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A REVIEW OF FUTURE ORBIT TRANSFER TECHNOLOGY.(U)
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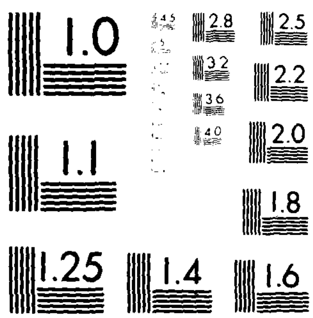

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ROYAL AIRCRAFT ESTABLISHMENT

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Technical Report 81080

June 1981

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A REVIEW OF FUTURE ORBIT TRANSFER TECHNOLOGY

by

D. G. Fearn

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Farnborough, Hants

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REPORT DOCUMENTATION PAGE

Overall security classification of this page

UNCLASSIFIED

As far as possible this page should contain only unclassified information. If it is necessary to enter classified information, the box above must be marked to indicate the classification, e.g. Restricted, Confidential or Secret.

1. DRIC Reference (to be added by DRIC)	2. Originator's Reference	3. Agency Reference	4. Report Security Classification/Marking
	RAE TR 81080	N/A	UNCLASSIFIED

5. DRIC Code for Originator	6. Originator (Corporate Author) Name and Location
7673000W	Royal Aircraft Establishment, Farnborough, Hants, UK
5a. Sponsoring Agency's Code	**6a. Sponsoring Agency (Contract Authority) Name and Location**
N/A	N/A

7. Title

A review of future orbit transfer technology

7a. (For Translations) Title in Foreign Language

7b. (For Conference Papers) Title, Place and Date of Conference

8. Author 1. Surname, Initials	9a. Author 2	9b. Authors 3, 4	10. Date	Pages	Refs.
Fearn, D.G.			June 1981	64	58
11. Contract Number	**12. Period**	**13. Project**	**14. Other Reference Nos.**		
N/A	N/A		Space 599		

15. Distribution statement

(a) Controlled by – Head of Space Department, RAE (RAL)

(b) Special limitations (if any) –

16. Descriptors (Keywords) (Descriptors marked * are selected from TEST)

Space transportation. Electric propulsion. Chemical propulsion. Orbit raising. Orbit transfer. Large payloads.

17. Abstract

Probably the most important factor in deciding the viability of future space projects will be the cost of transportation. This is particularly true of those missions categorised as 'space industrialisation', which include plans to place massive solar power satellites in geostationary orbit. Although a major proportion of the transportation costs can be attributed to the initial stage of reaching low earth orbit, in many cases the expense of transferring cargoes and personnel to outer orbits will be crucial. This paper reviews the technological options available for this second stage, with emphasis on large, long-term projects, such as solar power satellites. It is concluded that chemical propulsion systems will be needed for the transfer of personnel, but that the high specific impulse offered by electric propulsion provides an enormous economic advantage for the movement of non-priority cargo.

F5910/1

RAE Form A143

UDC 629.193.5 : 629.19.085 : 629.19.035.52 : 629.19.035.51

ROYAL AIRCRAFT ESTABLISHMENT

Technical Report 81080

Received for printing 30 June 1981

A REVIEW OF FUTURE ORBIT TRANSFER TECHNOLOGY

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SUMMARY

Probably the most important factor in deciding the viability of future space projects will be the cost of transportation. This is particularly true of those missions categorised as 'space industrialisation', which include plans to place massive solar power satellites in geostationary orbit. Although a major proportion of the transportation costs can be attributed to the initial stage of reaching low earth orbit, in many cases the expense of transferring cargoes and personnel to outer orbits will be crucial. This paper reviews the technological options available for this second stage, with emphasis on large, long-term projects, such as solar power satellites. It is concluded that chemical propulsion systems will be needed for the transfer of personnel, but that the high specific impulse offered by electric propulsion provides an enormous economic advantage for the movement of non-priority cargo.

Paper presented at the Symposium on "Space Transportation Systems for the 1990s - Requirements and Solutions", held at the Headquarters of the British Interplanetary Society, London, 15 April 1981.

Departmental Reference: Space 599

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1 INTRODUCTION

Since the beginning of the era of practical space flight, vast resources have been devoted to the development of rocket vehicles capable of raising ever-increasing payloads from the surface of the earth to orbits of a few hundred kilometres altitude, or to transfer orbits from which further manoeuvres can be commenced. Most of this effort has been concentrated upon multi-stage expendable systems employing conventional chemical propulsion technology, hence the very high costs incurred. This technology has also been adopted, perhaps in slightly modified form, for the later stages of the more exotic missions, including those of the Apollo programme and the exploration of the planets of the solar system.

Although the great expense of this approach was acceptable during the early stages of the conquest of space, the large-scale 'industrialisation'[1] and exploration expected during the remainder of this century and in the more distant future, heralded by the development of the Space Shuttle, demand significant cost reductions. This will be increasingly necessary as the orbiting structures required to accomplish more onerous tasks become larger and heavier, culminating perhaps in the deployment of spacecraft as large as solar power satellites[2] (SPS).

In the case of transportation to low earth orbit (LEO), the Shuttle and its derivatives should reduce costs very significantly, mainly by extensively re-using as much of the hardware as possible, but also by employing the latest propulsion, structural, electronics and aerodynamics technology. This advance will have a major impact on the economic viability of many tasks which can be performed in LEO, such as materials processing in a near-zero g environment. However, a very large proportion of envisaged missions require the transportation of payloads to geostationary earth orbit (GEO), or to highly elliptical orbits or escape trajectories. No new low-cost technology is being actively developed at present to cater for this need, although the inertial upper stage (IUS) is a first step in this direction.

This Report covers the technological options available for this important additional transportation task, with particular emphasis on the LEO to GEO mission. Both moderately sized and large payloads are considered, up to those appropriate to SPS deployment, and chemical, nuclear and electric propulsion technologies are discussed. If it is accepted that cost is the most important factor in the decision-making process, the conclusion is unavoidable that solar-powered electric propulsion (EP) is virtually mandatory for the transportation of cargo from LEO outwards. However, the resulting orbit transfer times are lengthy, perhaps 100-300 days for the trip to GEO[3,4], thus necessitating the use of high thrust chemical systems for personnel or priority cargo. Consequently, there is an important future for both forms of propulsion.

Although nuclear concepts are promising, particularly for powering EP systems[5], there is considerable doubt as to their political and environmental acceptability. Consequently, except in very special cases, they must be disregarded at the present time.

2 FUTURE ORBIT TRANSFER MISSIONS

It is convenient for the present discussion to define three categories of mission by qualitative reference to both in-orbit mass and development timescale. These categories are described below, with the assistance of examples of representative spacecraft.

2.1 Near-term/low mass missions

These missions are arbitrarily taken to be those for which transportation to LEO can be accomplished by a single launch of an expendable rocket, such as Ariane, or of the Shuttle. Many such spacecraft have already been placed into orbit, and an increasing number will be developed in the remaining years of this century. Of those requiring a significant degree of in-orbit propulsion, primarily for LEO to GEO transfer, communications spacecraft are likely to form the majority.

An interesting example of an advanced communications spacecraft destined for the 1990 timescale is depicted in Fig 1. This design, due to Spar Aerospace of Canada, is intended to serve mobile ground communications systems. It has a 50m diameter antenna, a very long boom carrying the transponders at its far end, a 5-6kW solar array, and a life of 7-10 years. Its predicted mass, without apogee motor and north-south station-keeping (NSSK) propellant, is 3091 kg, which is at the limit of the capabilities of the Shuttle/IUS combination. It is interesting to note that a single Shuttle launch will suffice to accomplish this mission only if EP is used for NSSK[6], assuming that the IUS is employed for the orbit transfer manoeuvre.

An improved performance may be obtained by replacing the IUS by a liquid fuelled upper stage; plans here have concentrated on using the Centaur upper stage, perhaps modified to take advantage of the large diameter payload capability of the Shuttle. Under the latter circumstances, it has been estimated[9] that about 6400 kg can be delivered to GEO. Consequently, a single Shuttle launch can be expected to be able to cater for spacecraft significantly larger than that shown in Fig 1.

These mass limitations can be raised to much higher levels still, by utilising electric propulsion for the orbit transfer manoeuvre[3], although at the cost of long trip times. This principle also applies in the case of expendable launch vehicles, as was recognised during the ELDO programme[7]. Its application to Ariane[8] is currently under investigation.

2.2 Medium-term/moderate mass missions

The rapid expansion of communications traffic handled by spacecraft has already led to difficulties in providing adequate separation between satellites in certain arcs of the geostationary orbit. Although this problem can be alleviated somewhat by improved station-keeping and antenna design, the ultimate solution is likely to be the use of large, multi-purpose platforms or 'antenna farms'. It is probable that these will be deployed initially during the last decade of this century.

Large platforms of this type will require several launches of the Shuttle to place their component parts in LEO. It is probable that they will be assembled in LEO, possibly by astronauts, and that they will then be propelled to GEO via a spiral trajectory, using relatively low thrust chemical rockets or EP systems. A low thrust is necessary because the structures involved should not be any stronger (and heavier) than required for operation in GEO[10].

As an illustration of the type of spacecraft that might be developed, a communications platform described by Kunz[10] is shown schematically in Fig 2. The overall length of the structure is 240 metres and its basic width is 12 metres. An erectable pentahedral truss construction technique is employed, using graphite-epoxy composite material. The mass in GEO is estimated to be about 60640 kg, including a silicon solar array of 1312 m^2 area. The linear configuration, with a length to width ratio of 20:1, results from various constructional constraints associated partly with use of the Shuttle, but is is also of benefit in minimising the effects of thrusting during orbit transfer. Of course, many other platform configurations are equally valid; that depicted in Fig 2 is given merely as an example.

2.3 Long-term/large mass missions

Solar power satellites[2] are the largest and most massive structures that are likely to be placed in orbit within the next several decades. Although they represent an enormous technical challenge, no scientific, environmental or technical reasons have been identified which cause the concept to be rejected. Consequently, SPS development can proceed, once the necessary political and financial decisions have been made, and it is possible that the first demonstration spacecraft could be undergoing trials early in the next century.

The magnitude of the transportation task, to both LEO and GEO, can be illustrated by examining the DOE/NASA reference SPS designs[11] shown in Fig 3. These were evolved through very extensive studies of all possible alternatives by many contractors, and therefore represent highly refined concepts. In both cases, a microwave power transmission system is employed, with an electrical output from the ground rectenna of 5 GW. This power level, which is largely dictated by the physics of the transmission process, determines the huge dimensions of the spacecraft. The masses are correspondingly large, 51000 tonnes for the planar silicon solar array option and 34000 tonnes for the alternative GaAlAs design. The final choice between these two, or indeed, between them and competing designs employing heat engines to convert solar radiation energy into electricity[12], will depend on future developments of solar cell technology.

With masses as large as these to be transported to GEO, it is evident that the cost of the orbit transfer process will be crucial to the economic viability of any SPS system. For this reason, a great deal of effort has been devoted to the detailed study of this topic, taking into account the operational phase, lasting perhaps 30 years, as well as the construction task. It has been generally concluded[4,13] that EP is essential for the transportation of cargo or sections of spacecraft constructed in LEO, but that personnel and priority cargo must employ vehicles depending on chemical propulsion.

3 PROPULSION TECHNOLOGY

If the case of personnel transportation is excluded, cost will probably be the most important factor in deciding between the various options available for constructing large spacecraft and for transporting them to their intended orbits. As an illustration of the dominant role of this factor in many situations, it has been estimated[4] that about 25% of the $11.3B unit cost of a 5GW SPS can be attributed to transportation, whereas only 7.7% is due to construction and to associated support activities. Transportation costs are even more dominant during operation, amounting to about 60% of the annual charge of $203M per 5GW satellite. These very large figures have been derived assuming the use of the most efficient transportation technology, so any reduction of efficiency will increase the quoted values substantially.

The cost of the LEO to GEO transportation task depends critically upon the consumption of propellant by the propulsion systems used. This is because the propellant must first be placed in LEO by either a conventional rocket, the Shuttle, or, in the more distant future, a heavy lift launch vehicle (HLLV). One of the most recent estimates[13] for HLLV payload costs is $32 to $35 per kg, whereas, in 1977 prices, Shuttle charges will be in the region of $900 per kg; these and other estimates[14] are given in Fig 4. Consequently, anything that can be done to reduce in-orbit propellant consumption is of considerable financial benefit.

The propellant consumption for a given manoeuvre, requiring a velocity increment ΔV, is determined by the specific impulse (SI) of the propulsion system, which is closely related to the effective exhaust velocity v_e. Here the SI, I_{sp}, is defined as $T/\dot{m}g_0$, where T is the thrust, $\dot{m}$ is the total rate of use of propellant, expressed in terms of units of sea level weight, and g_0 is the acceleration due to gravity. The advantage of using a high value of I_{sp} or v_e can be illustrated by reference to the rocket equation,

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

where M_0 is the initial mass of the vehicle, and ΔM is the propellant mass consumed during a manoeuvre giving velocity increment ΔV. It is evident that, for a given value of ΔV, a relatively small increase in v_e will enable ΔM to be reduced substantially.

Unfortunately, the value of v_e attainable with a chemical system is limited by the energy liberated in the chemical reactions taking place in the combustion chamber. Very little can be done here to advance beyond the performance level reached by the Shuttle's main engines, unless even more exotic fuels than the liquid oxygen/liquid hydrogen (LO_2/LH_2) combination are envisaged. Such limitations do not exist if electric propulsion techniques are employed. In these, the propellant is electrically charged, usually in a gaseous discharge process, and intense electromagnetic or electrostatic fields are used to accelerate it. Apart from the velocity of light, there is no fundamental restriction to the values of v_e and I_{sp} that may be achieved; currently, an SI of 3000-6000 seconds is commonplace, and 20000 seconds should be attainable in the near future[15]. These figures should be compared to the 475 seconds representative of the most advanced LO_2/LH_2 systems.

Intermediate values of SI can be realised by the use of resistojets, in which a propellant gas exhausting through a conventional nozzle is electrically heated. If hydrogen is used[16], I_{sp} can reach about 800 seconds, but power consumption is high. Similarly, gas core nuclear systems[17] should be able to attain about 2200 seconds. These alternative concepts are discussed in more detail in sections 3.4 and 3.5 below.

3.1 Transfer from LEO to GEO

Since this mission is likely to continue to be of very great importance, particularly if an SPS system is deployed, it will be considered in a little more detail. If the mass delivered to GEO by a single-stage orbit transfer vehicle (OTV) is M_f, and the semi-major axes of the initial and final orbits are a_0 and a_f respectively, it can be shown[3] that

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

where μ is the gravitational constant of the earth (3.986×10^{14} m^3/s^2) and g_0 is the acceleration due to gravity at sea level. This expression applies to equatorial orbits and ignores the relatively small effects due to residual air drag and eclipse. It can be shortened to $\Delta M = M_f \beta$, where β is a function of I_{sp}.

If it can be assumed that ΔM is a very small fraction of M_f, so that $M_0 \sim M_f$, an approximate expression can be derived for ΔM by equating the total impulse required for the mission to the momentum transferred to the spacecraft by the propulsion system[15]. This gives

$$\Delta M = \frac{\Delta V M_0}{g_0 I_{sp}} . \tag{2}$$

Equation (1) was used to derive the values of ΔM and β plotted against I_{sp} in Fig 5 for various SPS designs. It can be seen that ΔM falls dramatically as I_{sp} rises above a few thousand seconds, and that it becomes prohibitively large at an SI typical of chemical rocket motors. In addition, equation (2) was employed to calculate values of ΔM for an SPS having a mass of 34000 tonnes, assuming ΔV = 5700 m/s, which is appropriate to an LEO altitude of 500 km and 31.6 degrees initial inclination[18]. The result, the dashed curve in Fig 5, is above that derived from equation (1) at high SI, owing to the inclined orbit, but, at low SI, grossly underestimates the propellant requirement.

Similar results to those depicted in Fig 5 can be derived for other payloads. For example, equivalent curves for payloads launched into LEO by Ariane are plotted in Ref 3. In all cases, the advantages of increasing v_e are clearly evident.

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The cost savings of using a high SI have been quantified in many analyses of orbit transfer techniques. Recently, such work has been extended to the SPS, where the financial benefits have been shown to be enormous. For instance, Hanley[19] reported that the sum of $1.2B might be saved in the construction of a single 5GW SPS, using GaAlAs solar cells and concentrators, by employing EP rather than chemical propulsion for orbit transfer tasks. This assumes an SI well within current achievements. A further desirable consequence is a large reduction in the number of HLLV trips required; in the case quoted, this was 453 rather than 1092, assuming construction in GEO. Similar results were obtained by Davis[20] for a 10GW silicon SPS design originated by Boeing; as shown in Fig 6, it was deduced that the use of EP would approximately halve the number of HLLV trips to LEO required. As well as cost advantages, there are also significant environmental benefits to be gained.

3.2 Chemical propulsion

With a development history stretching over many decades, chemical rocket motors have almost reached a pinnacle of performance, as exemplified by the Space Shuttle Main Engine (SSME). However, this is intended to operate in the atmosphere as well as in space, so its design represents a compromise, although a very effective one. For an OTV application, it can be optimised for vacuum conditions, raising its SI by an appreciable amount; ultimately, values in excess of 470 seconds should be attained.

NASA's advanced LO_2/LH_2 design[18], shown in Fig 7, indicates the present direction of progress. This engine provides a thrust of 9100 kg at an SI of 473 seconds. It has a staged combustion cycle capable of idle-mode operation, and the nozzle can be extended to its full length after injection into orbit to minimise packaging length requirements. The total mass is about 245 kg.

The high value of SI is achieved by raising the combustion chamber pressure to 136 atmospheres (2000 psi) and selecting an expansion ratio of 400; both values are considerably in excess of current practice. Further progress here is unlikely to be significant, owing to the materials problems inherent in raising combustion chamber pressures and temperatures to higher levels, and to the mass penalty incurred by using a larger expansion ratio.

Other similar engines have been proposed to suit specific applications. For example, Kunz[10] has reported a scaled-down version of the device shown in Fig 7, designed specifically for the propulsion of the communications platform illustrated in Fig 2. This engine has a thrust of 2273 kg, a length of 1.32 metres, a nozzle exit diameter of 0.76 metre, and a mass of 50 kg. The expansion ratio remains at 400, but the combustion chamber pressure is reduced to 102 atmospheres (1500 psi), resulting in an SI of 467 seconds.

3.3 Electric propulsion

During the last two decades, many different EP devices have been developed, mainly in the USA and Europe, but only four of these have advanced to the point where they can be considered suitable for missions of the type discussed in this Report. Of these four, the Kaufman ion thruster is the most highly developed, having reached, several years ago, the stage at which immediate application to orbit raising, NSSK, or interplanetary missions was possible. It is the only device which can currently be considered definitely suitable for SPS transportation and orbit control tasks[15].

