[4]. Additional capabilities can be provided by combinations of individual instruments and by the communications systems[8]. A realistic aim for the total mass of the payload is 15 kg. Taking this as the starting point, a possible payload is indicated in Table 1.

Experiment	**Reference**	**Origin**	**Mass (kg)**	**Power (W)**
Multispectral camera	6	NEAR	5	10
X-ray spectrometer	8	SMART-1	3	5
Radio science	8	SMART-1	0	0
Magnetometer	6	NEAR	1	1.5
IR spectrometer	8	SMART-1	2	2
Laser rangefinder	4	Muses-C	2	17
Penetrator		-	2	5
Total			**15**	**40.5**

Table 1. *List of possible payload instruments, with masses and power consumptions.*

MISSION PROFILE

In this mission, the dominating constraints are the initial mass of 200 kg and the use of an Ariane 5 auxiliary payload launch to GTO. Although target selection is best left to later scientific assessments, it is represented by a worst case ΔV of 8 km/s if only a single asteroid is to be visited. To this must be added the 4 km/s needed for Earth escape, giving a total of 12 km/s. It is proposed that the orbit-raising phase be based on the SMART-1 mission[8]. However, thrusting over a wide angle about the apogee to raise the perigee to geostationary altitude would seem to be the optimum initial procedure, followed by continuous thrusting to achieve escape, taking into account solar array degradation due to the Earth's radiation environment[1]. Attitude control would be provided by thrust vectoring, aided by momentum wheels, with momentum dumping by use of auxiliary thrusters, which may be hollow cathode arcjets (HCAs)[9].

For comparison purposes, the Pegasus-XL, SHTIL-2/Volna and Ariane 5 launch options were analysed to establish what can be accomplished using small ion thrusters. The altitudes to which the first two will reach were assumed to be 2000 and 300 km, respectively. The results, shown in Table 2, assume that gridded ion thrusters operating with an SI of 4500 s and thrust of 25 mN would be employed.

Launch Altitude (km) and Vehicle	ΔV (km/s)	Fuel Mass ΔM (kg)	Total Thruster On Time (hr)	Mission Time (days)		
				1 Thruster	2 Thrusters	3 Thrusters
300/Pegasus	7.73	32.1	15,746	800	400	267
2000/Volna	6.90	28.9	14,176	683	341	228
GTO/Ariane	4.00	17.3	8486	354	177	118

Table 2. *Characteristics of the orbit-raising phase of the mission.*

All three options are viable, although the launches into LEO require the greatest propellant mass, ΔM. Assuming the use of carbon grid technology, the operating times from GTO are acceptable. However, it is likely, for single thruster operation, that a spare might be needed to achieve the total time required, with a switching system to permit a single power conditioning unit (PCU) to be connected to either device. Although an SI of 4500 s was assumed, this is not necessarily the optimum, which can be derived by adding the total mass of the propulsion and power systems. This is because a change in SI at constant thrust alters both the propellant mass required for the mission and the power needed by the thrusters.

As an example of the effect of SI, ΔM was calculated as a function of SI for the orbit-raising manoeuvre, assuming the use of T5 gridded ion thrusters[10] operating at 25 mN. The range covered was determined by limiting the ion accelerating potential to 2.3 kV, which is representative of present grid systems[11]. The results (Fig 1) indicate that acceptable values of ΔM are required for all values of SI for a GTO launch, but that there is a mass advantage of about 8 kg in going from 3200 to 5000 s. As expected, ΔM is much larger for the LEO launches. The time required to reach escape velocity, assuming the use of two thrusters simultaneously, is plotted against SI in Fig 2. Unfortunately, the use of higher values of SI does not reduce this time, since this is largely determined by the thrust available.

