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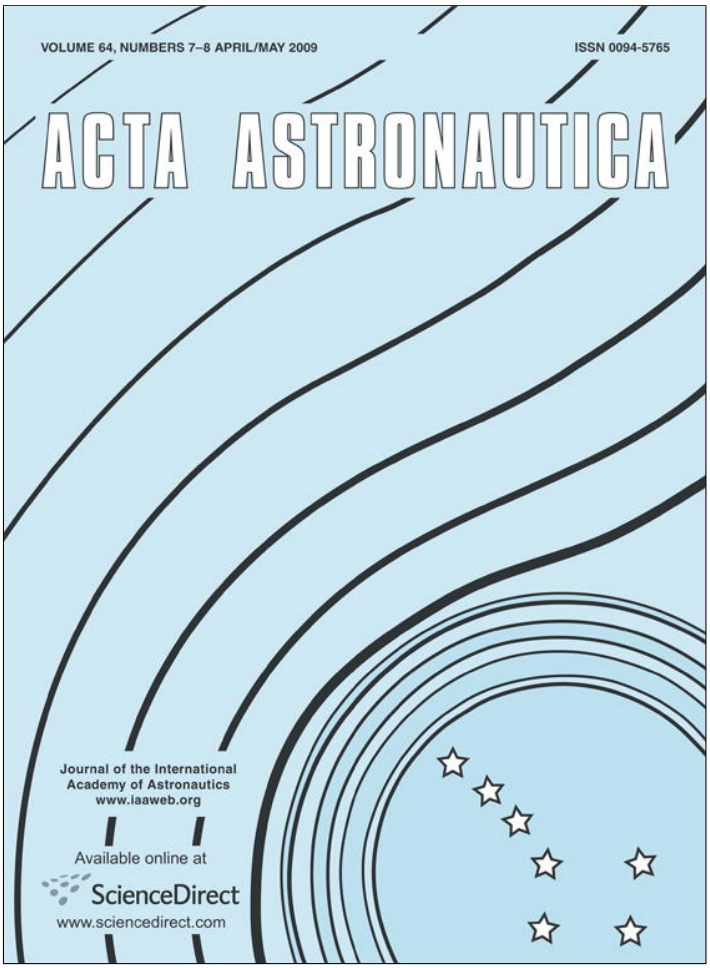

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Very high delta-V missions to the edge of the solar system and beyond enabled by the dual-stage 4-grid ion thruster concept

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Abstract

A new and innovative type of gridded ion thruster, the "Dual-Stage 4-Grid" or DS4G concept, has been proposed and its predicted high performance validated under an ESA research, development and test programme. The DS4G concept is able to operate at very high specific impulse and thrust density values well in excess of conventional 3-grid ion thrusters at the expense of a higher power-to-thrust ratio. This makes it a possible candidate for ambitious missions requiring very high delta-V capability and high power. Such missions include 100 kW-level multi-ton probes based on nuclear and solar electric propulsion (SEP) to distant Kuiper Belt Object and inner Oort cloud objects, and to the Local Interstellar medium. In this paper, the DS4G concept is introduced and its application to this mission class is investigated. Benefits of using the DS4G over conventional thrusters include reduced transfer time and increased payload mass, if suitably advanced lightweight power system technologies are developed.

A mission-level optimisation is performed (launch, spacecraft system design and low-thrust trajectory combined) in order to find design solutions with minimum transfer time, maximum scientific payload mass, and to explore the influence of power system specific mass. It is found that the DS4G enables an 8-ton spacecraft with a payload mass of 400 kg, equipped with a 65 kW nuclear reactor with specific mass 25 kg/kW (e.g. Topaz-type with Brayton cycle conversion) to reach 200 AU in 23 years after an Earth escape launch by Ariane 5. In this scenario, the optimum specific impulse for the mission is over 10,000 s, which is well within the capabilities of a single 65 kW DS4G thruster. It is also found that an interstellar probe mission to 200 AU could be accomplished in 25 years using a "medium-term" SEP system with a lightweight 155 kW solar array (2 kg/kW specific mass) and thruster PPU (3.7 kg/kW) and an Earth escape launch on Ariane 5. In this case, the optimum specific impulse is lower at 3500 s which is well within conventional gridded ion thruster capability.
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1. Introduction

In recent years, very intriguing and exciting data has been gathered by ground-based and space-based telescopes and deep space probes on the objects and environments at the edge of our solar system. Such data has placed important constraints on theories of the formation of the solar system and given tantalising hints on the dimensions of the heliosphere with the interstellar medium. Although some questions have been answered, the many new ones posed have established the very outer solar system as a high priority for exploration.

As optical and IR imaging telescopes have become ever more sensitive and image processing ever more powerful, the existence of the Edgeworth-Kuiper

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Belt (EKB) has been truly confirmed with the discovery of over 500 Kuiper Belt Objects (KBOs) between 30 and 50 AU distance in the last decade, including the largest of these, "Quaoar" (2002 LM60), a 1300 km diameter icy body about half the size of Pluto [1] and 2003 UB313, a 2700 km body about 20% larger than Pluto [2]. A population of over 70,000 KBOs larger than 100 km are predicted to exist and contain primitive accretion disk material from the early solar system.

The discovery in 2004 of a new object of 1200–1800 km diameter called Sedna at 90 AU distance and with an orbit well beyond the EKB has, for the first time, given a possible glimpse at the inner part of the hypothesised Oort cloud, a predicted shell of billions of icy proto-comets extending halfway to the nearest star. Sedna is currently the most distant known object and is near the 76 AU perihelion of its highly eccentric orbit. Its bright red colour does not fit with observations of icy bodies such as Pluto and other KBOs and so its composition is completely unknown [3].

