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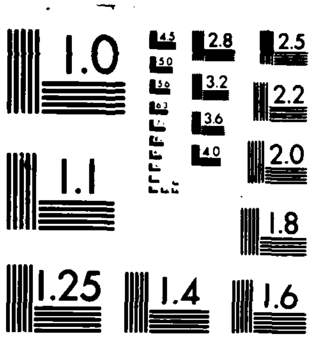
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ROYAL AIRCRAFT ESTABLISHMENT

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Technical Report 81131

October 1981

ELECTRIC PROPULSION OF SPACECRAFT

by

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SUMMARY

The Report commences with a discussion of the merits of electric propulsion technology, explaining that the very high exhaust velocities attainable allow the propellant masses required for most missions to be drastically reduced. The various types of electric thruster are then described briefly. The most highly developed and potentially useful thruster, the Kaufman electron bombardment ion thruster, is covered in greater detail, with particular reference to the T5 device developed in the UK. Candidate missions are discussed, ranging from attitude and orbit control functions to the application of ion propulsion to the deployment of solar power satellites. Important terrestrial applications of electric propulsion technology are also mentioned.

This Technical Report expresses the opinions of the author and does not necessarily represent the Official view of the Royal Aircraft Establishment.

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1 INTRODUCTION

Despite the outstanding successes of modern rocket technology in launching astroauts to the moon and complex unmanned spacecraft to Mars, Jupiter, Saturn and the inner planets, a large amount of international effort has been devoted to the development of more exotic forms of space propulsion, notably electric propulsion (EP). In the latter field, this effort commenced in the USA in the late 1950s[1,2], and significant contributions have also been made by teams in France, Germany, Italy, Japan, the UK and the USSR, and by the European Space Agency. Many advanced propulsion concepts have been studied during hundreds of man-years of endeavour, indicating a general conviction that substantial benefits were to be expected from any practical applications resulting from this work.

These predicted benefits arise because the effective exhaust velocity v_e of an EP system is not limited, as with a conventional rocket motor, by the energy released in a chemical reaction. The only physical limitation is the velocity of light, assuming that an adequate power source can be made available. As will be shown below, the very high values of v_e that can be provided by EP systems allow the payloads carried by spacecraft of a given total mass to be increased substantially, thus enhancing overall mission capability. In commercial space activities, notably the provision of communications services of all kinds, this can have a large beneficial impact on revenue and profitability.

The Report briefly describes the various types of electric thruster available, and concludes that the electron bombardment ion thruster, devised by Kaufman[1], has reached the most advanced state of development and is closest to operational applications. This device is discussed in greater detail, with particular reference to the T5 thruster developed by a UK team headed by the Royal Aircraft Establishment at Farnborough.

An account is then given of some of the missions for which EP systems are suited, concentrating on the application of ion thrusters to north-south station-keeping (NSSK) of geostationary spacecraft[3], orbit transfer manoeuvres[4], and interplanetary objectives. A future, very large scale mission also covered is the propulsion, both during manufacture and in operation, of solar power satellites[5].

In general, EP technology is finding, usually in a simplified form, increasingly wide terrestrial applications[6], ranging from paint spraying to the manufacture of advanced integrated circuits. Some of these applications are discussed in the final part of the Report.

2 THE ROCKET EQUATION

It is usual to characterise a space mission by reference to the velocity increment ΔV required to complete it. Typical values of this parameter, which is independent of spacecraft mass, are:

- Impulsive escape from a 480 km altitude orbit 3.15 km/s
- Spiral escape from a 480 km altitude orbit 7.59 km/s
- Earth orbit to Mars orbit and return 14 km/s
- Earth orbit to Jupiter orbit and return 64 km/s

and, in geostationary earth orbit,

- North-south station-keeping for 1 year 41-51 m/s.

By equating the instantaneous rate of change of momentum to the force applied to the spacecraft by its propulsion system and integrating with respect to time, it may be shown that:

$$\Delta V = v_e \log_e\left(\frac{M_0}{M_f}\right) = v_e \log_e\left(\frac{M_0}{M_0 - \Delta M}\right) \tag{1}$$

where M_0 and M_f are the initial and final masses of the vehicle and ΔM is the mass of propellant consumed. This, the rocket equation, clearly illustrates the importance of obtaining as high a value of v_e as possible. For a given velocity increment, a low exhaust velocity must be compensated for by an enormous increase in ΔM, and therefore of the size and cost of the vehicle. It is this dependence of ΔV upon the logarithmic term, for constant v_e, which provides the incentive for EP development.

As an illustration of the magnitude of this effect, consider a hypothetical mission in which a spacecraft is to be sent from low earth orbit to an orbit around Mars and then returned. If the payload is 1000 kg, a single-stage chemical rocket would, ignoring its own mass, require a ΔM of 32115 kg for an effective exhaust velocity of 4 km/s. An EP system with v_e increased to 40 km/s, which is within the current state of the art, would need a propellant mass of only 419 kg. Other, more realistic examples of the advantages of using EP will be given later.

Although equation (1) involves the effective exhaust velocity of the propulsion system, it is more usual to refer to its specific impulse (SI), I_{sp}, which is defined as the ratio of thrust T to the total rate of use of propellant, $\dot{m}$, expressed in terms of units of sea-level weight. Thus

$$I_{sp} = \frac{T}{\dot{m} g_0} \tag{2}$$

where g_0 is the acceleration due to gravity at sea level.

Taking $T = \dot{m} v_e$, equation (2) becomes $I_{sp} = v_e/g_0$. Typical values of v_e and I_{sp} are given in Table 1, from which it will be seen that EP systems are intrinsically superior to all forms of chemical propulsion, when judged on the basis of these parameters.

Table 1

Typical values of specific impulse and effective exhaust velocity

Propulsion system	I_{sp} (s)	v_e (km/s)
Cold gas jets (*eg* nitrogen, propane)	66	0.65
Solid propellants	210-330	2.1 to 3.2
Liquid propellants and oxidisers	300-460	2.9 to 4.5
Exotic bipropellants	410-610	4.0 to 6.0
Liquid hydrogen/liquid oxygen	$\leq$475	4.66
Monopropellants (*eg* hydrazine)	170-300	1.7 to 2.9
Power augmented hydrazine	285-330	2.8 to 3.2
Hydrogen resistojets	$\leq$815	$\leq$8.0
Electric propulsion (state of the art)	$2\text{-}20 \times 10^3$	20-200

It should, of course, be pointed out that EP systems are not without disadvantages. These include generally greater overall complexity, the need to provide large electrical power supplies, and a low thrust per device. The latter is a particularly serious limitation, and it essentially restricts the use of EP to long duration missions which can be accomplished with low acceleration.

3 THE PRINCIPLES OF ELECTRIC PROPULSION

There are two basic forms of EP, and both have been subjected to considerable development effort in several different countries; they rely on either electromagnetic (EM) or electrostatic (ES) acceleration processes. A third type of device, the resistojet[7], is sometimes included under the EP heading. However, this does not involve the production and acceleration of charged particles, so it will not be discussed in this paper, beyond stating that it achieves an enhanced SI by injecting thermal energy into a gaseous exhaust, using an electrical resistance heater.

3.1 Definitions and figures of merit

In discussing the performance of electric thrusters, a number of important parameters are of particular significance, in that they can be used as figures of merit in comparing the various devices. The SI has already been defined; the others are:

(a) Mass utilisation efficiency, η_m ; this is the proportion of the total propellant fed to a thruster which is usefully employed in producing thrust.

(b) Electrical efficiency, η_e ; this is the ratio of the directed energy in the exhaust to the total input energy.

(c) Total efficiency $\eta_T = \eta_e \eta_m$.

(d) Beam divergence; this is arbitrarily defined, in most analyses, as the semi-angle of the cone bounding 95% of the beam current or energetic plasma exhaust.

3.2 Electromagnetic acceleration

Very many types of electromagnetic propulsion device have been studied, involving a wide variety of geometrical configurations, current sources, and plasma production and acceleration mechanisms. However, they all have in common a source of dense, electrically neutral plasma and a method of electromagnetically accelerating this or a part of it. To generalise, the plasma may be produced in the following ways:

(a) a self-ignited pulsed discharge, with current coming from a capacitor;

(b) a pulsed discharge initiated by a separate trigger discharge;

(c) an auxiliary dc or RF discharge.

In such devices, the plasma may be formed from a gas or vapour injected into the discharge chamber by means of a special valve, or it may be ablated from a solid block of propellant or evaporated from the surface of a liquid propellant. In (a) and (b) above, the discharge may be lengthened to the stage where it may be considered quasi-steady state. This is usually the case in the magnetoplasmadynamic (MPD) arc thruster[8], which is possibly a significant long-term contender for primary propulsion applications[9].

Several techniques have been employed to accelerate the plasma to a high velocity, producing thrust. These include the following:

(i) a high current pulsed discharge between a pair of electrodes;

(ii) a high current pulsed inductively-coupled discharge;

(iii) a travelling RF wave in a separately produced plasma;

(iv) the Hall effect in a steady-state plasma.

In (i) and (ii), the pulsed discharge current can both produce and accelerate the plasma.

Two examples of EM thrusters are depicted in Figs 1 and 2. A schematic of an MPD device[9] is shown in Fig 1 and a pulsed plasma rail gun[10] in Fig 2. The former can only operate at reasonable efficiency if it is pulsed at MW power levels, so it is likely to be suitable for primary propulsion only. Conversely, the rail gun is a contender for attitude control functions, having been designed as a low power, low thrust device.

In the MPD thruster, the central discharge to the cathode pinches down to a small radius as the current increases, owing to the compressive force exerted by its own azimuthal magnetic field. This magnetic field then interacts with the radial current sheet at the anode, producing an expulsive force which accelerates the plasma away from the thruster. Joule heating of the plasma also contributes to the expulsion process.

A similar mechanism occurs in the rail thruster (Fig 2), in which an auxiliary discharge both evaporates propellant from a reservoir of mercury and initiates the main discharge. The azimuthal magnetic fields of the currents flowing in reverse directions in the two rails reinforce each other in the centre of the device, and the total field there interacts with the discharge current through the plasma to produce a large expulsive force.

