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THE APPLICATION OF ION PROPULSION TO THE TRANSPORTATION AND CONTROL OF SOLAR POWER SATELLITES

by

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SUMMARY

A common feature of all proposed solar power satellites is their enormous mass, perhaps approaching 100000 tonnes for a 10 GW version. The methods of transporting such masses to geostationary orbit are reviewed. It is concluded that electric propulsion techniques offer very considerable technical and financial advantages, and that ion thrusters currently represent the most suitable technology to employ. It is also shown that the use of ion propulsion for attitude and orbit control would be of great benefit. An advanced form of ion thruster, which offers a very high beam velocity and current density, is proposed for these applications.

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LIST OF CONTENTS

		Page
1	INTRODUCTION	3
2	TRANSPORTATION FROM LEO TO GEO	4
	2.1 Chemical propulsion	4
	2.2 Self-propulsion to GEO	5
	2.3 EP orbit transfer vehicle	6
3	THE CASE FOR ELECTRIC PROPULSION	7
	3.1 Orbit transfer mission	9
	3.2 Orbit and attitude control	10
4	CHOICE OF THRUSTER SYSTEM	11
	4.1 Kaufman thrusters suitable for the SPS application	12
	4.2 An alternative design approach	15
5	CONCLUSIONS	18
References		19
Illustrations		Figures 1-10
Report documentation page		inside back cover

1 INTRODUCTION

A major problem facing the world today is the finite size of the fossil fuel reserves still available for exploitation. At the present rate of consumption, most of the readily accessible gas and oil will be exhausted by the end of the century, and coal, although relatively plentiful, will be increasingly difficult to mine. It is also relevant to point out that, even if fossil fuels were still abundantly available, it might not be possible to use them at an ever increasing rate, owing to the climatic dangers inherent in allowing the carbon dioxide content of the atmosphere to continue to rise unchecked[1].

The major alternatives are nuclear energy and the so-called renewable resources, which include solar radiation. Unfortunately, the latter is heavily attenuated by the atmosphere and by clouds, and is only available during day-time. Consequently, Glaser[2] proposed placing a solar power satellite (SPS) in earth orbit to collect continuously the full flux of radiation. Using either photovoltaic or heat engine technology to produce electricity, the power generated would then be transmitted to a ground rectenna via a microwave beam or, in the more distant future, perhaps by a laser system. This proposal has been very extensively studied[3-7], with various alternative technologies being considered. No technical or environmental reasons have been found for eliminating the SPS as a future valuable source of energy, including its application in Europe[8].

Assuming that a 2.45GHz microwave power transmission system is adopted for reasons of efficiency and safety, it has been generally concluded that the physics of the transmission process enforce a lower limit on the power emitted from each orbiting antenna of about 5 GW. This immediately dictates that the dimensions and mass of each SPS be enormous, typical values being 5-20 km and 20000-100000 tonnes[3-5]. As a specific example, the reference designs adopted by NASA and the US Department of Energy[5] (Fig 1) will each produce 5 GW at the output terminals of the ground rectenna system. The planar silicon solar array option has an estimated mass of 51000 tonnes and that using GaAlAs technology 34000 tonnes.

With masses and dimensions as large as these, it is evident that the cost and efficiency of the space transportation techniques adopted will have a large, perhaps dominant bearing on the economic viability of any SPS project[3,4]. This transportation task can be broken down into two phases, surface to low earth orbit (LEO) and LEO to geostationary earth orbit (GEO), and in each of these the movement of both goods and personnel must be considered. This paper is concerned with the LEO to GEO phase, and, in particular, with the application of electric propulsion (EP) to this and to the attitude and orbit control (AOC) of an SPS in geostationary orbit.

It is shown that, irrespective of the type of SPS considered, it is far more cost-effective to transport all goods, materials, components and so on to GEO using EP, and that ion thrusters offer, at present, the greatest advantages. The same conclusions apply to the AOC functions.

2 TRANSPORTATION FROM LEO TO GEO

As the dimensions and masses of all proposed SPS designs are so large, construction in space from individual components and materials will be mandatory. Fundamentally, it is possible to construct the spacecraft in either LEO or in GEO, but an additional option is also available, that of partial construction in LEO. In every case, it will be necessary to deploy large teams of workers in space, together with extensive manufacturing and assembly facilities, and living accommodation for the crews[3,6]. If electric propulsion is used for transporting everything but construction crews and urgently required supplies to GEO, all possible options will remain open. This is not so if conventional rockets are employed, because the relatively flimsy SPS structures will not be able to withstand the thrusts involved during orbit transfer, unless greatly strengthened; only GEO construction will be feasible, unless a large mass penalty is to be incurred.

Using EP, two basic concepts have been proposed, and these are considered in more detail below. Possibly the most attractive is to manufacture large sections of the satellite in LEO, then use part of the electrical output from each section to power EP modules, which are attached for the orbit transfer mission[3,6]. Final assembly of the SPS would then take place in GEO, where only a small astronaut crew would be needed. The alternative is to transport all components and materials to GEO for manufacture there, using a special-purpose cargo orbit transfer vehicle (COTV)[5,9].

It would also be possible to construct the complete SPS in LEO, then use EP to propel it to GEO. However, this would not be cost-effective, owing to the much higher gravity gradient forces in the lower orbit, which would require the structure to be some 200 times stronger than necessary for its ultimate purpose[7]. In addition, the mass of attitude control propellant consumed during orbit transfer would be much greater for a complete spacecraft, because the propellant needed varies as the cube of the length of the structure[9].

It should be mentioned in passing that a nuclear power source is a very attractive proposition for the orbit transfer mission[10]. It can be used either to generate electricity to power an EP system[11], or to produce thrust via a thermally heated gaseous exhaust. Unfortunately, although technically feasible, such proposals are unlikely to be adopted, owing to opposition on safety and political grounds.

2.1 Chemical propulsion

There is no doubt that chemical propulsion, in its present-day form, could be used to undertake the orbit transfer mission. Indeed, where short transfer times are mandatory, as is the case with personnel, there is no alternative. In such cases, orbit transfer times of the order of 10 hours may be achieved by spacecraft employing derivatives of Space Shuttle propulsion technology[6].

Much larger versions of the personnel vehicle have been proposed, carrying payloads of up to about 400 tonnes[12]. Almost all designs use LH_2/LO_2 rocket motors, with specific impulse (SI) limited to below about 470 seconds, and it is generally concluded that about 1.9 to 2.1 kg of propellant are needed per kilogramme of payload to complete the mission. A careful analysis of published estimates suggests that the surface to LEO transportation will cost \$16 to \$33 per kilogramme, at 1978 prices. Thus the propellant for the COTV will cost \$32 to \$66 per kilogramme of payload; to this must be added all other charges, such as refurbishment, return on capital, and so on, which may be about \$11 per kilogramme[12].

However, these studies may have been over-optimistic, in that more recent work[13], assuming the use of motors with an SI of up to 475 seconds and specially developed lightweight structure (Fig 2), has suggested total costs of \$120 to \$180 per kilogramme of payload. The lower value refers to the higher payload versions of about 200 tonnes (Fig 3), with a surface to LEO cost of \$50 per kilogramme.

It may be concluded that, depending upon the financial assumptions made, the propellant costs usually dominate LEO to GEO transportation prices. Consequently, any method of reducing the amount of propellant consumed is to be welcomed; this is the reason for turning to EP, as covered more fully in section 3.

2.2 Self-propulsion to GEO

Possibly the most detailed investigation of the self-propulsion technique has been carried out by Boeing[3], with reference to their 10GW SPS design which uses a planar array of silicon solar cells without concentrators. In effect, one half of this forms the upper reference design depicted in Fig 1. It is proposed that the spacecraft would be constructed in eight separate sections, each of equal length, but two having 5GW microwave antennas attached. An array of ion thrusters placed at each corner, as shown in Fig 4, would provide the propulsion. Each array would be gimballed about two perpendicular axes for thrust vector control during orbit transfer[3,6]. It is also proposed that LO_2/LH_2 thrusters should be carried to provide attitude control, but it would be better to use ion thrusters for this purpose, with power being derived from batteries during eclipse[4].

The characteristics of the self-propelled sections are given in Table 1, where reference is made to both the Boeing 120cm diameter thrusters[6,14] and an uprated version providing higher SI and greater thrust density. It will be seen that the increased SI drastically reduces the propellant requirements, at the expense of a much larger power consumption. At an optimistic earth to LEO transportation cost of \$20 per kilogramme, this will save over \$200M per SPS, and the elimination of the chemical system would save a further \$220M. The increased thrust per thruster allows a major reduction in system complexity to be achieved, the number of devices needed being a factor of 10 below that originally proposed. In its original form, the overall propellant cost of the concept is about 0.35 kg per kilogramme of payload, including the LO_2/LH_2. This is reduced to 0.25 kg at the higher SI and to about 0.16 kg for an all-EP system.

Table 1

Characteristics of Boeing 10GW SPS self-propelled sections

Thruster type	Boeing 120 cm thrusters[6,14]		Uprated 120cm thrusters*	
Type of SPS section	Non-antenna	Antenna	Non-antenna	Antenna
Number of sections	6	2	6	2
Transfer time (days)	160	180	180	180
Mass (10^6 kg)	8.7	23.7	8.7	23.7
Total thrust (N)	4500	12200	4500	12200
Specific impulse (seconds)	7000	7000	13000	13000
Thrust per thruster (N)	1.89	1.89	18.7	18.7
Number of thrusters	2400	6400	240	652
Thruster power (kW)	125	125	1680	1680
Total power (MW)	300	810	403	1095
Per cent of solar array used	13	36	18	50
Mass of argon propellant (10^6 kg)	2.0	5.6	1.08	3.02
LO_2/LH_2 propellant (10^6 kg)	1.0	2.8	1.0	2.8

* Based on design in Ref 9.