As such, these extremely interesting minor planets represent prime targets for in situ characterisation by interplanetary spacecraft. However, the delta-V requirements are high for reasonable mission durations. This is provided by a combination of launcher escape velocity and on-board propulsion. The NASA New Horizons Pluto/KBO flyby mission will be the first to attempt this challenge when it reaches Pluto in 2015 and proceeds to encounter other KBOs in 2016–2020 [4]. Recent technology assessment studies have indicated that missions to rendezvous with Pluto or other members of the KBO family are within the propulsive capabilities of humankind using a combination of a powerful launcher giving a high escape velocity and a 1 kW-class RTG-powered spacecraft with conventional gridded ion propulsion to accomplish the mission within 12–17 years from launch [5,6]. However, to flyby or rendezvous with Sedna at more than double the distance of Pluto and a higher ecliptic declination within a reasonable mission duration is even more challenging. A flyby could be achieved within 21 years, assuming a small RTG-powered probe with minimal on-board propulsion, and launched at a high escape velocity of 9.3 km/s on a ballistic trajectory with multiple swings (Earth–Venus–Venus–Earth–Jupiter). In the case of a Sedna rendezvous mission, the delta-V requirements and hence desired specific impulse may be on the boundary of performance of current ion propulsion systems.

At the end of 2004 after 28 years, the Voyager 1 spacecraft reached a distance of 94 AU from the Sun and sent back data indicating that it had reached a crossing of the "termination shock" of the solar wind at the edge of the Sun's magnetosphere. Beyond the shock is the heliopause, the pressure balance interface between the plasma of the solar wind and that of the Very Local Interstellar Medium (VLISM). Despite this data, models of the heliopause distance are still open to large uncertainty and precise properties of the interstellar medium remain very unclear. The only way to determine these properties and truly understand the nature of interstellar space is again by in situ measurements, particularly of the magnetic field and low-energy cosmic rays, beyond the heliopause. This is at the boundary of technological capability on many mission elements such as launcher escape velocity, power, propulsion, communications, on-board autonomy and extremely long-duration component/system reliability. Nonetheless, recent studies on an Innovative Interstellar Explorer [7] have indicated that it could be feasible with certain technology developments and aggressive mass savings to reach the predicted closest VSLIM at a distance of 200 AU within 30 years from launch (foreseen as the maximum "scientific career lifetime"). The approach again uses very high escape velocity (over 11 km/s) provided by a heavy launcher with multiple solid boost stages, and a 1000 kg, 1 kW-class RTG-powered spacecraft with current ion propulsion performance (4000 s specific impulse) and small scientific payload in order to meet the high delta-V requirements of over 17 km/s.

Alternative approaches for an Interstellar Heliopause Probe (IHP) involving solar sail technology have also been proposed in the frame of the ESA Science Directorate's Technology Reference Studies [8] in order to assess likely technology development needs for fulfilling the objectives of Europe's Cosmic Vision 2015–2025. The concept aims to deploy a 1 μm thick, 280 m diameter, spinning disk sail after an Earth escape launch with a Soyuz launcher. The sail would perform a close solar flyby of 0.25 AU, and then gain sufficient solar system escape velocity to release a 213 kg probe at 5 AU, which reaches 200 AU in 25 years after launch.

2. Dual-stage 4-grid (DS4G) thruster

A new concept for an advanced 'dual-stage' gridded ion thruster has been proposed by Fearn which has the potential to deliver substantial improvements in propulsive performance over the current state-of-the-art [9,10]. Four grids are used instead of the usual three-grid arrangement in order to separate the ion extraction and acceleration processes (done simultaneously in current systems). This enables very high ion beam

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potentials on the grids in the acceleration stage, thereby significantly increasing exhaust velocity, specific impulse, power density and thrust density. Operating at a beam potential of 30 kV with Xenon propellant (as opposed to a maximum of 5 kV in present ion engines), the predicted performance of this advanced thruster is a specific impulse of 19,000 s, power density of over 600 W/cm^2, thrust density of over 6 mN/cm^2, and beam divergence $< 5°$. This high performance comes at the expense of a high power-to-thrust ratio, which at this operating point is approximately 110 W/mN. Thus, the thruster is foreseen to be compatible with lightweight power systems only.

The very high specific impulse and thrust density make the DS4G concept a potential candidate for use on very high delta-V missions such as those discussed above, with potential pay-offs in transfer time, scientific payload mass and relaxed Earth escape launch requirements.

In order to demonstrate the feasibility of this new thruster concept, a small experimental laboratory prototype has been designed and constructed. The experimental test campaign comprised two successful test phases which were conducted in a vacuum facility at ESTEC during November 2005 and May 2006, with the aim of demonstrating the practical feasibility of the 4-grids concept, verifying the high performance predicted by the analytical and simulation models, and investigating critical design issues and technological challenges [11,12]. Total accelerating potentials of up to 30 kV were demonstrated. Narrow beam divergences of the order of 2–4° were also achieved. The SI reached 14,000–15,000 s and the open area thrust and power densities 8.4 mN/cm^2 and 740 W/cm^2, respectively.

Due to the high power density of the thruster technology, it is ultimately scaleable to very high power levels of nearly 1 MW (50 cm beam diameter), and hence suitable for the ambitious missions discussed in this paper.