For low thrust applications, the main competitor of the Kaufman thruster is the RF ion thruster[21] developed in Germany, which is similar in operation, apart from the use of an RF ionising discharge instead of a dc discharge. Another competitor is the magneto-electrostatic containment (MESC) thruster[22], which employs a much stronger magnetic field in the discharge chamber than the Kaufman device. At the present time, when overall performance, potential lifetime and thruster/spacecraft integration problems are taken into account, the Kaufman concept is preferable for most low thrust applications.

Both the Kaufman and RF thrusters have been scaled up in size, although the performance of the former becomes increasingly superior as diameter increases. Kaufman thrusters as large as 1.5 metres in diameter[23] have been tested and a 30cm device[24] is almost ready for flight. RF thrusters of up to 35 cm in diameter have been operated successfully[25]. Although it is conceivable that both types could be applied to the propulsion of large space platforms and SPSs, only the Kaufman thruster shows real promise in this context. Consequently, there has been general agreement in studies of SPS technology that Kaufman thrusters, using argon propellant, should be employed. It has, however, been acknowledged that the relatively low thrust density of current devices should, if possible, be increased substantially; methods exist for accomplishing this[15,18].

It is possible to obtain a much higher thrust density immediately by using magnetoplasmadynamic (MPD) arc thrusters, which depend upon the electromagnetic acceleration of a dense plasma. However, these have so far demonstrated only low efficiency and poor lifetime, so that they cannot yet be considered seriously for operational applications.

The EP systems mentioned above are described in more detail below, and their major characteristics are summarised in Table 1. Also included, for completeness, are three other low thrust devices, the colloid, field emission and contact ionisation thrusters.

(i) Kaufman ion thruster

In this device, the propellant gas is ionised by a dc electrical discharge between a hollow cathode and a cylindrical, concentric anode. The discharge chamber is enclosed at one end by a pair of carefully aligned, closely spaced grids. A large electric field imposed between these grids extracts and accelerates the positive ions from the plasma in the discharge chamber, forming an intense ion beam and producing thrust. The positive space charge of the ion beam is neutralised by electrons from an external cathode, and a magnetic field applied to the discharge chamber enhances the efficiency of the ionisation process. The exhaust velocity, and thus the SI, is limited only by the power available and the inter-grid breakdown voltage.

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The thruster shown schematically in Fig 8 illustrates the principles of operation of this device; it is assumed there that the propellant is mercury vapour, but almost any material that is in gaseous or vapour form at about 400°C or below may be used. Argon is certainly satisfactory.

The Kaufman thruster has reached an advanced stage of development, a considerable amount of flight experience having already been accumulated[27,28]. Several devices are almost ready for application; these range in size up to 30 cm diameter[24], and, as already mentioned, tests have been carried out on a thruster of 1.5 metres diameter[23].

Thruster life-testing experience in the laboratory extends to at least 15000 hours[29] and critical components have operated for longer times[30]. Successful space operation of one thruster experiment extends over a period of 9 years and thousands of hours of running[28].

(ii) <u>MPD arc thruster</u>

In an MPD arc thruster, an intense dc discharge, with current densities many orders of magnitude greater than in the ion thruster, both ionises and accelerates the gaseous propellant, producing an electrically neutral exhaust plasma and thrust. A typical device, shown schematically in Fig 9, includes an axial cathode with a downstream conical or cylindrical anode. The propellant may be introduced either around the cathode, through holes in the cathode itself, or at some downstream location.

Two processes are involved in the acceleration of the exhaust plasma. The simple Joule heating of the plasma is probably dominant in low energy thrusters, operating in the kilowatt power regime. However, electromagnetic acceleration rapidly increases in importance as current and power levels rise. In this mechanism, the central discharge to the cathode pinches down to a very small diameter, owing to the compressive force exerted by its own magnetic field B_θ, and this field interacts with the radial current component flowing to the anode, J_r, to produce an extremely large $J_r \wedge B_\theta$ force which accelerates the plasma away from the thruster. An external magnetic field can be applied to assist this process, but this is most effective at lower power levels.

Although MPD thrusters are comparatively simple, the electrical efficiency η_e (kinetic energy in the exhaust divided by total input power) of most devices has proved to be disappointingly low[31], usually 20-30%. In addition, their development status is primitive in comparison with that of ion thrusters. A major problem is that of electrode erosion and, therefore, of durability, at the megawatt power levels envisaged for space use. At these power levels, continuous operation may not be possible, introducing switching problems; indeed, all laboratory testing is currently done in a pulsed mode.

Despite present limitations, recent work has shown that these devices offer considerable future promise. For example, the Princeton group[32] has found that the major loss, the heat transmitted to the anode following dissipation in the anode sheath, falls from 50% at 200 kW to only 10% at 20 MW, representing considerably increased efficiency.

It has been found that a critical parameter is $I^2/\dot{m}$, where I is the discharge current. The best configuration limits this to about 100 kA^2-s/g, corresponding to a specific impulse with argon of roughly 2000 seconds. An increase by a factor 3 will be necessary to achieve 5000 seconds.

(iii) RF ion thrusters

As already mentioned, these devices are similar to Kaufman thrusters, but make use of an RF plasma instead of a dc discharge[21]. Although they perform in a similar manner to Kaufman thrusters, at least in the smaller sizes (about 10 cm diameter), they suffer from a number of disadvantages, which include the basic inefficiency of the RF coupling mechanism, lifetime limitations caused by shorting of the RF field by metallic coatings sputtered onto the inside of the cylindrical discharge chamber, and control problems. There seem to be no significant compensating advantages, although the RIT-10 device has been developed to near flight status[8].

Most of the work in this field has been concentrated on a 10cm diameter device. However, preliminary investigations have been reported of larger thrusters, up to 35 cm diameter, intended for primary propulsion[25]. Efficiency does not increase with size so rapidly as with the Kaufman thruster, owing largely to the problem of uniformly injecting energy into the discharge chamber for the ionisation process; the skin effect tends to cause most of the ionisation to occur around the periphery of the discharge chamber, leaving a volume of relatively low plasma density in the centre.

(iv) MESC thrusters

As already stated, the MESC thruster is similar in operating principle to the Kaufman device in most respects, apart from using a totally different magnetic field configuration in the discharge chamber[22]. A divergent field of a few tens of Gauss is applied in the Kaufman thruster, whereas the MESC concept employs a localised cusp-shaped field, of thousands of Gauss, around the periphery of the discharge chamber. This can promote, under some circumstances, a higher value of n_e, but, to a potential user, the characteristics of the two devices are very similar.

Most of the work on MESC thrusters has been concerned with small, low thrust devices, and many of these have used cesium propellant, which has very restricted applications. Although possibly suitable for NSSK and similar missions on present communications spacecraft, assuming that a more suitable propellant can be employed, there is no clear evidence that scaled-up versions can be produced to give the high thrust and thrust density needed for large-scale projects such as the SPS.

(v) Colloid thrusters

A colloid thruster depends for its operation on the electrostatic spraying of minute droplets of a liquid from the ends of fine needles, tubes or slits. The liquid must be slightly conducting, and is typically glycerol doped with sodium iodide. The droplets are formed because the liquid meniscus at the spraying site becomes unstable in the presence of an intense electric field, set up between the metallic needle, tube or slit and an external electrode[33].

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Such devices give moderate efficiency and specific impulse. They have also reached an advanced stage of development, but a planned flight test was abandoned owing to lifetime problems. These, together with the low thrust attainable per emitter and environmental problems associated with the propellant, make them unsuitable for the applications discussed here.

(vi) Field-emission thrusters

These are similar in some respects to colloid thrusters, but the spraying process causes the emission of positive ions rather than droplets, owing to the use of very high electric fields and propellants with low ionisation potentials. The emission site is a sharp point or edge covered, by a wetting process, with a thin film of the propellant, or sharp cusps formed by instability processes on the meniscus at the end of a tube or slit[33].

As the exhaust velocity can be very high, and therefore the SI, the electrical efficiency can be good. However, much propellant is wasted by evaporation, so the mass utilisation efficiency η_m is relatively poor at present. Here, η_m is defined as the fraction of the propellant mass fed to the thruster which is actually employed in the exhaust in producing thrust. Additional disadvantages include the low thrust obtained from each emitter, the primitive development status, and the unacceptable chemical characteristics of most propellants (nearly all published work has been done using cesium).

(vii) Contact ionisation thruster

In this device, ionisation of the vapour of a material having a low ionisation potential, typically cesium, is achieved by passing it through a heated porous tungsten plate. The positive ions emerging from the plate are subsequently accelerated by a high electric field, giving a very large specific impulse. However, the thermal radiation loss from the tungsten surface gives a low η_e, and other problems result from cesium condensation on ceramic insulators. The development status is not very advanced and adequate durability has not been proven.

(viii) Other devices

Many other thrusters have been investigated in the past, but work on most has now ceased, owing to the discovery of serious problems or disadvantages. At least three broad classes can be identified: very high current pulsed discharge thrusters, such as conical z-pinches and coaxial plasma guns, steady-state $J \wedge B$ acceleration of plasmas produced by other mechanisms, and steady-state RF acceleration by applying travelling waves to a separately derived plasma.

All these devices suffer from low electrical or mass utilisation efficiency, lack of durability, or requirements that would be difficult to meet in a space application. The latter include the need for large, high voltage condenser banks for the very high current pulsed devices, although it should be pointed out that very small versions of these thrusters have been successfully developed and used in applications where efficiency was not important.

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Table 1

Summary of characteristics of electric thrusters

Type	Kaufman ion	RF ion	MESC ion	MPD*	Colloid	Field-emission	Contact ionisation
References	24, 34	25	22	31	33	33	-
Acceleration mechanism	E/S	E/S	E/S	E/M, Joule heating	E/S	E/S	E/S
Propellant - usual - acceptable alternative	Hg Argon	Hg Argon	Cs Argon	Argon N_2	Glycerol† None	Cs None	Cs None
Exhaust exit dimensions	30 cm diameter	35 cm diameter	12 cm diameter	10 cm diameter	3 cm diameter annulus	3 cm linear emitter	5.1 cm × 0.6 cm rectangle
Thrust (mN)	200	160	17	140×10^3	$0.5^{\dagger\dagger}$	$\sim 2.5^{\dagger\dagger}$	1.5
Power (kW)**	5.4	3.6	0.34	6000	~0.01	~0.16	~0.12
Usual maximum specific impulse (second)	5000	3360	3270	2400	1000-2000	9000	6700
η_e	0.93	0.79	0.81	0.31	0.7	~0.9	~0.4
η_m	0.90	0.88	0.97		0.2-0.8	~0.7	0.99
Maximum life-test duration (hour)	10000	None‡‡	600^{ϕ}	None	$475^{\ddagger}$	None	None$^{\phi\phi}$
Environmental acceptability	Excellent	Excellent	Excellent	Excellent	Poor	Very poor	Very poor
Development status	Flight ready	Very poor	Medium	Very poor	Medium	Very poor	Low

* Values given apply to ms pulsed operation
** Excluding power dissipation in power conditioning electronics
† Doped with NaI
†† Easily stacked to increase thrust by a factor of 10 to 100
‡ But 6500 hours on modular multiple needle thruster
‡‡ But 8000 hours on 10 cm diameter thruster
ϕ But 2600 hours on smaller device
ϕϕ But components to about 5000 hours
E/S = electrostatic
E/M = electromagnetic

It may be concluded that, at the present time, only ion thrusters are suitable for carrying out the propulsion tasks considered in this Report, and that, of the three types available, the Kaufman device is the most promising. This conclusion has also been reached by many other authors and study groups[11,18,26], who have paid particular attention to existing technology.

3.4 Resistojet systems

Resistojets use electrical energy to increase the flow velocity of a gas exhausting from a conventional divergent nozzle, thereby raising the specific impulse to a value intermediate between that attainable by a chemical rocket and that produced by an electric propulsion device. If hydrogen is used[16], the specific impulse can reach about 800 seconds, nearly twice that attainable with LH_2/LO_2 rocket motors, but the power consumption can be large. Although the heater technology is not easy, there should be no difficulty in developing a hydrogen resistojet for advanced missions, should one be needed; the use of argon or of any other gas is not advocated, as it would seriously reduce the SI.

The use of hydrogen resistojets for orbit raising missions was suggested as long ago as 1970, with emphasis on combining these devices with ion thrusters to optimise the effective impulse in power limited cases[35]. Re-usable space tugs for carrying large payloads to geostationary orbit, using a similar combination of resistojets and ion motors, have also been studied. However, despite encouraging results, the resistojet fails to meet performance criteria necessary for the economic success of the missions considered here.

3.5 Nuclear systems

Solid core nuclear reactors may be used to heat exhaust gases, which subsequently expand through a conventional nozzle. Unfortunately, they are limited in the same way as are resistojets, by the physical and chemical characteristics of the materials that can be used for the reactor and heat exchanger. It is evident that melting points cannot be closely approached, so the exhaust, assuming the use of hydrogen, cannot have a velocity much in excess of 8 km/s [16], giving a specific impulse of around 800 seconds[17].

The results are different if these materials limitations can be avoided, as in the gas core reactor system[17]. A specific impulse of about 2200 seconds can then be realised, although the technology is much more difficult. It has been estimated that a system only 7 metres long could produce a thrust of 41 tonnes for a mass of 32 tonnes. Such a device would allow total transportation costs, for SPS assembly in geostationary orbit, to be reduced to possibly $88 per kg. Its nearest competitor would be an MPD thruster system powered by a nuclear reactor, which could achieve $110 per kg. For comparison, a re-usable chemical vehicle would give transportation costs of $230 per kg, well above the economic limit[14].

The alternative way to employ a nuclear reactor is to use its thermal output to generate electricity, and then feed this power to electric thrusters[5]. It has been estimated that, using thermionic converters, a re-usable tug that could be launched by the Space Shuttle would be able to transport payloads to geostationary orbit for around $130 to $150 per kg, the propulsive effort being produced by ion thrusters. Much larger nuclear electric systems, using MPD thrusters, have also been advocated for SPS missions, with about $120 per kg being achievable[36].

Despite reservations about the accuracy of published data, it does appear that nuclear electric and gas core reactor systems could find extensive applications in the long-term future. It is unlikely, however, that they can be considered for early missions because of their low development status and present day concern regarding the safety of nuclear systems in orbit.

3.6 High thrust ion propulsion systems

Although the orbit transfer of relatively small payloads can be accomplished using Kaufman ion thrusters developed for NSSK applications[3], which produce thrusts in the 5-20 mN range, much larger devices will be needed for communications platforms and SPS systems. The most highly advanced thruster likely to be suitable for early missions is the 30cm device developed by NASA Lewis and Hughes RL[24,34]. The 1.5 metre diameter thruster[23] 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 2, its design thrust is about 130 mN, with an SI of 2985 seconds, but these values can be increased[34] to 290 mN and 6298 seconds by raising both beam accelerating potential V_T and beam current I_B. With argon, these values would be 260 mN and 14270 seconds. The 35cm RF thruster is included for comparison.

The equivalent design data for thrusters proposed specifically for the SPS mission are given in Table 3. The 120cm diameter concept due to Grim[26], which is depicted in Fig 10, is relatively conservative, in that the thrust and beam current densities are only slightly larger than the values currently being achieved. Conversely, the 60cm thruster proposed by Byers and Rawlin[37] assumes that the beam 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*[38] 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 2

Performance data from large ion thrusters

Thruster type	Existing NASA/Hughes 30 cm 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(amp)	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	24	34	34	-	-	25

* Calculated data

Table 3

Performance data for advanced Kaufman argon ion thruster designs

Beam dimensions (cm)	60 cm diameter	76 cm diameter	100 cm diameter	120 cm diameter	100 cm × 150ccm
V_T (kV)	4.2	2.0	5.0	6.0	1.88
I_B (amp)	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/cm2)	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	37	18	38	26	39

Conversely, the rectangular 100 cm × 150 cm concept[39] and the 76cm diameter device[18] 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 10. 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[40]. 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[41]. 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 40-50 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, perhaps with certain performance or durability penalities. However, few systematic investigations have been reported, so much work remains to be done, with emphasis on potential life-limiting areas.

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 40 kW 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[18], 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.

Several major objectives should guide these developments. 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,

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

Most of these can be achieved by using straightforward developments of the Kaufman thruster principle. This gives, as already mentioned, long life, high efficiency, flexibility of operation and 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 should therefore be mentioned that an alternative design approach is possible, which should enable very large values of thrust and thrust density to be achieved without raising I_D and P_D to excessive levels[15]. The principle adopted here is to use very high beam accelerating potentials to increase simultaneously the thrust density and the SI, whilst placing no additional demands upon discharge chamber or cathode technology. Although this concept requires a very much greater power input to the thruster, this should be acceptable for many of the missions under discussion, particularly SPS transportation. There is an additional benefit to be gained, in that the values of SI possibly further reduce propellant requirements (Fig 5).

The exhaust velocity of an ion thruster is given by:

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

where m_i is the ion mass. The SI is then:

$$I_{sp} = \frac{\eta_m v_e}{g_0} .$$

These expressions, which assume that all beam ions are singly charged, have been used to plot the curve of v_e and I_{sp} against V_T shown in Fig 11. This curve is for argon and $\eta_m = 0.88$, a value representative of advanced present day thrusters[6]. It can be seen that a very high SI is immediately accessible; for example, at $V_T \sim 5$ kV, which is easily attainable, $I_{sp} \sim 14000$ seconds, a factor of 30 better than can be reached with chemical rockets. If 40 kV could be used, the SI would rise to about 40000 seconds, a factor of 84 above that of the best chemical system. The thrust, which is given by:

$$T = \eta_m v_e \dot{m}$$

increases correspondingly, without further modification to the thruster. In particular, the discharge chamber plasma and the cathode conditions need not be changed.

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To achieve values of V_T of around 40 kV is probably not possible using the twin grid systems depicted in Figs 8 and 10. Apart from electrical breakdown problems, difficulties would result from the dual role of this type of grid system; it both extracts ions from the discharge chamber plasma and accelerates them. It is therefore proposed[15] that these two functions be separated by substituting a four grid system, as employed very successfully in ion accelerators designed for application to controlled thermonuclear fusion experiments; such devices[42] have produced 4-7 A beams at 70 keV, from grids of only 10 cm diameter with an open area ratio of as little as 31%. Using this ion extraction/acceleration method, a comparable performance should be obtainable from a Kaufman thruster.

4 APPLICATIONS

Having described the various propulsion technologies likely to be available in the next few decades, possible applications will now be discussed, with reference to the missions mentioned in section 2. These will be treated as examples, and the case of LEO to GEO transportation will be emphasised. Of course, many other missions and final destinations may be considered, but space does not permit this here.