Several variations have been suggested, such as splitting an SPS into smaller 6 × 1km sections, each propelled by two banks of thrusters[14]. In almost every case, however, large argon ion thrusters have been adopted, operating at high values of SI. In general, it has been assumed that radiation damage to solar arrays can be repaired by an annealing process[5,6], and that the thrusters will be employed in the AOC role once GEO is reached.

2.3 EP orbit transfer vehicle

The technology referred to above can be utilised directly for propelling a COTV having a very impressive payload capability. Moreover, SPS manufacturing processes and facilities can be immediately transferred to the task of constructing the vehicle. The characteristics of four of the published designs are summarised in Table 2, from which it can be seen that remarkable payload to empty mass ratios can be achieved, the design described in Refs 4 and 15 and depicted in Fig 5 reaching a value of 5. The propellant mass for the complete mission can be as low as 0.07 kg per kilogramme of payload, for an SI of 13000 seconds[9].

Other EP powered COTV designs have been proposed, ranging in capability from a few tonnes payload to a huge device able to transport an SPS in a single trip[9]. They all provide very large cost benefits compared to chemically propelled vehicles, particularly when operated at high SI.

Table 2

Summary of electrically propelled COTV proposals

Origin and reference	Caluori, *et al*[16]	Weddell, *et al*[9]	DOE/NASA[5]*	Hanley[4,15]
Length (km)	0.566**	2.25	1.14	1.70
Width (km)	0.099	1.30	1.14	1.30
Height (km)	~0.022	1.13	0.164	0.563
Empty mass (10^6 kg)	0.055	1.70	1.10	1.09
Payload (10^6 kg)	0.227	4.27	4.00	5.17
Payload/empty mass	4.13	2.51	3.64	4.74
Solar cell type	Si	GaAlAs	Si	GaAlAs
Concentration ratio	1	2	1	~2
Solar array area (km^2)	0.054	0.90	>0.92	0.90
Solar array power (MW)	~8.7	384	>150	336
Thruster diameter (cm)	50	100	120	76
Thruster thrust (N)	0.70	13	0.189	69.7
Number of thrusters	206	268	1184	144†
Specific impulse (seconds)	8000	13000	7000	8213
Mission time, LEO-GEO-LEO (days)	180	180	160	150
Argon propellant mass for mission (10^6 kg)	0.03	0.30	0.985	0.667

* Assuming use of 120cm thrusters in Refs 6 and 14
** Array only
† Including 25% spares

3 THE CASE FOR ELECTRIC PROPULSION

Over the past 10-15 years, the use of EP for a wide variety of missions has been strongly advocated, usually on the grounds that significant quantities of propellant can be saved, allowing costs to be reduced or mission capabilities to be enhanced. However, owing to the complexity of most EP systems and to the lack of recent space demonstrations of their advantages, no operational flights have so far taken place. This reluctance to accept a new technology, which can offer enormous benefits, may well change in the near future, because those benefits will increase dramatically as spacecraft masses increase and as more electrical power becomes available.

The advantages of EP arise because the SI of a conventional rocket is limited by the energy available in the chemical reactions producing the exhaust gases. Here the SI, I_{sp}, is defined by

$$I_{sp} = T/\dot{m}g_0$$

where T is the thrust and $\dot{m}$ is the total rate of use of propellant, expressed in terms of units of sea-level weight. The SI is unlikely to exceed 475 seconds, even using advanced LO_2/LH_2 systems[13].

No such limit applies when electrostatic or electromagnetic forces are used to accelerate electrically charged propellants. Theoretically, an exhaust velocity v_e approaching the velocity of light is attainable, assuming an adequate power level can be provided. To take a particular example, a Kaufman ion thruster has a specific impulse

$$I_{sp} = \eta_m v_e / g_0$$

where η_m is the propellant utilisation efficiency and g_0 is the acceleration due to gravity at sea level. Assuming that all beam ions are singly charged,

$$v_e = (2eV_T/m_i)^{\frac{1}{2}}$$

where V_T is the net accelerating voltage, e is the electron charge, and m_i is the ion mass.

Using the above expressions, for argon propellant, v_e and I_{sp} have been plotted against V_T in Fig 6. It has been assumed that $\eta_m = 0.88$, a value representative of advanced thrusters[17], taking into account the presence of doubly-charged ions. It can be seen that it is simple to achieve an SI which is a factor 20 greater than attainable by chemical means ($V_T \sim 2.3$ kV), an increase of a factor of 30 is readily achievable ($V_T \sim 5.0$ kV), and much higher values should be accessible. As an indication of possible trends, ion accelerators designed for controlled thermonuclear reaction (CTR) experiments have achieved acceleration potentials of 70-110 kV[18-19]; the upper value represents, for argon, an SI of more than 65000 seconds, or a factor 138 greater than the best predicted for chemical systems.

Each manoeuvre undertaken by a spacecraft can be characterised by the total impulse I required, which is equivalent to a definite velocity increment ΔV, assuming that the spacecraft's mass M remains effectively constant. Thus

$$I = \Delta VM \ .$$

By equating this to the momentum transferred to the spacecraft by the propulsion system, it follows that the mass of propellant consumed is

$$M_p = \frac{\Delta VM}{g_0 I_{sp}} \ . \tag{1}$$

This expression confirms that enormous propellant savings are possible by using EP. For example, it was mentioned in section 2.1 that about 2 kg of propellant are required per kilogramme of payload for a chemically propelled COTV. Increasing the SI by a factor of 30, for which $V_T \sim 5.0$ kV, reduces this to about 0.07 kg per kilogramme of payload, representing a considerable financial gain.

3.1 Orbit transfer mission

Although equation (1) can be used to determine M_p when EP is used, this expression does not yield accurate results if the SI is lowered towards the values appropriate to chemical rockets, because the mass of the vehicle then changes appreciably as the mission progresses. A more exact analysis gives[20]

$$\frac{M_f}{M_0} = \frac{M_f}{M_f + M_p} = \exp\left[\frac{\mu^{\frac{1}{2}}}{g_0 I_{sp}}\left(\frac{1}{a_f^{\frac{1}{2}}} - \frac{1}{a_0^{\frac{1}{2}}}\right)\right] \qquad (2)$$

assuming that only single-stage vehicles are considered. Here M_0 and M_f are the initial and final masses, a_0 and a_f are the semi-major axes of the initial and final orbits, and μ is the gravitational constant of the earth (3.986×10^{14} m^3/s^2). This expression applies to equatorial orbits and ignores the relatively small effects due to residual air drag and eclipses. It simplifies to:

$$M_p = M_f \beta \ ,$$

where β is a function of I_{sp}.

Equation (2) has been used to derive the values of M_p and β plotted against I_{sp} in Fig 7. It can be seen there that the propellant mass required rises dramatically as SI falls below about 5000 seconds and that it is prohibitively large for I_{sp} = 400-500 seconds.

As a check on these results, equation (1) was employed to calculate values of M_p for an SPS having a final mass of 34000 tonnes (Fig 1), using ΔV = 5700 m/s, which is appropriate to an LEO altitude of 500 km and 31.6^{o} inclination[15]. The result is shown as the dashed curve in Fig 7; this curve is above that derived from equation (2) at high SI, owing to the inclined parking orbit but, at low SI, equation (1) grossly underestimates the propellant requirements.

The cost savings to be obtained by adopting EP are spectacular, as may be illustrated by reference to the 5GW Rockwell SPS concept[7], which employs GaAs solar cells and concentrators. It has been estimated that the use of EP in the construction of one satellite will save $1.2B, despite a very conservative assumption concerning the achievable SI. The very large reduction in propellant mass has a further important consequence; the number of heavy lift launch vehicle (HLLV) flights to LEO is very much reduced, from 1092 to 453[7] if construction is in GEO. As well as cutting costs, this will help to minimise any atmospheric pollution caused by launching operations. Similar results were obtained by Davis[21] for the Boeing 10GW SPS; it was found that the use of EP approximately halved the number of launch vehicles required.

It should be mentioned that all the orbit transfer options discussed in which EP is used assume that solar cell degradation caused by the impact of high energy particles[20]

043

can be eliminated by annealing. This can be done by operating GaAlAs cells at high temperatures, using concentrators[7,15], or by subsequent heating of Si cells[6].

3.2 Orbit and attitude control

The AOC functions are vital for the successful operation of an SPS, and they require a significant propulsive effort, owing to the enormous mass of the spacecraft and the substantial disturbing forces. The largest force is that due to solar radiation pressure on the huge area of the satellite; if uncorrected, this can produce a $\pm 2.5^{\circ}$ cyclical change in longitude with a period of 1 year, which is unacceptable[22]. This is unlike the situation with conventional spacecraft, where the greatest perturbation is that due to solar/lunar gravitational forces, necessitating north-south station-keeping (NSSK)[17]. Of smaller magnitude still is the disturbance caused by the triaxiality of the earth's gravitational field, which requires east-west station-keeping (EWSK).