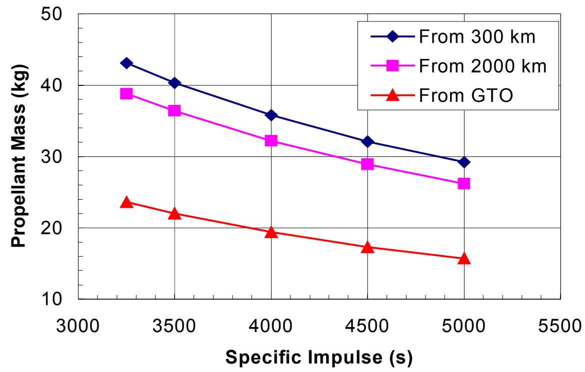

Figure 1. *Propellant mass as a function of SI for the orbit-raising phase of the mission*

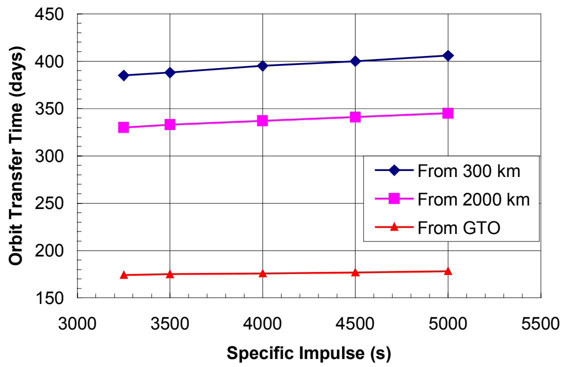

Figure 2. *Manoeuvre time as a function of SI for the orbit-raising phase of the mission.*

The orbit-raising manoeuvre will be followed by the interplanetary trajectory, represented by the ΔV required and the maximum distance from the Sun. Thrusting with the IPS to maximum velocity (to minimise trip time) will then be followed by a retardation phase, again using the IPS, with the aim of a rendezvous with the target. Depending upon the target, various coast phases might be introduced. The ΔV required is between 4.6 and 8.0 km/s, giving a total of 8.6 to 12.0 km/s, including the initial orbit-raising manoeuvre. Bearing in mind the low gravitational force exerted by all likely targets, the thrust required for capture into orbit should be well within the capabilities of the IPS. A more significant challenge involves terminal navigation, since the target will almost certainly have a low albedo. It may therefore be difficult to acquire it visually at a large distance, so a very slow approach, with the possibility of needing to thrust perpendicular to the velocity vector, may be necessary. However, the value of ΔV required for this and for maintaining the orbital parameters around the target is negligible compared to the maximum of 12 km/s quoted above.

PROPULSION REQUIREMENTS

Assuming the need for a total ΔV of 12 km/s, reasonable values of ΔM can be attained only if the SI is large. Consequently, only gridded ion thrusters or field emission EP (FEEP) [12] devices can be used, with an SI determined by the thrust required and the available power. This conclusion is illustrated by Table 3, which assumes the use of two T5 thrusters[10], with each operating at 25 mN during most of the mission.

SI (s)	ΔM (kg)	Thrusting time (hr)
2500	77.4	10,544
3000	67.0	11,445
3500	59.0	11,256
4000	52.7	11,491
4500	47.6	11,674
5000	43.4	11,831
5500	39.9	11,969
6000	36.9	12,059

Table 3. *Propellant mass and thrusting time for a range of values of SI, using two T5 ion thrusters.*

The attainment of adequate thruster durability is crucial to the success of this mission, with the need to reach the operating times indicated in Table 3. The estimated thruster lifetime[13] 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 available using carbon, so much enhanced mission durations can be achieved with this material.

As the power required increases with SI, the mass of the solar array rises as ΔM falls, and there will normally be an optimum value of SI at which the total mass is a minimum. To determine this minimum, performance data for a flexible array[1] based on high efficiency triple junction solar cells were adopted; these are 2 kg/m^2, 125 W/kg, and 250 W/ m^2. An allowance was also made for degradation due to the Earth's radiation belts during orbit-raising. Assuming the use of two thrusters, from Fig 2 the transfer time will be about 6 months and the worst case array degradation[1] is about 3%, assuming that the whole of this period is spent in GTO; a value of 5% was taken to provide a suitable margin.