4.1 Low payload missions

In this context, a low payload is defined as one which can be placed in GEO using a single launch of the Shuttle or an expendable rocket. As far as chemical propulsion is concerned, conventional apogee motors or the IUS would be appropriate with, in the longer term, a higher performance being achieved through the use of liquid-fuelled upper stages, such as the Centaur[9].

Although a re-usable solar electrically propelled space tug[3] could be designed to raise payloads in this category to GEO, most relevant applications of EP will involve the use of thrusters integrated into the spacecraft itself, and retained with it in GEO. Many proposals for such spiral orbit-raising missions have been made[3,7,8,35], spanning nearly 20 years. One of the larger designs, SERT-C[43], is shown in Fig 12. This spacecraft was intended to demonstrate the application of three 30cm ion thrusters to the orbit-raising mission, and of four 8cm devices to the attitude control and station-keeping tasks. It was also planned to use these thrusters to conduct a rendezvous experiment in GEO.

Characteristics of the spacecraft and mission, which never proceeded beyond the detailed planning stage, were as follows:

Spacecraft mass	- 818 kg
Launch vehicle	- Thor Delta 2910
Initial orbit	- 3150 km altitude, inclined at 28.5 degrees
Power source	- two solar arrays of 2.74 × 16.15 metres
Power level	- 9 kW, beginning of life, 4.5 kW end of life
Orbit-raising thrusters	- three 30 cm, 128 mN devices
Specific impulse	- 2955 seconds

Attitude/orbit control thrusters - four 8 cm, 4.5 mN devices
Orbit transfer time - 290 days
Propellant consumption - 123 kg.

4.2 Medium payload missions - chemical propulsion

Reference has already been made to the analysis reported by Kunz[10]. This will be used as an example of a possible approach to the orbit transfer of moderate payloads, such as the communications platform shown in Fig 2, which has an estimated mass of 60636 kg. This type of structure is relatively flimsy, so the thrust that can be applied to it must be limited to a level dictated by its strength and mass. Kunz shows that a critical parameter is the ratio of the thrust to the initial mass.

A low value of T/M_0 has beneficial effects, in that a stronger and more massive structure is not needed. However, as indicated in Fig 13, the overall mission ΔV increases as T/M_0 falls, due to the larger gravity losses associated with the longer trip time. The additional propellant required causes the overall initial spacecraft mass to increase rapidly at low T/M_0 (assuming the use of chemical propulsion). There is, consequently, an optimum value of T/M_0 at which the initial spacecraft mass is a minimum; this is illustrated in Fig 14. The minimum occurs at $T/M_0 = 0.28$ for the spacecraft under consideration, a very low value by present standards.

A further conclusion is that motor burn times must be much longer than currently employed, typically 15-30 minutes (Fig 15). In addition, the need to keep the acceleration below the critical level will require active throttling of the motors.

The LO_2/LH_2 motor to be employed is the low thrust version of that depicted in Fig 7. It is envisaged that four of these would be mounted in the propulsion module shown in Fig 16. This is the largest that can be carried in a single Shuttle launch. Its inert mass is 3493 kg; fully loaded with propellants this becomes 28864 kg. A single module can deliver a payload of 12700 kg to GEO. It is envisaged that five would be used to propel the communications platform to GEO, mounted at the end of the structure, as indicated in Fig 2. Three of these modules provide initial first stage perigee thrust, then the remaining two are fired, as a second stage, to give the additional perigee ΔV required. These two later provide apogee thrust to circularise the orbit at GEO. At all times, T/M_0 is kept below the critical value by shutting down or throttling motors in balanced pairs as propellant is consumed.

Although not unique, this concept gives a good indication of how a very effective medium-payload OTV system might be designed. Its main advantage stems from the use of propellants giving high I_{sp} ; this is emphasised in Table 4, in which it is shown that far less shuttle launches are required to orbit this system than competing chemical designs. However, this table also suggests that EP should be superior, because less than one shuttle payload would be needed to launch an ion propelled OTV capable of the type of task being considered. An additional advantage of the EP system is that the low acceleration removes all structural constraints, and allows full deployment of antennae and solar arrays in LEO, where astronaut assistance can perhaps be provided in case of problems.

Table 4

Orbit transfer propulsion comparison for raising 50 tonnes payload from LEO to GEO

Propulsion system type	Cryogenic	Storable	IUS	SEP
Specific impulse (seconds)	467	290	295	>5000
Maximum thrust/mass	0.2	0.2	1.15	$<10^{-4}$
Number of modules	4	11	20	N/A
Number of STS launches to place in orbit	4	11	10	<1
Structural limitations on spacecraft arrays, antennae	Partial deployment		No deploy	Full deploy

4.3 Medium payload missions - electric propulsion

The analysis presented by Kunz[10] clearly showed the advantages of using EP, assuming that long orbit transfer times can be tolerated. This conclusion may be emphasised by pointing out that the propellant load in the propulsion module shown in Fig 16 is 25371 kg. Thus propellant to payload mass ratio is almost 2.0 kg per kg, a value typical of advanced chemical systems. Using ion thrusters, with V_T no higher than 5 kV, this can be reduced to below 0.07 kg per kg, so the propellant requirement for the same 12700 kg payload falls dramatically to only 846 kg.

Many proposals have been made to take advantage of these benefits by the deployment of a solar electrically propelled space tug[3,44,45], having very wide applications. One of the most advanced of these is a modular concept[45] using banks of NASA Lewis/Hughes 30 cm ion thrusters[24,34]. Each of these thrusters is mounted separately, with its associated tank, power processing unit (PPU) and waste heat radiator (Fig 17), forming a thruster sub-system module (TSSM). The thrusters are gimballed to provide thrust vector control, and the propellant tanks are placed as near to the spacecraft centre of mass as possible to minimise mass distribution changes as a mission proceeds. Total module mass is 54.5 kg, its dimensions are 0.61 × 0.61 × 2.29 metres, and a major design aim was to minimise spacecraft interface problems.

It is envisaged that up to 10 modules of this type could be assembled together and joined to an appropriate solar array to form a tug vehicle suitable for a very wide range of applications. Fig 18 illustrates how this might be done in the case of a mission needing eight modules. The solar array is, of course, a very important feature of such a vehicle, and much effort has been expended in developing a 25 kW fold-out system[46]. This has a power to mass ratio of 66 W/kg and each wing has dimensions of 32.0 × 4.06 metres.

Unfortunately, the availability of adequate power seriously limits the possible applications of solar EP; it is for this reason that nuclear EP systems have been proposed[5]. To illustrate the effects of this limitation, it can be shown, using the simple theory yielding equation (2), that the propellant required to transfer the communications platform in Fig 2 to GEO is 12460 kg, if ΔV is taken from Fig 13 and propulsion is by means of six TSSMs with power being supplied from an 18kW solar array. The hardware mass of such a system[45] is only 734 kg, giving a total propulsion system mass of 13190 kg. This is an enormous improvement over the 137000 kg needed for chemical propulsion[10]. However, when the transfer time is evaluated, it is found that several years are needed for the trip, which is unacceptable. Thus, with payloads of this size, the thrust must be increased very considerably, and this can be done only by providing more power, assuming that a greater propellant load is not permissible. This option is considered in section 4.5 below.

4.4 Large payload missions - chemical propulsion

As mentioned before, the advent of SPS technology will eliminate the present limitations on solar EP orbit transfer vehicles, by making copious quantities of electrical power available in space comparatively cheaply. Thus the propellant requirement for a chemical OTV, of at least 2 kg per kg of payload, will be prohibitively large for all but priority cargo and personnel, leading to the development of electrically propelled cargo OTVs (COTVs), which will need below 0.1 kg per kg of payload.

However, the demand for a chemical OTV is not trivial, as can be seen from Fig 19, in which the number of personnel in orbit is plotted against time for the commercial exploitation phase of an SPS programme placing 60 spacecraft in orbit in 33 years[47]. This programme requires about 22 flights per year of an OTV carrying personnel for construction purposes, with an increasing additional number for maintenance. The total rises to 40 flights per year after 13 years (20 satellites) and 80 after 33 years (60 satellites).

Many different chemically-propelled OTV systems have been proposed for SPS projects, but most have similar characteristics. In general, they utilise LO_2/LH_2 engines, often based on Space Shuttle technology, and transfer times are relatively short, of the order of 10 hours. This is ideal for the movement of construction crews, but, with a specific impulse of 450-470 seconds, the consumption of propellant is relatively high. The two-stage example shown in Fig 20 is a fully re-usable system that can transport 75 passengers plus crew supplies for 90 days, or a payload of 65 tonnes, to the geostationary orbit[48]. It could return a payload of 41 tonnes to low orbit, and would have a mass of 560 tonnes at the beginning of a mission. An enlarged version, carrying 160 passengers or 151 tonnes of payload, has also been described[4], as has a very much bigger development having a payload of 400 tonnes[49]. A rather different, single-stage vehicle is shown in Fig 21; this can accommodate 80 passengers[4].

The single-stage concept[50] depicted in Fig 22 is perhaps more advanced from the structural point of view. Although shown as carrying a payload of 50 tonnes, versions accommodating up to 200 tonnes have been considered[50], also an application to personnel transport, with a payload of 12 tonnes[13]. In the latter case, an SI of 476 seconds has been claimed for the engines used.

A recent cost analysis[50], based on the design in Fig 22, suggests that the orbit transfer mission will require $110-180 per kg of payload. It was shown that costs are strongly dependent on a number of parameters and assumptions, which include the following:

(a) Payload

The range 20 to 200 tonnes was considered. A sharp decrease in cost, of about 25%, results from increasing this parameter from 20 to 100 tonnes. Thereafter, the advantage is less.

(b) Staging philosphy

Cost is insensitive to the relative sizes of the stages in a two-stage vehicle, but a single-stage version is definitely more expensive (Fig 23).

(c) Engine specific impulse

There is a strong dependence on this parameter. Increasing it from about 440 to 475 seconds causes a reduction in cost of about 16%, as shown in Fig 23.

(d) HLLV costs

There is a linear relationship between orbit transfer costs and the cost of transporting materials and propellants into LEO. In this analysis, the HLLV cost was assumed to be $10-50 per kg.

(e) Re-usability

It is clear from this analysis that re-usable vehicles provide the most economic option, despite the fact that they must be returned to LEO at the end of each mission.

(f) Trust-to-weight ratio

The start-of-mission thrust to weight ratio has a moderate influence for low thrust orbit transfers, where this ratio is in the range 10^{-3} to 10^{-1}; going from 10^{-3} to 10^{-2} might decrease cost by 7-8%.

(g) Number of rocket burns during orbit transfer

This parameter also has a moderate influence for low thrust missions, with a saving of perhaps 8% resulting from increasing the number of burns from two to five.

4.5 Large payload missions - electric propulsion

Reference has already been made to the immense transportation costs that will be incurred during the construction and maintenance of an SPS system. The OTV phase has been estimated[4] to contribute 20.7% of this total cost insofar as cargo is concerned, but only 0.05% for personnel, so it is fortunate that EP can be used to minimise the expense of moving huge quantities of materials or large SPS sections from LEO to GEO.

Basically, the total SPS mass must be transported to GEO, and this could amount to 51000 tonnes per 5GW unit (Fig 3) if silicon solar cell technology is used. In addition, construction machinery and facilities, workshops and living quarters must all be transported to GEO, although the extent of this task will depend upon the SPS construction philosophy adopted.

Using EP, two concepts have been proposed. Possibly the most attractive is to manufacture large sections of the SPS in LEO, then use part of its own electrical output to power attached EP modules to propel it to GEO, via a spiral trajectory. Final assembly would then take place in GEO, where only a relatively small astronaut crew would be needed[51]. The alternative is to deliver all components, materials, and construction equipment to GEO, using a COTV, and manufacture the spacecraft there[11,38].

It would also be possible, theoretically, to construct the complete SPS in LEO, then use its electrical output to power EP systems 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[19]. 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[38].

As already mentioned, a nuclear power source is a very attractive proposition for the orbit transfer mission[17]. It can be used to either generate electricity to power an EP system, 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.

Possibly the most detailed investigation of the self-propulsion technique has been carried out by Boeing[51], with reference to their 10 GW 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 3. 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 24, would provide the propulsion. Each array would be gimballed about two perpendicular axes for thrust vector control during orbit transfer[51]. 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[52].

The characteristics of the self-propelled sections are given in Table 5, where reference is made to both the Boeing 120cm diameter thrusters 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 kg, this will save over $200M per SPS, and the elimination of the chemical system would save a further 080

\$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 kg 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 5

Characteristics of Boeing 10GW SPS self-propelled sections

Thruster type	Boeing 120cm thrusters[51,26]		Uprated 120cm thrusters*	
Type of SPS section	Non-antenna	Antenna	Non-antenna	Antenna
Number of sections	6	2	6	2
Transfer time (days)	180	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
Percent 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 38

Several variations have been suggested, such as splitting an SPS into smaller 6 × 1 km sections, each propelled by two banks of thrusters[26]. 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[51], and that the thrusters will be employed for orbit and attitude control once GEO is reached.

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 6, from which it can be seen that remarkable payload to empty mass ratios can be achieved, the design described in Ref 18 and depicted in Fig 25 reaching a value of 5. The propellant mass for the complete mission can be as low as 0.07 kg per kg of payload, for an SI of 13000 seconds[38]. The design in Fig 25 uses GaAlAs solar cells, with concentrators. An alternative, employing Si cells, is depicted in Fig 26.

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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[38]. They all provide very large cost benefits compared to chemically propelled vehicles, particularly when operated at high SI.

Table 6

Summary of electrically propelled COTV proposals

Origin and reference	Caluori *et al*[53]	Weddell *et al*[38]	DOE/NASA[11]*	Hanley[18,52]
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)	0.030	0.30	0.985	0.667

* Assuming use of 120cm thrusters in Refs 26 and 51

† Including 25% spares

‡ Array only

5 ORBIT TRANSFER STRATEGY

Although a considerable amount of analysis has been carried out to determine optimum spiral orbit raising strategies using electric propulsion[3], structures as large as an SPS have been considered only rarely[20]. Their size introduces added complexity, especially at the start of a mission, when the aerodynamic drag of the residual atmosphere[54] on large, lightweight areas can cause orbit decay and attitude control problems. In addition, the need also exists to rotate continually the active solar arrays so that they are always perpendicular to the sun's direction. Both of these difficulties make the operation more complex and add to propellant requirements. The effect of solar radiation pressure must also be taken into account. This becomes comparable to aerodynamic forces at an altitude of 650 km. A further effect that must be included is the shadowing of the solar cells by the earth, which occurs for an appreciable time on each orbit at low altitude[3].

It is difficult to predict many of these effects, owing to their dependence on the geometry and mass distribution of the objects in question. However, as regards aerodynamic drag, it can be shown that this is a function of the parameter $C_D A/W_0$ where C_D is a non-dimensional drag coefficient, which depends upon geometry, A is the frontal area of the object, and W_0 is its mass. A large, low mass object has a high value of this parameter, and thus its orbit decays rapidly. It has been estimated[54] that a complete SPS might lose 4 km of altitude per orbit due to residual drag, at an altitude of 500 km. Consequently, this effect is very important initially, but it falls off rapidly as altitude is gained, the residual atmospheric pressure dropping by more than an order of magnitude between 500 and 800 km.

The problem of maintaining the array perpendicular to the sun's direction during the spiral orbit-raising manoeuvre has been considered in detail by Boeing[55] for the case of the transportation of a 10GW SPS in 16 sections, each measuring 4.15 × 2.5 km. This SPS concept employs silicon solar cells, and about 22% of the array is needed for the orbit transfer, using argon ion thrusters mounted at each corner, as in Fig 24.

The way in which the thrust vectors must be changed throughout each orbit to counteract gravity gradient torques and to increase the orbital velocity is illustrated in Fig 27. This diagram is appropriate to the first few days of a 180 day transfer, and it will be seen that, at this low altitude, much of the propulsive effort has to be devoted to keeping the array correctly aligned. In fact, within the assumptions of this study, the thrust available is inadequate for this purpose if the ion thruster system is sized for a transfer time exceeding 200 days.

Although the ion thrusters can usually carry out all attitude control functions, this is not normally true when the solar array is in earth shadow. Batteries must then be provided or chemical thrusters, assumed to be LH_2/LO_2 , must be used to maintain correct attitude. If this was not done, the satellite section could accelerate to a 0.1 degree per second rotation so that, upon emerging into sunlight again, it would have rotated almost 180 degrees, and be facing away from the sun. However, as this effect diminishes rapidly as altitude is gained, these auxiliary systems need not be operative for the complete mission.

Apart from causing the spacecraft to be equipped with batteries or a chemical propulsion system, the repetitive crossing of the earth's shadow obviously lengthens the duration of the mission[3]. The time of eclipse during any one orbit is a function of the angle between the earth-sun line and the normal to the orbital plane, as well as of the attitude of the spacecraft. It thus depends on the time of year and on the orbital inclination. However, it has been shown[54] that, in the worst case, where the sun is continuously in the orbital plane, the time in darkness is only about 18% of the total time[38].

Orbit transfer strategies have been considered for types of spacecraft other than that depicted in Fig 24, but the basic principles are always the same. For example, Boeing have analysed[55] the equivalent mission for a section of a heat engine SPS powered by three arrays of ion thrusters.

Although the pointing requirements may be rather more stringent in this case, the smaller dimensions and the concentration of much of the mass near the centre of mass cause the effects of gravity gradient torques to be much less. Consequently, a smaller proportion of the propulsive effort has to be devoted to attitude control, and the transfer time can therefore be shorter. In addition, less propulsion is needed during eclipse, 0.27×10^6 kg of LO_2/LH_2 being required, rather than 0.38×10^6 kg. Another advantage is that the spacecraft energy collection system does not suffer from degradation, as do solar cells, during its passage through the earth's radiation belts.

6 ENVIRONMENTAL EFFECTS DURING ORBIT TRANSFER

The two main environmental effects encountered during orbit transfer are potentially very severe for large, electrically propelled spacecraft. They are solar array degradation through impact by high energy particles, and collisional damage caused by other orbiting objects and meteorites. Both problems are particularly important when EP is used, owing to the long orbit transfer times, and collisional damage is obviously more likely for the largest satellites. Consequently, although these effects can be relevant to all types of spacecraft to some extent, the present discussion will be limited to those most vulnerable, solar power satellites. For these, the cost implications of both kinds of damage are very significant, and considerable attention has been paid to possible remedies.

It should also be pointed out that emplacement of large spacecraft in GEO may significantly damage the environment of the ionosphere and magnetosphere, due to the exhausts of any propulsion systems used. Such effects will also be considered in this section, again with particular reference to the largest potential source of difficulty, the SPS.