3. Advanced power systems

3.1. Nuclear power sources

Beyond the technology developed and flown since the 1960s with the low-power US Radioisotope Thermoelectric Generators (RTGs) and moderate-power Russian TOPAZ reactors, there has been little advancement in nuclear power systems to much higher power levels. Many high-power space fission reactor concept studies have been performed worldwide, however, which enables us to at least get an indication of typical masses

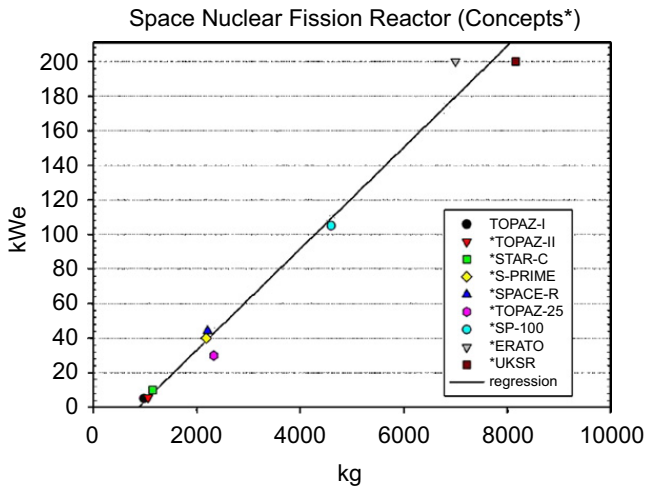

Fig. 1. Nuclear reactor mass for different electrical power levels [13].

at different power levels. Fig. 1 shows these values for the different concepts.

Despite radically different reactor core, energy conversion and heat rejection techniques used, the distribution of total reactor system mass versus electrical power generated shows a remarkably close linear regression fit (solid line in the figure). Hence, reactor power versus mass can be expressed by an analytic approximation:

$$P(\mathrm{kW}_e) = 0.0293M(\mathrm{kg}) - 25.63 \quad (1)$$

Using this relationship, the reactor specific mass levels off to 42 kg/kW for power levels above 100 kW. This is significantly higher than for some of the more advanced nuclear power concepts, which employ high-efficiency dynamic energy conversion systems such as Brayton cycle devices to reach reactor specific masses approaching 25 kg/kW. Reactor specific masses are an order to magnitude higher than lightweight solar generators, however, the NEP spacecraft benefits from constant power and thrust throughout the mission (especially at larger solar distances) and reduced operational constraints.

3.2. Solar generators

The solar generator is an essential part of a spacecraft. It converts the solar energy, which is available (about 1.35 kW/m^2 at 1 AU), directly into electric power. Conventional space technology provides a specific power (power per unit mass) at solar array-level of 50 W/kg; this involves a structure weight of ca. 1.5 kg/m^2, a protective glass layer of about 0.5 kg/m^2 and the solar cells which are roughly 2 kg/m^2. Development is ongoing to provide high efficiency, lightweight solar arrays by reducing structural weight, introducing innovative actuators, thinner solar cell wafers and of course higher

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Table 1
Solar generator specific power and technology maturity.

Technology	Specific power (W/kg)	Alpha (kg/kW)	Maturity
State of art[14,15]	50	20	In flight
State of art plus Improvement in structures	80	12.5	Near term
CIGS [16]	100	10	Near term
Multi junction on 140 μm Ge substrate	108	9.25	Near term
Multi junction on 30 μm Ge substrate	200	5	5 years term
Stretched lens solar array (SLA)[17]	45	22.2	On fight
Stretched lens solar array (SLA)	330	3	Near term
Stretched lens solar array (SLA)	500	2	10 years term
Amorphous silicon and CIGS	$\gg 100$	$\ll 10$	Long term

conversion efficiencies. Further important developments are achieved with the stretched solar array technology SLA, which is a system incorporating a solar cell and a refractive Fresnel lens concentrator. Deep Space 1 was the first spacecraft using an Ion engine and a solar concentrator array.

Other promising developments are expected from very light thin film and flexible solar cells based on Cu(In,Ga)Se2—*CIGS* which would enable large lightweight structures. Table 1 shows the state of art and the expected development for different technologies. In general, the space sector benefits a great deal from the booming terrestrial solar cells industry, which will lead to increases in conversion efficiency and volume production techniques (and therefore lower cost for high power solar generators) in particular in the future. Development and flight model costs are much lower than nuclear reactors, and the specific masses are over an order of magnitude lower. However, the disadvantage is that they are only useful for powering interstellar missions at close distances from the Sun due to the rapid drop off in solar flux intensity at increasing solar distances.

4. Mission analysis scenario

The focus of this paper is on the technical assessment of the DS4G thruster concept for Nuclear Electric Propulsion (NEP) and Solar Electric Propulsion (SEP) missions to the local interstellar medium at 200 AU, since this represents the most demanding scenario for both the power and propulsion technologies, as well as many others, in terms of performance, mass, autonomy and reliability. It is noted that the results of this paper are also applicable to shorter duration flyby missions of KBOs and inner Oort cloud objects such as Sedna [18].