3.3 Electrostatic acceleration

In an ES device, positive ions or charged liquid droplets are extracted from a plasma or a conducting liquid by a strong ES field, and are then accelerated by the same field. External charge neutralisation of the emerging ion or droplet beam is necessary, and involves the use of a heated filament or hollow cathode[11] to generate the required electrons.

Many different devices have been developed making use of these concepts. Possibly the most simple is the colloid thruster, in which charged droplets are produced by electrostatic spraying from a liquid meniscus at the end of a fine needle, tube or slit[12]. The liquid must be slightly conducting, and is usually glycerol doped with sodium iodide. The droplets are formed because the liquid meniscus becomes unstable in the presence of the intense electric field set up between the needle, tube or slit and an external electrode. In the example shown in Fig 3, a linear raised edge fed with liquid from slits on either side provides the emission source.

Such devices give moderate η_T and SI. They have also reached an advanced stage of development, but a planned flight test[13] was abandoned owing to lifetime problems. They could, however, prove useful in future low thrust applications requiring rapid response and small power consumption.

The field-emission (FE) ion thruster is a further development of the colloid concept, in which the spraying process causes the emission of positive ions rather than charged droplets[14]. The propellant is invariably a liquid metal, such as cesium, which wets the emitting edge or needle. Alternatively, emission can be from sharp cusps formed on the liquid's meniscus by instability processes caused by the very high applied electric field. The European Space Agency (ESA) has sponsored a major FE thruster development programme[14], in which it has been shown that very high values of v_e and η_e may be achieved, but that η_m is relatively low due to propellant evaporation. In addition, cesium can cause considerable problems to the spacecraft designer, although possible alternatives are being investigated.

Another device employing cesium is the contact ionisation thruster[15], in which the propellant vapour is ionised by passing it through a porous tungsten plug heated electrically to about 1200°C (Fig 4). The ions are subsequently accelerated and focussed by an electrode system. Although providing an excellent η_m, the thrust level is low, the thermal radiation losses from the tungsten cause η_e to be poor, and other problems result from cesium condensation on insulators and electrodes. Development has now virtually ceased.

A thruster which is much more promising than those described above, and which is being prepared for a flight test[16,17], is the RF ion thruster developed in Germany. In this device, the propellant gas or vapour is fed into a quartz discharge chamber where it is ionised in an RF discharge at typically 1 MHz (Fig 5). One end of the discharge chamber consists of a perforated quartz grid, the holes in which are aligned with those drilled in a pair of external metal grids. A strong electric field imposed between these grids and the discharge chamber plasma both extracts and accelerates the ions away from the thruster in a well-collimated beam. The positive space charge of the beam is neutralised by electrons emitted by an external hollow cathode[11] fed with the same propellant as employed by the thruster.

The RF thruster as developed to date uses mercury as a propellant, but it has been demonstrated that gases such as argon and xenon are acceptable alternatives. The flight test thruster[17] is of 10 cm diameter, but versions up to 35 cm have been studied[18]. Both η_e and η_m are good, as is durability, but the latter factor appears to be limited by shorting of the applied RF field by metallic coatings sputtered onto the inner wall of the discharge chamber.

The most highly developed ion thruster is the Kaufman type[1,19], which is described in more detail in section 4. It is similar in configuration to the RF thruster, with the major difference that the RF discharge is replaced by a dc discharge between a hollow cathode and a cylindrical anode. Consequently the discharge chamber can be made of a metal such as stainless steel, and the quartz grid at its end can be eliminated, greatly simplifying constructional and space qualification problems. It can employ a variety of propellants, although most work has been done with mercury, and sizes of up to 1.5 m diameter have been operated. In general, it demonstrates a higher η_e than the RF thruster and is usually considered to be more durable. It has received a very extensive flight test in the SERT II mission[20], in which two thrusters were operated intermittently in space for a period of about 10 years. Ground tests have exceeded 15000 hours and components have been tested for much longer times.

The ionisation efficiency of the Kaufman thruster is enhanced by the application of a cusp-shaped magnetic field to the discharge chamber. This contains the energetic primary electrons within the region where ionisation is required and also encourages, via the Hall effect, the drift of ions towards the grid system. This field allows η_m to approach 90%. A somewhat different device has also been developed, the magnetoelectrostatic containment (MESC) thruster[21], in which this field is replaced by a very much stronger series of cusp-shaped fields on the periphery of the discharge chamber. This can, under some circumstances, lead to a higher value of η_e, but large-diameter versions have not yet been tested.

3.4 Comparison between thrusters

An attempt has been made in Table 2 to compare some of the more significant thruster parameters. In making reference to this, it should be recalled that the data are for specific devices at single operating points; very considerable variations are possible in almost all cases, many of these designs being extremely flexible in their tolerance to parameter changes.

4 THE KAUFMAN ION THRUSTER

Since the Kaufman ion thruster is the most highly developed of ES electric propulsion systems, it will be described in a little more detail in this section. Specific reference will be made to devices using mercury as a propellant, but it should be recalled that many other gases and vapours would be suitable. For example, argon, xenon, cesium and nitrogen have been investigated.

A schematic diagram of a Kaufman thruster is shown in Fig 6. This is based on the UK T4/T5 series of thrusters[19,22], so it differs in some respects from other devices. For instance, the NASA Lewis/Hughes SIT-8 thruster[25] incorporates only a single propellant feed system to the discharge chamber and makes use of permanent magnets to provide the cusp-shaped magnetic field. However, the principles of operation remain the same.

Liquid mercury is stored under pressure in a stainless steel tank, separated from the pressurising gas by a bellows or a flexible diaphragm. The liquid is fed to three vaporisers, via simple on/off valves, where porous tungsten plugs act as phase separators. The liquid is unable to pass through the pores in these plugs, whereas any vapour can do so. The amount of vapour flowing is determined by plug temperature, which is regulated by varying the power fed to a heater surrounding the vaporiser.

The mercury feed system is at spacecraft potential, whereas the thruster operates at a high positive voltage, typically 1 kV or greater. This is possible because the two propellant feed lines to the thruster body incorporate alumina electrical isolators, in which porous internal structures inhibit electrical breakdown of the mercury vapour. The central vapour flow passes through a hollow cathode electron source[11] into the discharge chamber, via an annular gap between the inner magnetic polepiece and a circular baffle disc. The other, larger flow goes into an annular distribution chamber, from whence it emerges in a radial direction.

The hollow cathode[11] consists of a cylindrical tantalum tube surrounded by a bifilar tungsten heater encapsulated in sprayed alumina. The downstream end of the tube is closed by a circular tungsten disc having a small central orifice. The complete assembly is surrounded by a radiation shield consisting of a tightly wound strip of thin dimpled molybdenum foil inside an outer stainless steel casing. Internally, the cathode contains a dispenser of low work function material which, in the case of the T5 thruster, is a hollow cylinder of porous tungsten impregnated with barium calcium aluminate.

The discharge is initiated by heating the cathode to above 1000°C and applying a potential of several hundred volts to the keeper electrode, which is situated adjacent to the cathode tip. Once a discharge current is flowing between this electrode and the cathode orifice, it immediately transfers to the cylindrical anode, which is at 40-50 V relative to the cathode. The cathode heater can then be turned off, operating temperature being maintained subsequently by ion bombardment heating of the cathode tip.

The primary electrons generated by the cathode cause some ionisation within the cylindrical inner polepiece, thus forming the 'coupling plasma'. To emerge into the discharge chamber, these electrons gain energy from the applied electric field in crossing the fringing magnetic field in the annular polepiece/baffle disc gap. The design of this region is crucial to obtaining high efficiency. The electrons then spiral along the field lines, as shown in Fig 6, causing ionisation by collision, before eventually reaching the anode.

The ions so formed drift towards the grid system, which consists of a pair of inwardly dished molybdenum ectrodes ch containing many closely spaced and carefully

Table 2

Comparit ,n of parameters of ion and plasma thrusters

Type reference	Kaufman ion 25	Kaufman ion 24	RF ion 16, 17	RF ion 18	MESC 21	Colloid 22	Field emission 14	Contact ionisation 15	Rail* Gun 23, 26	MPD* arc 8
Acceleration mechanism	ES		ES		ES	ES	ES	ES	EM	EM
Usual propellant-alternative	Hg Argon, etc		Hg Argon, etc		Cs Hg, Argon	Glycerol † none	Cs None?	Cs None	Teflon Hg	Argon N_2
Spacecraft acceptability	Good		Good		Poor (Cs) Good	Fair	Poor	Poor	Good	Good
Exhaust exit dimensions (cm)	8 dia	30 dia	10 dia	35 dia	12 dia	3 dia annulus	3 cm linear	5 × 0.6 rectangle	7.5 × 3.4 rectangular	10 dia
Accelerating (A) Discharge (D) potential	1.2 kV (A)	1.1 to 5.0 kV (A)	1.5 kV (A)	~3.7 kV (A)	760 V (A)	10-16 kV (A)	2-4 kV (A)	4 kV (A)	2.5 kV (D)	100-400 V (D)
Power (kW)**	0.125	2.6 to 10.4	0.310	3.6	0.34	~0.01	~0.16	~0.12	0.153	6000
SI (s)	2801	2000-6300	3089	3360	3270	1000-2000	9000	6700	1794	2400
η_e	0.69	0.84 to 0.96	0.64	0.79	0.81	0.7	~0.9	~0.4	0.32	0.31
η_m	0.84	0.89 to 0.95	0.80	0.88	0.97	0.2 to 0.8	~ 0.7	0.99		
Thrust (mN)	4.9	130 to 290	10	160	17	0.5††	~ 2.5††	1.5	4.5	140×10^3
Life test (h)	15000	10000	8000	None	600φ	475≠	Few 100	None φφ	2×10^6 pulses	None
Development status	Flight ready	Flight ready	Flight ready	Poor	Medium	Good	Fair	Fair	Flight operational	Fair#

* Pulsed operation.
** Excluding dissipation in power supply electronics.
† Doped with Na I.
†† Easily stacked to give increase by a factor of 10 to 100.