6.1 Solar array degradation

Solar array degradation during orbit transfer has been dealt with in detail on a number of occasions[14,54,38]. The magnitude of the effect on gallium arsenide cells[38], which is due almost entirely to energetic protons, is illustrated in Fig 28. It will be seen that large losses of power are inevitable, especially for long transfer times; similar results are obtained with silicon solar cells[3].

There are three possible approaches to overcoming the degradation problem. The most obvious is to protect the solar cells with thick cover glasses, but, for GaAs cells, these would have to stop all protons with energy up to 10 or 12 MeV[38]. Such cover glasses would be prohibitively heavy, and would also reduce cell efficiency via absorption of solar radiation. The same is true for silicon cells.

The second method is to start the mission with an over-sized array, but this leads to substantial cost and mass penalties[55]. It can also distort the optimisation of thruster specific impulse, causing a drastic reduction in the optimum value, so that only a relatively small array area is used during the orbit transfer. As an example of the mass penalty, it is of the order of 10^6 kg for an array sized to give around 600 MW at beginning of life. Of course, this mass increase is doubly undesirable, because it results in the expenditure of more attitude control propellant.

However, the third proposal offers hope that the degradation can be repaired by a thermal annealing process[14,48]. For silicon cells, it is anticipated[14] that annealing at a temperature of 500°C can eliminate all damage; this has been demonstrated experimentally[56]. It has been found that the best results are obtained using an oven; electron beams preferentially heat the surface of a cell, and CO_2 lasers are effective only with thick cells. Although an earlier NASA experiment which indicated that GaAs cells do not degrade appreciably under particle radiation[56] has been contradicted by more recent results[38], it is generally accepted that successful annealing can be performed at a much lower temperature than with silicon[19]. In fact, it is claimed that complete annealing can be carried out at only 125°C. Consequently, if a solar array is operated at a temperature exceeding this value, as would be the case with a concentration ratio of 2 or more, GaAs cells would be totally self-annealing[19]. Such high temperature operation is feasible because the efficiency of GaAs cells falls only slowly with increase in temperature. Thus the available evidence suggests that annealing will be possible for either type of cell.

6.2 Orbital collision damage

The problem of collision damage has been analysed in detail[17,54,55]. It has been shown that the flux of man-made objects in near-earth space, although small, is large enough to present a potential hazard to any SPS. It is orders of magnitude greater than the flux of natural objects of comparable relative momentum or kinetic energy, and is continually increasing. The flux of man-made objects is much greater in low orbit than at geostationary altitude, so this must be considered in selecting a construction site and an orbit transfer technique. For example, in 1973 over 90% of all space debris occupied orbits intersecting likely SPS construction orbits between 500 and 1500 km[54].

An estimate of the flux of objects likely to be encountered by an SPS is given as a function of altitude in Ref 55. This includes known orbiting satellites and debris, as in 1975, and predicted values for the year 2000. The resulting calculated total number of collisions per complete SPS is shown in Fig 29, as a function of altitude, for a 10GW satellite self-propelled to geostationary orbit by argon ion thrusters. It can be seen that, even with the present population of this region of space with man-made objects, there will be a large number of damaging impacts by items at least 3 m^2 in size.

There are, basically, two answers to this problem. One is to detect and avoid objects that might cause damage, the other is to clear away such debris. In general, the SPS will have available sufficient propulsive effort to enable avoiding action to be taken successfully in the majority of cases[55]. This propulsive effort can produce an acceleration of about 5×10^{-4} m/s, which is adequate to move an SPS section transversely, during orbit transfer, through a distance equivalent to its own size in about an hour. Thus, it should be possible to avoid orbit transfer collisions.

Collisions during construction in low orbit and during operation could be dealt with by clearing up the debris by use of a special purpose vehicle[55]. An alternative would be to protect the SPS or construction base by an interceptor vehicle, which would place itself in the way of the approaching object. It would be equipped with special energy-absorbing materials to avoid impact damage.

6.3 Effects on the environment

Although perhaps not causing as much concern as the pollution from an HLLV in the troposphere and stratosphere, careful investigation is undoubtedly required of the effects of orbital transfer vehicles on the ionosphere and magnetosphere. Both chemical rocket and ion thruster effluents must be considered.

Preliminary estimates[57] suggest that high energy beams from ion thrusters used for orbit transfer propulsion and, in geostationary orbit, for station-keeping and attitude control of the completed SPS, will significantly perturb the magnetosphere, particularly the plasmaspheric region. Both particle density and total energy content will be increased, the latter via Coulomb collisions between the energetic ions and ambient plasma. The significance of these effects must be assessed.

Two other effects of SPS operation are worth mentioning in the present context. The first is the reflected sunlight and infra-red emittance from the spacecraft during night-time. It has been shown that an SPS will be the brightest object in the night sky, apart from the moon, and that 100 satellites will produce a fraction of 0.1 of the moon's visible radiation. *This may interfere with optical astronomy.* The infra-red emission is less likely to be troublesome, because the main component in this region of the spectrum, thermal emission from the lower atmosphere, will dominate.

The second effect is that due to the gas and particular debris resulting from orbit manoeuvres, orbit transfer, and construction and operation activities. Possible results[58] could include the absorption of solar radiation, the modification of the interaction of the magnetosphere with the solar wind, and the formation of an artificial 'meteroid belt'. Although these could, in turn, influence the weather and interfere with other spacecraft or radio communications, such effects would require average emissions to be between 100s of kg per day and 100s of kg per second. This is extremely unlikely.

7 CONCLUSIONS

This review of future orbit transfer missions and technology has indicated that electric propulsion will have an increasingly dominant role to play as spacecraft to be placed in geostationary orbit become more massive, especially if solar power satellites are developed and deployed. The inherent advantage of an electric thruster, the acceleration of the exhaust by means of electrostatic or electromagnetic fields, leads to values of specific impulse of one to two orders of magnitude greater than attainable by chemical means. Thus the propellant mass required to transport a payload to GEO can be reduced from about 2 kg per kg to well below 0.1 kg per kg, representing an enormous financial saving. It has been shown that Kaufman ion thrusters are currently the best suited devices for this task; suggestions have been made as to how they might be improved for application to SPS and similar projects.

Despite these advantages, EP systems are not suitable for all missions, especially those requiring short transfer times, or when inadequate electrical power is available. Consequently, chemical vehicles have a significant role to play in any future scenario, particularly for personnel transportation. Although some advances in structural and propulsion technology can be predicted in this field, it is unlikely that vehicles much superior to those described, which are based on Space Shuttle liquid oxygen/liquid hydrogen propulsion concepts, can be developed.

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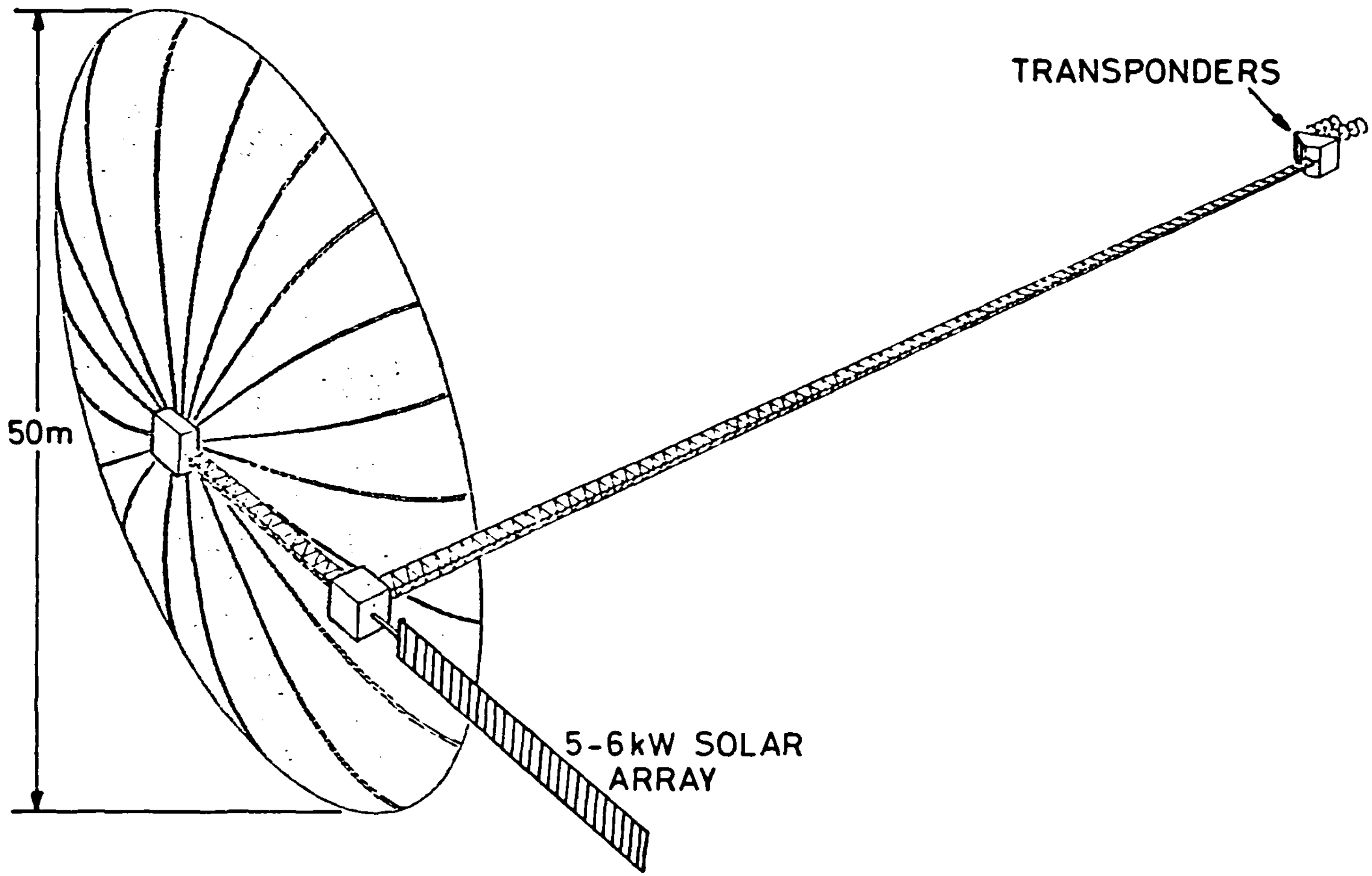

Fig 1 Spacecraft designed to serve mobile communications systems in the 1990s (Spar Aerospace Ltd)

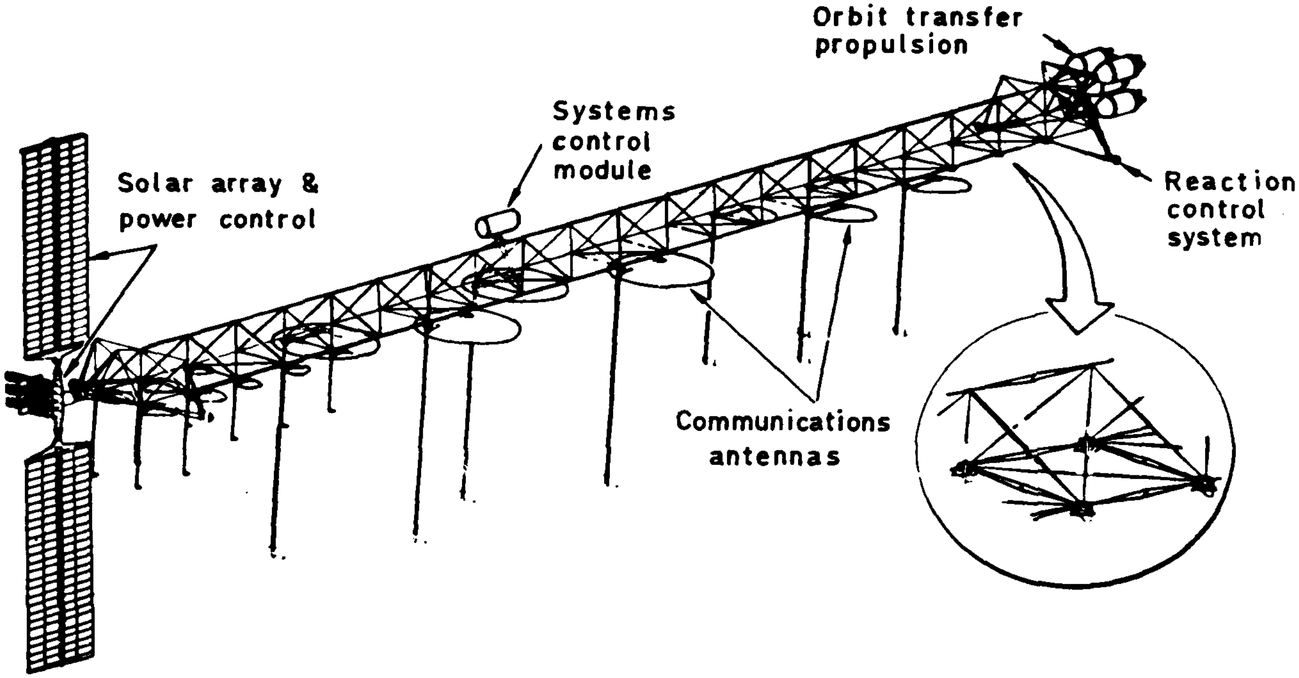

Fig 2 Schematic of advanced communications platform (Kunz, Ref 10)

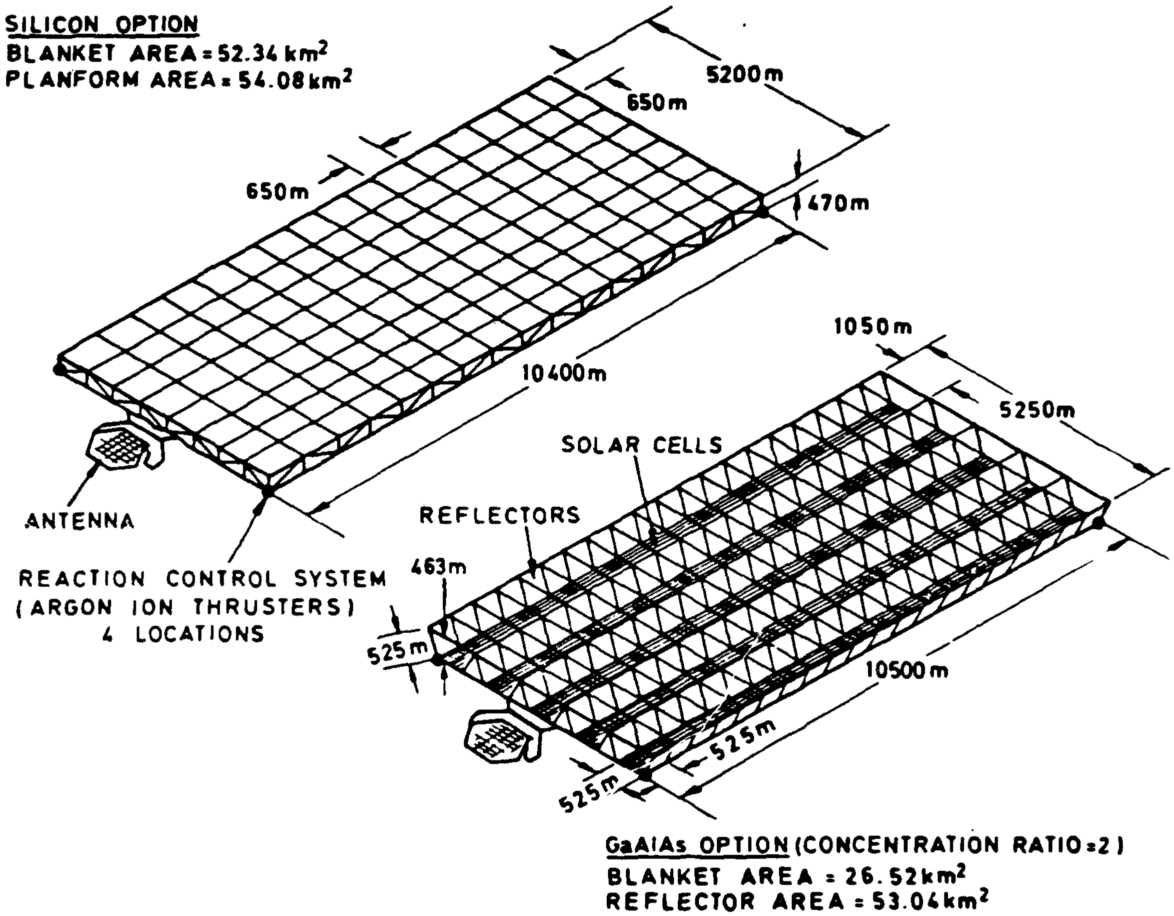

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Fig 3 DOE/NASA reference SPS designs (DOE/NASA, Ref 11)

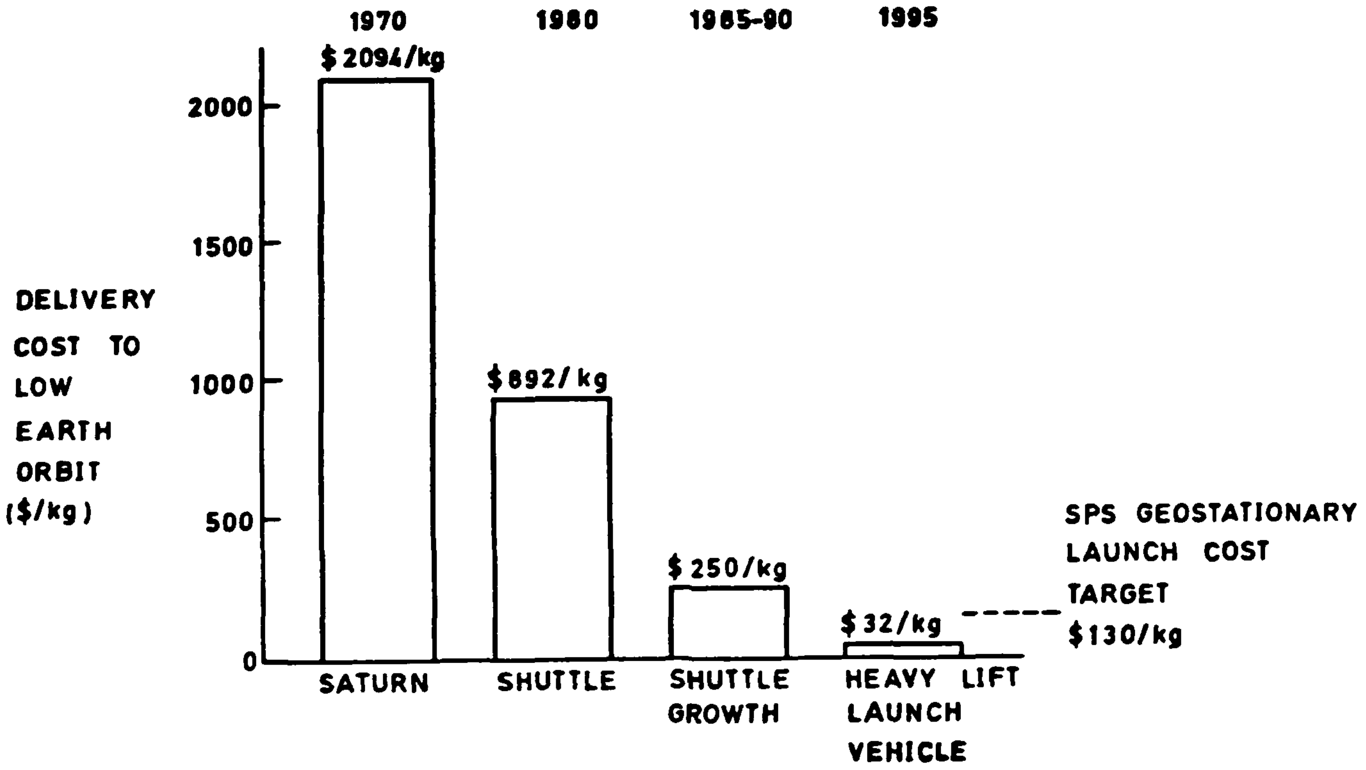

Fig 4 HLLV operational costs, in 1977 prices, compared to earlier launchers (Grumman, Ref 14)