Owing to the fact that these forces and the gravity gradient apply torques to the SPS, tending to rotate it, it is necessary to control its attitude at all times, to ensure that the solar array is approximately perpendicular to the sun's rays and that the microwave antenna is properly directed towards the ground rectenna. The latter must be accurate to 0.05°, with the phased array being used for vernier control of the beam, and the former should have a precision of about 0.5°, for a concentration ratio of 2[22].

If dealt with separately, these attitude control requirements would demand dedicated thruster systems, or alternative complex arrangements, such as spin or gravity gradient stabilisation, solar pressure vanes or large momentum wheels. However, a thruster system installed to provide station-keeping can, if properly mounted and gimballed, also carry out the attitude control task, with very little additional expenditure of propellant[22]. A throttling capability, either by switching thrusters on and off or by varying their thrust, will assist in providing fine control.

As an example of what can be achieved, the NASA/DOE baseline design using GaAlAs technology (Fig1) has been carefully analysed[22,23]. The recommended AOC system consists of four banks of thrusters, one at each corner. These operate continuously to counteract the force due to solar pressure; gimballing and differential throttling provide the remaining forces and torques for conventional NSSK, EWSK, and attitude control. The values of ΔV required annually are shown in Table 3, together with the propellant masses required to provide these, according to equation (1), using both EP and chemical propulsion. Cost savings resulting from the use of EP are also shown, assuming an LEO to GEO transportation charge of \$100 per kilogramme[6,13]; they are very substantial.

The thrusters selected in this study[22,23] were argon ion devices of 100cm diameter and producing an SI of 13000 seconds, with $T = 13$ N. Although only 36 are needed, it was decided to install 64 for reasons of redundancy and to assist servicing. Their mass is only 0.08% of M , and they consume 34 MW.

Table 3

Summary of AOC requirements for DOE/NASA baseline GaAlAs SPS design

Orbit perturbation	ΔV† (m/s per year)	Propellant mass**				Cost saving over SPS life* ($M)
		Annual (tonnes)		Over SPS life* (% of M)		
		Chem	EP	Chem	EP	
Solar pressure (E-W)	282.5	2061	75.3	181.9	6.65	5957
Solar/lunar gravitation (N-S)	53.3	389	14.2	34.3	1.25	1124
Earth triaxiality (E-W)	2.1	15.3	0.56	1.4	0.05	44
Microwave radiation pressure (E-W)	~0	~0	~0	~0	~0	~0
Station changes (E-W)	1.1	8.0	0.29	0.7	0.03	23
Totals	339.0	2474	90.40	218.3	7.98	7151

* Spacecraft mass = 34000 tonnes. Life ~30 years

** I_{sp} = 475 seconds for chemical systems, 13000 seconds for EP

† From Ref 22

As a more general result, Fig 8 shows the ratio M_p/M as a function of SPS life, for a variety of different propulsion systems. The advantages of ion propulsion are clearly evident.

4 CHOICE OF THRUSTER SYSTEM

Most of the EP systems studied during the past 10-15 years have been more suitable for low thrust applications, such as the station-keeping of moderately-sized communications satellites[17]. The thrusts of individual devices have tended to be low, in the 5-150 mN range, and only limited ability to scale up this factor has been demonstrated, apart from the clustering of many separate thrusters. The latter approach leads to great complexity and a very low thrust density. Consequently, devices such as the colloid thruster and the field-emission thruster can be rejected in this context, although they may be eminently suitable for other missions.

Of the higher thrust devices, the Kaufman ion thruster is the most advanced, and it shows considerable promise for the SPS application. Past experience indicates that it can be increased in size to at least 1.5m diameter[24], assisted by known scaling laws[25], and its SI can be raised to a very high value, certainly to 15000-20000 seconds, using argon propellant. The RF variety pioneered in Germany[26] can be made as large as 35 cm in diameter, but physical limitations probably restrict its development potential. The magnetoelectrostatic containment (MESC) thruster perhaps offers greater promise, but no experience yet exists of large diameter devices.

As at present designed and operated, all the above ion thrusters provide only low thrust density of the order of 0.2 to 0.3 mN/cm^2 of grid area. Much higher values can be achieved using the magnetoplasmadynamic (MPD) concept and, for this reason, the use of

043

MPD thrusters has been advocated for the SPS application[14]. A further advantage is that of relative mechanical and electrical simplicity. However, they are at an early stage of development, with only very short demonstrated lifetimes, and electrical efficiency, η_e, defined as kinetic energy in the exhaust divided by total power input, is low, typically[27] 20-30%. Megawatt power levels are necessary, causing severe electrode erosion and requiring quasi-steady pulsed operation, with the need to store electrical energy between pulses.

Despite present limitations, recent work has suggested that these devices offer considerable promise, particularly if the power level can be increased further. It has been shown[28], for example, that the major energy loss, heat transmitted to the anode following dissipation in the anode sheath, falls from 50% at 200 kW to 10% at 20 MW. Assuming that it will be possible to overcome a current limitation on SI, which is about 2000 seconds for argon, it should be feasible[14] to attain an efficiency of 60% at the 1 MW power level, with an SI of 5000 seconds. However, if 2000 seconds cannot be exceeded, an efficiency of below 40% is likely to be the limit.

At present, it is reasonable to conclude that the Kaufman ion thruster is the only serious contender for early SPS missions. It has significant advantages as regards development status, overall efficiency and durability, and its only drawbacks concern complexity and thrust density. As indicated below, the availability of large amounts of electrical power allows the latter disadvantage to be overcome for SPS applications, and electrical complexity can be reduced somewhat by the partial direct operation of the thruster from the solar arrays[9].

4.1 Kaufman thrusters suitable for the SPS application

The most highly advanced Kaufman thruster likely to be suitable in any way for an SPS mission is the 30cm device developed by NASA Lewis and Hughes RL[29,30]. The 1.5m diameter thruster[24] previously referred to, although potentially more appropriate, was never developed beyond the early experimental phase. In its standard form, using mercury propellant, the 30cm device has demonstrated a lifetime of approaching 10000 hours (416 days), which is more than adequate for the spiral orbit raising mission. As shown in Table 4, its design thrust is about 130 mN, with an SI of 2985 seconds, but these values can be increased[30] to 290 mN and 6298 seconds by raising both V_T and beam current I_B. With argon, these values would be 260 mN and 14270 seconds. The 35cm RF thruster[26] is included for comparison.

The equivalent design data for thrusters proposed specifically for the SPS mission are given in Table 5. The 120cm diameter concept due to Grim[14], which is depicted in Fig 9, is relatively conservative, in that the thrust and beam current densities are only slightly larger than the values currently being achieved. Conversely, the 60m thruster proposed by Byers and Rawlin[31] assumes that the mean current density J_B can be increased by a factor of about 6 and that the thrust density can be raised by a factor of 3 to 4. The 100cm design by Weddell *et al*[9] also assumes that considerable advances can be made, although less so in the area of mass utilisation efficiency. The aims characterised by the 60cm and 100cm designs do not seem to be unreasonable, given that the availability of power is no longer a severe restraint.

Table 4

Performance data from large ion thrusters

Thruster type	Existing NASA/Hughes 30cm diameter					Giessen 35cm diameter RF thruster*
	Standard	Large I_B	Large V_B	Large V_B and I_B*		
Propellant	Hg	Hg	Hg	Hg	Argon	Hg
V_T (kV)	1.1	1.28	5	5	5	1.5
I_B (ampere)	2	4	2	4	4	1.8
Mean current density (mA/cm^2)	2.8	5.7	2.8	5.7	5.7	1.9
I_{sp} (seconds)	2985	3400	6293	6364	14272	3360
Total power (kW)	2.6	6.0	10.4	20.9	20.9	3.27
Thrust (Newton)	0.13	0.29	0.29	0.58	0.26	0.14
Thrust density (mN/cm^2)	0.18	0.41	0.41	0.82	0.37	0.15
η_m	0.90	0.95	0.89	~0.90	~0.90	0.87
η_e	0.84	0.85	0.96	0.96	0.96	0.80
Reference	29	30	30	-	-	26

* Calculated data

Table 5

Performance data for advanced Kaufman argon ion thruster designs

Beam dimensions (cm)	60cm diameter	76cm diameter	100cm diameter	120cm diameter	100 cm × 150 cm
V_T (kV)	4.2	2.0	5.0	6.0	1.88
I_B (ampere)	96	1887	225	80	1500
Mean current density (mA/cm^2)	34.0	416.0	28.7	7.1	100.0
I_{sp} (seconds)	13000	8213	13000	16000	7763
Total power (kW)	421	4151	1175	500	~3120
Thrust (Newton)	6.4	69.7	13.0	5.8	56.3
Thrust density (mN/cm^2)	2.26	15.4	1.66	0.51	3.75
η_m	~0.90	0.82	~0.82	~0.89	~0.80
η_e	0.96	0.91	0.96	0.97	0.90
Reference	31	15	9	14	32

Conversely, the rectangular 100 cm × 150 cm concept[32] and the 76cm diameter device[15] require an enormous step forward in discharge chamber and cathode design and performance to achieve the quoted values of J_B ; in the smaller of these thrusters, the discharge current I_D is likely to be 3000-4000 A, and the discharge power P_D about 400 kW. These high values result partly from operating at low SI, whilst designing for large J_B and T .