Approximate analytical expressions were derived for the power consumed by the thrusters as a function of SI and thrust, taking into account the losses in the PCU. This then provided the array mass, using the data mentioned above. The PCU mass, which also depends on the power level, was calculated and doubled to take into account dual thruster operation. The result was then halved to allow for advanced power processing technologies[1]. The resulting mass was added to that of the array to give the total dependent upon SI, recognising that the thruster itself and the propellant feed system are independent of this parameter.

The results are given in Fig 3, in which ΔM, the total mass of the solar array and PCU, and the aggregate of these are plotted against SI. Although the upward trend of the mass of the power systems is evident, this is less rapid than the rate at which ΔM falls with increase of SI, so there is no minimum in the curve for the total mass, despite extending the range covered to 6000 s. Thus there is no optimum value of SI within this range. To check the sensitivity of this conclusion to the assumptions made about the power systems, 20% was added to their mass, giving the upper curve in Fig 3; this also does not display a minimum.

In order to proceed, it was decided to limit the SI to 5000s, for which ΔM is 43.4 kg. At this value, the ion thruster grid operating point is acceptable, with a beam accelerating potential of 2.25 kV. If an allowance of 300 W is made for platform systems during thruster operation, together with a 5% degradation due to radiation damage, an array power to 2.4 kW at the start of the transfer from GTO to Earth escape is a reasonable objective, with some additional margin. The array mass is then 19.2 kg and the area 9.6 m^2.

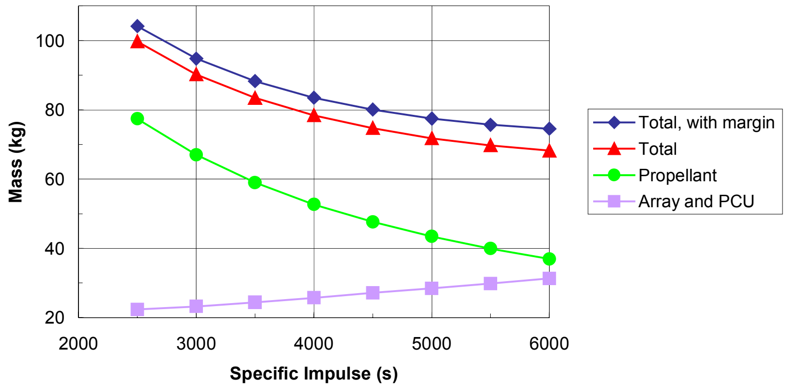

Figure 3. *Propellant, power system and aggregate masses as functions of SI.*

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 and a total of 50 mN, the propellant mass needed is 47.7 kg if a margin of 10% is included. The thruster on-time is less than 12,000 hours, which is well within the capabilities of carbon grids. Of this, about 4300 hours are required for the orbit-raising phase of the mission. For redundancy, it is assumed that three thrusters will be flown, mounted on individual gimbal platforms, and supplied by two PCUs, together with a switching system to permit either of these to operate any of the thrusters. Bearing in mind the fact that thrusting will not take place continuously, the total mission duration will be at least 2.5 years, assuming an operational stay of about 1 year at the target asteroid.

SPACECRAFT DESIGN

Constraints

There are several important constraints which affect the design of the spacecraft and the way in which it can be operated. Some of these will have an impact on the power required, on the overall mass and on trajectory optimisation, so must be considered at an early stage.

Distances from the Sun and Earth As an example of the importance of the distance to the Sun, in the Muses-C study of the mission to Nereus[14] this distance became as great as 2.017 AU. Thus the power generated by the array decreased to only 25% of its initial value, and the planned thrust was modulated to take these variations into account. From the point of view of communications, the distance to the Earth increased to a maximum of 2.979AU. Thus the signal strength at Earth was very low, implying the need for a relatively large diameter high gain antenna (HGA) on the spacecraft and a high power transmitter.

Variation of the Sun Vector Angle This will partly determine the design of the solar array and, in particular, its ability to move about one, two or three axes in order to maximise power output. For simplicity, the advantages of multi-axis alignment capabilities must be ignored and it is suggested that the array be capable of rotation about its long axis only. It should be noted that it will be necessary to rotate the spacecraft through 180° in order to thrust against the velocity vector prior to the rendezvous with the target. In addition, in this context, the possible need to apply large thrust vector angles when nearing the target must be accomplished by rotation of the complete spacecraft.