We assume that the NEP spacecraft starts its mission from an Earth escape launch by the Ariane 5 ECB

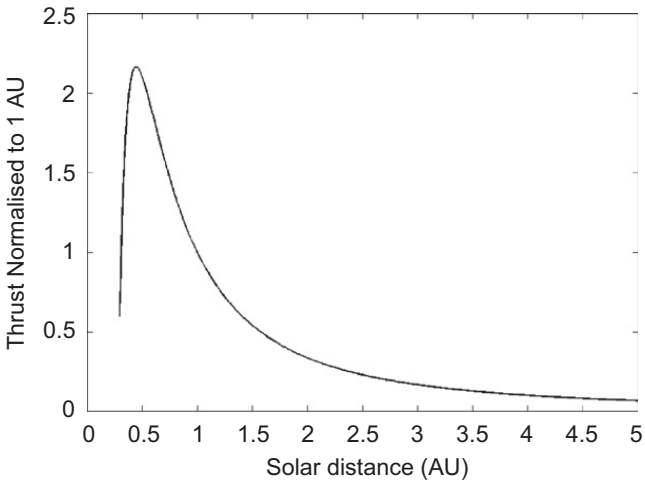

Fig. 2. Variation of thrust with solar distance normalised to 1 AU (amorphous Si cells) [13].

launcher, and then thrusts continuously until propellant burnout, at which point it then ejects a small scientific probe which coasts with a fixed solar system escape velocity until 200 AU. No gravity assist manoeuvres are considered, thus giving a wide launch window. The direct injection capability of the Ariane 5 ECB for an escape launch energy C3 of 16 km^2/s^2 is predicted in [19] to be 8200 kg (assuming an Earth escape declination of 0°), 10,900 kg at 1 km^2/s^2, and 4200 kg at 50 km^2/s^2, respectively. A curve representing injected mass versus Earth escape energy has been used in the mission-level optimisation.

The SEP spacecraft employs a similar mission scenario to the NEP spacecraft-based mission, except that the power (and hence the thrust) varies according to the solar distance of the SEP spacecraft in order to account for solar flux intensity and cell working temperature effects on the power output of the solar array (see Fig. 2). After a distance of 5 AU, when practically all propellant has been depleted and there is negligible

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thrust generated, the solar array is jettisoned to allow the small scientific probe payload (similar to the 213 kg probe described in [8]) to reach 200 AU with reduced attitude control demands. Analyses were performed for both Ariane 5 ECB and Soyuz–Fregat launch vehicles for Earth escape.

5. Mission performance optimisation

The main objective of the analysis is to find the optimum launch energy, spacecraft system design (payload, power, propellant), ion thruster performance (optimum specific impulse, thrust) and trajectory (thrust vectors) that gives the minimum time to reach 200 AU. A simple analytical model is used in order to derive the main spacecraft parameters of interest to the trajectory optimisation software, namely the power generated and the thrust produced. The power to be generated by the power subsystem can be expressed by the formula:

$$P = \frac{M_{p+p}}{(\alpha_{thr}/\eta_{ppu} + \alpha_{ppu})N_{T,per} + \alpha_{pow}} \tag{2}$$

where α_{thr}, α_{ppu}, and α_{pow} are the specific masses (kg/kW) of the thruster, PPU (thruster power processing unit) and power system, respectively. η_{ppu} is the efficiency of the PPU and $N_{T,per}$ is the normalised thrust factor at the minimum perihelion distance. $N_{T,per}$ is used in the SEP case to size the propulsion system to accept the higher power available at the spacecraft's closest distance to the Sun, if perihelion is < 1 AU. For the NEP case, $N_{T,per} = 1$ since power and thrust are constant and independent of solar distance. M_{p+p} is the power and propulsion system mass available after accounting for the propellant, payload and other subsystem masses, and is given by

$$M_{p+p} = (M_{wet} - M_f)(1 - m_{s/s} - m_{p/l}) - m_{\tan} M_f \tag{3}$$

where M_{wet} is the spacecraft wet mass, $m_{s/s}$ is the mass fraction of other subsystems (ADCS, structure, avionics etc), $m_{p/l}$ is the scientific payload mass fraction, and $m_{\tan}$ is the tank mass fraction. Finally, M_f is the fuel mass varied by the trajectory optimisation code. Once the power has been calculated, then the thrust produced by the thruster can be determined by

$$T = \frac{2\eta_{thr}}{g_o I_{sp}} \eta_{ppu} P \tag{4}$$

where η_{thr} is the thrust efficiency, which (as the PPU efficiency) we assume to be constant. g_o is specific gravity at Earth surface. Hence, the thrust is entirely dependent on power available and the specific impulse of the thruster (varied by the trajectory optimisation code). The thrust, propellant mass and specific impulse, total spacecraft mass and launch escape energy C3 are used to determine the trajectory during the continuous thrust phase until the burnout condition, and then the trajectory is propagated to 200 AU. Together they determine the transfer time. These parameters are varied (for a specific spacecraft design) to find the optimum specific impulse and fuel mass fraction that gives the minimum transfer time. Here, the subsystem specific masses are $\alpha_{thr} = 0.25$ kg/kW, α_{ppu} and α_{pow} are input parameters varied during the analysis. EP system efficiencies are$\eta_{ppu} = 0.95$ and $\eta_{thr} = 0.7$, respectively. Subsystem mass fractions are $m_{s/s} = 0.28$, $m_{\tan} = 0.15$, and $m_{p/l}$ is another input parameter in the analysis.