Past experience suggests that thruster lifetime is limited by cathode and grid durability, particularly the former. Both diminish as I_D and I_B rise. Consequently, the large thrusters discussed here will, if conventional technology is employed, probably utilise many cathodes operating in parallel, as indicated in Fig 9. To achieve a stable and uniform discharge chamber plasma under these circumstances will require a significant amount of experimental and theoretical effort, particularly in the transition region between the plasmas within the cathode polepiece and in the main chamber[33]. It is essential that the primary electrons accelerated through this region are uniformly distributed and have identical velocity distributions everywhere around the baffle disc/polepiece annulus.

The extent of this problem can be gauged from the fact that the nominal emission current for the 30cm thruster cathode is 12 A; a lifetime of at least 18000 hours is available at this value[34]. Higher currents, of up to 20 A, have been achieved without difficulty, probably without significantly sacrificing durability. So it is reasonable to assume that up to 200 such cathodes might be needed in the discharge chamber of the 76cm thruster, presenting seemingly insurmountable mechanical and thermal problems. Even the 100cm thruster would need 40 to 50 cathodes, its discharge power and current being about 40 kW and 800-1000 A.

The cathodes so far developed and life-tested have operated on mercury vapour, although it has been shown that other gases may be successfully employed[35], perhaps with certain performance or durability penalties. However, few systematic investigations have been reported, so much work remains to be done. Emphasis must be placed on potential life-limiting areas, notably depletion of low work function material from the internal dispenser, orifice erosion, tip weld integrity, and resistance of the heater assembly to thermal stresses and adverse chemical reactions.

The other major area of development work should be concerned with thermal aspects of the thruster's design. Of particular importance here is the achievement of adequate dimensional stability of the large diameter grid system, despite wide temperature variations. On the other hand, the change from mercury propellant to argon will simplify the design somewhat, in that the requirement to avoid mercury condensation during start-up will be eliminated. Consequently, auxiliary heaters will not be needed and it may be possible to employ a propellant distribution system more closely tailored to the aim of high utilisation efficiency.

Although the overall thermal load on the discharge chamber will be much higher than in present devices, this should not present insurmountable materials problems. However, an elementary calculation suggests that the 40kW discharge chamber power quoted for the 100cm diameter device can be dissipated by radiation only if the outer walls of the chamber and the grids are at 600-900°C, depending upon the physical arrangement of the mountings, the temperature of the surroundings, and the emissivity of the surfaces. The situation will be much more severe in the 76cm device, which must dissipate 400 kW; it was assumed, in designing this[15], that a grid temperature of 1900 K would be permissible. It is evident that strenuous efforts will be necessary to radiate the discharge power away as effectively as possible, probably using heat pipe technology.

To develop large thrusters of advanced design, considerable investment will be needed in test facilities. Although the use of argon will eliminate the materials problems associated with mercury, such facilities will be very costly, mainly because the cryo-pumping systems currently employed to remove mercury vapour will no longer operate with liquid nitrogen. To achieve the necessary high vacuum conditions, at least down to 10^{-7} torr, it will thus be essential to rely upon very large diffusion pumping capacity, or liquid hydrogen or helium cryo-pumping. In addition, the high heat loads, both from the thruster itself and from the beam, will demand very carefully designed cooling systems. The target will have to cater for an input energy of over 1.1 MW in the case of the 100cm thruster, and sputtering action will be very severe, so this component also calls for an imaginative design approach.

4.2 An alternative design approach

Several major objectives should guide the development of ion thrusters intended for application to the propulsion of large space structures such as the SPS. These include the attainment of:

(a) high thrust per device, to reduce overall system complexity,

(b) high thrust density, to reduce system dimensions and mass,

(c) high specific impulse, to minimise propellant consumption,

(d) long life, to give reliability and low maintenance costs.

(c) flexibility and stability, to simplify operational use.

Most of these can be achieved by using straightforward developments of the Kaufman thruster principle, as exemplified by the NASA Lewis/Hughes 30cm device[29,30,34]. This gives, as already mentioned, long life, high efficiency, flexibility of operation and moderately high values of SI. However, in developing this to larger sizes, major difficulties are likely to be encountered in achieving the necessary discharge currents and in coping with the additional thermal load to be expected.

It is also possible that attempts to increase the SI much beyond present levels may meet problems caused by excessive penetration of the electric field E between the grids into the discharge chamber plasma. Such penetration will cause the plasma sheath on the upstream side of the screen grid to become more concave, resulting in changes to the ion trajectories and greater impingement on the acceleration grid. This is a consequence of combining, in the twin grid system, the ion extraction and acceleration functions. Although an immediate solution is available, in that the grid separation can be increased to reduce E , this has adverse repercussions on beam focussing and overall perveance.

043

In this section, an alternative design approach is suggested, in which the ion extraction and acceleration mechanisms are separated by using the 4-grid systems developed for CTR applications[18,19]. These certainly work well at energies of 100 keV or more and at high values of J_B. Using this approach, the thrust per unit area can be drastically increased, without placing excessive demands upon discharge chamber and cathode technology.

A 4-grid system is shown schematically in Fig 10, together with a section through a shaped system used for extracting a 70keV beam[19]. The electric field applied between the plasma grid A and the extraction grid B extracts the ions from the edge of the discharge chamber sheath, and they are accelerated by a separate field applied between grids B and C. The latter is equivalent to the acceleration grid in the usual 2-grid system. It is negative to repel beam electrons. The fourth grid, D, defines the neutralisation plane. Extremely low beam divergence can be achieved[18] by careful selection of aperture diameters, grid spacings, cross sectional shapes, and dishing geometry.

Table 6 gives an indication of the performance that can be achieved by a 4-grid system. It is clear that V_T, I_B and J_B are all within the range of interest of SPS thrusters, although it should be pointed out that the data all refer to pulsed operation. However, in the case of the 10cm device, the pulses were of 10 seconds duration; they were limited to this by heat dissipation in the grids. As the design had paid little attention to methods of reducing this, the results can be regarded as promising. Such methods include increasing the open area ratio τ of the plasma grid and making it much thinner, thereby reducing the plasma ion current to it and improving the extraction efficiency[25]. This would almost entirely eliminate the ~1.9kW plasma heating of this grid. Similarly, the production of secondary electrons from grid C could be much reduced by improving the ion optics and using more appropriate materials; this would, in turn, reduce the power input from the beam to grids A and B, which is 1.8 and 0.4 kW respectively for a beam power of 375 kW. The power recorded at grid D, 2.3 kW, could also be reduced substantially by improved optics and by operating in a better vacuum environment. In addition, raising η_m to the level of Kaufman thrusters would provide considerable benefits, in that charge-exchange collisions in the spaces between the grids would be much reduced in number.

It appears, therefore, that J_B can certainly exceed 100 mA/cm^2 with τ as low as 31%. If the latter parameter can be increased to values common in small Kaufman thrusters[25], very much higher currents should be attainabie, perhaps reaching 500 mA/cm^2. Such currents must be supplied, by diffusion to the plasma sheath adjacent to grid A, from the discharge chamber plasma, which must therefore be much more dense than in a conventional thruster. Owing to the large value of I_D, the Hall effect resulting from the largely radial current crossing the axial component of the applied magnetic field will assist the diffusion process.

043

The diffusion ion current density J_i is given by the Bohm sheath criterion[25] and is proportional to $n_i(T_e/m_i)^{\frac{1}{2}}$, where n_i is the ion number density and T_e is the electron temperature. The change from mercury to argon will raise J_i by a factor of 2.24, but a very substantial increase of n_i will be necessary to approach CTR values of J_B . However, this should be quite feasible, by raising $\dot{m}$ and I_D , assuming that T_e is only marginally larger than with mercury, perhaps 15 eV.

If it is conservatively assumed that J_i can be increased by an order of magnitude to about 100 mA/cm^2 (n_i increased by a factor of 5), J_B becomes 80 mA/cm^2 if τ = 0.8. Selecting V_T to be 40 kV, to be compatible with SPS power generation and distribution systems[6,14], I_{sp} is 39470 seconds. It is possible that ambient plasma leakage currents[15] may prevent operation at such a high voltage when the SPS section or COTV is at low altitude. If that is found to be the case, relatively simple electrical insulation of conductor surfaces should provide a remedy.

A thruster diameter of 50 cm should be quite acceptable from a constructional point of view. With this, I_B = 157 A and the beam power is 6.28 MW. Adopting Kaufman thruster discharge chamber scaling relationships[25], I_D becomes 655 A and, at $V_D \sim 50$ V, $P_D \sim 33$ kW. The overall electrical efficiency is 99.5% and the discharge chamber value is a reasonable 208 eV/ion. The thrust is 28.6 N, which appears to be appropriate for all SPS missions. Assuming that the discharge chamber length equals its diameter, the discharge power can be radiated away at a temperature of about 650°C; although high, this should not be unacceptable.

Table 6

Characteristics of some CTR hydrogen ion sources

Grid diameter (cm)	10	12†	18	single 0.4cm hole
Grid form	flat	flat	dished*	flat
Open area ratio (%)	31	40	51	-
Beam energy (keV)	70	75	35-65	60-110
Beam current (A)	4-7	15	7-20	~0.04
Beam power (kW)	280-490	1125	245-1300	-
Beam divergence (degrees)	1.4	0.6	~2	<0.3
Mean J_B (mA/cm^2)	170-190	133	27-79	>300
Reference	19	37	36	18

* Focal length 4 m
† Design study

To achieve a discharge current of 655 A, using present cathode technology[34,35], about 30 cathodes would be run in parallel. Although mechanically feasible, an increase of current per cathode, without degradation of durability, is desirable. This can probably be achieved by increasing orifice diameter[35] and by changing from internal porous tungsten barium dispensers to lanthanum hexaboride or lanthanum molybdenum[39]. In fact, it may be quite feasible to revert to a single cathode using the latter materials;

043

cathodes using LaB_6 have run at 800 A and LMo emitters should allow at least 500 A to be reached[39]. Alternatively, an entirely separate plasma source could be employed with advantage; a design that may be applicable has recently been described by Chan *et al*[40].