Variation of the Earth Vector Angle Under normal circumstances, this will determine the design of the mounting for the HGA, since it must be capable of pointing at Earth during communications sessions. Alternatively, as accepted here, the spacecraft can be manoeuvred to point the antenna at the Earth at pre-determined communications times.

Thermal Design The absorption of solar radiation should be minimised by the use of reflectors on surfaces likely to be illuminated by the Sun. Thermal control will then rely on a balance between the waste heat produced internally, augmented as necessary by the use of heaters, and radiation to deep space from external surfaces not illuminated by the Sun. Louvres may be needed to reduce the total emissivity of the latter as the power produced by the arrays decreases with distance from the Sun. In addition, as the radiators must not see the Sun, the design might need to include appropriate shields.

Thruster Throttling   Owing to the variation of the power available during the mission, it will be necessary to throttle the thrusters, perhaps over a factor of 5 in thrust. When throttling, it is essential to maintain a very high SI if performance is not to be degraded unacceptably. When coupled with the SI requirement, this wide throttling range dictates the selection of a gridded ion thruster or a FEEP device.

Design Concept

The proposed on-orbit configuration of the spacecraft is shown in Figs 4 and 5. It is conventional, with solar arrays on either side of the platform, which contains all service systems and payload instruments and is dominated by the HGA. This is fixed in position and therefore requires the spacecraft to point towards the Earth for the transmission of data. Two omni-directional antennas are also included for low data rate communications. The array shown is of the hybrid type, although the flexible concept[1] is of lower mass. Only a single T5 ion thruster is indicated, mounted on the lower face of the platform. Although this could probably accomplish the mission if using carbon grids, it is recommended that two or three thrusters be employed. These can be mounted readily onto one face of a platform with major dimensions of about 1 m. As shown in the partially cut-away view depicted in Fig 5, the platform is modular, with the IPS situated within the launcher interface adapter; as mentioned above, only a single ion thruster is shown in this version.