≠ But 6500 h on multiple needle thruster.
φ But 2600 h on smaller device.
φφ Components up to 5000 h.
\# But note Japanese flight test (Ref 30).

aligned holes. The open area ratio of the inner, or screen, grid is as high as possible, again to maximise efficiency; values of 80-90% can be achieved. The screen grid is at a high positive potential relative both to the outer, accelerator grid and to space potential. Ions are thus accelerated through the holes, forming an energetic beam and producing thrust. Their energy is usually at least 1 keV, and values much higher are readily attainable. The grids are dished primarily for mechanical stability, but this also has an influence on the ion trajectories. Care in design is necessary to ensure that the beam divergence is not too large; values of $10-15^{\circ}$ can be achieved.

The positive space charge of the beam is neutralised by electrons supplied by an auxiliary hollow cathode mounted externally. This cathode is very similar in design to the other, although it operates at a much lower mercury flowrate and discharge current. As the body of the thruster is at a high positive potential, an earthed screen surrounds it to prevent backstreaming of electrons from the neutraliser.

4.1 Power supplies and control

In most applications, the power required to operate a Kaufman thruster would be derived from solar arrays. However, a special power conditioning electronics system is necessary to convert raw array power into the many specific voltages and currents required by the thruster. This power conditioning unit (PCU) is the most massive component of the EP system and often dominates in any analysis of reliability.

It is not generally possible to switch supplies on and off in a simple manner. This has to be done in a carefully programmed sequence, and microprocessor systems are ideal for this application, because they allow great flexibility of operation, alterations being made via software changes. Similarly, control functions can also be carried out using a microprocessor, with great advantage. These functions include control of thrust level and of mass utilisation efficiency, fault diagnosis and rectification, and throttling.

The type of system which has been implemented is depicted in Fig 7, which shows schematically the T5 thruster PCU and microprocessor arrangement[19]. This was extremely successful, particularly during the thruster start-up and shut-down sequences.

4.2 Performance

Many different Kaufman thrusters have been designed and developed, with performance levels covering a significant range. As an example only, full details of the performance of the T5 thruster[19] are given in Table 3. These may be considered typical of a very highly developed device, which also demonstrates excellent stability, durability and throttling range.

Table 3

Performance of T5 ion thruster

Thrust	10.5 (10.2)* mN
Thrust range	7-17 mN
Exhaust velocity for singly-charged ions	30 km/s
Specific impulse	2879 (2730) s
Total input power	223 W
Power/thrust ratio	21.4 (21.8) W/mN
Total mass flow rate	0.370 (0.382) mg/s
Energy cost per beam ion	233 (266) eV/ion
Electrical efficiency	70.4%
Mass utilisation efficiency	93.9 (87.6)%
Total thruster efficiency	66.1 (61.7)%
Semi-angle of beam at 95% of I_B	$<11^{\circ}$
Thruster potential	940 V
Beam current I_B	167 mA
Accel grid potential	300-400 V
Accel grid current	<0.5 mA
Anode potential	43 V
Discharge current	0.9 A
Anode - keeper potential	32.5 V
Keeper current	0.4 A
Doubly-charged ions (% I_B)	8%
Neutraliser flow rate	~0.012 mg/s
Neutraliser keeper current	0.3 A
Neutraliser bias potential	15 V
Discharge chamber temperature	$\sim 200^{\circ}C$
Backplate temperature	$\sim 250^{\circ}C$
Solenoid temperature	$\sim 180^{\circ}C$
Mass	1 kg
Start-up time	12 min

* Values in brackets have been corrected for doubly-charged ions, keeper power and neutraliser mass flow, where applicable.

5 MISSIONS

As already explained, EP systems are extremely desirable for a very wide range of missions because they provide a very high SI, allowing propellant requirements to be reduced substantially. These missions fall into two main categories, auxiliary propulsion and primary propulsion.

5.1 Auxiliary propulsion

The mission that has been studied in greatest depth is NSSK, with applications mainly to communications spacecraft[3]. The T5 [19], RIT-10 [16] and SIT-8 [25] ion thrusters were designed for this purpose, although they are also capable of performing other functions, such as east-west station-keeping, orbit changes, and the spiral orbit raising of small spacecraft[4,27].

The advantages of using ion thrusters for NSSK are clearly indicated by the data in Fig 8, in which propulsion system mass is plotted against operational life for a 960kg communications spacecraft. The use of T4 thrusters[28], electrothermal hydrazine thrusters (EHTs) and power-augmented EHTs (PAEHTs)[29] is compared, and it is evident that very large mass savings can be made if EP is employed, even if it is assumed that dedicated batteries or solar arrays must be provided.

It should be mentioned here that one EM EP system has reached the operational deployment phase, and this has been in the auxiliary propulsion field. This device is the Teflon-fuelled pulsed plasma rail gun[23,26], which has successfully carried out station-keeping and attitude control functions on several spacecraft over a number of years.

5.2 Primary propulsion

The two main near-term primary propulsion applications are to orbit transfer manoeuvres and interplanetary missions, with, again, propellant saving being the major driving factor. For the more difficult concepts, such as a rendezvous with the nucleus of a comet[31,32], the benefits to be gained from using EP are particularly important. They allow a very advanced mission to be flown employing a modest launch vehicle, thus reducing cost drastically and, perhaps, making the project economically feasible. A proposed Halley's Comet mission was an excellent example of what could be achieved; in this, an array of 30cm diameter ion thrusters[24] was to be used to propel a spacecraft to a rendezvous. In a later alternative, a fly-by of Halley's Comet would have been followed by a rendezvous with the Comet Tempel 2.

Other possible interplanetary missions[32] which have been studied extensively include flights out of the ecliptic plane, rendezvous trips to asteroids, surveys of the outer planets and, as a commercial example, the disposal of radioactive waste. The NASA Lewis/Hughes 30cm thruster[24] shown in Fig 9 has reached a flight-ready state of development, and would be suitable for these and many other missions.

Orbit transfer manoeuvres[4] are accomplished by thrusting tangentially to the orbital path. Very large payloads can be raised from low earth orbit (LEO) to geostationary earth orbit (GEO) with only a small expenditure of propellant, but at the cost of a long transfer time, typically 200 days or more. The thrusters used can be part of the spacecraft or can, with their solar arrays and orbit and attitude control system, constitute a separate re-usable space tug[33]. The latter has been studied in great depth, confirming its advantages; some numerical results are given in Fig 10, in which the performances of various propulsion systems are compared.

5.3 Future large-scale space operations

Plans for the 'industrialisation' of space call for the construction in orbit of structures of ever-increasing size and mass, with very long lives. Such structures include antenna 'farms', materials processing bases and, ultimately, huge solar power satellites[34]. In all cases, very considerable advantages would result from using EP for all in-orbit propulsion tasks, ranging from attitude control to LEO to GEO orbit transfers. In view of the large spacecraft masses involved, arrays of thrusters would be required, and, eventually, individual devices would have to be developed giving much more thrust per unit than the 30cm motor shown in Fig 9.

As an example of what may be possible, an orbit transfer vehicle designed by Rockwell[35] is shown in Fig 11. This enormous structure is intended to transport components and materials from LEO to GEO in preparation for the construction of solar power satellites in GEO. It is powered by 144 ion thrusters using argon propellant, each giving 70 N thrust at an SI of 8213 s. The empty mass is 1090 tonnes and the loaded mass is 5170 tonnes, so the payload/empty mass ratio is a remarkable 4.74. The LEO-GEO-LEO return mission takes 150 days and consumes 667 tonnes of argon. A vehicle such as this requires considerably less than 0.2 kg of propellant per kg of mass for the mission, whereas the best chemical propulsion system needs 2 kg of propellant per kg, or more.

6 TERRESTRIAL APPLICATIONS

Many important terrestrial applications of EP technology have emerged in recent years. Some are almost self-evident, such as the relevance of colloid thrusters to liquid spraying techniques, notably paint spraying, and the application of field-emission ion sources to ion microscopy. Others are less obvious; an example might be the use of pulsed plasma rail guns, with ablative propellant injection, to plate components with materials such as gold.

However, Kaufman-type ion thrusters and various derivatives probably have the widest range of potential applications[36], reflected to some extent in the sales of simplified commercially-orientated versions. These have been designed with low cost and ease of maintenance in mind; unlike the situation in space, efficiency, durability and low mass are of secondary importance. Such ion sources are attractive because all beam characteristics can be controlled independently; these are velocity (*ie* energy), current density, divergence, density profile and composition.

Ion beams are capable of altering the surface properties of materials, through both implantation and sputtering processes. Parameters that can be changed in this way include optical emissivity and reflectivity, field-emission, corrosion resistance, surface hardness, and the adhesion of coatings. The possibilities offer enormous scope for experimentation. One example, shown in Fig 12, is the use of sputtering to modify the adhesive properties of surgical implants[37]. In many cases, this adhesion is inadequate, resulting, for instance, in artificial hip joints becoming loose, due to failure of the cement used to bond them to bone. There is some evidence that sputtering treatment, as depicted in Fig 12, can both eliminate the need for a cement and improve adhesion. The sputtering depth can be carefully controlled so that the living tissue penetrating into the pits so formed is not so far away from its source of nutrients that it dies.

A potentially very rewarding application of ion thruster technology is to ion beam machining, particularly of semiconductors[38]. In the construction of integrated circuits there has always been great emphasis on making each device and all interconnections as small as possible, to minimise size and improve speed of operation. This process is severely limited, in chemical etching methods, by under-cutting beneath the mask of photo-resist. At present, there is a great deal of effort being expended in reducing interconnection sizes to the 1 μm level and below; it has been suggested that this could be improved to the 10^{-2} μm range by employing ion beam machining techniques, which eliminate under-cutting. Such a dramatic improvement would have a massive impact on semiconductor technology.

7 CONCLUSIONS

The Report has shown that very significant advantages may be gained by using electric propulsion systems, due to the very high exhaust velocities that they provide. The various types of system available have been reviewed, and it has been concluded that the Kaufman electron bombardment ion thruster offers the greatest promise, although the RF ion thruster is also highly developed in its smaller version, and pulsed plasma thrusters have performed operational missions successfully.