5 CONCLUSIONS

It may be concluded that electric propulsion technology has a major role to play in the construction and operation of any SPS system. On a cost basis alone, its use is virtually mandatory for the LEO to GEO orbit transfer mission for all but top-priority cargo and personnel transportation. The mass savings resulting from the availability of high specific impulse thrusters are such that the number of launches from the earth to LEO, during SPS construction, can be approximately halved, with massive financial benefits. The same conclusions are inescapable when considering the AOC functions during SPS operation; electric propulsion is vital to success.

An examination of present technology immediately shows that only the Kaufman ion thruster is sufficiently developed to be acceptable for SPS applications, although MPD devices may ultimate prove suitable. The NASA/Hughes 30cm thruster could be used immediately, if necessary, but a specially developed version having higher thrust, specific impulse and thrust density is desirable. A review of proposed concepts has led to the view that a modified approach would be desirable, taking advantage of 4-grid ion extraction/acceleration systems developed for CTR purposes. An outline design of such a thruster has been proposed, which produces a thrust of 28 N from a diameter of 50 cm, at a specific impulse of over 39000 seconds.

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043

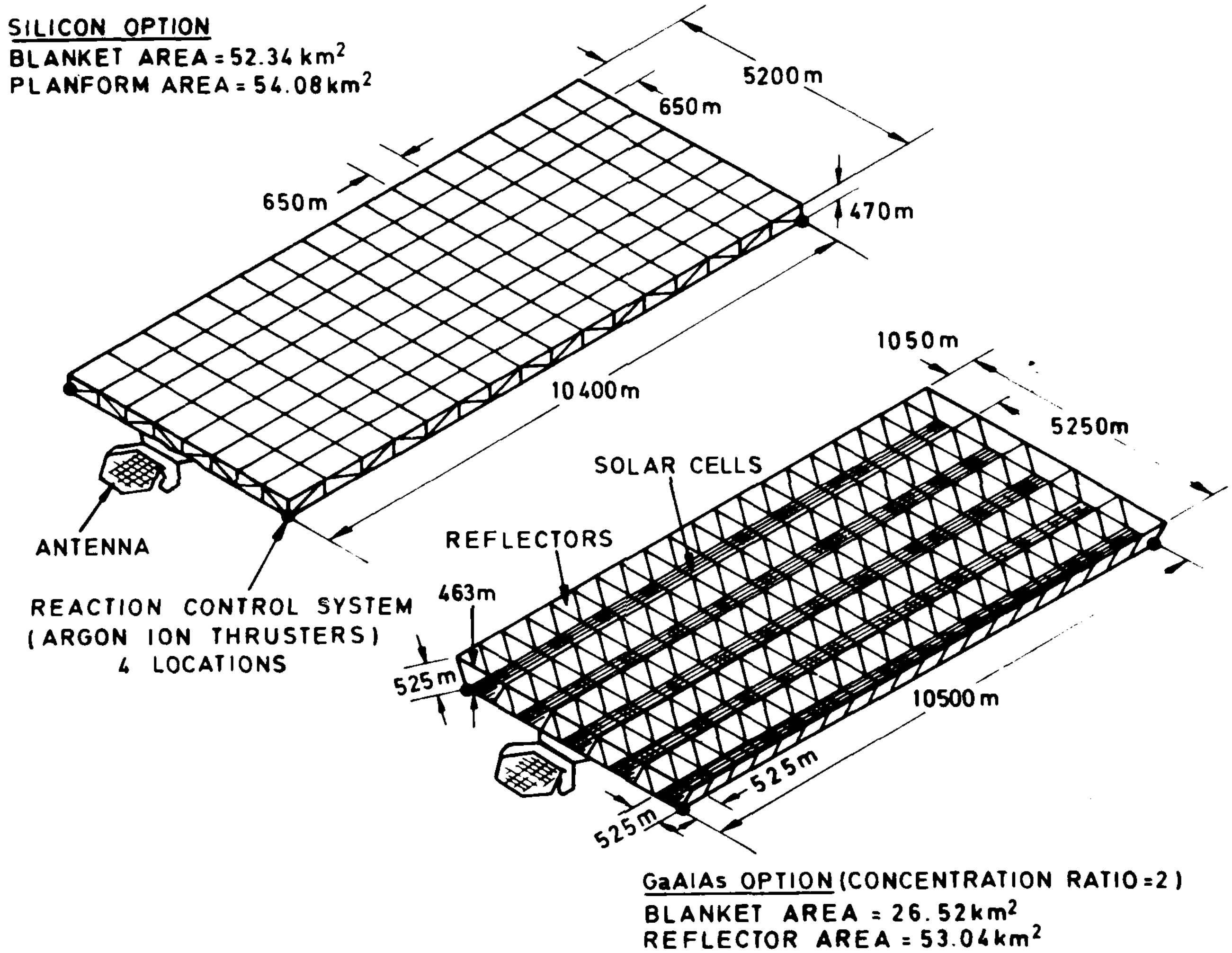

Fig 1 DOE/NASA reference SPS systems (from Ref 5)

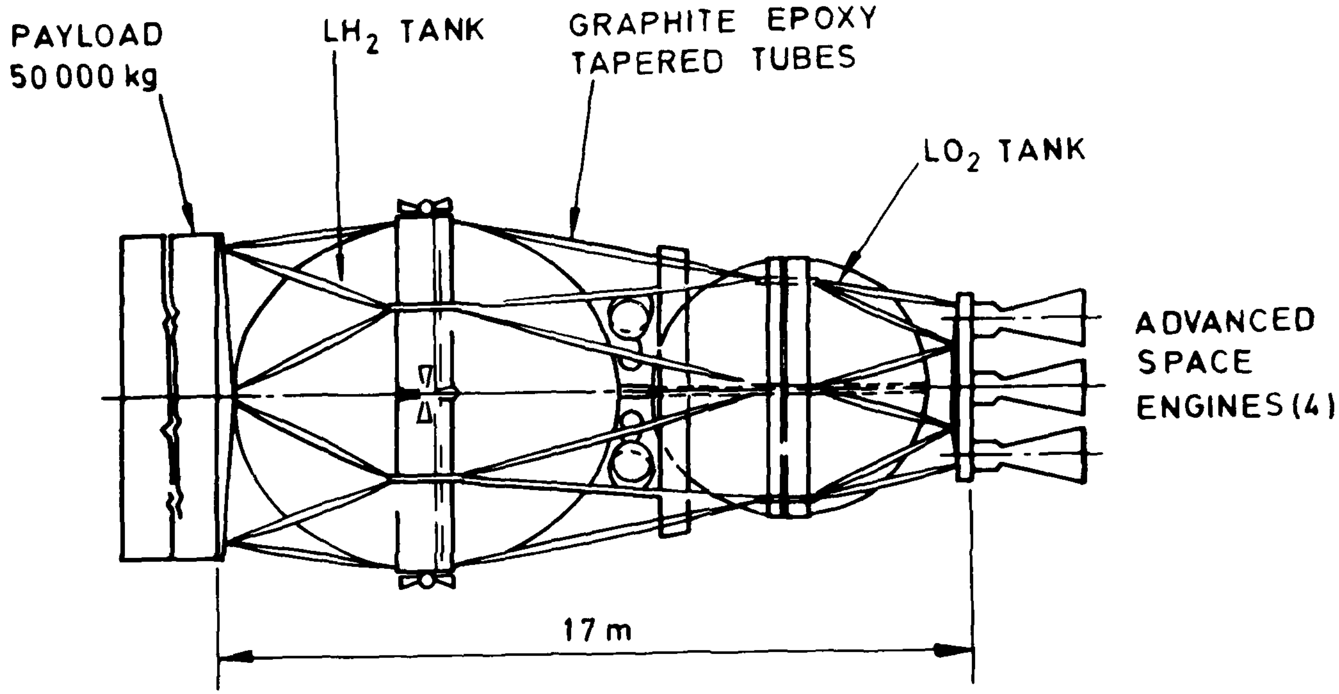

Fig 2 Schematic of advanced chemically propelled COTV (from Ref 13)