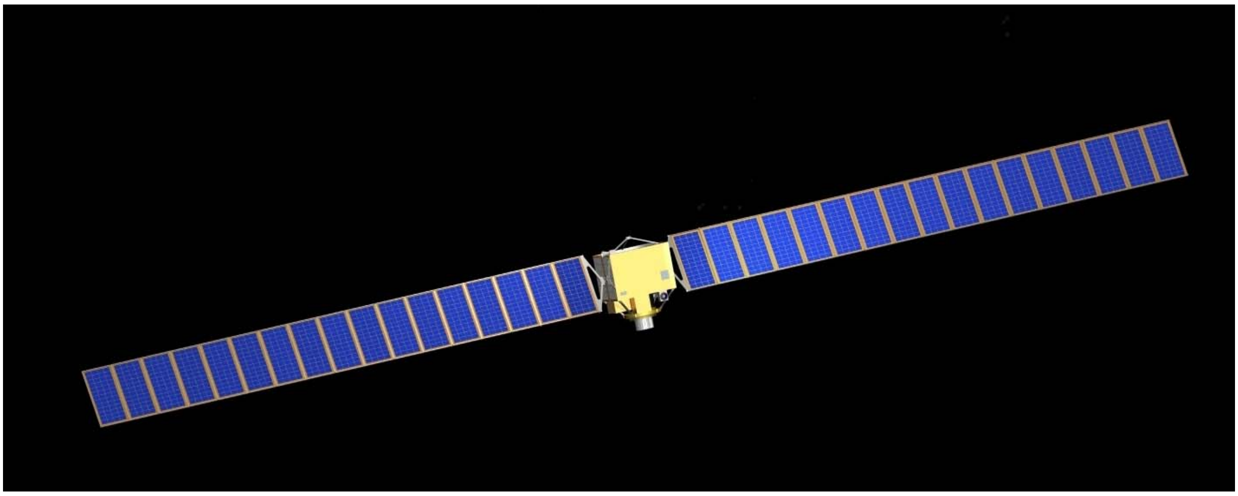

Figure 4. *Conceptual configuration with deployed hybrid array.*

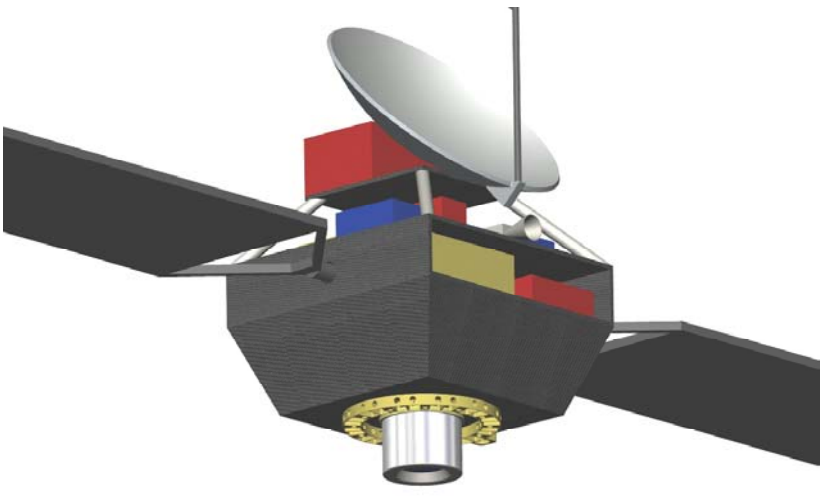

Figure 5. *Close-up, partially cut-away view of the underside of an early version of the spacecraft.*

Ion Propulsion System

The masses of the separate components of the IPS can be deduced largely from the Artemis programme[15], taking into account existing and expected advances in electronics and propellant feed technologies[1]. The latter assume the use of MEMS components for certain functions, such as temperature and pressure sensors. It has been assumed that three T5 thrusters will be flown, with two operated simultaneously from two PCUs. A switch system is incorporated to enable any thruster to be connected to either PCU. The individual items are listed in the mass budget in Table 4, together with their estimated masses and appropriate margins.

To the total of 30.2 kg must be added the mass of the propellant tank. The preferred tank is a recent QinetiQ development and is a filament-wound toroid, which makes the best use of the volume available within the platform and has a mass of about 10% of ΔM. Assuming that the attitude control system (ACS) will also use xenon and that it requires 1.5 kg, the total ΔM will be 44.9 kg for a 12 km/s capability. This becomes 49.4 kg with a margin of 10%. The tank will therefore weigh 5 kg and the total IPS dry mass will be 35.2 kg.

Power Generation, Storage and Distribution

Assuming the use of a pair of T5 engines operating simultaneously at 25 mN thrust and an SI of 5000 s, the power consumption of each is about 830 W. With an overall PCU efficiency of 87%, the input power to the thruster becomes 954 W. To this must be added the 20 W consumed by the propellant feed system, giving a total of 974 W. Thus, an array power of about 2 kW should be adequate for the IPS. If 300 W covers the needs of the platform systems and payload, the total requirement becomes 2248 W. To this must be added an allowance for array degradation; it is estimated that a worst case figure[1] will be 5%. Thus the required capability is 2366 W, which has been rounded up to 2400 W. Assuming flexible array technology[1], the mass becomes 19.2 kg. If the array must be packaged within the outline of the spacecraft body, with a side of length 1 m, it is likely to have dimensions of about 0.9 m × 5.5 m. Small, commercially-available, array drive systems weigh about 1.1 kg, so a dual redundant unit will require perhaps 2.5 kg.

To minimise mass and complexity, the power bus will be unregulated. In estimating the mass of this system, the maximum power to be handled by the battery charge and discharge regulators is limited to the peak demands on the battery. This will be determined by the sum of the payload and platform power consumptions, which is less than 400 W. The total mass of these regulators using present technology was estimated to be 2.5 kg. The shunt regulator must be able to handle the full design output power of 2400 W. Assuming a bus potential of 50 V, the maximum current is 48 A and the predicted mass is 1.1 kg. Thus the total mass is 3.6 kg, which can be reduced to 1.8 kg using advanced electronics technologies[1].