6. Mission optimisation results

6.1. NEP interstellar probe

The results of the mission optimisation can be seen in Figs. 3–6. Fig. 3 shows the time to 200 AU and related optimised values as a function of spacecraft mass for a power system specific mass of 25 kg/kW, and payload mass of 400 kg. The power specific mass is quite an optimistic assumption, but is achievable if high efficiency power conversion technologies such as Brayton cycle turbines are developed with over 20–30% efficiency. Initial conditions for the trajectory, i.e. the C3 launch energy vary according to the spacecraft mass due to Ariane 5 ECB capabilities, and the thrust is assumed as always tangential. We can see that all spacecraft wet masses between 4000 and 11,000 kg give similar times to 200 AU of 23–24 years, with 8000 kg being the optimum. The optimum values at this point are $I_{sp} = 10{,}212$ s, time to 200 AU = 23.03 years, propellant mass = 3914 kg, $T = 0.9$ N, power = 64.64 kW. The trajectory for this optimum is presented in Fig. 4.

Fig. 5 provides the variation of minimum time to 200 AU with the scientific payload mass on a spacecraft of 8000 kg wet mass. As we might imagine, this has non-negligible impact on the time to 200 AU. Increasing the payload to 1400 kg still allows the mission to complete within 30 years, 7 years longer than with 400 kg payload. Reducing the payload to 200 kg only reduces the transfer time by 1 year. For larger payloads, the optimum specific impulse increases towards 12,000 s, still well within the capabilities of the DS4G.

Finally, Fig. 6 displays the variation of minimum time to 200 AU and optimum specific impulse with the power system specific mass for an 8000 kg spacecraft with 400 kg payload. Specific mass has a strong impact

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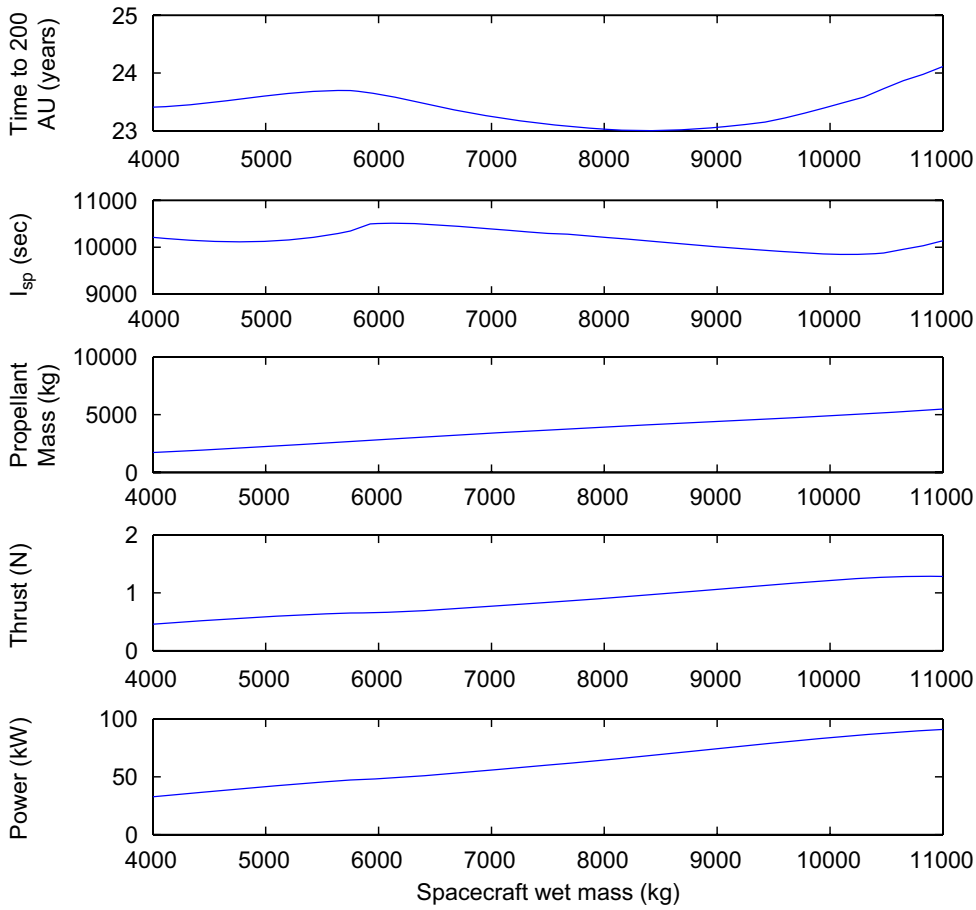

Fig. 3. Variation of optimum mission design parameters with spacecraft mass (assuming power system specific mass of 25 kg/kW and payload mass of 400 kg).

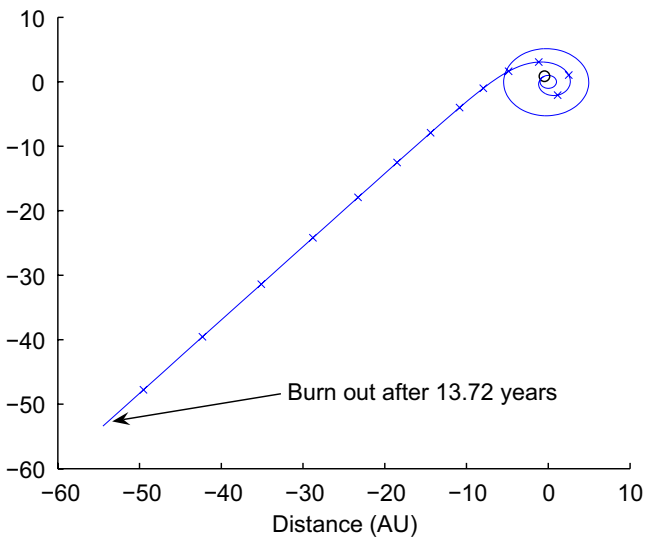

Fig. 4. The optimal trajectory is shown up to the burnout. The crosses are per year.

on both the transfer time and I_{sp}. For heavier nuclear reactors with values approaching 50 kg/kW (present day performance), we see that time to 200 AU increases to 32 years and optimum I_{sp} down to below 8000 s (within the upper limits of 3-grid thrusters). This is also the reason why the DS4G is not compatible with smaller RTG-powered missions, since RTGs have a specific mass of the order 212 kg/kW. If specific masses of order 10 kg/kW are used (very optimistic), then the trip time would reduce to as low as 18 years and optimum I_{sp} rises to over 12,000 s. Hence, nuclear reactor technology development would be warranted for interstellar probe missions.