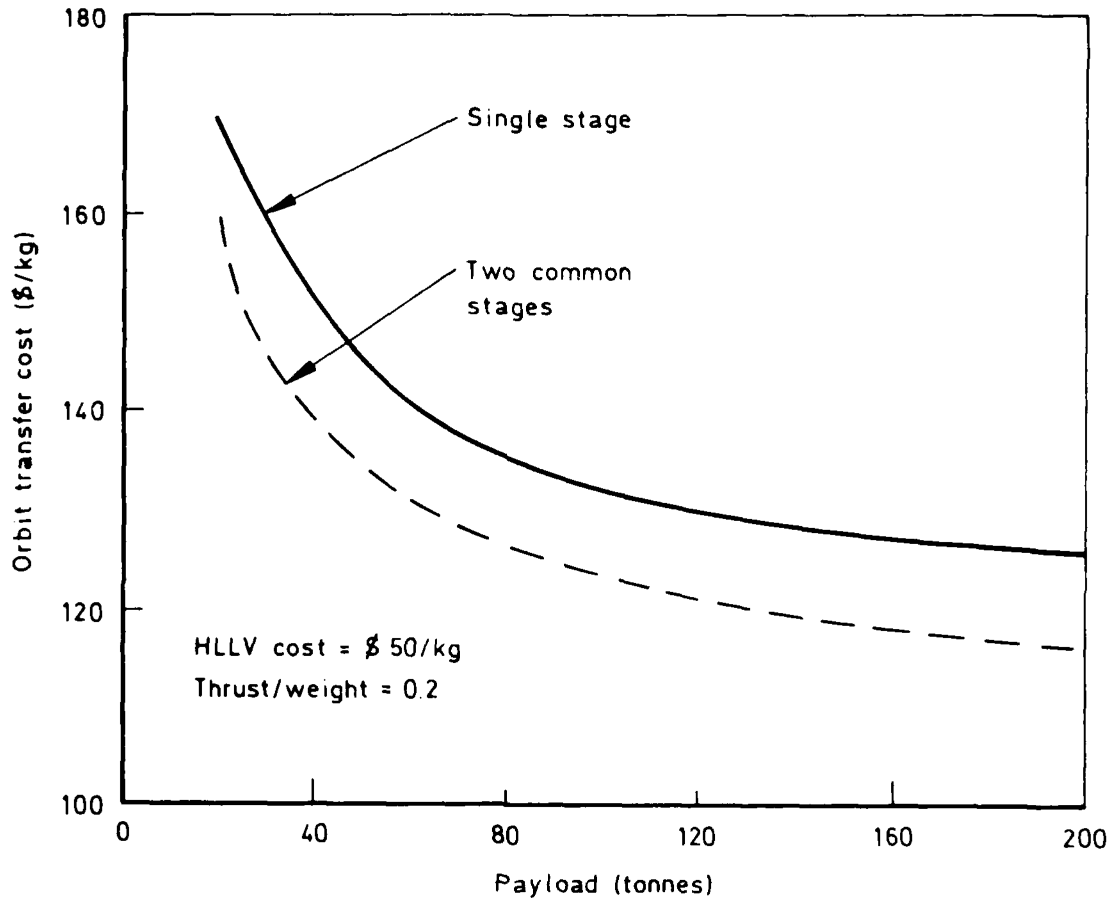

Fig 3 LEO to GEO transportation cost for chemically propelled COTV as function of payload (from Ref 13)

TR 81043

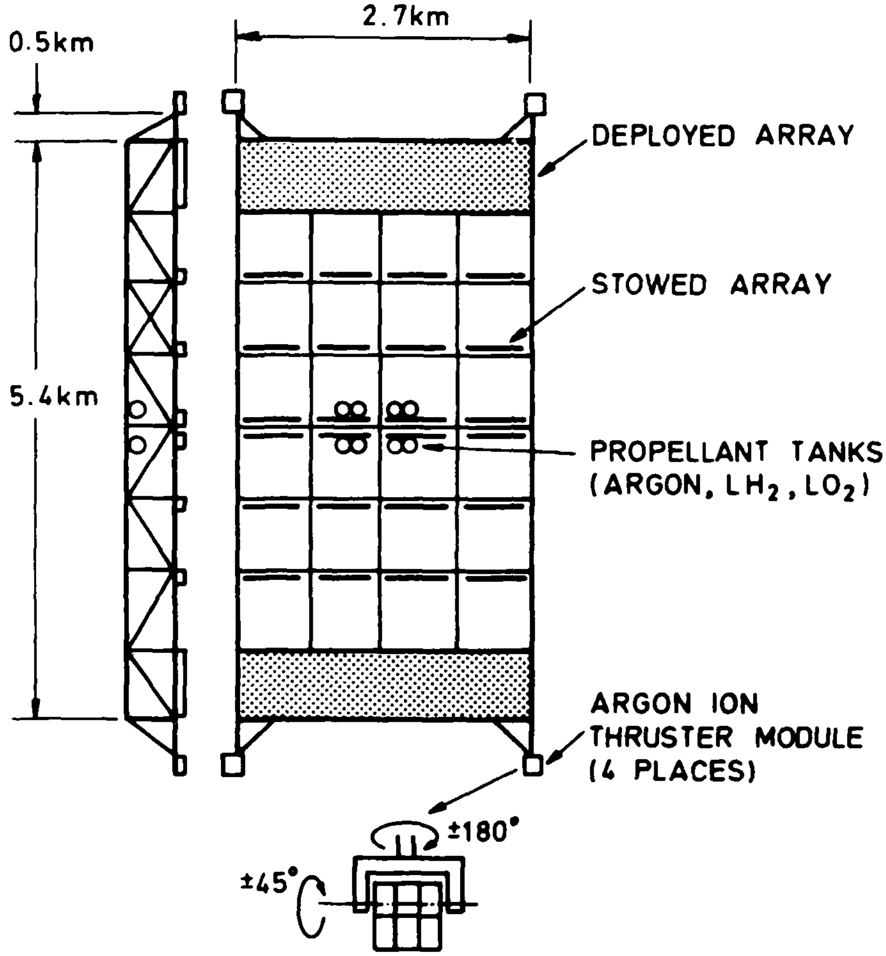

Fig 4 Boeing concept for transporting sections of a 10 GW SPS to GEO (from Ref 6)

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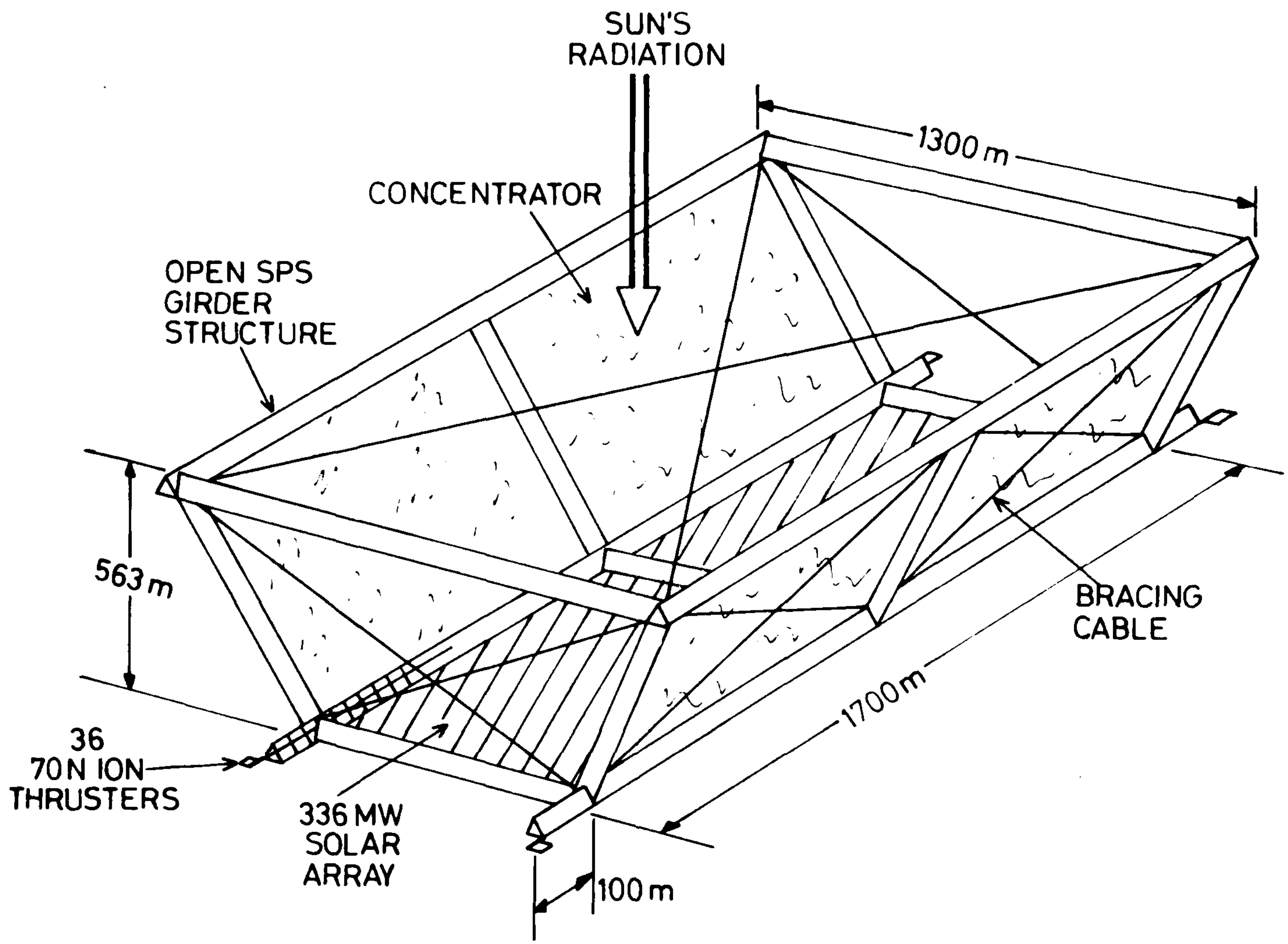

Fig 5 Rockwell COTV concept using argon ion propulsion (from Ref 15)