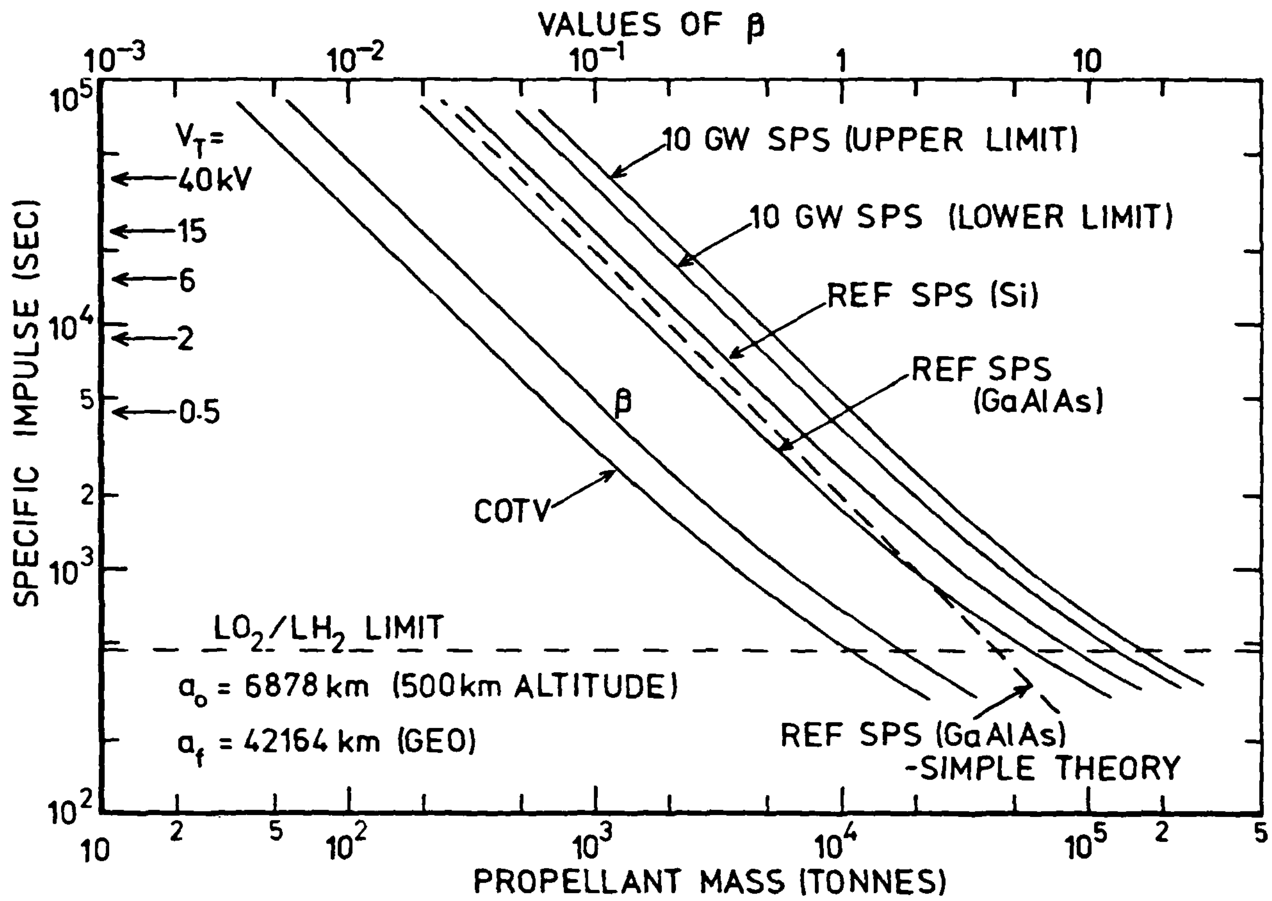

Fig 5 Propellant mass as a function of specific impulse for various SPS designs

Fig 6

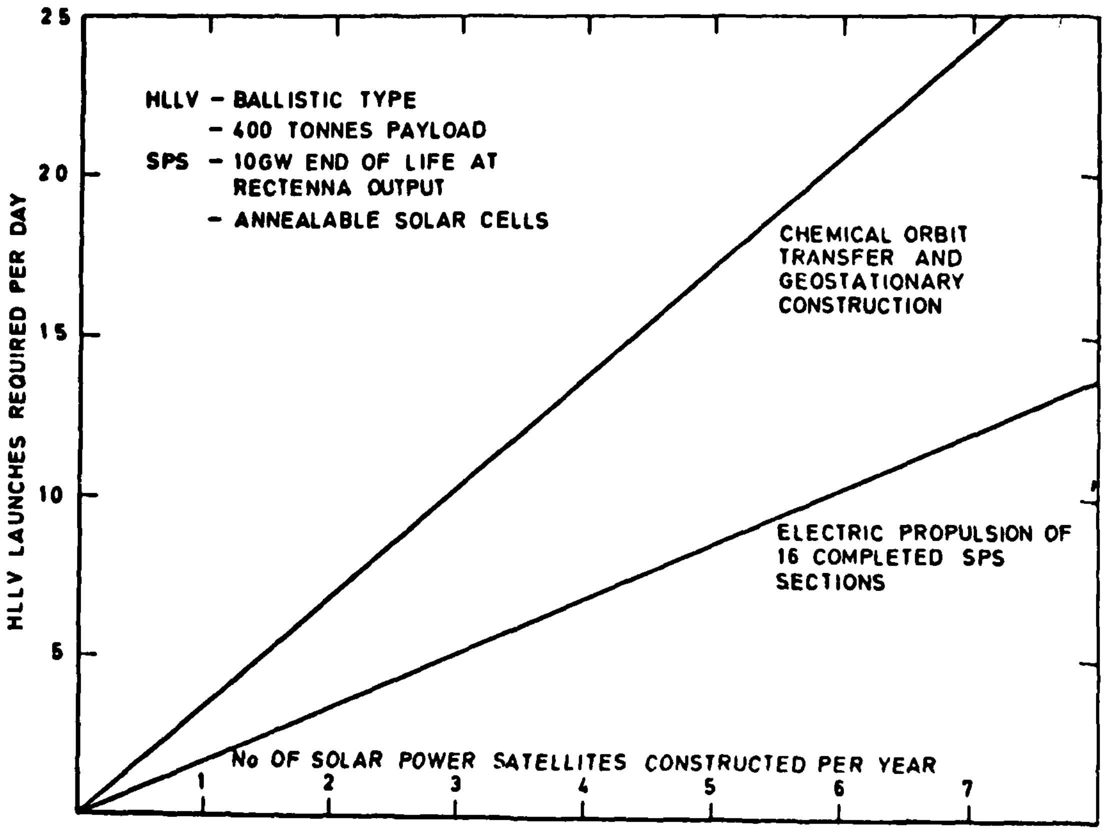

Fig 6 Number of HLLV launches required per day as a function of the number of 10 GW solar power satellites constructed per year, for both chemical and electric propulsion orbital transfer systems (Davis, Ref 20)

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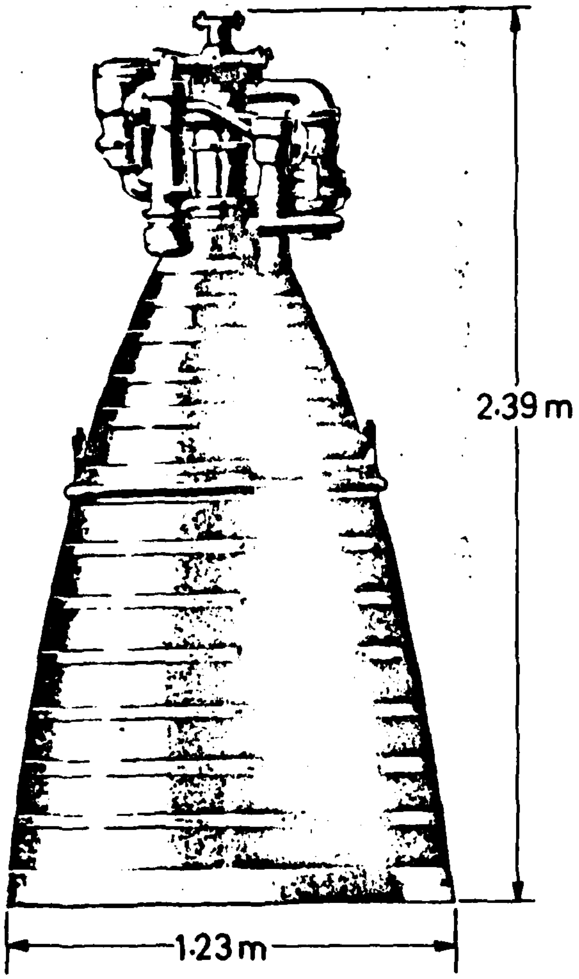

THRUST (LB)	20,000
CHAMBER PRESSURE (PSIA)	2000
EXPANSION RATIO	400
MIXTURE RATIO	6.0
SPECIFIC IMPULSE (SEC)	473.0
DIAMETER (IN.)	48.5
LENGTH (IN.)	
NOZZLE RETRACTED	50.5
NOZZLE EXTENDED	94.0

Fig 7 Artist's impression of NASA's advanced space engine (Hanley, Ref 18)

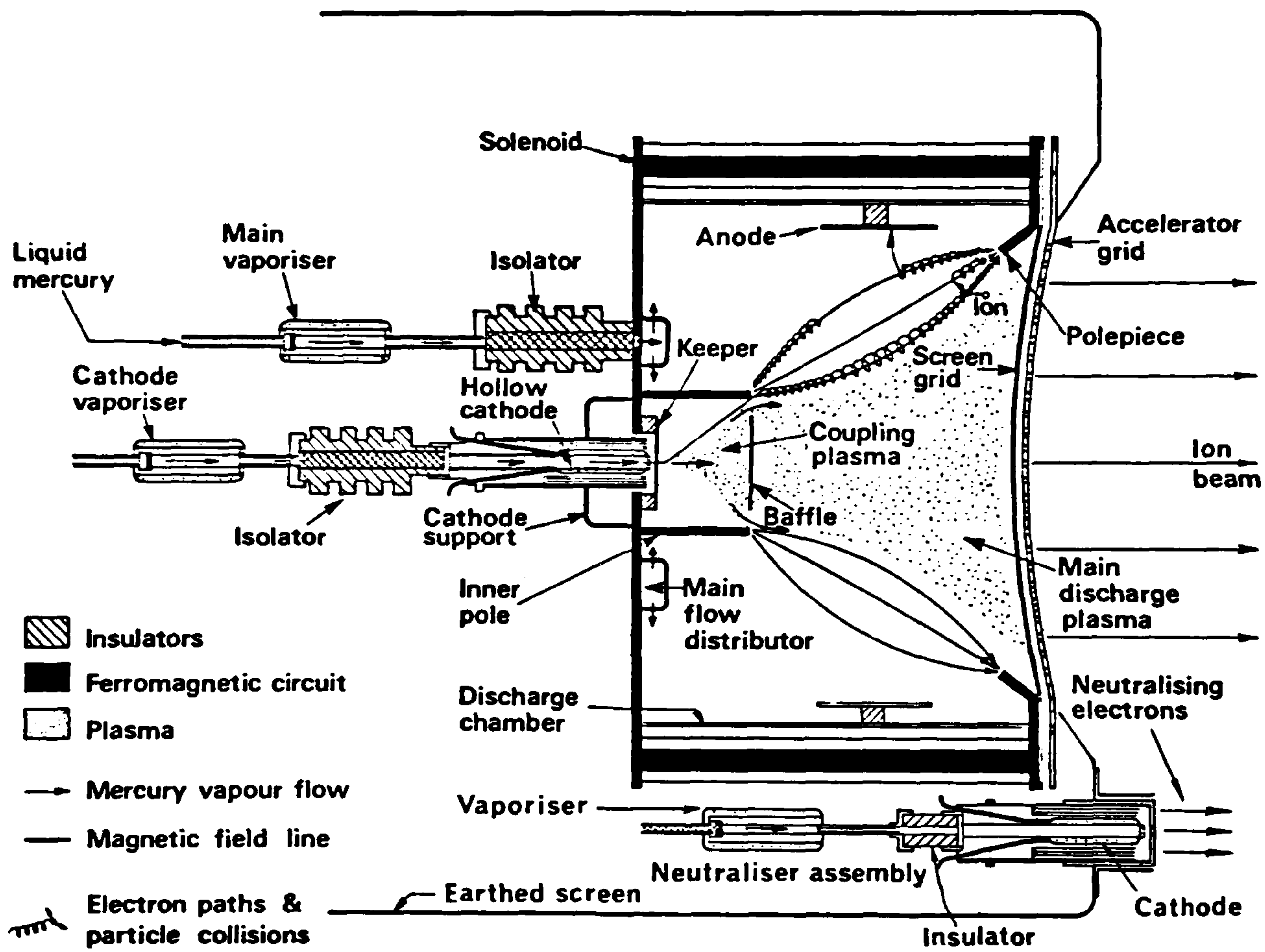

Fig 8 Schematic of Kaufman ion thruster, using mercury propellant (to use argon, flow control valves would replace the vaporisers)

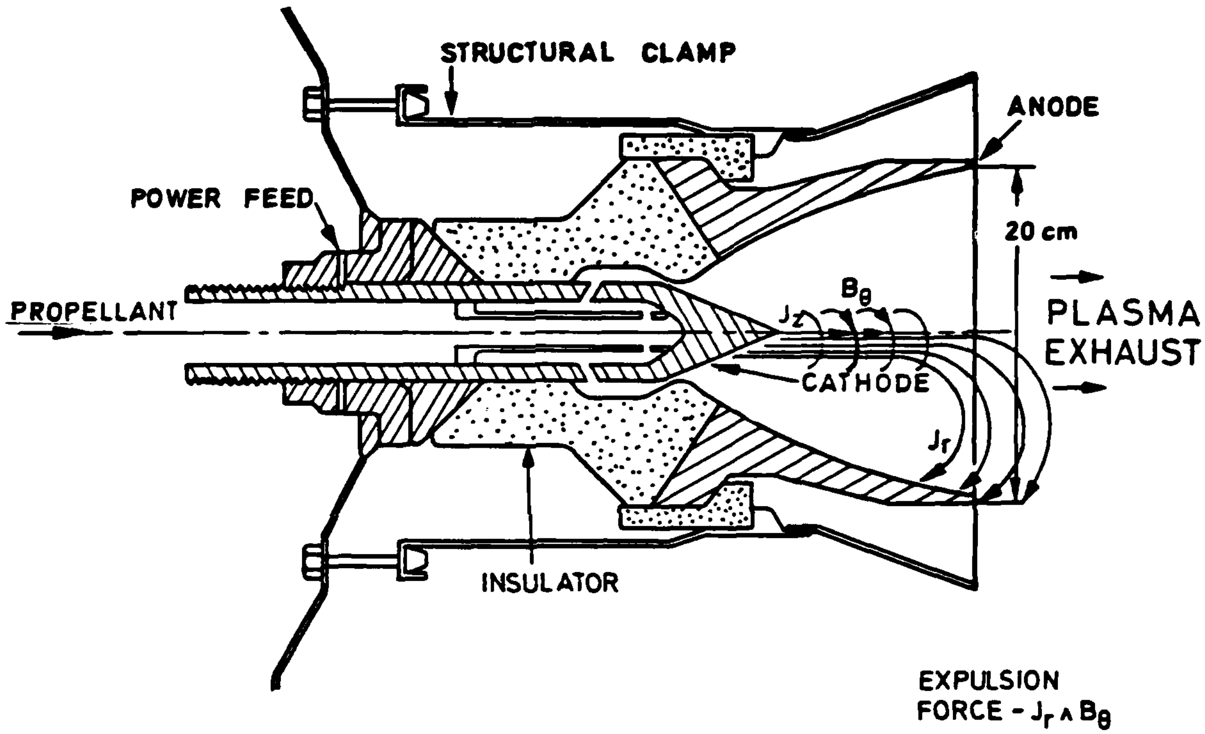

Fig 9 Schematic of 20 cm diameter MPD thruster (Grim, Ref 26)