6.2. SEP interstellar probe

The results of the mission optimisation are displayed in Table 2. Twelve different optimisation cases have been considered for Ariane 5 and Soyuz launchers, for missions to 40 AU (Kuiper Belt Objects) and 200 AU (VLSIM), and for different technology assumptions on the specific masses of the power subsystem and the ion thruster PPU. In particular, for the latter, three different sets of specific masses are considered according to near-, medium- and long-term future developments in

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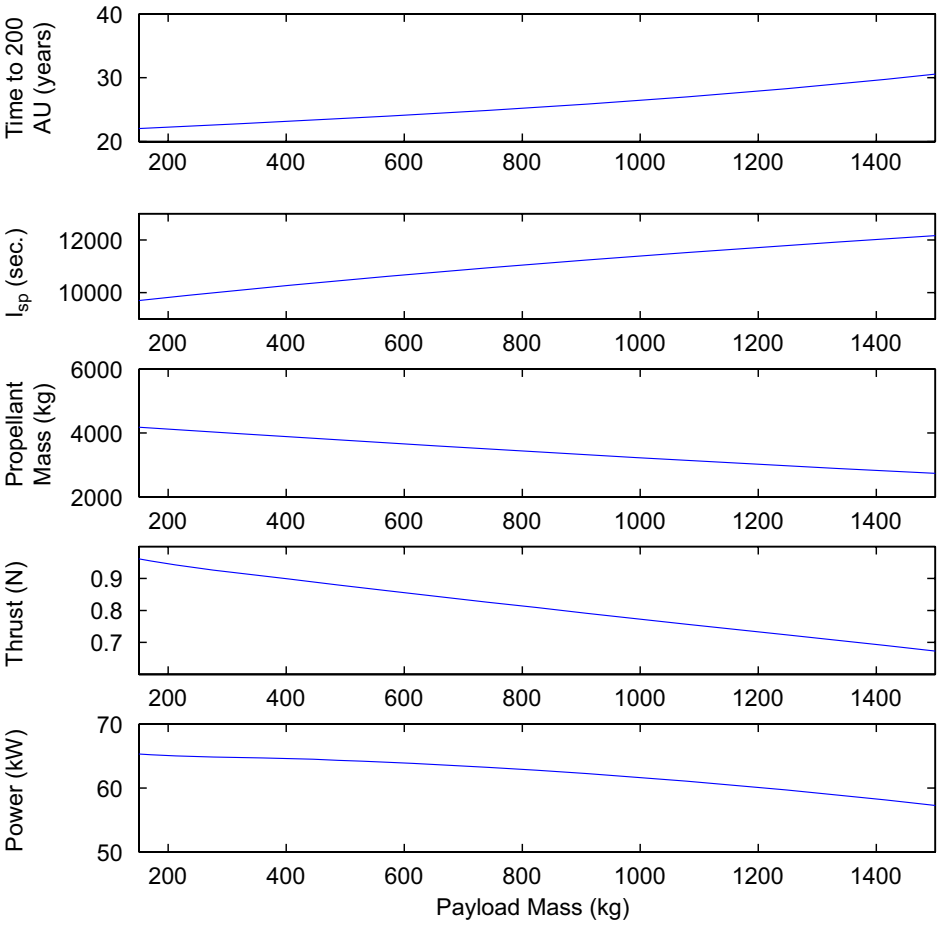

Fig. 5. Variation of optimum mission design parameters with payload mass (assuming power system specific mass of 25 kg/kW and spacecraft mass of 8000 kg).

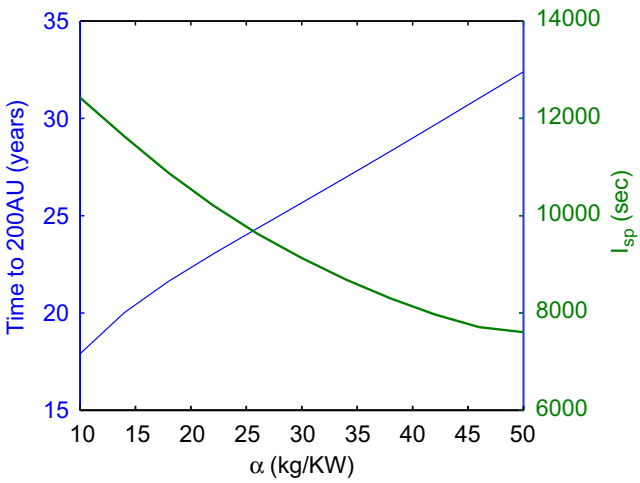

Fig. 6. Variation of time to 200 AU and optimum specific impulse with power system specific mass (assuming payload mass of 400 kg and spacecraft mass of 8000 kg).

those technologies. The long-term case is very ambitious for both power and PPU, and such specific masses are only likely to be realisable with the introduction of new materials and very lightweight structures for solar arrays such as inflatable rigid booms, and high temperature electronics for the voltage conversion in PPUs. Nonetheless, we were interested in assessing the possible mission-optimised performance in order to see the possible benefits of these developments on the transfer time to 200 AU. For these results, again the thrust was assumed to be tangential, i.e. along the spacecraft velocity vector, and this time the initial flight path angle γ at Earth departure was taken as another control variable in the optimisation process to minimise the transfer time. This resulted in optimal trajectories that initially went inside 1 AU to a minimum solar distance of 0.7–0.8 AU in order to pick up a higher power and hence thrust a little closer to the Sun. The minimum solar distance could not go lower than this since it was constrained by the departure C3 of the launch vehicle.