After describing the Kaufman thruster in greater detail, the Report covered some of the missions that can be performed by electric propulsion systems, ranging from station-keeping to the transportation of solar power satellites to geostationary orbit. Finally, various important terrestrial applications of this technology were briefly mentioned.

REFERENCES

No.	Author	Title, etc
1	H.R. Kaufman	An ion rocket with an electron bombardment ion source. NASA TN D-585 (1961)
2	M. Alperin G.P. Sutton (Editors)	Advanced propulsion systems. Pergamon Press, Symposium Publications Division (1959)
3	T. Rees D.G. Fearn	Use of the UK 10cm ion thruster for north-south station-keeping of the proposed European TVBS spacecraft. AIAA Paper 76-1064 (1976)
4	D.G. Fearn	The use of ion thrusters for orbit raising. *J. Brit. Interplan. Soc.*, 33, 4, 129-137 (1980)
5	D.G. Fearn	The application of ion propulsion to the transportation and control of solar power satellites. AIAA Paper 81-0760 (1981)
6	B.A. Banks	Ion beam applications research - Lewis Research Center 1981. AIAA Paper 81-0669 (1981)
7	J.A. Donovan W.T. Lord	Preliminary testing of a high performance 3kW hydrogen resistojet. AIAA Paper 72-449 (1972)
8	L.K. Rudolph R.G. Jahn K.E. Clark W.F. von Jaskowsky	Performance characteristics of quasi-steady MPD discharges. AIAA Paper 76-1000 (1976)
9	D. Grim	Electric propulsion for solar power satellites. AIAA Paper 78-695 (1978)
10	B.C. Barber D.G. Fearn	Measurements of the thrust and exhaust characteristics of a pulsed rail thruster using mercury propellant. Proc. DGLR - Symposium Elektrische Antriebssysteme, Braunschweig, June 1971. DLR Mitteilung 71-21, pp 331-349 (1971)
11	D.G. Fearn C.M. Philip	An investigation of physical processes in a hollow cathode discharge. AIAA Paper 72-416 (1972)
12	A.G. Bailey	Investigation of a single spraying site on a colloid thruster. *J. Phys. D.*, 6, 276-288 (1973)
13	S. Zafran F.B. Mead	Constant power, constant mass flow control colloid propulsion system. AIAA Paper 76-1043 (1976)

REFERENCES (continued)

No.	Author	Title, etc
14	ESA	Field emission day. ESA SP-119 (1977)
15	G. Eschard A. Pelissier J. Paulin J.F. Bonnal	Experimental studies of a linear strip cesium contact ion thruster. *J. Spacecraft and Rockets*, 13, 4, 193-197 (1976)
16	H. Bassner W.U. Klein	Recent progress in development of RF-ion thruster system RITA and application aspects. AIAA Paper 79-2101 (1979)
17	H. Bassner H.P. Berg W. Birner H. Mueller P. Mueller W.U. Klein	Space flight test of RIT 10 ion thruster system on European Heavy Satellite. AIAA Paper 78-660 (1978)
18	W. Pinks J. Lensin H.W. Loeb	The Giessen RF-primary propulsion engine RIT 35. AIAA Paper 76-1008 (1976)
19	D.G. Fearn R.C. Hughes	The T5 10cm mercury ion thruster system. AIAA Paper 78-650 (1978)
20	W.R. Kerslake L.R. Ignaczak	SERT II 1980 extended flight thruster experiments. AIAA Paper 81-0665 (1981)
21	E.L. James W.D. Ramsey	A prototype 12cm cesium ion thruster for north-south stationkeeping. AIAA Paper 75-387 (1975)
22	D.G. Fearn	Electric propulsion in the UK. Proc. Third European Electric Propulsion Conference, Hinterzarten, 1974. DGLR - Fachbuchreihe Band 5, pp 22-30, (1974)
23	J. Lazar	Electric propulsion status and development plans - NASA programs. AIAA Paper 73-1143 (1973)
24	R.L. Poeschel R.P. Vahrenkamp	High power operation of a 30cm mercury bombardment ion thruster. AIAA Paper 76-1007 (1976)
25	B.G. Herron J. Hyman D.J. Hopper	Development of an 8-cm engineering model thruster system. AIAA Paper 76-1058 (1976)
26	W.J. Guman D.J. Palumbo	Pulsed plasma propulsion system for North-South stationkeeping. AIAA Paper 76-999 (1976)

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REFERENCES (concluded)

No.	Author	Title, etc
27	G. Krülle	Payload capacity of Ariane launched geostationary satellites using an electric propulsion system for orbit raising. *Acta Astonautica*, 7, 633-652 (1980)
28	D.G. Fearn D. Stewart P.J. Harbour G.L. Davis J. Williams	The UK 10cm mercury ion thruster development program. AIAA Paper 75-389 (1975)
29	G.E. Keyte R.E.W. Jansson	The use of augmented electrothermal hydrazine thrusters in station-keeping applications. Proc. Internat. Conf. on properties of hydrazine and its applications as a source of energy, Poitiers, France (1974)
30	K. Kuriki S. Morimoto K. Nakamura	Flight performance test of MPD thruster system. AIAA Paper 81-0664 (1981)
31	P. Hong R. Boain	SEPS navigation requirements for 1981 Encke rendezvous mission. AIAA Paper 75-423 (1975)
32	K.L. Atkins	Mission applications of electric propulsion. AIAA Paper 74-1085 (1974)
33	R. Holdaway R.E.W. Jansson	The feasibility of geostationary emplacement of satellites by solar electric tug. *J. Brit. Interplan. Soc.*, 26, 220-227 (1973)
34	P.E. Glaser	Solar power satellite developments. *J. Astronautical Sciences*, 26, 2, 101-127 (1978)
35	G.M. Hanley	Satellite power systems (SPS) concept definition study. Vol.IV. Transportation analysis, NASA CR-3321 (1980)
36	B.A. Banks	Ion beam applications research - Lewis Research Center 1981. AIAA Paper 81-0669 (1981)
37	B.A. Banks A.J. Weigand C.A. Babbush C.L. van Kampen	Potential biomedical applications of ion beam technology. NASA TM X-73512 (1976)
38	D.D. Robertson	Advances in ion beam milling. *Solid State Technology*, 57-60, December 1978

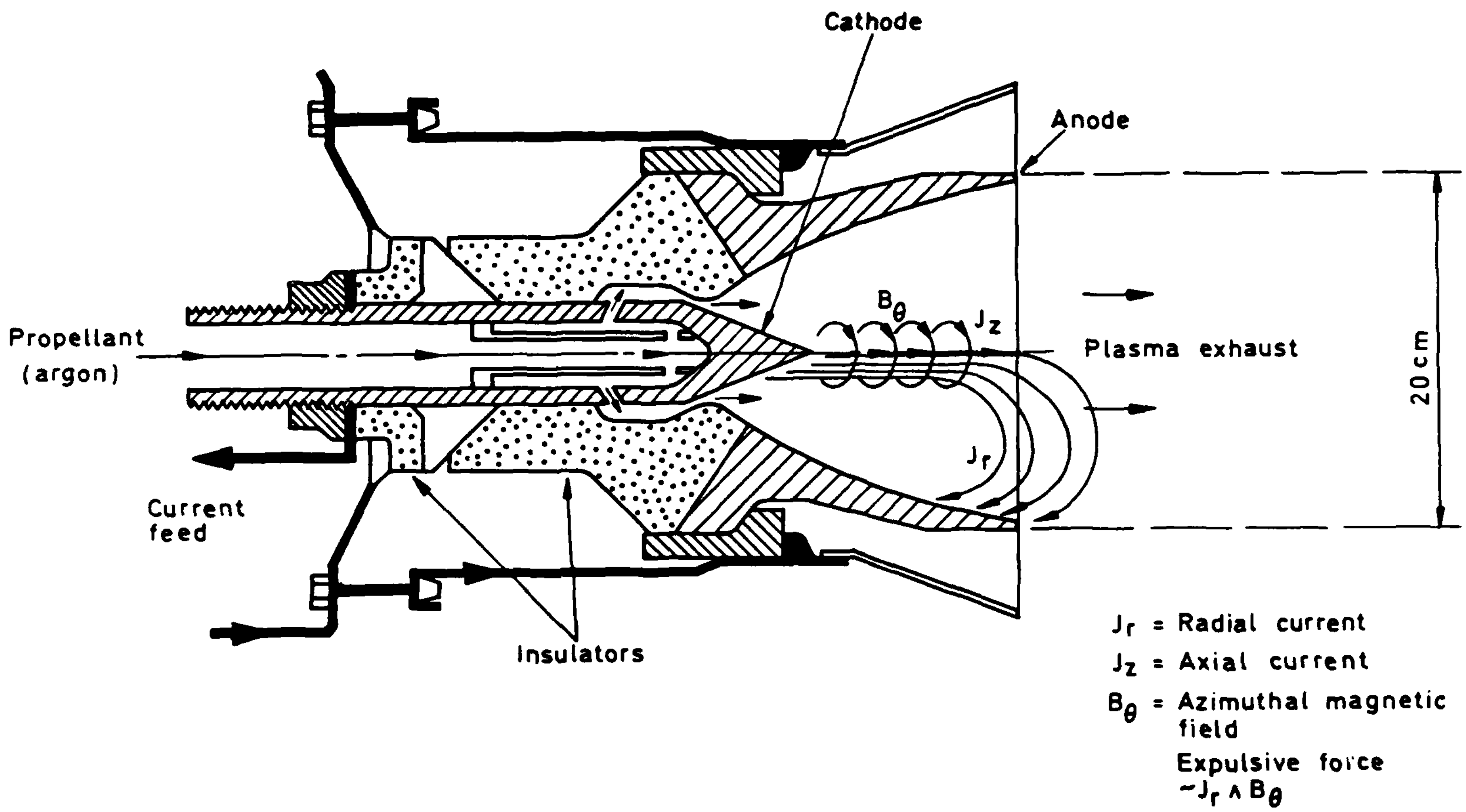

Fig 1 Schematic of 20 cm diameter multi-MW MPD thruster (based on Ref 9)