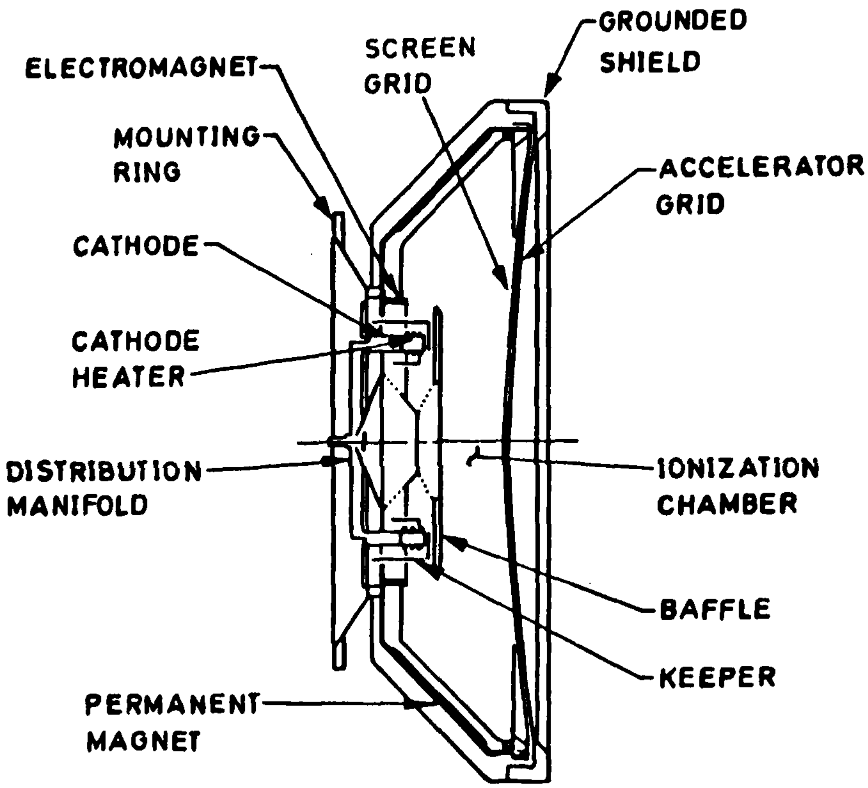

Fig 10 Schematic of 120 cm diameter argon ion thruster (Grim, Ref 26)

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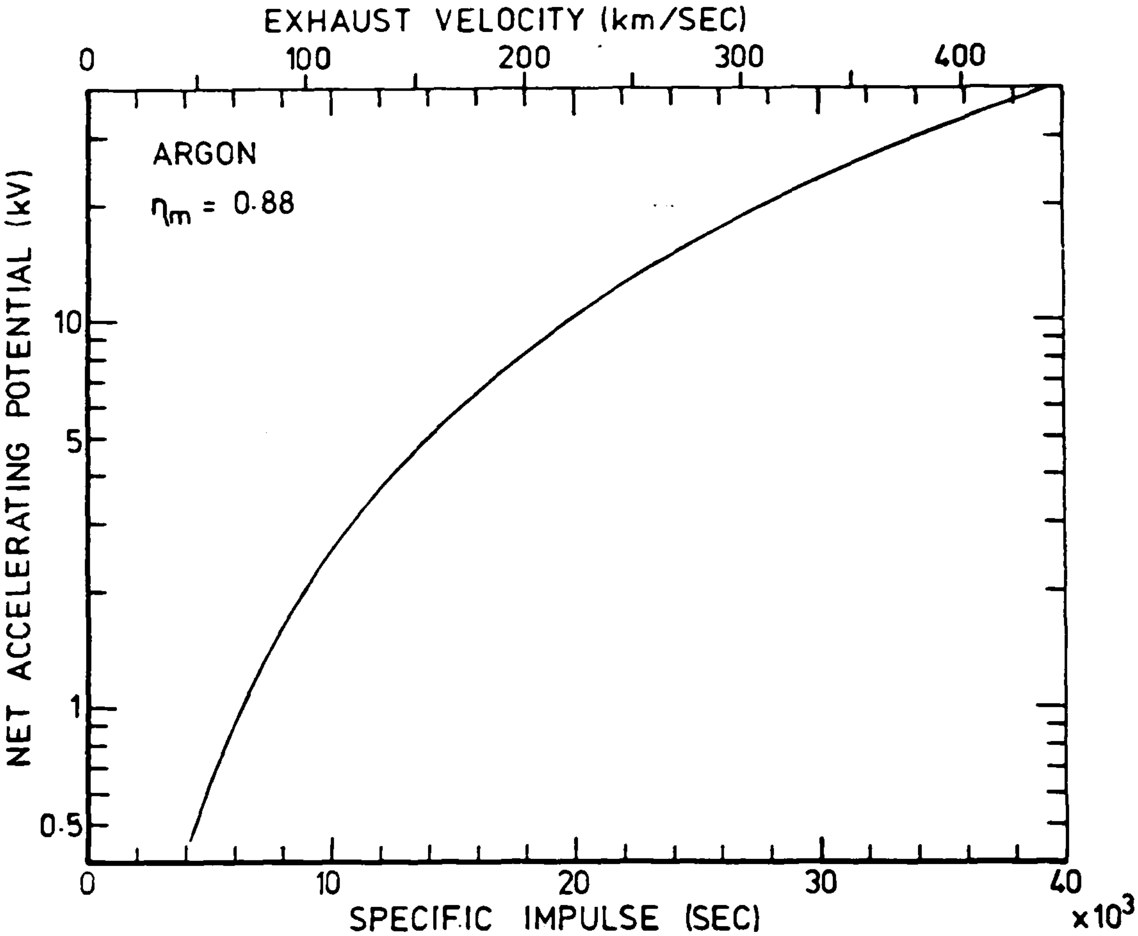

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Fig 11 **Specific impulse and exhaust velocity as functions of accelerating voltage for singly-charged argon ions**

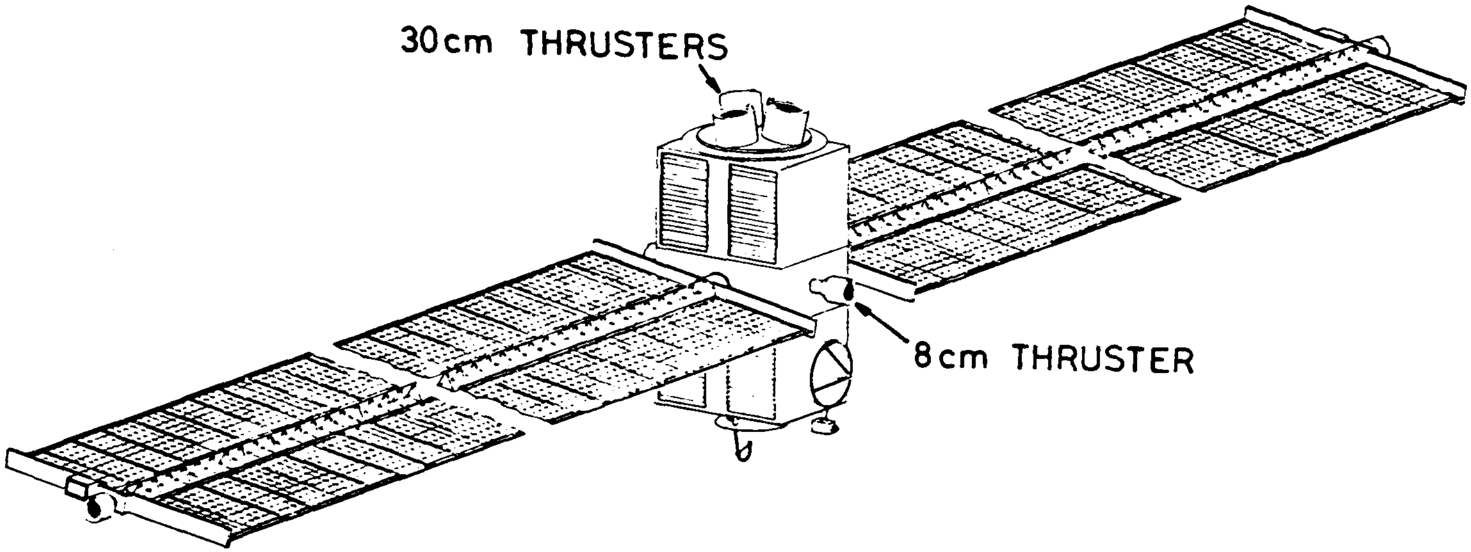

Fig 12 SERT–C spacecraft (NASA, Ref 43)

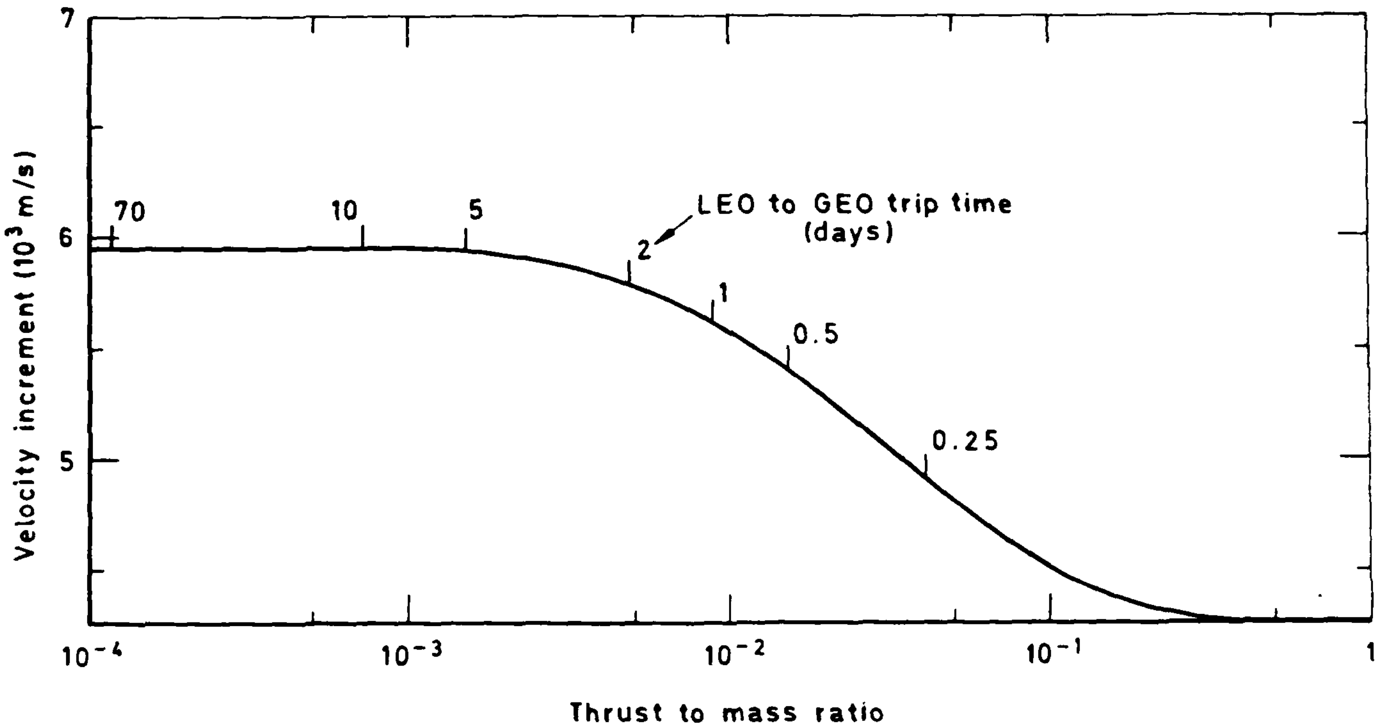

Fig 13 Velocity increment required for orbit transfer to GEO of the communications platform shown in Fig 2, as a function of T/M_0 (Kunz, Ref 10)

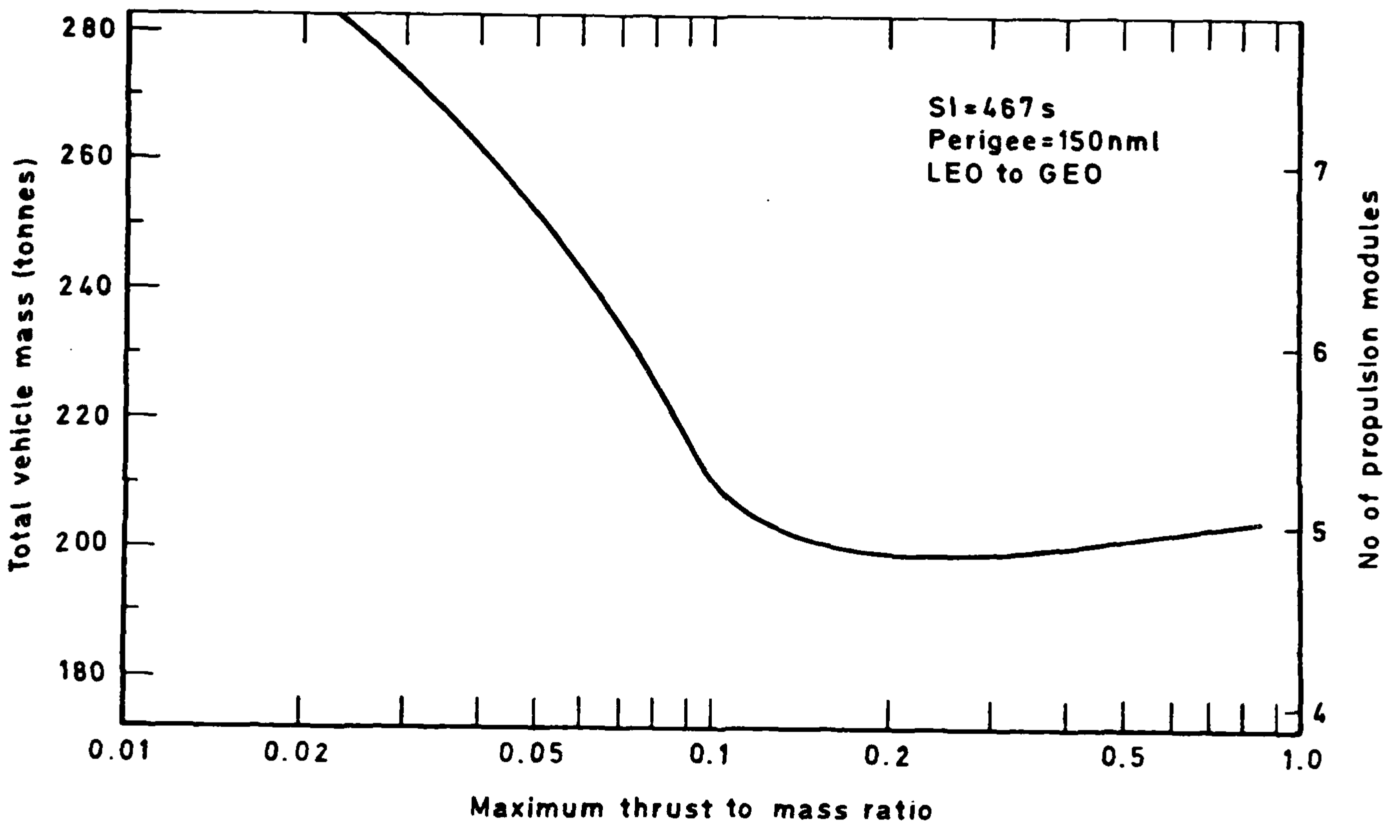

Fig 14 Total vehicle mass, as a function of T/M_0, assuming chemical propulsion (Kunz, Ref 10)

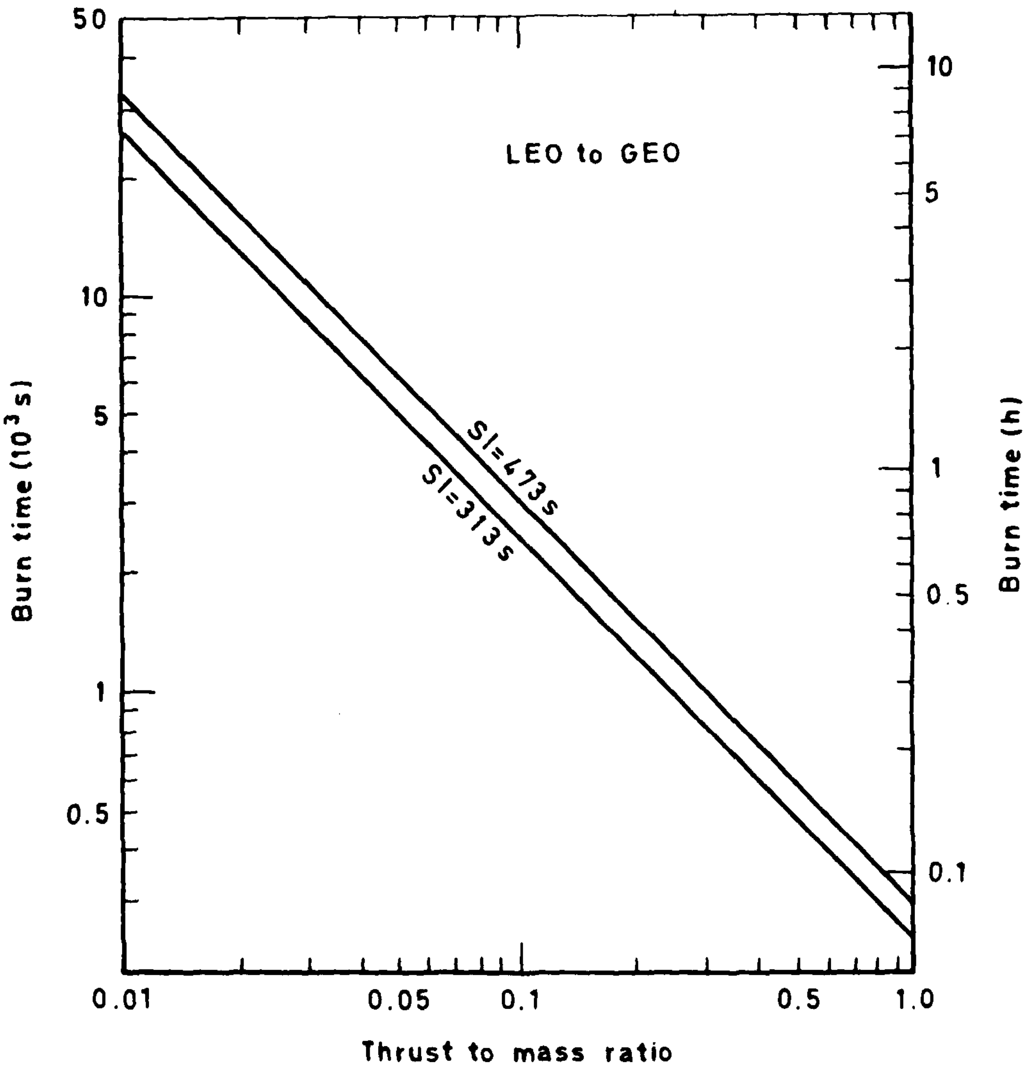

Fig 15 Motor burn time for orbit transfer to GEO, as a function of T/M_0 (Kunz, Ref 10)