We can see from Table 2 that a mission to 200 AU could be accomplished within 27 years using near-term developments for power and PPU (2 and 5 kg/kW, respectively) and an Ariane 5 ECB launcher. In this

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Table 2
Comparison of optimal solutions for high power SEP missions to the outer solar system and beyond using different launchers and different assumptions for power and PPU specific masses.

Launch	Distance (AU)	α_{pow} (kg/kW)	α_{ppu} (kg/kW)	I_{sp} (s)	Thrust (N)	Power (kW)	$M_{s/c}$ (kg)	M_f (kg)	γ (rad)	$T_{transfer}$ (years)
Ariane 5	200	2	5	3244	5.37	122	3683	1799	0.45	27.06
		2	3.7	3508	6.32	155	3892	1926	0.54	25.17
		1	1	5578	10.93	427	4033	1995	1.15	17.9
	40	2	5	3233	2.76	62.5	1924	833	0.34	5.53
		2	3.7	3519	3.07	75	1924	836	0.37	5.2
		1	1	5354	5.23	196	1924	843	0.62	3.88
Soyuz–Fregat	200	2	5	3995	1.94	54.37	1721	732	0.78	38.4
		2	3.7	4274	2.27	68	1778	767	0.82	34.25
		1	1	6026	4.23	178.5	1785	781	1.1	21
	40	2	5	3768	1.97	52.1	1682	718	0.75	7.46
		2	3.7	4131	2.31	66.8	1735	740	0.81	6.75
		1	1	5887	4.36	180	1798	781	0.98	4.5

case, the optimum specific impulse for the mission was determined to be just over 3200 s, which can easily be achieved using current gridded ion thruster technology. The DS4G concept operates in the range of 8000–20,000 s, so would not be suitable for such a mission. The power requirement for the EP system at 1 AU is 122 kW in order to provide 5.4 N of thrust, which could be met by four 30 kW thrusters operating simultaneously in a cluster arrangement. When the power and propulsion system alphas are reduced for this mission scenario, then we can see that much shorter flight times can be achieved (18 years for the long-term technology developments) with a growth in spacecraft wet mass, a lower departure C3, and a consequent increase in optimum specific impulse to nearly 5600 s. Propellant mass fraction remains almost constant, but the power and hence thrust increase dramatically due to the smaller alphas, which leads to the shorter transfer time to 200 AU. The optimum specific impulse does not increase dramatically from its low level, most probably because the thrust duration is shorter than NEP and thrust magnitude is tailing off rapidly with solar distance, so most of the thrusting is performed close to the Sun.

When optimising the combined trajectory and spacecraft design and EP system performance for a 40 AU mission using Ariane 5, it is observed that this leads to a much lower spacecraft and propellant mass than the 200 AU cases, although the propellant mass fraction remains very similar. This is because the shorter distance leads to a minimum transfer time solution that is more favourable to much higher launcher departure C3. (with the effect of reducing total launch mass of course). In these cases, transfer times of 4–5.5 years to 40 AU are achievable, depending upon the power and PPU technology. The total power requirement is also driven to lower values of 60–200 kW. This dramatic reduction in spacecraft wet mass when reducing the target distance from 200 to 40 AU does not occur with the Soyuz–Fregat cases because the total launch mass of the Soyuz is much lower than Ariane 5 for the same C3 launch energy. This means that is not possible to move to a higher C3 (and hence a lower launch mass) with Soyuz, due to the fact that there is a fixed payload mass of 213 kg for the interstellar science probe which would drive down the power and propulsion dry mass to unacceptably low levels for the transfer time.

For the Soyuz–Fregat cases, it is possible to perform a 34-year mission to 200 AU using medium-term power and PPU technology developments. With further aggressive reduction in alphas in the longer-term, it might be possible to reduce the transfer time further to 21 years with a Soyuz. Optimum specific impulses are very similar to the respective values produced for the Ariane 5 cases, given the similar trajectory profile and delta-V requirement, and the same alphas used. For 40 AU missions, the Soyuz could allow transfer times in the order of 4.5–7.5 years, again depending on the alphas employed.