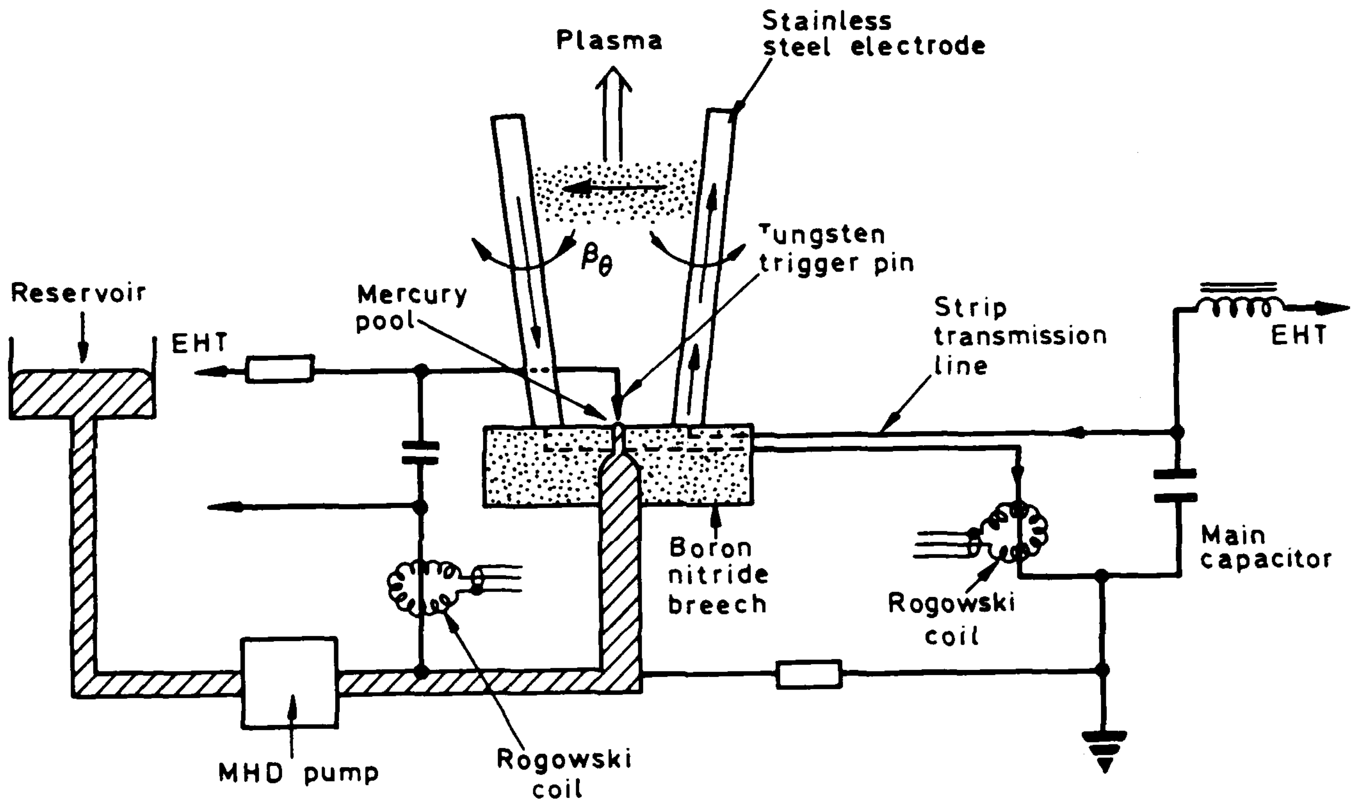

Fig 2 Schematic of pulsed plasma rail thruster developed by RAE Farnborough

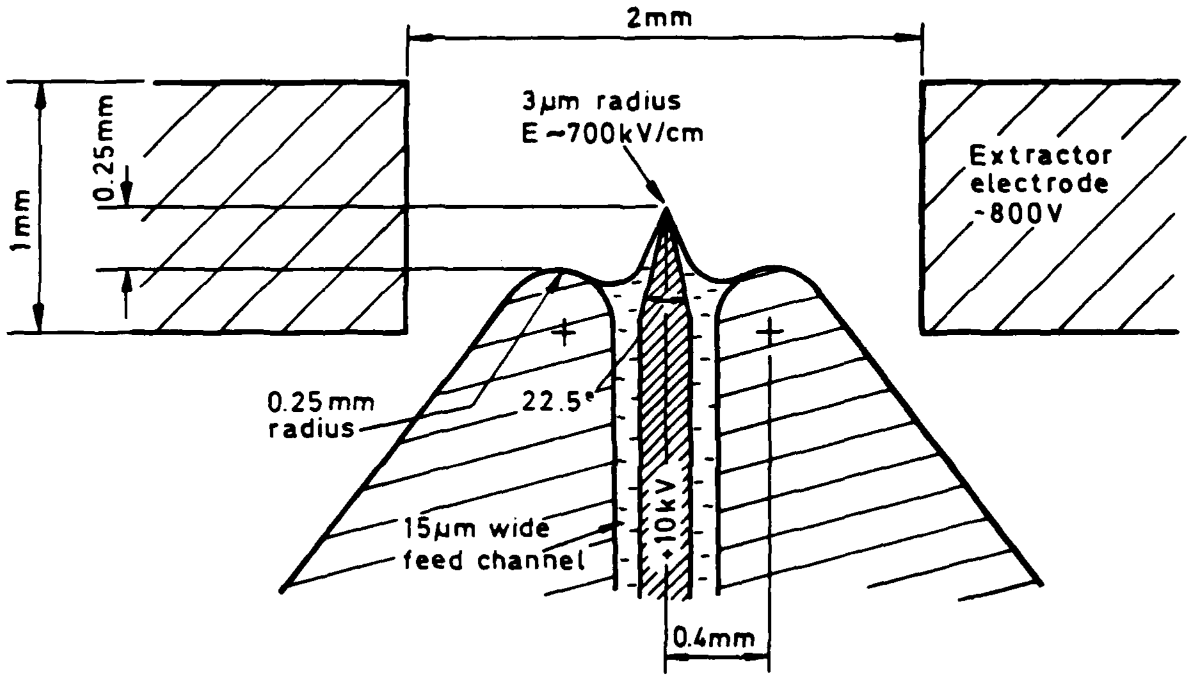

Fig 3   Schematic of linear raised edge emitter colloid thruster

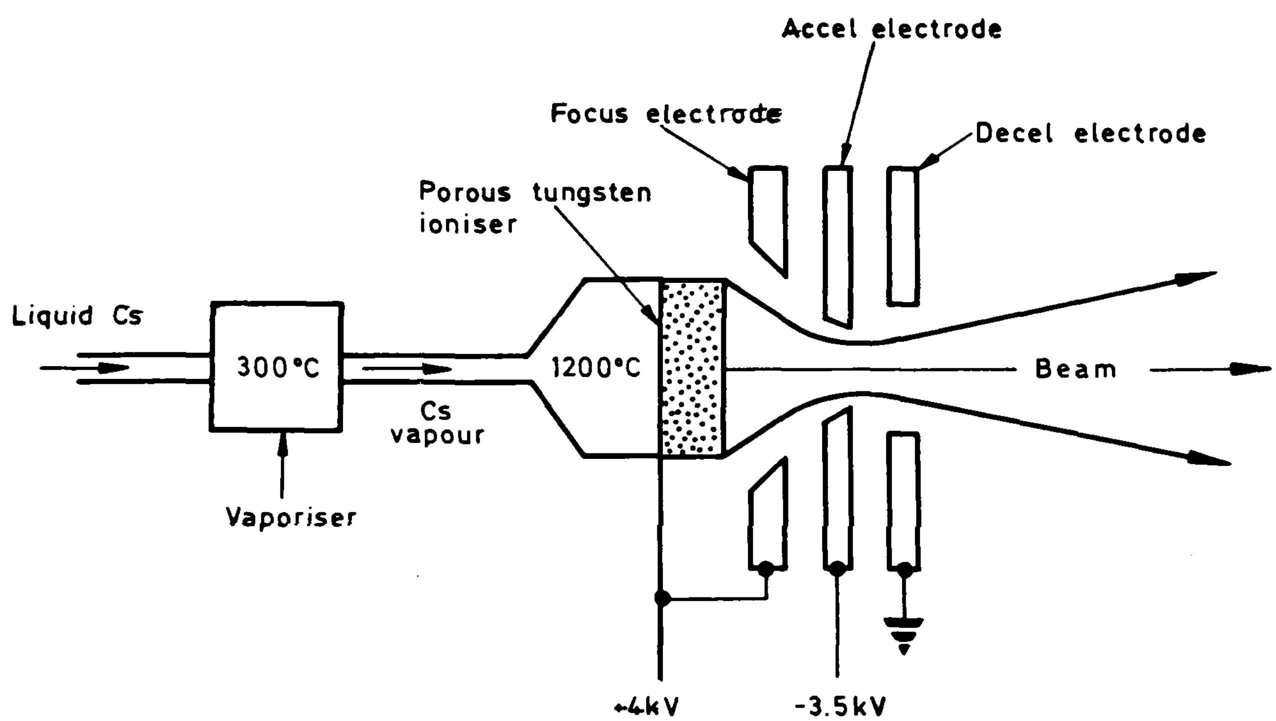