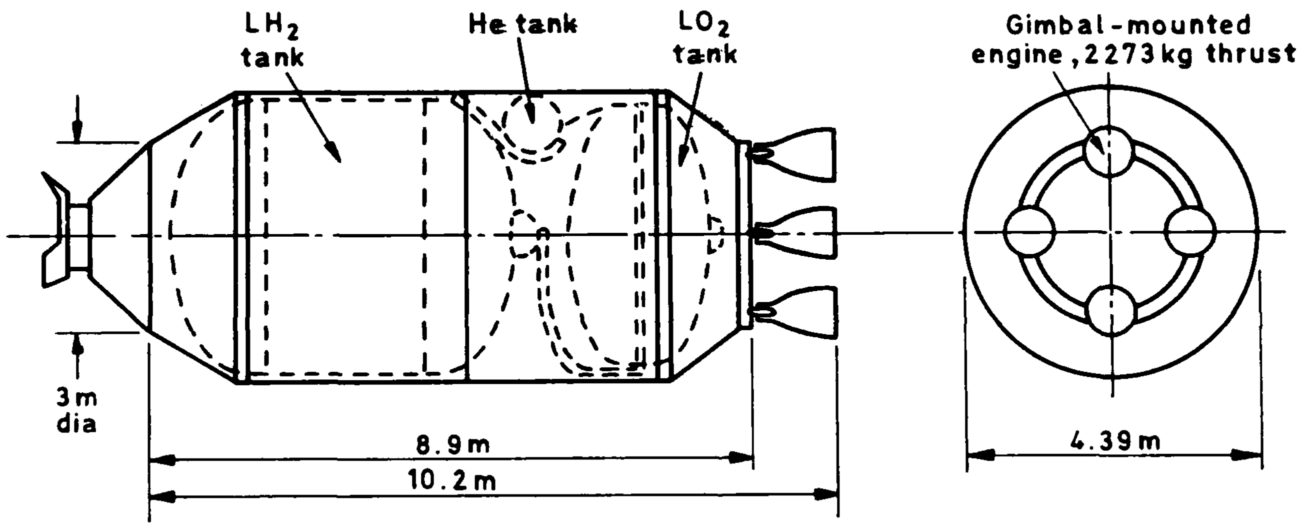

Fig 16 Low thrust propulsion module, using four advanced LO_2/LH_2 motors of 2273 kg thrust each (Kunz, Ref 10)

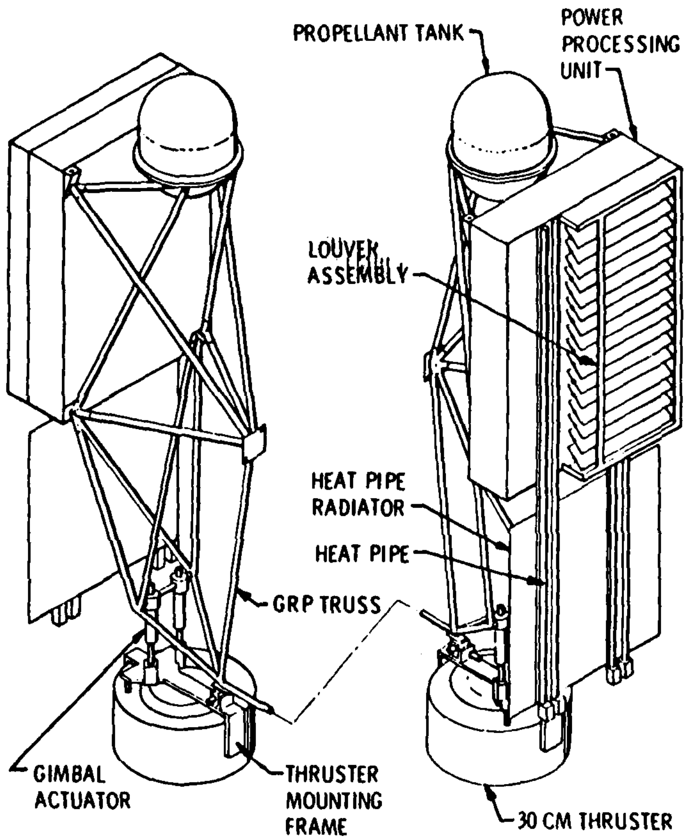

Fig 17 Ion thruster propulsion module designed by NASA Lewis (Cake, et al, **Ref 45)**

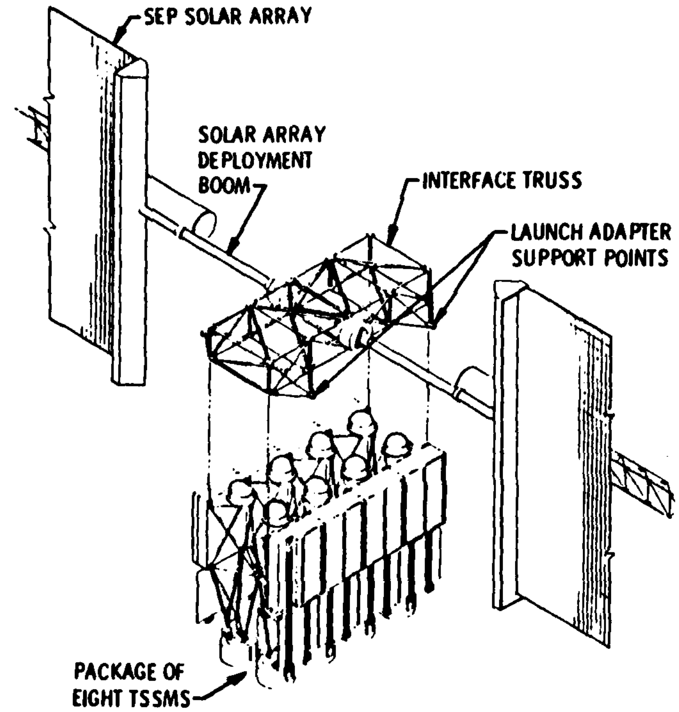

Fig 18 Space tug vehicle constructed from 8 TSSMs (Cake, et al, Ref 45)

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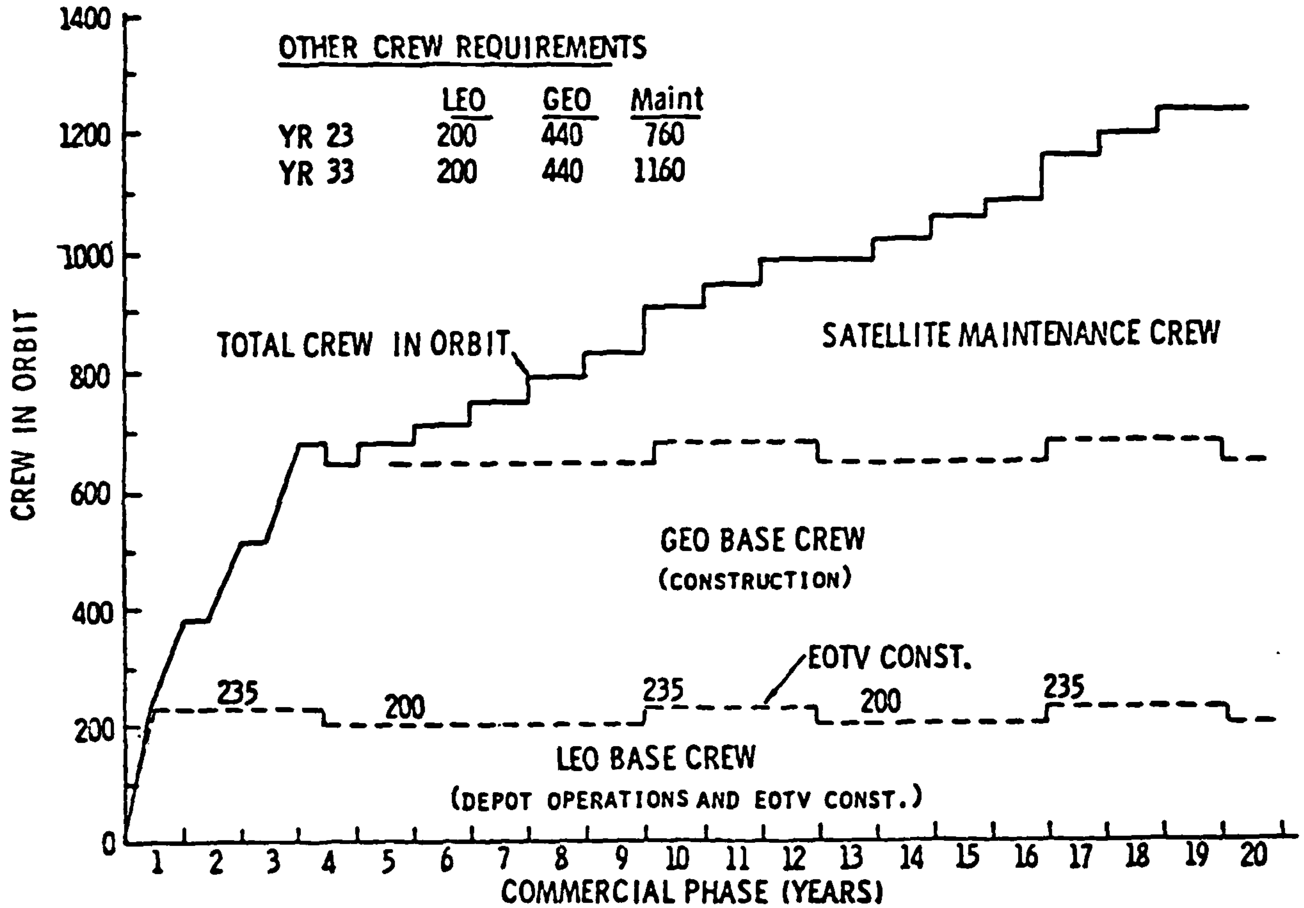


Fig 19 Number of crew in orbit as a function of time for an SPS programme of 60 spacecraft in 33 years (DOE/NASA, Ref 47)

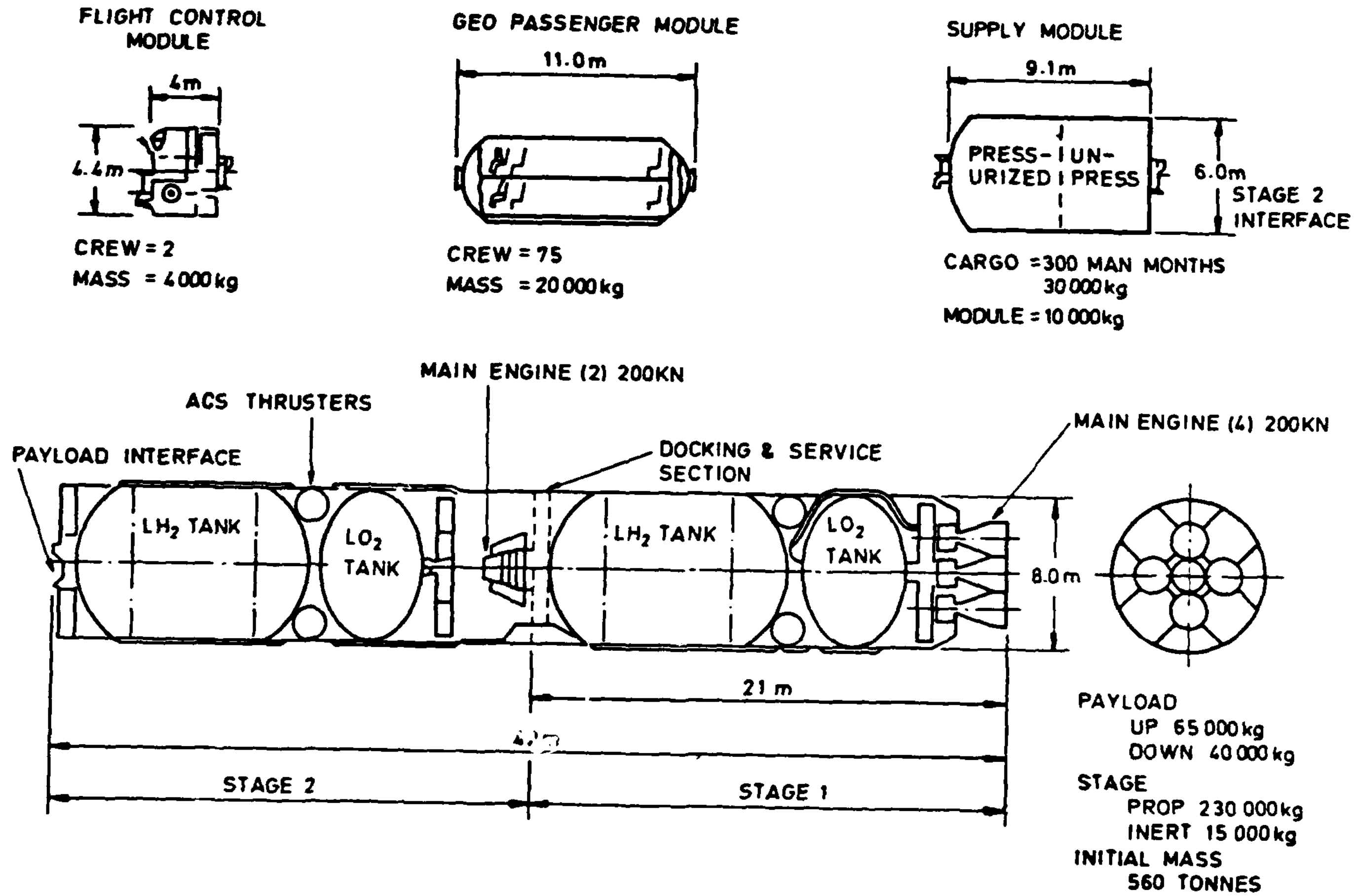


Fig 20 Chemically propelled OTV, indicating configurations of personnel and cargo versions (Woodcock, Ref 48)

TR 81060

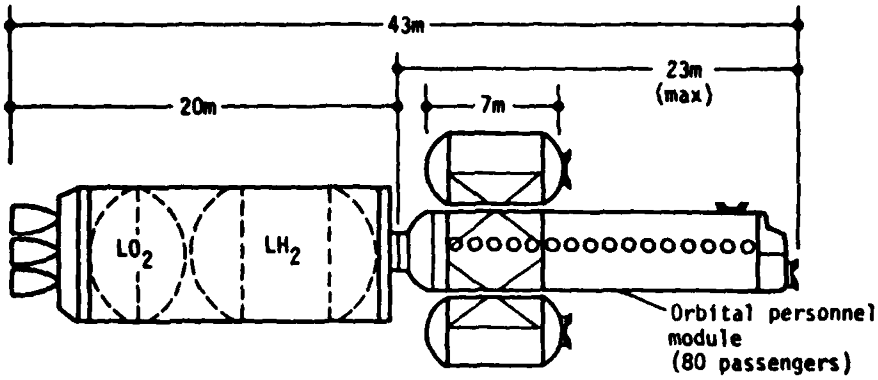
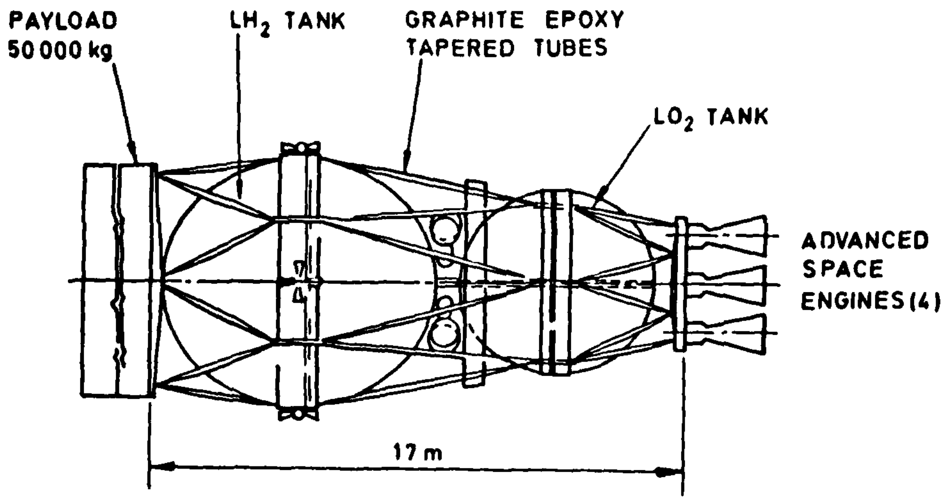


Fig 21 Single-stage chemically propelled OTV (DOE/NASA, Ref 4)



PROPELLANT MASS = 132 000 kg DRY MASS = 6800 kg

Fig 22 Schematic of advanced, chemically propelled cargo OTV (Rehder, et al, Ref 50)

Fig 23

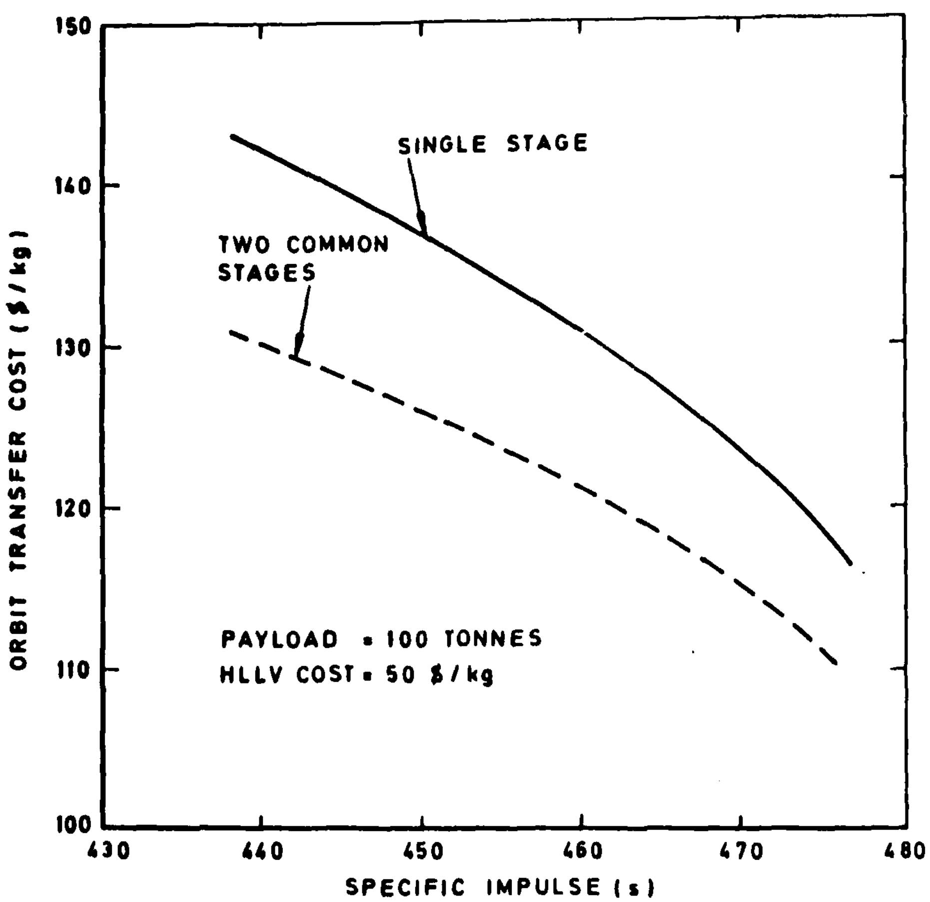


Fig 23 Effect of SI and staging on chemically-propelled OTV costs (Rehder, et al, Ref 50)

