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ELECTRIC PROPULSION IN 1964 -- A STATUS REVIEW

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Introduction

The year 1964 has not yet brought the expected flight test of an electric thrust engine. However, the development of electric propulsion as a new branch of rocket technology has made significant progress in many ways. Power efficiency, propellant efficiency, and lifetime of thrustors have increased. Studies of propulsion systems for specific applications have resulted in realistic appraisals of performance figures and optimization schemes.

Desirable features of power sources have been identified to an extent that guidelines for power supply development can be specified. Mission studies have further demonstrated the superiority of electric propulsion systems over other systems on a number of unmanned and manned missions. Most significant, however, is probably the growing recognition of the fact that the development of electric propulsion systems should not be shelved until an indisputable "requirement" arises. Our national space program is evolving under a charter which makes unmanned and manned exploration of space a national goal. In this light, electric propulsion is one very promising propulsion system of great potentialities. It is not a competitor, but a complement to other systems, and it will extend our payload capabilities considerably with respect to mass, flight times, and mission feasibility. It should be noted that none of the decisive accomplishments in our space program so far was achieved as fulfilment of an immediate "requirement," but rather in response to the challenge of a new technology which had come within reach.

Electrothermal Thrustors

Electrothermal thrustors, because of their inherent simplicity in design and operation, are of interest for mission applications which require an optimum exhaust velocity on the order of 10 000 to 15 000 m sec^{-1} ($I_{sp} \approx$ 1000 to 1500 sec). Power efficiencies of 40 to 50%, and operating times of weeks, have been achieved with 30 kw engines.

In its simplest form, an electrothermal engine is a "resistojet" which heats a flow of hydrogen while it passes along hot tungsten surfaces. An engine with concentric jackets as heater elements, and an engine with heated, spirally wound filaments, were tested recently at the NASA-Lewis Research Center, both at 30 kw power input. Another version of the resistojet is under investigation at Giannini Scientific Corporation. In this engine, heating occurs in a small region of contact resistance between a tungsten rod and a loosely fitting, concentric tungsten cylinder.

The theory of arc jet thrustors is still not well understood. Processes of heat conduction and radiation, momentum transfer, wall effects, magnetic pinch phenomena, and thermodynamics of non-equilibrium gases are so complex that prediction of arc jet behavior, correlation between different gases, or even scaling laws are not yet within reach [1].

Uncooled thrustors, whose external chamber and nozzle walls assume high temperatures, are distinguished by relatively uniform gas temperature profiles and good efficiencies, but by limited specific impulses. Thrustors with cooled walls allow higher temperatures in the axial region and therefore higher specific impulses, but their efficiencies are lower because of the relatively "cold" regions near the walls.

Problem areas of electrothermal thrustors include electrode lifetime [2], heat losses [3], and storage of hydrogen. However, progress in these areas is being made, and there is a possibility that electrothermal thrustors will find application in missions of a few week's duration, provided that the electric power source is needed for other purposes during non-propulsive periods.

A very exciting new development in electric thrustors which may possibly turn out to be a breakthrough was reported recently by Giannini Scientific Corporation [4]. By modifying the design and lowering the chamber pressure of a hydrogen arc jet engine, exhaust velocities up to 100 000 m sec^{-1} ($I_{sp} \approx 10\,000$ sec) were obtained. Power efficiency at this I_{sp}, defined [19] as

$$\eta = \frac{F^2}{2 \dot{M} W_e}$$

where

F = thrust (newtons)

$\dot{M}$ = mass flow rate (kg sec^{-1})

W_e = electric input power (watt)

was 47%. The chamber operated at a pressure of about 10^4 newtons m^{-2} (0.1 atm.); its nozzle was negligibly small (Fig. 1).

The stagnation temperatures corresponding to the measured exhaust velocities would be above 100 000°K. It is obvious that gases at such temperatures could not be contained in a tungsten chamber of relatively small dimensions. The only plausible reason why exhaust velocities of this magnitude can occur is an acceleration by magnetic forces, caused by the very high discharge currents. The theory of this process, which may be considered as a self-excited Hall current effect, is not yet developed. Exhaust velocities as high as 200 000 m sec^{-1} ($I_{sp} \approx 20\,000$ sec) at power efficiencies above 80% are forecast for this system which was given the name "thermo-ionic propulsion system" by its developer, A. C. Ducati at Giannini Scientific Corporation. Results similar to those quoted above have been reported by AVCO RAD, and others. One obvious feature of the thermo-ionic system is the lack of an expansion nozzle. As a consequence, there will be no recovery of dissociation and ionization energy. However, as Fig. 2 indicates, this loss is almost irrelevant at exhaust velocities on the order of 100 000 to 200 000 m sec^{-1} ($I_{sp} \approx 10\,000$ to 20 000 sec).