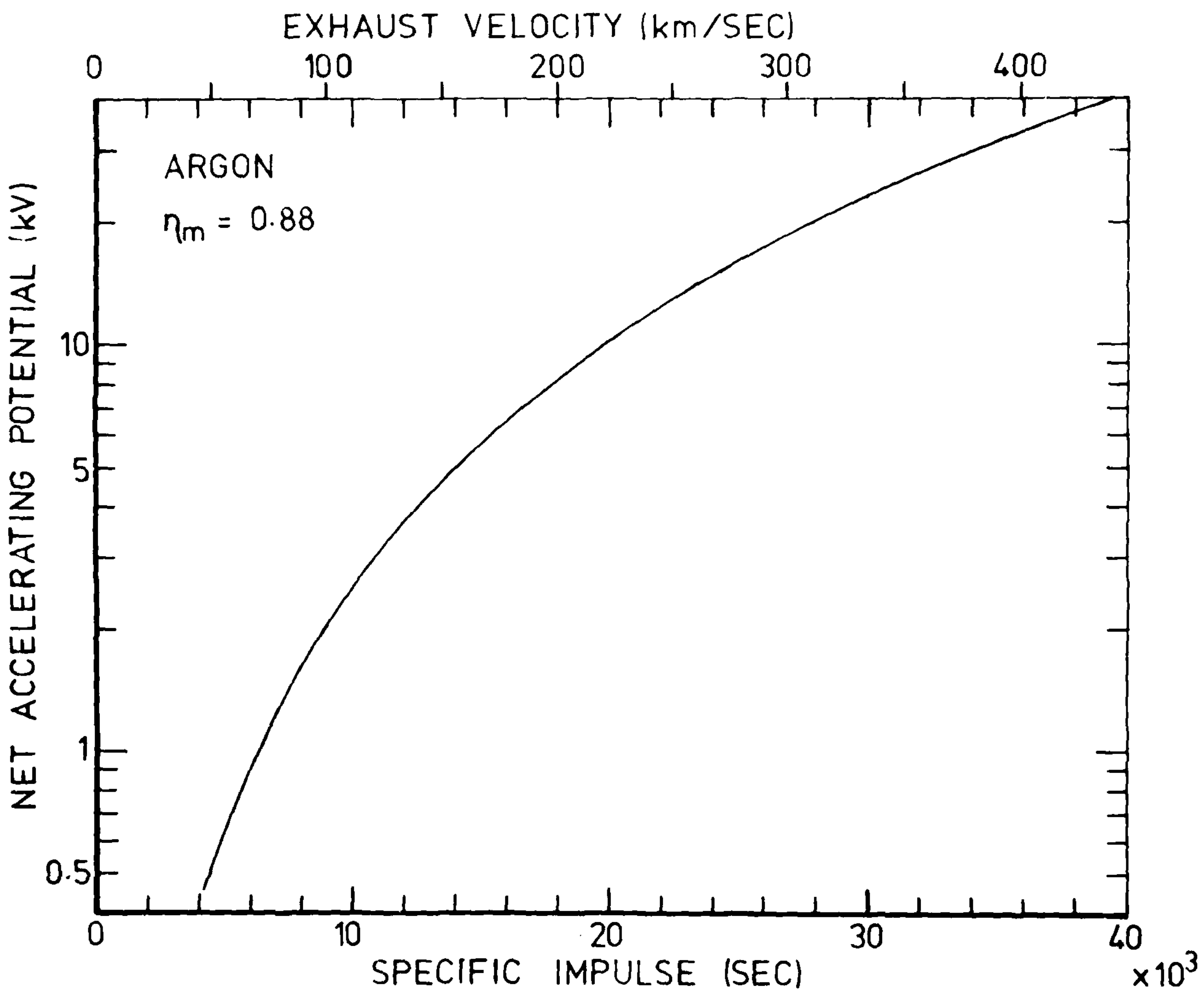

Fig 6 **Specific impulse and exhaust velocity as functions of accelerating voltage for argon ions**