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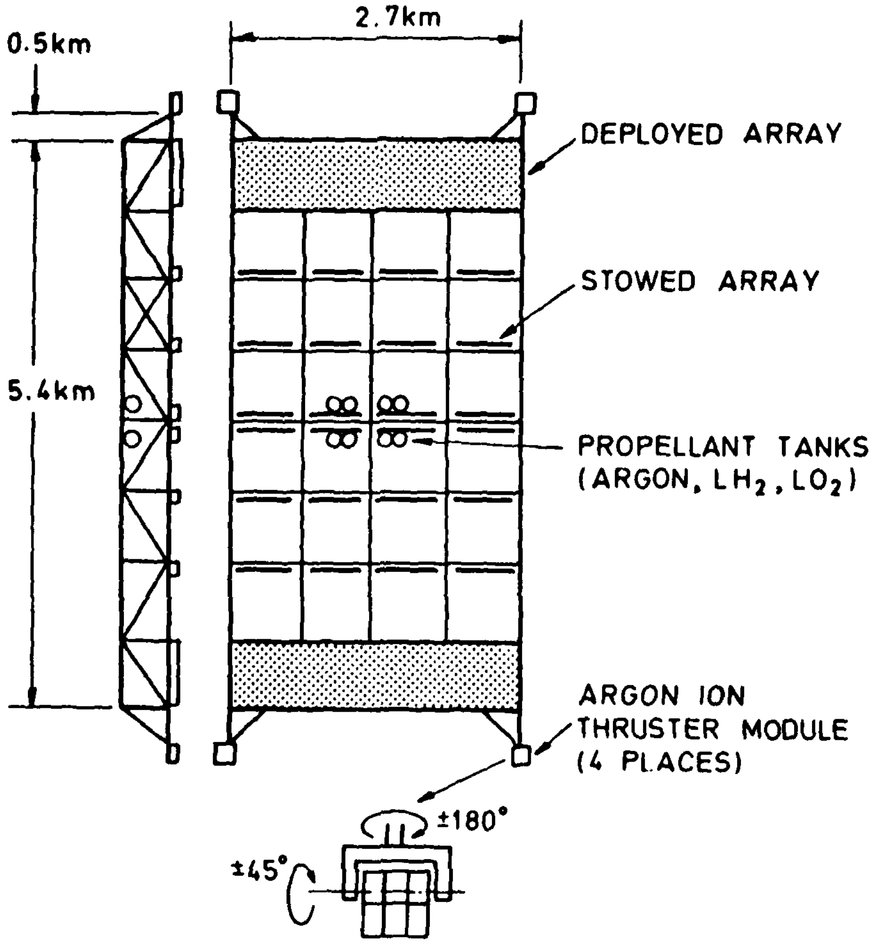

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Fig 24 Boeing concept for transporting sections of a 10 GW SPS to GEO (Piland, Ref 51)

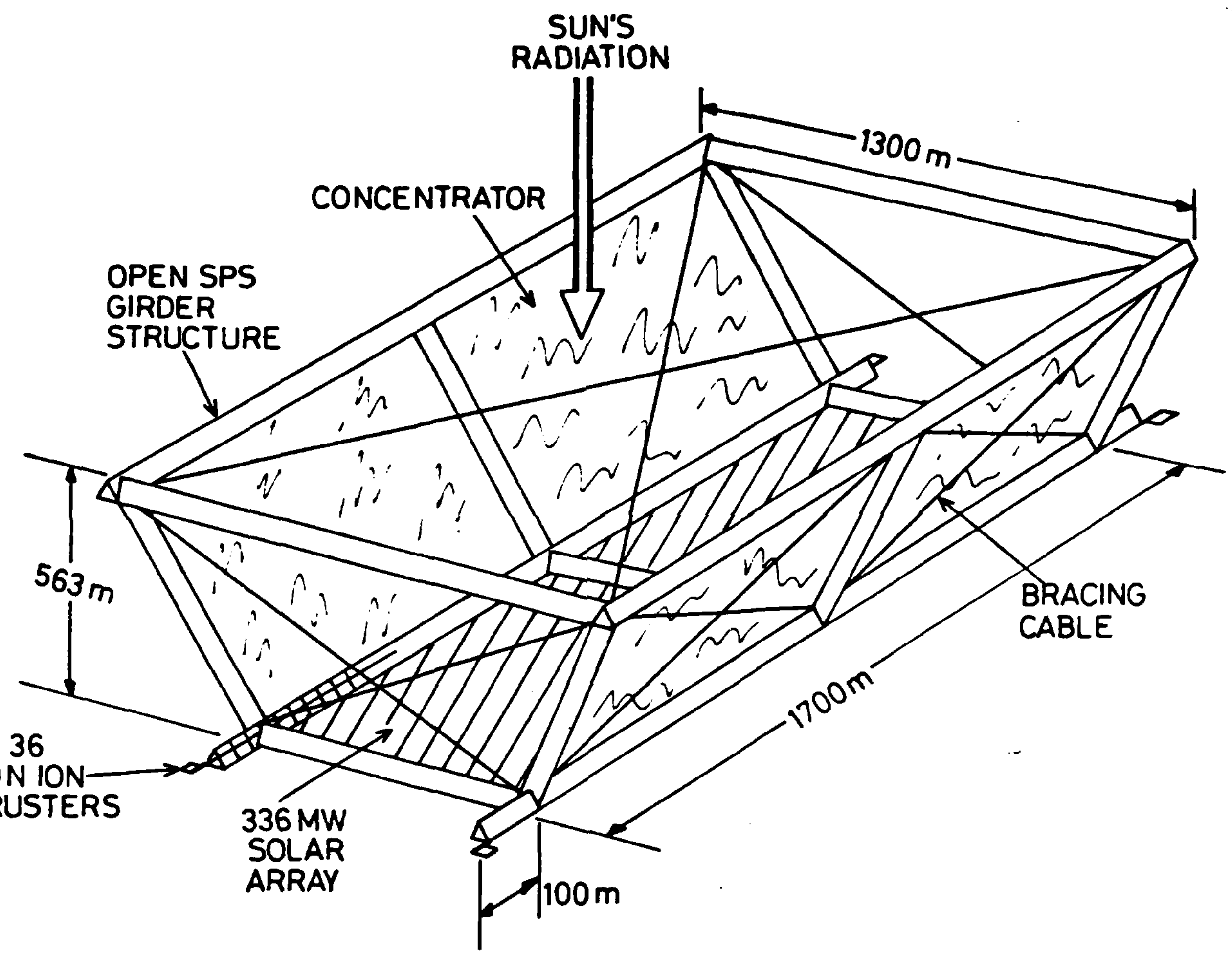

Fig 25 Rockwell COTV concept, using argon ion propulsion (Hanley, Ref 18)

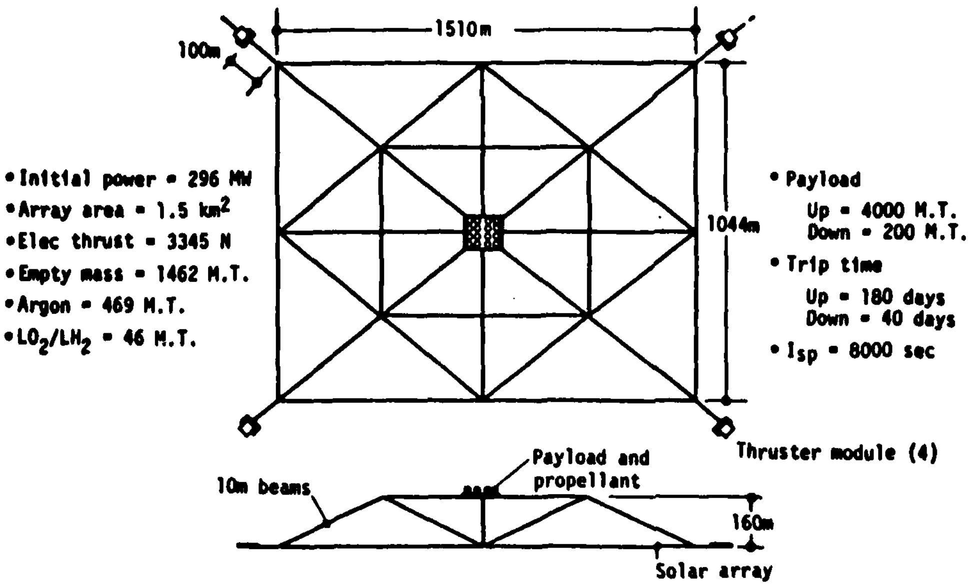

Fig 26 COTV employing silicon solar cells (MT = metric tonnes) (DOE/NASA, Ref 4)

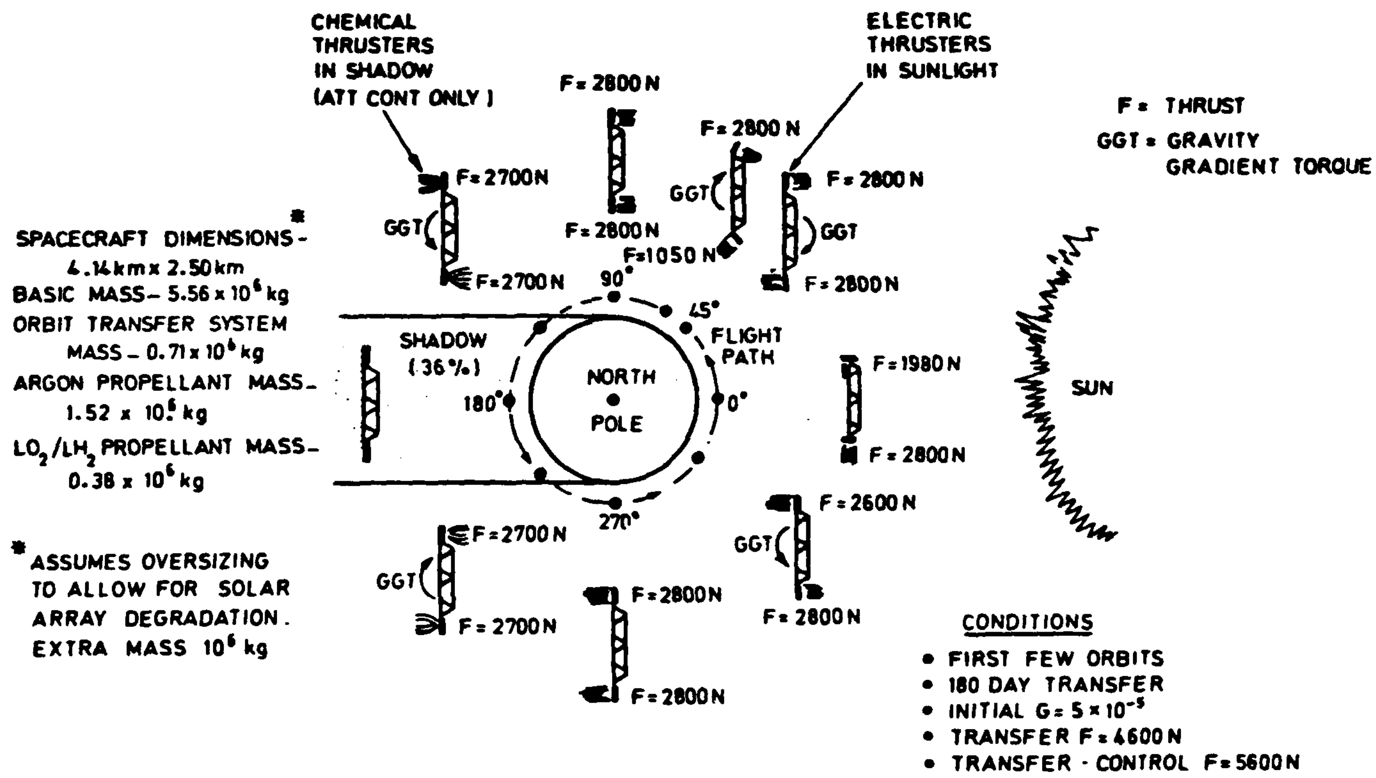

Fig 27 Thrusting strategy for rectangular section of 10 GW SPS during spiral orbit raising (Boeing, Ref 55)

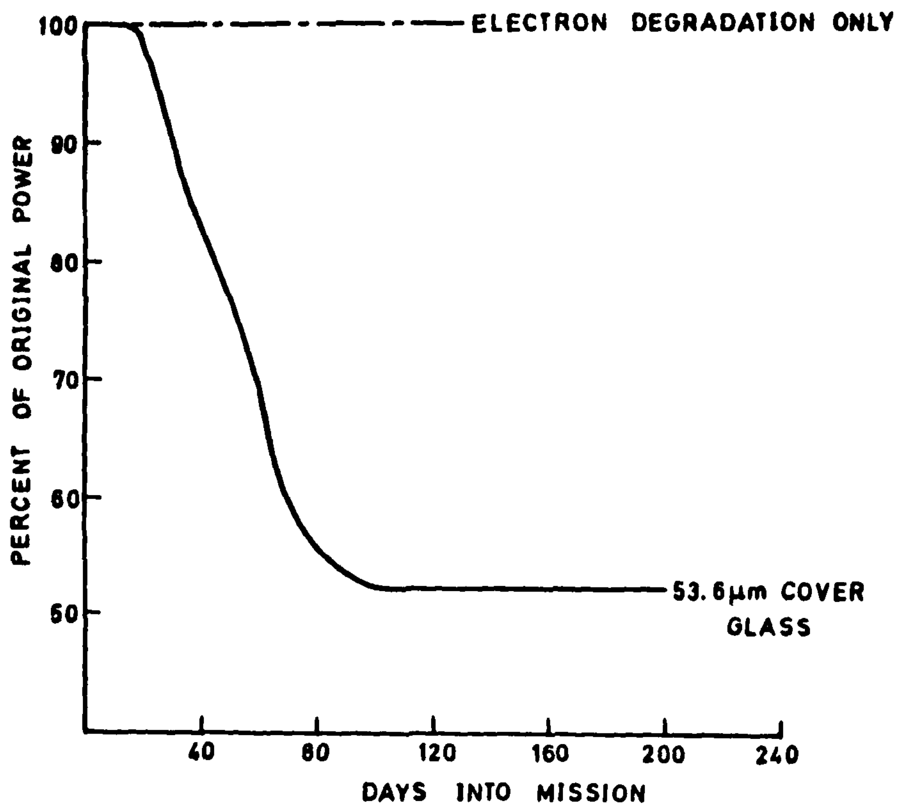

Fig 28 Degradation of gallium arsenide solar cells during orbit-raising (Weddell, et al, Ref 38)

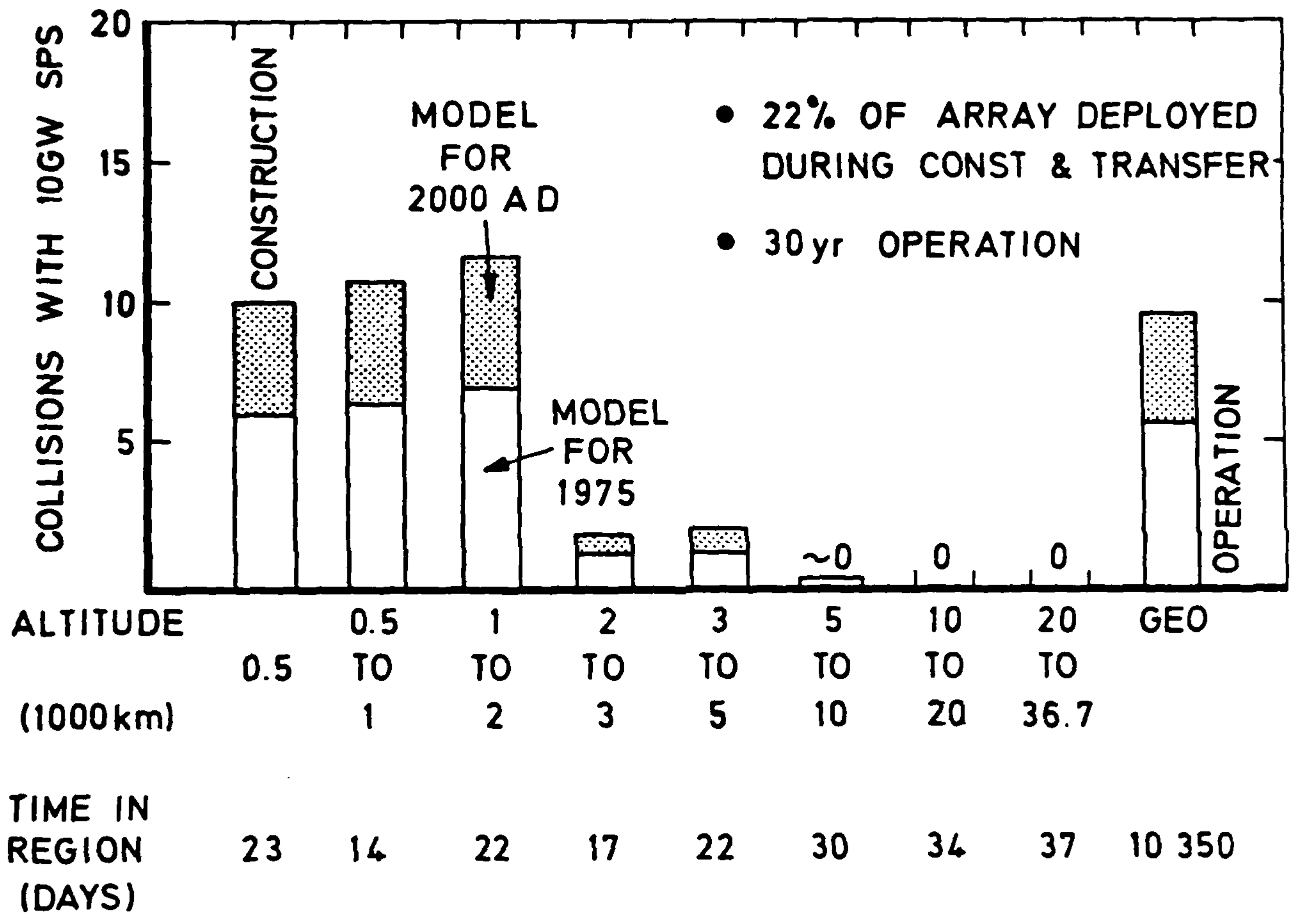

Fig 29 Collision risk during construction, orbit transfer and operation of a 10 GW SPS (Boeing, Ref 55)

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