The battery will be lithium-ion to capitalise on the high Whr/kg that this technology offers[1]; 150 Wh/kg should be achievable with a DoD of 80%. A high capacity is not needed, as the only eclipses that will be experienced will be during orbit-raising from GTO, and the operation of the IPS will not occur during these periods. It was assumed that 500 Whr will be fully adequate, so the mass becomes 3.4 kg.

Data Handling Unit (DHU) and Data Storage

A single DHU is required to perform all of the spacecraft and payload functions, including housekeeping, autonomous navigation and control, and operation of the ion thrusters and of the payload. The ERC-32 (Sparc) processor is likely to be the best candidate to meet these requirements and is commercially available in a radiation-tolerant form. An additional digital signal processor could be accommodated if required for the data processing of the payload. It was estimated that the mass of a computer using this technology and having a memory capacity of 1 Gbit will be 2.4 kg, assuming a radiation shielding thickness of 7 mm.

Communications

The frequency bands of choice are either Ka- or X-band and, to minimise cost, it is assumed that 12 m diameter ground antennas will be used. As a baseline, the system flown very successfully on DS-1[3] operated in Ka-band and had a mass of 3 kg. Allowing 5 kg for the HGA and cabling, the total was about 8 kg. However, at X-band, the latest microwave power module technology could be used; these are smaller and lighter than those of the Ka-band system. Thus an X-band system was selected, with a 0.5 m diameter reflector, a horn feed, a continuously powered low noise amplifier, an X-band receiver, and a transmitter. Small omni-directional antennas are also provided to allow for emergency access to the spacecraft and for low-rate housekeeping data to be downloaded.

It is estimated that this system, operating at 8.5 GHz, will provide a data rate of 600 bps at a distance of 2 AU from the Earth,. The required 9.6 dB margin (equivalent to a bit error rate of $1{:}10^5$) is exceeded by 1.7 db by the use of standard coding techniques. The beamwidth of the HGA will be less than 1° and will therefore require accurate Earth-pointing. The transmitter RF power output of 180 W requires about 400 W of DC power. As mentioned previously, a constraint on the mission is that the transmitter is not operated whilst thrusting. A mass of 7 kg was assumed for the complete system.

Attitude and Orbit Determination and Navigation

The spacecraft requires 3-axis sensing for all mission phases. Attitude determination during orbit raising will utilise digital sun and Earth sensors, based on MEMS technology, augmented as necessary by data from the star sensor/tracker. A number of MEMS devices currently in development weigh only 10s of grammes each, so a total allocation of 1 kg for sun and Earth sensors is generous. The star tracker can be used during both orbit-raising and the cruise phase and provides an essential autonomous navigation capability. This is vital owing to the large distances from Earth and the need to minimise ground segment costs. Excellent results from DS-1[3] suggest that this concept is entirely feasible. Indeed, this mission provided en-route accuracy of better than 400 km and 0.2 m/s, which improved to better than 3 km at the encounter.

The star sensor will be aligned near the array axis, which places the sun at 90°, avoiding blinding. This allows the device to be small and reduces the requirement for a baffle around the head. The star tracker used on the Clementine mission[7] weighed only 0.6 kg, and consumed 4.5 W; this should be suitable in the present case, although an allocation of 1.5 kg has been made to provide redundancy.

Orbit determination whilst in Earth orbit will be by standard spacecraft ranging through the transponder. This is likely to be augmented by use of a GPS system, although the utility of this in and beyond GTO has yet to be demonstrated. Bearing in mind that commercial units for terrestrial use are currently available which have a mass of a few hundred grammes, an estimate of 1 kg for this would appear to be reasonable.

Attitude Control System (ACS)

DS-1 has demonstrated[3] that the consumption of ACS propellant can be minimised by vectoring the thrust of an ion engine. Assuming that this will be possible in the present case, a ΔV allowance[14] of 90 m/s should be included for injection errors; this is trivial in the context of the overall 12 km/s that is needed and can be provided during orbit-raising.