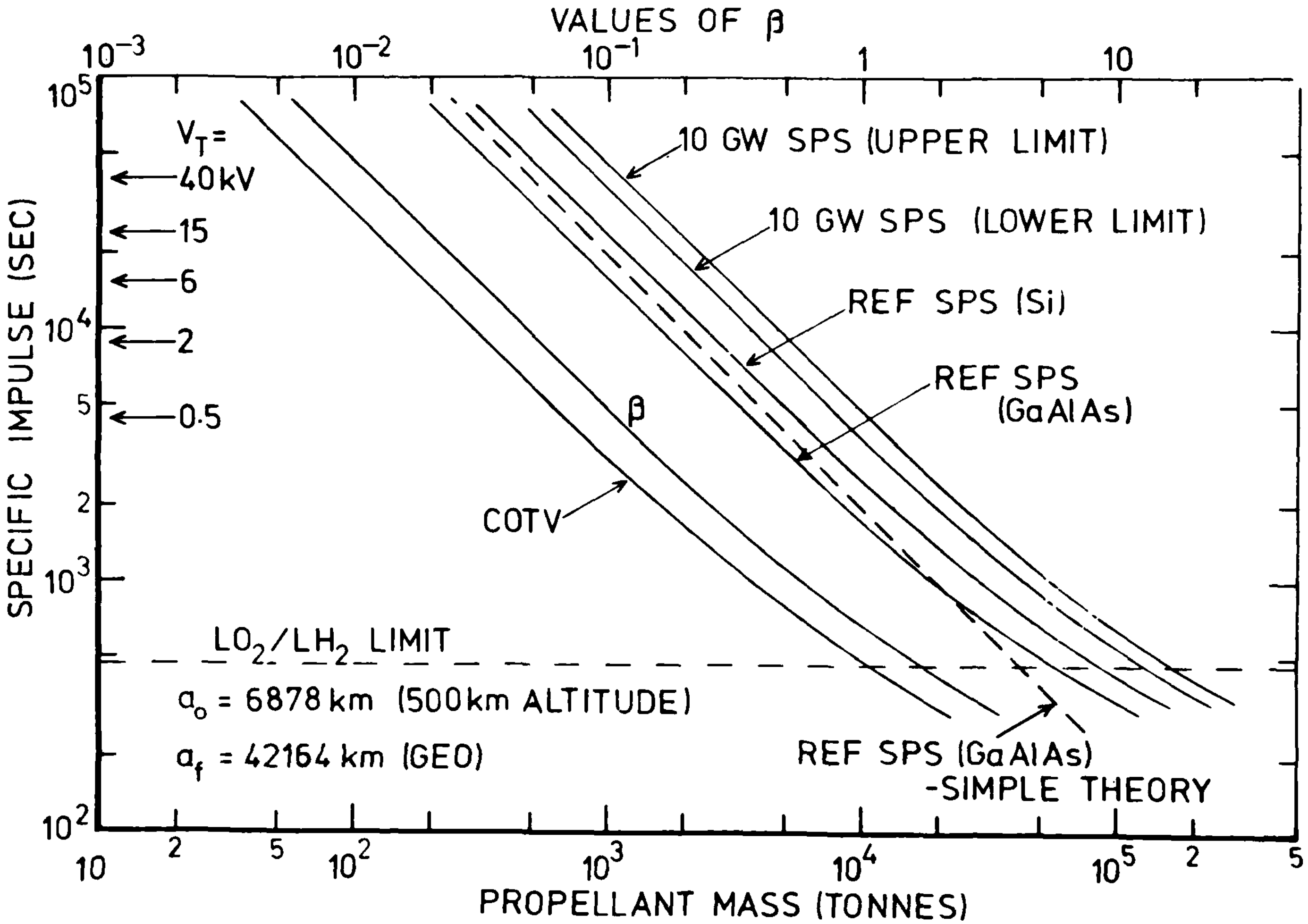

Fig 7 Propellant mass as a function of specific impulse for various SPS designs

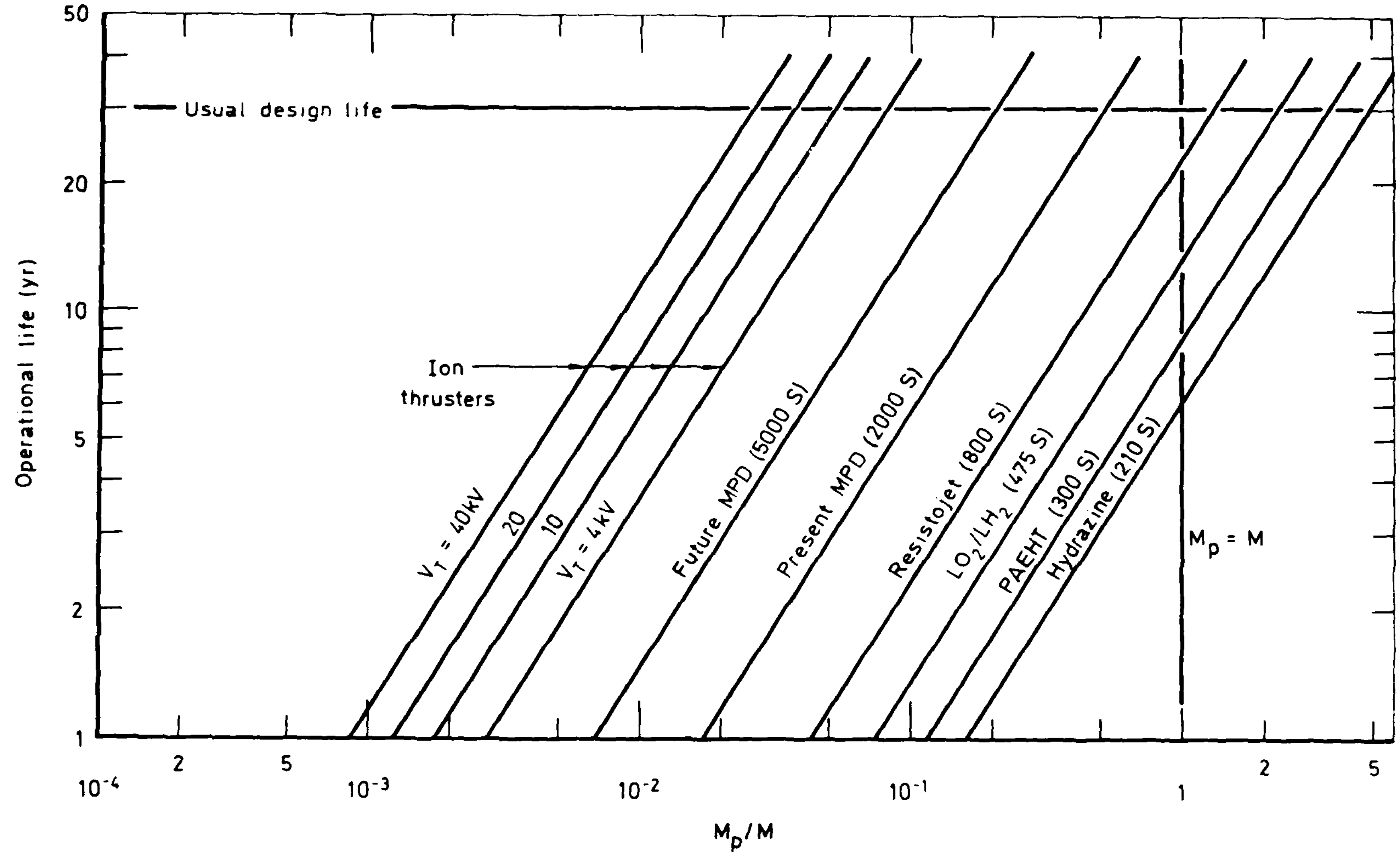

Fig 8 Ratio of propellant to spacecraft mass as a function of SPS life, for a variety of AOC propulsion systems

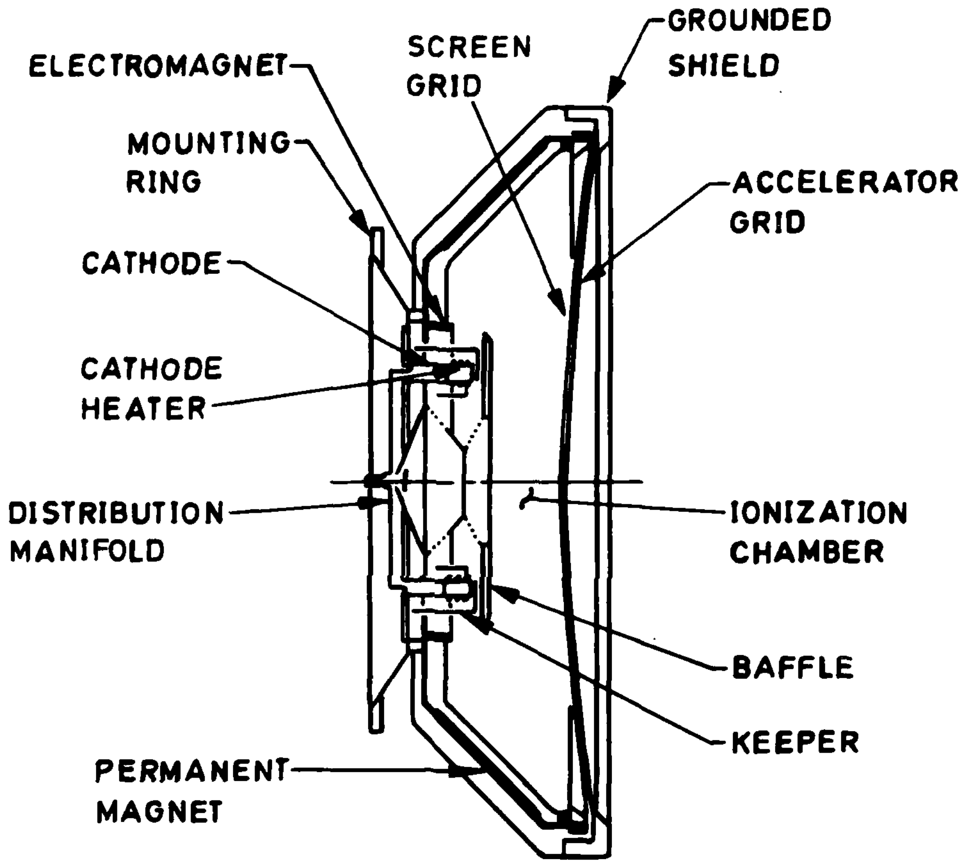

Fig 9 Schematic of 120 cm diameter argon ion thruster (from Ref 14)

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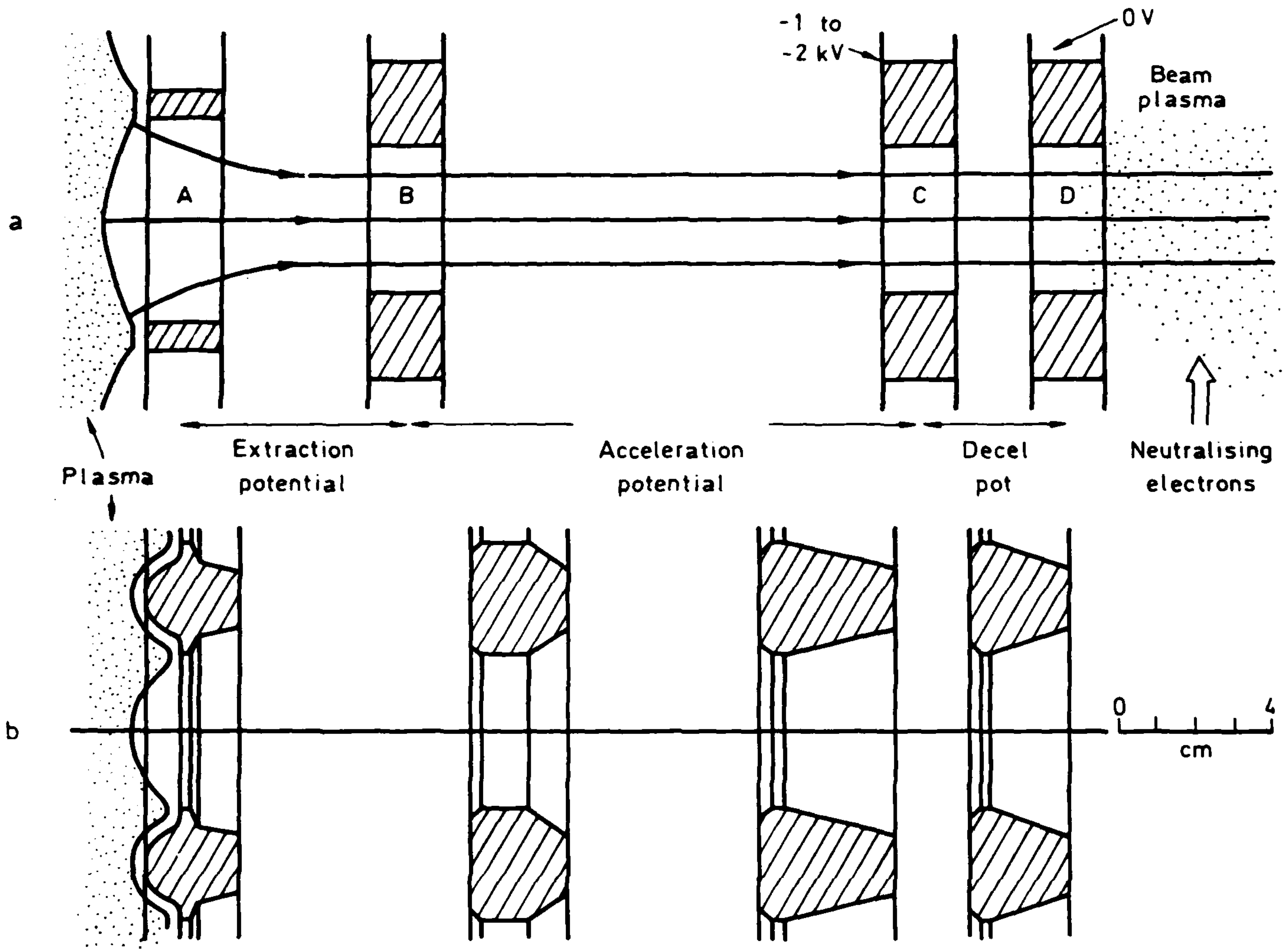

Fig 10 **Schematic of 4-grid systems: (a) basic configuration, (b) design used for 5 A, 75 keV beam (Ref 19)**

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17. Abstract

A common feature of all proposed solar power satellites is their enormous mass, perhaps approaching 100000 tonnes for a 10 GW version. The methods of transporting such masses to geostationary orbit are reviewed. It is concluded that electric propulsion techniques offer very considerable technical and financial advantages, and that ion thrusters currently represent the most suitable technology to employ. It is also shown that the use of ion propulsion for attitude and orbit control would be of great benefit. An advanced form of ion thruster, which offers a very high beam velocity and current density, is proposed for these applications.

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