Fig 4 Schematic of contact ionisation ion thruster, using Cs propellant

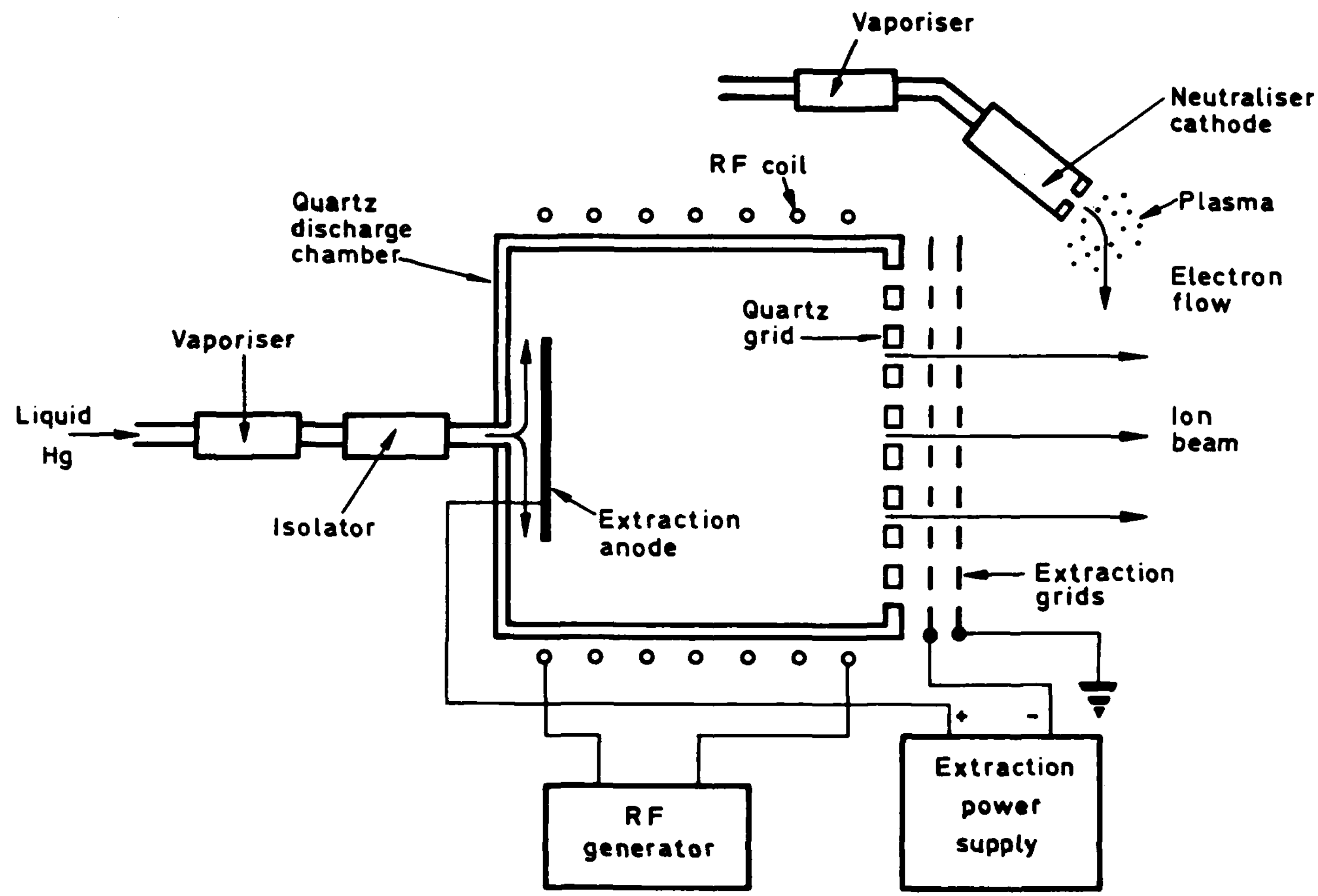

Fig 5 Schematic of RF ion thruster

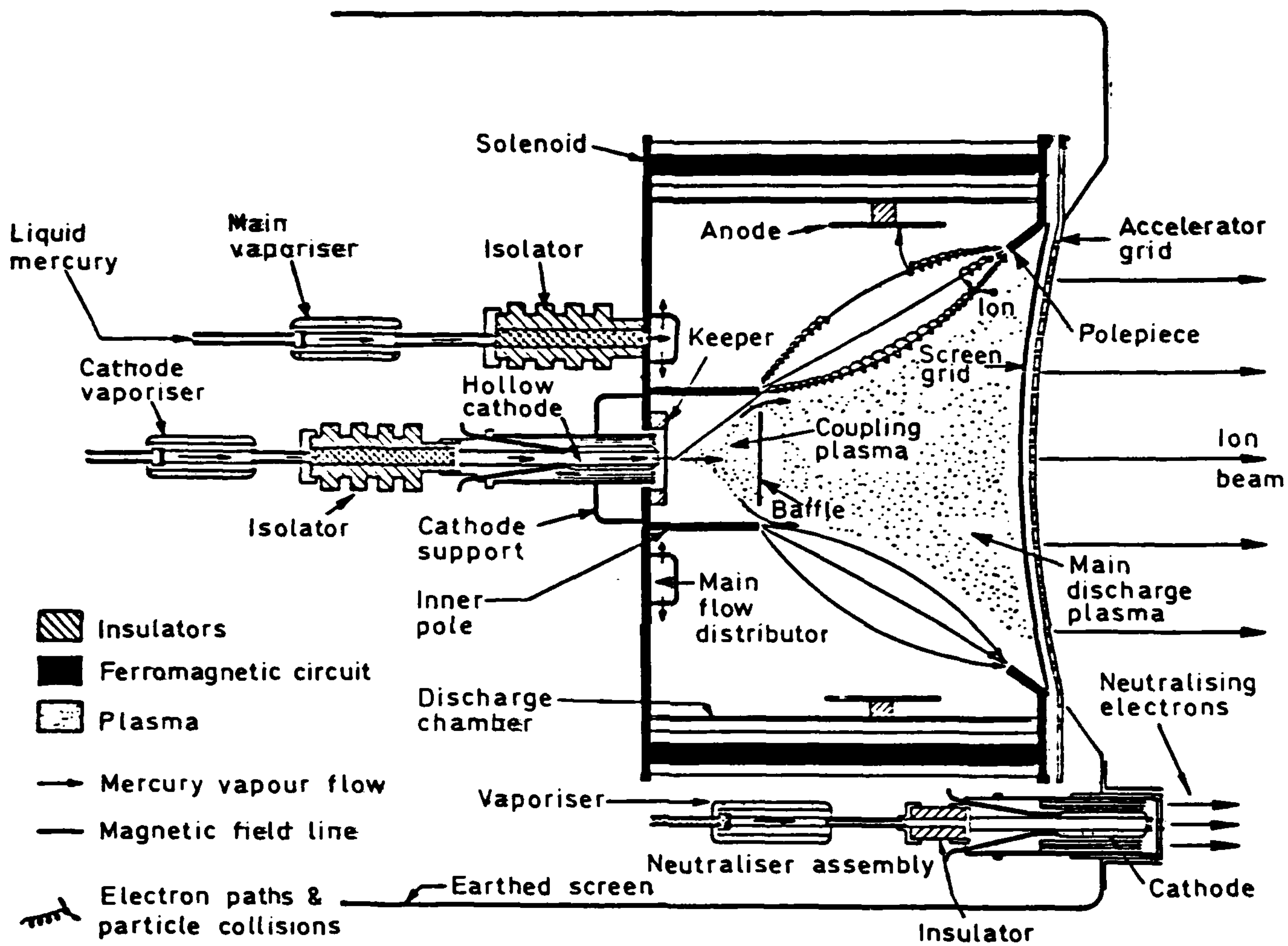

Fig 6 Section through Kaufman electron bombardment ion thruster (based on T5 device, developed by RAE Farnborough)