It is suggested that basic attitude control is accomplished by four control moment gyros (CMGs) arranged in a tetrahedron and providing redundancy against the failure of one wheel. However, these gyros must periodically be offloaded by the attitude thrusters. Including redundancy, this requires a minimum of eight thrusters, although degraded control could be accomplished with six, still with redundancy, but would incur significant additional fuel use and reorientation manoeuvres. Although hydrazine or ammonia could be considered as ACS propellants, they would require a second set of propulsion equipment. However, recent developments of HCAs[9] suggest that relatively high values of SI will be achievable, perhaps of the order of 500 s, and these devices have the advantage of using xenon as the propellant. Eight HCAs were therefore selected. The T5 cathode has a mass of 70 g, to which must be added flow control valves at 100 g each, mounting brackets, pipework, and other items, giving a total of 2 kg.

Although it might be assumed that the thruster PCUs could be used to operate these devices, this will only be possible if the thrusters are not in use. Thus an allowance must be added for power units for two HCAs to operate simultaneously, plus a switch system to interconnect the 8. The combined heater and anode supplies for an HCA are estimated to have a mass of 0.5 kg. If the switch system weighs 0.3 kg, the total is 1.3 kg.

The stability required will depend on the payload selected. However, at a range of 10 km from the target body, an estimated pointing stability of 200 mrads^{-1} is considered adequate. Commercially available CMGs of this standard weigh of the order of 1 to 3 kg; the mean figure of 2 kg is adopted in this case. This corresponds well with the 2 kg system flown on the Clementine spacecraft[7].

Structure, Launch Adaptor, Thermal Control and Harness

The structural configuration employs both carbon fibre reinforced plastic struts and honeycomb panels. A single toroidal filament-wound fuel tank is situated around a central thrust/torsion cylinder, which also houses the ion thrusters and their flow control units. 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. A preliminary assessment suggests that 12% of the launch mass is a reasonable allocation for this. To this must be added about 3% to allow for the components of the launch adaptor fixed to the spacecraft.

Removal of heat from both the communications equipment and each ion thruster PCU will require additioanl mass. These dissipate about 220 W and 124 W, respectively, although they will not operate simultaneously, easing the problem. A heat shield will also be required between the ion thrusters and the adjacent spacecraft structure. An additional 3% of the launch mass, which amounts to 6 kg, is allocated for these purposes. The harness is likely to require about 1% of the launch mass. This takes into account the separate allocation of 2 kg to cover the part of the harness connecting the thrusters, switch system and PCUs.

Mass Budget

Table 4 gives the preliminary mass budget, including estimated margins which have been selected in the range 5 to 20% according to the maturity of the technology involved. It is clear that the spacecraft can be constructed within the 200 kg limit, including the margins identified in this analysis, and that the 12 km/s velocity increment can be achieved. However, this budget does not include any scope for an overall 20%

ITEM	**ESTIMATED MASS (kg)**	**MASS MARGIN (%)**	**(kg)**
Payload	15.0		
Propellant (including ACS)	44.9	10	4.5
Tank	5.0	20	1.0
Pressure regulator system (redundant)	3.0	10	0.3
T5 thrusters (3)	4.8	5	0.3
Gimbal mounts (3)	4.5	20	0.9
PCUs (2)	12.4	20	2.5
Switch unit	1.0	20	0.2
FCUs (3)	1.5	10	0.2
IPS harness	2.0	20	0.4
IPS pipework and fittings	1.0	20	0.2
Solar array (including deployment)	19.2	20	3.9
Array rotation mechanism (redundant)	2.5	10	0.3
Array electronics	1.8	20	0.4
Battery	3.4		
DHU (computer and memory)	2.4	10	0.3
Communications system	7.0	10	0.7
Sun and Earth sensors	1.0	10	0.1
Star tracker (redundant)	1.5	10	0.2
GPS system	1.0	10	0.1
ACS thrusters	2.0	10	0.2
ACS power supplies and switches	1.3	10	0.2
CMGs	2.0	10	0.2
Structure	24.0	10	2.4
Launch adaptor	6.0	20	1.2
Thermal control	6.0	20	1.2
Harness	2.0	20	0.4
TOTALS	**178.2**		**22.3**

Table 4. *Preliminary mass budget for the interplanetary spacecraft.*

margin, as often required at the start of many projects; this amounts to 31 kg in this case, since the propellant mass of 45 kg would not be included in the calculation.