In additional optimisation was performed in order to investigate the effects of removing the constraint of the tangential thrusting on the combined trajectory+SEP spacecraft optimal design solution. This was a challenging step, since the optimisation software had to be redeveloped in order to accept a varying thrust angle as an additional optimisation parameter. This was

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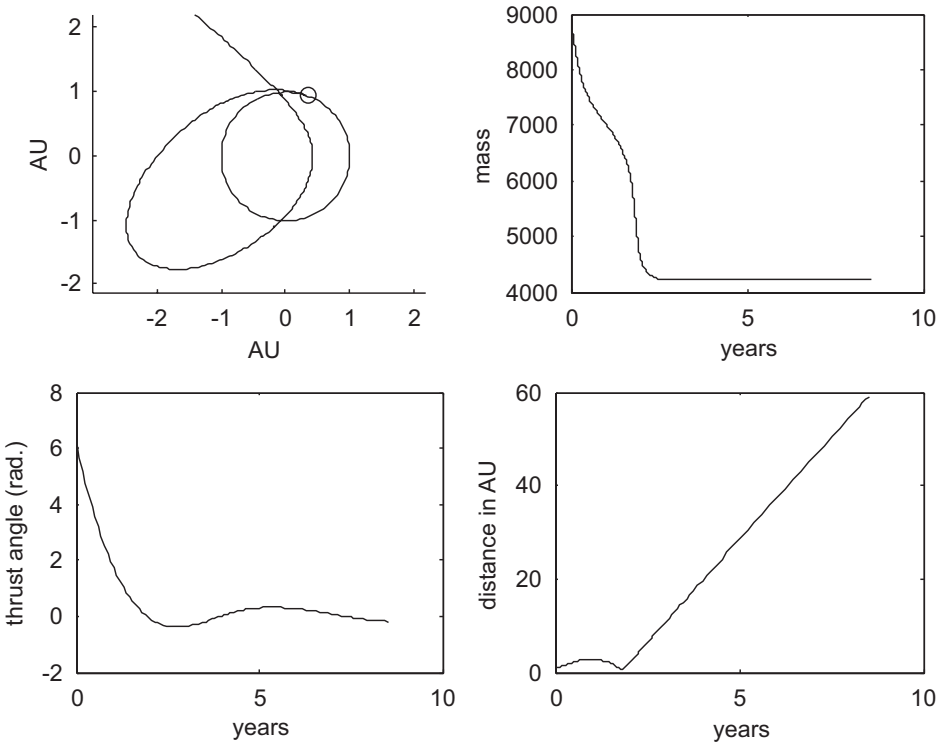

Fig. 7. Optimal solution from the system design optimisation process (optimal trajectory+SEP spacecraft design), based on an Ariane 5 ECB Earth escape launch, and specific masses of alpha_power = 2 kg/kW and alpha_ppu = 3.7 kg/kW; the SEP spacecraft reaches 200 AU in 24.6 years.

successfully performed, however, and the results are shown in Fig. 7 for the Ariane 5 ECB launch case for a 200 AU mission with alphas of 2 and 3.7 kg/KW for the power and PPU, respectively. As we can see, the transfer time was cut from 25.2 to 24.6 years due to the thrust angle variation. The optimum design parameters were found to be:

$I_{sp} = 5400$ s
Propellant mass = 4696 kg
Spacecraft wet mass = 8925 kg
Thrust time = 2.52 years
Minimal solar distance = 0.42 AU
Starting flight path angle = −0.27 rad
Departure C3 = 11.78
Power = 288 kW
Thrust (1 AU) = 7.61 N

The largest impact on the optimised solution of relaxing the thrust angle constraint is a different trajectory strategy which initially thrusts out to 2 AU to achieve an eccentric orbit around the Sun which brings the SEP spacecraft closer to the Sun at 0.4 AU minimum distance (instead of 0.8 AU previously). This closer solar flyby allows the maximum power and thrust to be much higher (288 kW compared 155 kW). An additional consequence of this more optimised strategy is to also increase the optimum specific impulse of the EP system to 5400 s instead of 3500 s. It is expected that for lower values of alpha, the optimum specific impulse would be much higher than this.

7. Conclusions

Preliminary technology assessment and mission optimisation studies suggest that significant benefits for local interstellar probe missions to 200 AU and KBOs or inner Oort cloud objects can be obtained by employing the Dual-Stage 4-Grid ion thruster concept for NEP spacecraft. Such benefits include reduced trip time (7 years compared to previous studies), increased payload mass (several hundreds of kg may be available), lower C3 launch energy (16 km^2/s^2 delivered by Ariane 5 instead of 140 km^2/s^2), and wider launch windows/lower radiation dose (no close Jupiter swing by needed). However, the development of a low specific mass space nuclear reactor would be required. An optimum design solution was found for an 8000 kg spacecraft wet mass, 400 kg payload, with 65 kW reactor and DS4G operating at 10,200 s specific impulse and thrust of 0.9 N. A single 25 cm diameter DS4G thruster with a beam

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potential of 13,000 V may be sufficient for these requirements.

It is also found that an interstellar probe mission to 200 AU could be accomplished in 25 years using a SEP system with a lightweight 155 kW solar array (2 kg/kW specific mass) and thruster PPU (3.7 kg/kW), and an Earth escape launch on Ariane 5 ECB. In this case, the optimum specific impulse is lower at 3500 s which is well within conventional gridded ion thruster capability. In this scenario, the DS4G thruster concept is found not to be a suitable technology. However, such as mission could be achieved in the "medium-term" future with development of lightweight solar power generation and conversion systems. This approach may be less expensive than the development of a lightweight nuclear reactor. In both NEP and SEP spacecraft designs considered, the payload capacity would be sufficient to deliver a 200 kg scientific probe to either 40 AU for Kuiper Belt Object flybys or 200 AU for local interstellar in situ characterisation.

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This paper is published in the memory of David Fearn, who sadly died on 29 August 2007 before being able to see this paper published. He will be greatly missed by the co-authors, his many friends, collaborators and colleagues in the electric propulsion community to which he gave so much.

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