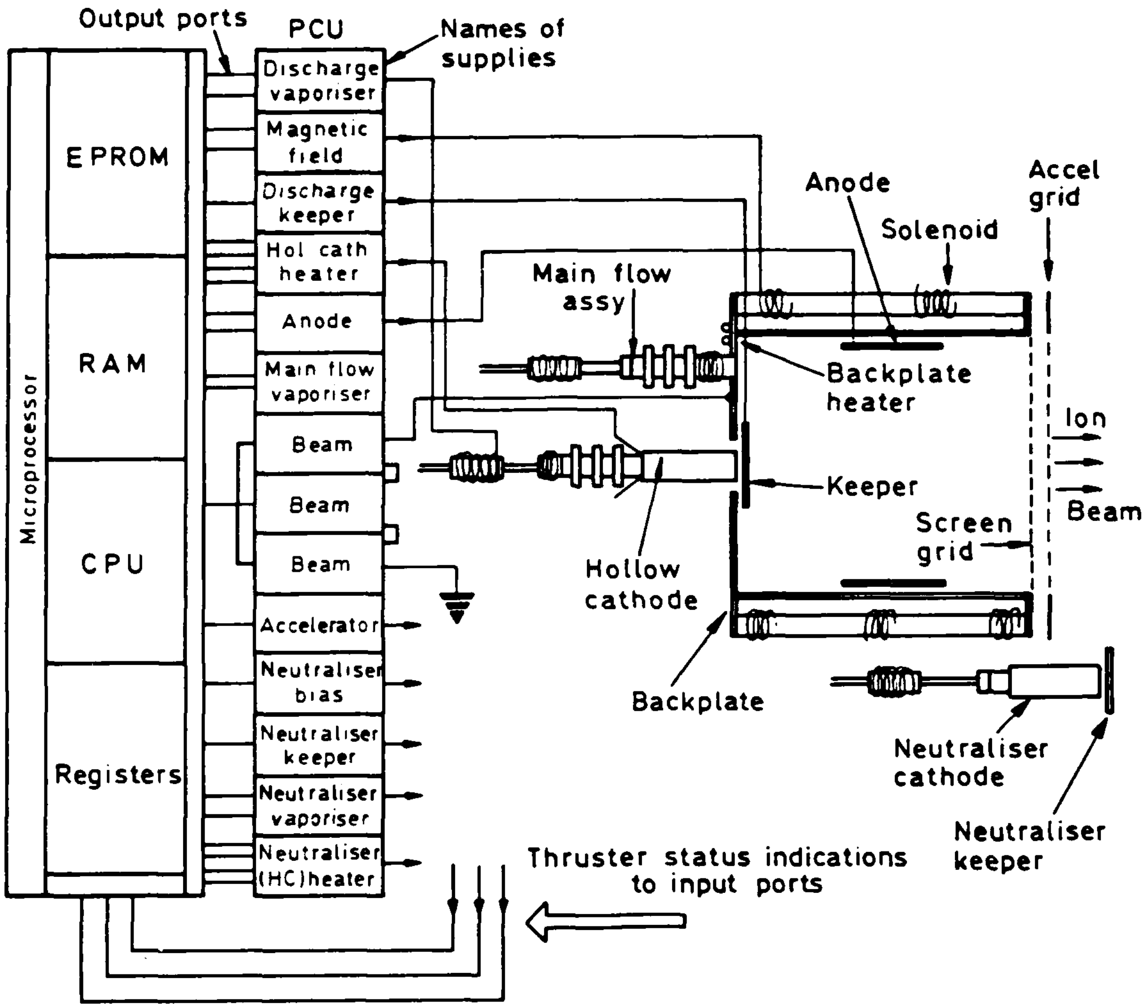

Fig 7 T5 thruster power supply and control systems

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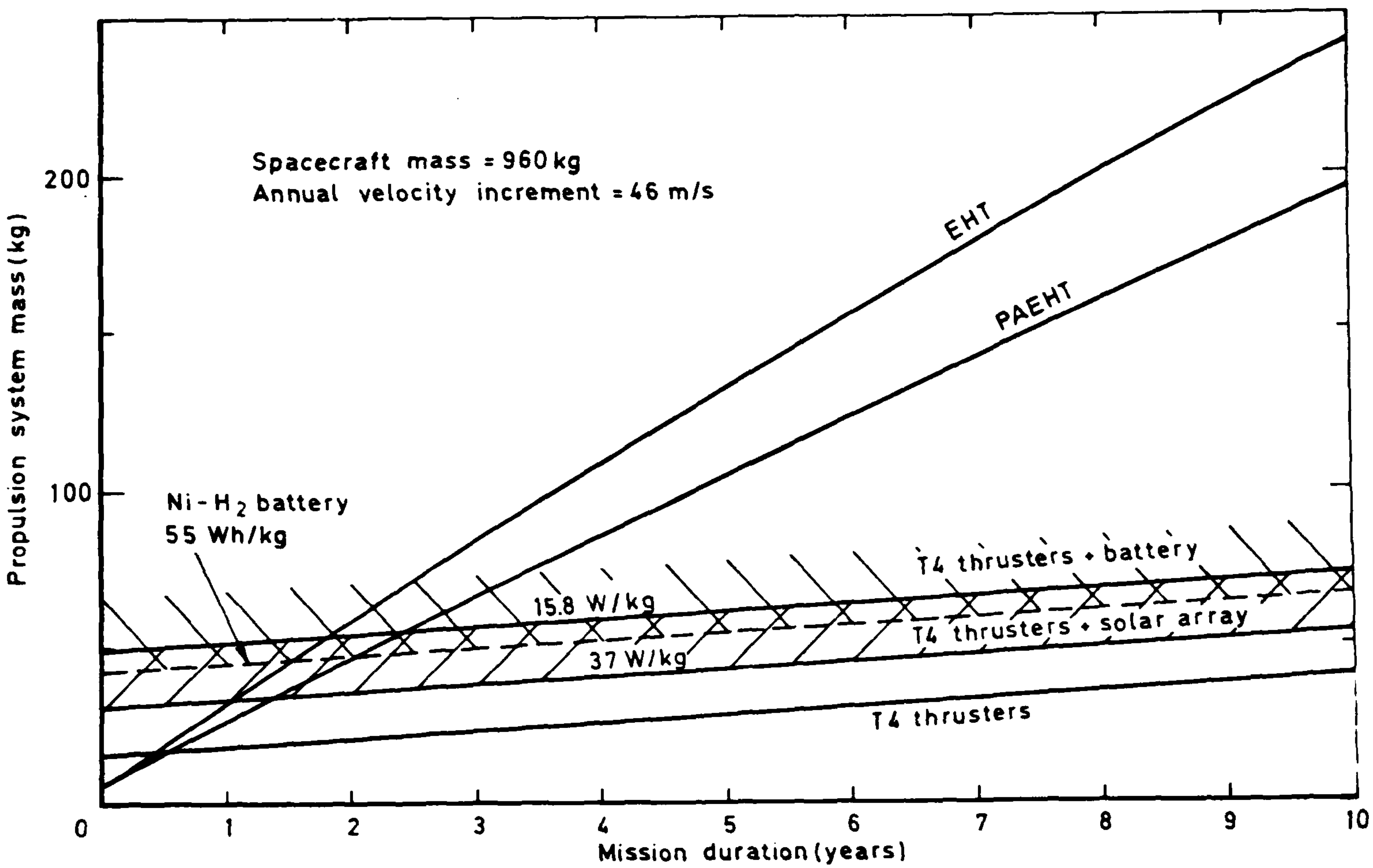

Fig 8 NSSK propulsion system mass as a function of mission time for EHT, PAEHT and ion thruster systems