CONCLUSIONS

This study aimed to minimise the size and mass of an interplanetary mission to an asteroid or comet, to meet a 200 kg requirement while using an economical launch vehicle. After consideration of a range of targets, it was decided that the selection process should be left to the scientific community, since the necessary expertise lies there. Consequently, a velocity increment (ΔV) of 8 km/s was assumed for the interplanetary phase of the mission to ensure that a very wide range of asteroids and comets will be accessible. The assumption of an Ariane 5 auxiliary launch into geostationary transfer orbit (GTO) increased ΔV to 12 km/s, to provide an orbit-raising manoeuvre to achieve escape from the Earth's gravitational field.

Assuming the use of two 25 mN gridded ion thrusters operating simultaneously with an SI of 5000 s, together with one spare thruster, the total predicted mass of the proposed spacecraft is 178 kg. This includes a solar array with a 2.4 kW capability at the beginning-of-life and a payload of 15 kg, and enables a ΔV of12 km/s to be achieved. This is within the 200 kg limit. If appropriate margins are included for individual components and systems, the 200 kg limit remains feasible. However, a 20% global margin, which is normally required for such missions, cannot be accommodated.

REFERENCES

1. Clark, S D, Fearn, D G and Marchandise, F, "A study into the techniques for miniaturised electric propulsion systems, and mission categories, for small spacecraft", IEPC Paper 2003-0216, (March 2003).

2. Wells, N and Fearn, D, "Minimising the size and mass of interplanetary spacecraft", IAF Paper IAF-01-Q.5.07, (October 2001).

3. Rayman, M D, Varghese, P, Lehman, D H and Livesay, L L, "Results from the Deep Space 1 technology validation mission", IAF Paper IAA-99-IAA.11.2.01, (October 1999).

4. Kawaguchi, J and Uesugi, K T K, "Technology development status of the Muses-C sample and return project", IAF Paper IAF-99-IAA.11.2.02, (October 1999).

5. Barbera, R and Bianchi, S, "Vega: the European small-launcher programme", *ESA Bulletin*, No 109, 64-70, (February 2002).

6. Santo, A G, Lee, S C and Gold, R E, "NEAR spacecraft and instrumentation", *J of the Aeronautical Sciences,* **43**, 4, 373-397, (October-December 1995).

7. Rustan, P L, "Clementine: mining new uses for SDI technology", *Aerospace America*, **32**, 1, 38, (1994).

8. Racca, G D, Foing, B H and Rathsman, P, "An overview on the status of the SMART-1 mission", IAF Paper IAA-99-IAA.11.2.09, (October 1999).

9. Gessini, P, Gabriel, S B and Fearn, D G, "The hollow cathode as a micro-ion thruster", IEPC Paper 01-233, (October 2001).

10. Fearn, D G and Smith, P, "A review of UK ion propulsion - a maturing technology", IAF Paper IAF-98-S.4.01, (Sept/Oct 1998).

11. Martin, A R and Latham, P M, "High thrust operation of the UK-10 rare gas ion thruster (T4A)", IEPC Paper 88-062, (October 1988).

12. Bianco, P, Moratto, C and Tucci, L, "FEEP propulsion system for small spacecraft interplanetary space missions", IAF Paper IAF-01-S.4.05, (October 2001).

13. Fearn, D G, "Life-testing of the UK-10 ion propulsion system", Report DRA/CIS(CSC3)/CR/94/43, (Dec 1994).

14. Kawaguchi, J, Yamakawa, H and Matsuoka, M, "Synthesis of Muses-C low thrust sample and return trajectory", IAF Paper IAF-98-A.4.01, (September/October 1998).

15. Gray, H L, "Development of ion propulsion systems", *GEC Rev*, **12**, 3, 154-168, (1997).