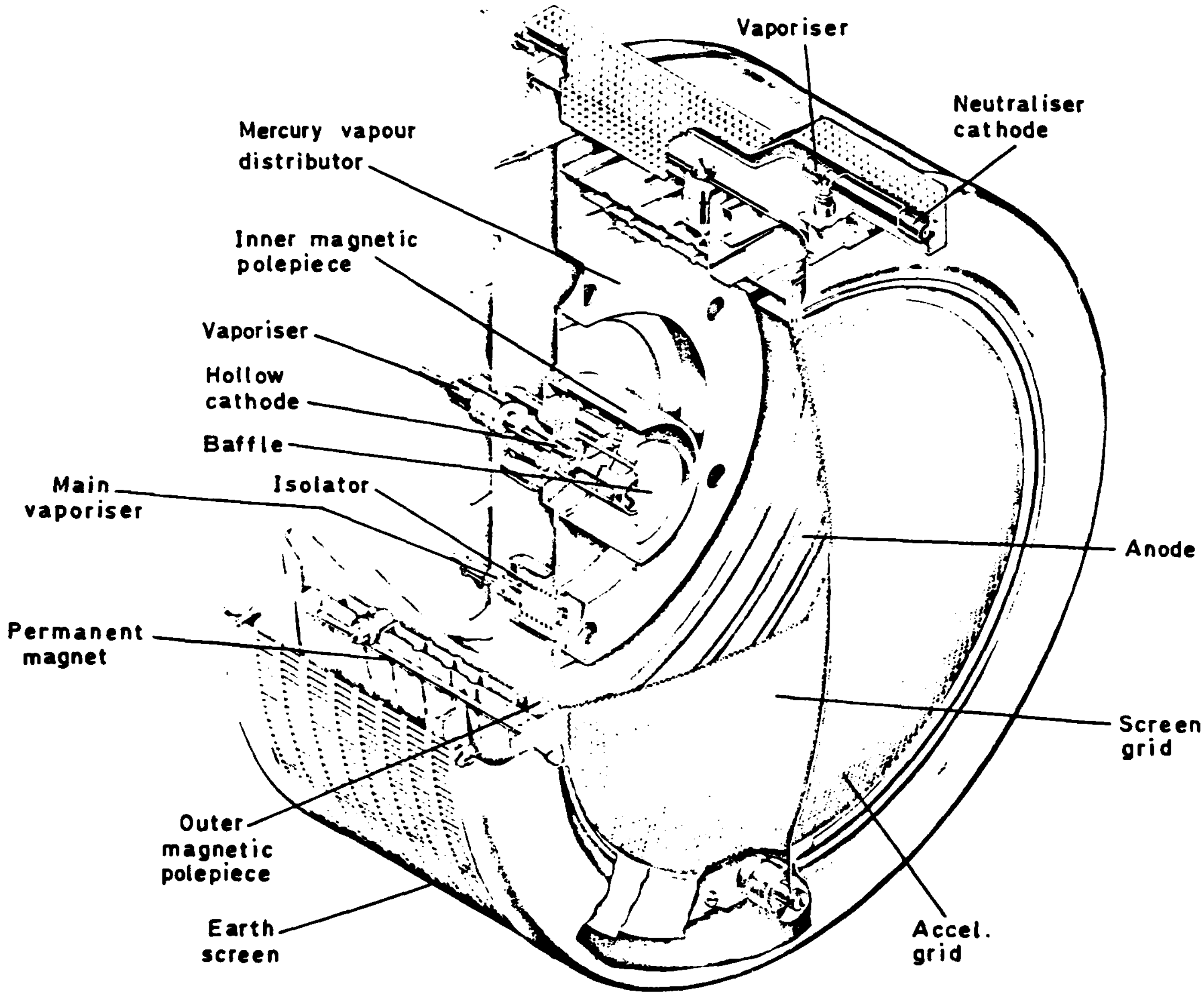

Fig 9 Section of NASA Lewis/Hughes R.L. 30 cm Kaufman ion thruster

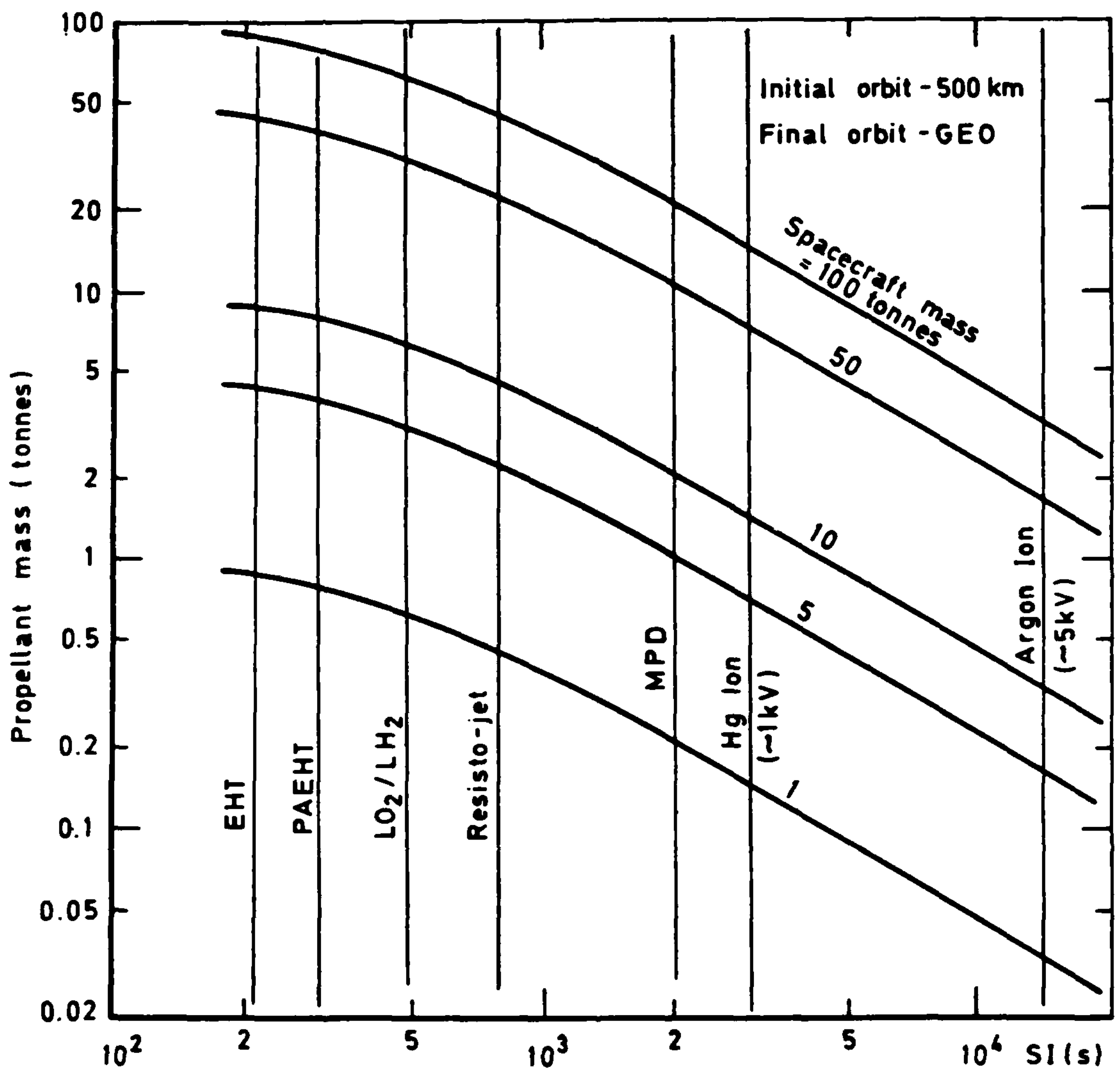

Fig 10 Propellant mass required for the LEO to GEO mission as a function of propulsion system SI, for a range of initial spacecraft masses

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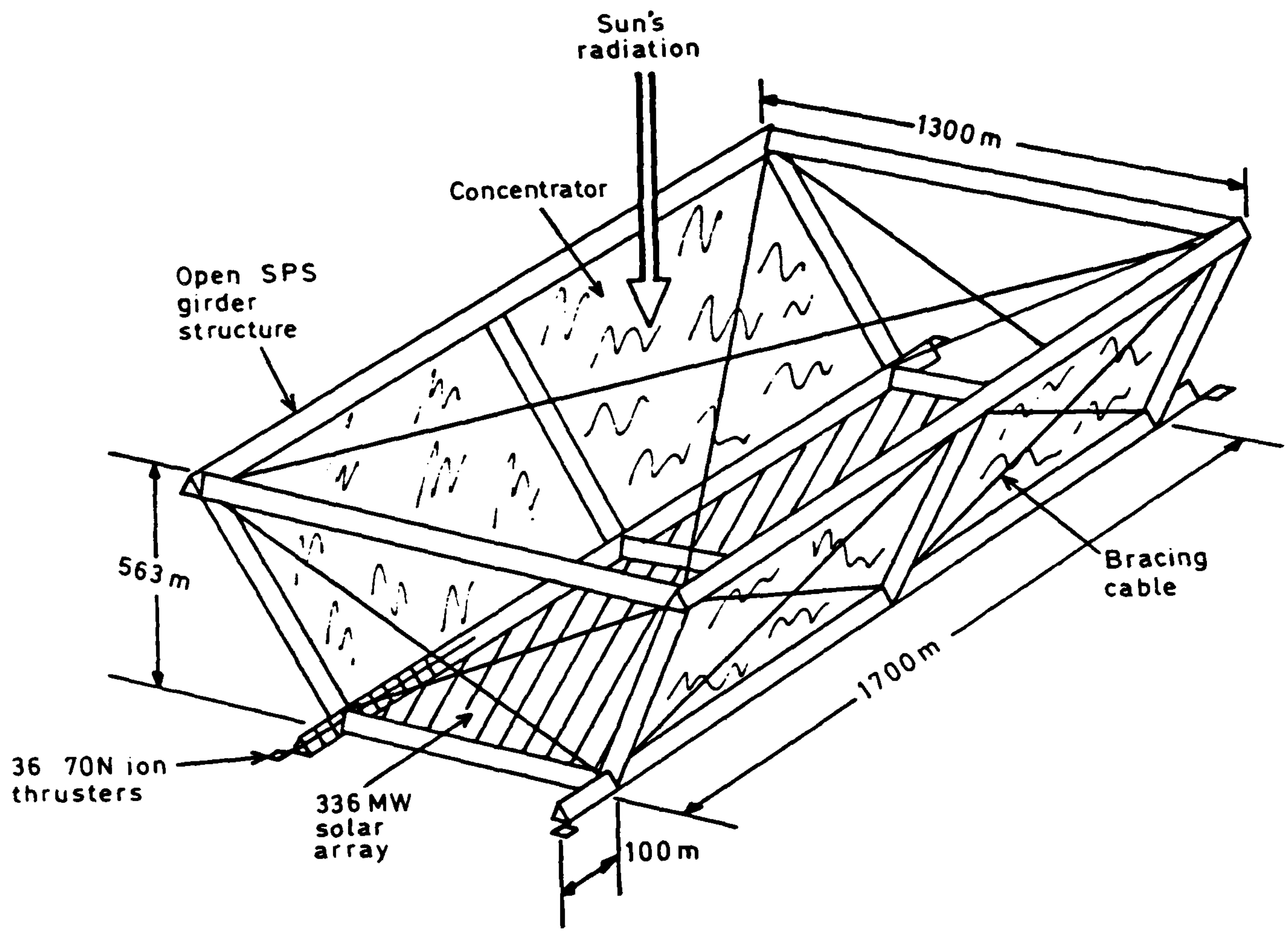
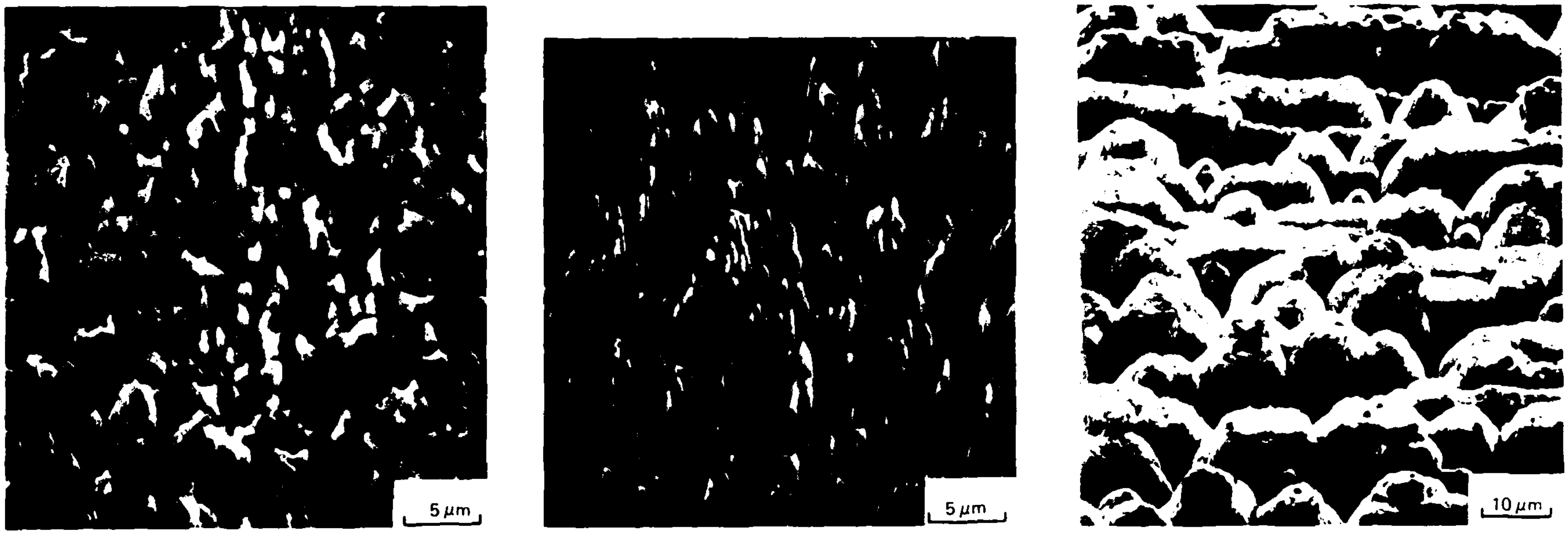


Fig 11 Solar electrically propelled orbit transfer vehicle designed for solar power satellite construction (Ref 35)



a b c

Fig 12a-c Scanning electron micrographs of surgical alloys used for orthopedic prosthesis, typically for artificial hip joints, after ion beam texturing:
(a) Titanium—6%Al—4%V, (b) Surgical stainless steel (ASTM F55—71)
(c) Cobalt—Chromium alloy (Ref 37)

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17. Abstract

The Report commences with a discussion of the merits of electric propulsion technology, explaining that the very high exhaust velocities attainable allow the propellant masses required for most missions to be drastically reduced. The various types of electric thruster are then described briefly. The most highly developed and potentially useful thruster, the Kaufman electron bombardment ion thruster, is covered in greater detail, with particular reference to the T5 device developed in the UK. Candidate missions are discussed, ranging from attitude and orbit control functions to the application of ion propulsion to the deployment of solar power satellites. Important terrestrial applications of electric propulsion technology are also mentioned.

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