Electrostatic Thrustors

A very good survey of electrostatic thrustor development was published by Mickelsen and Kaufman [5]. Electron bombardment-type thrustors (Kaufman engines) have been developed at the Lewis Research Center to a point where not much further improvement can be expected except for the lifetime of the cathode which is still limited to about 1600 hours with present designs. Units up to 50 cm diameter and 30 kw electric power were successfully operated in the exhaust velocity range from 40 000 to 100 000 m sec^{-1} ($I_{sp} \approx 4000$ to 10 000 sec). Total thrustor efficiency, defined as the product of propellant mass efficiency times power efficiency, reached 79% at an exhaust velocity of 91,000 m sec^{-1} ($I_{sp} \approx 9100$ sec). Neutralization of the ion beam appears to be no problem. The lifetime of this thrustor type is limited by the endurance of the electron-emitting cathode. A different kind of cathode, developed at Electro-Optical Systems, Inc., shows great promise for lifetimes in excess of 10 000 hours. It uses a low-voltage arc in a cesium atmosphere as electron and ion source; the cesium provides a low work function of the cathode, and also the ions that impinge upon the cathode for heating. Erosion and cathode sputtering are not problems in this cathode because of the low arc voltage. Contact ionization thrustors are under development and testing at Electro-Optical Systems, Inc., at Hughes Research Laboratories, Inc., at Space Technology Laboratories, and at Lewis Research Center. All of them use porous tungsten as surface, and cesium as propellant. The Hughes Research Laboratories team concentrated much effort recently on the linear strip tungsten-cesium ion source which gives better results than the former annular design [6].

Reliability and performance of the contact ion source has greatly increased during recent years. Ion optical problems appear to be solved; however, problems still persist with respect to neutral efflux at high current densities. High current density on the order of 20 to 30 mA cm^{-2} is desirable because it permits smaller ion emitter surfaces, and hence smaller heat losses of the source. Neutral efflux not only reduces the propellant efficiency; more important even is the danger of charge exchange of ions and atoms, and subsequent impingement on surfaces. Recent data reflecting the progress achieved in contact ionization thrustors cannot be reported because of classification. Several types of modified ion thrustors, among them the divergent-flow thrustor with increased power efficiency, the circular-flow thrustor with greatly reduced thermal losses, and the reverse-feed thrustor with simple surface contact ionization, are under theoretical and experimental investigation at the NASA-Lewis Research Center [5]. One type of ion source, the duoplasmatron, has experienced some decrease of emphasis recently, although it is capable of high current density and high propellant utilization. This source suffers from erosion of electrodes, caused by charge exchange in and near the exit orifice.

Colloidal-particle thrustors were further investigated at several places (See Ref. 5.). Vapor condensation, preforming of particles, electric spraying, and other methods were applied to produce particles of the right sizes. Charging resulted from the process of particle production, from corona discharges, or from other external means. While these experiments provided interesting and valuable data on colloid propulsion, they are still far from a colloid thrustor useful for vehicle propulsion.

One requirement of electrostatic thrustors, beam neutralization, was considered a problem several years ago. At present, concern about this problem has subsided. Experiments by Sellen and Kemp in 1961 [7], and subsequent theoretical analysis, have shown that beam neutralization under space conditions should occur without difficulty. Under physically restricted laboratory conditions, neutralization is always achieved by interaction of the beam with walls, residual gases, etc.

Overall performance figures of electric propulsion systems using ion thrustors still suffer from the relatively large masses of thrustors, and of power conversion and switching equipment. In spite of these shortcomings, ion thrustors of present design could be used with advantage as auxiliary thrust systems for attitude and orbit control of satellites, and as primary thrust systems for specific missions such as Van Allen Belt mapping.

Electrodynamic Thrustors

Research and development of electrodynamic or plasma thrustors has continued at many places, and on many systems. There are at present about a hundred different devices, each of which represents a plasma engine. Most of them can be categorized into six basic types: the J x B or constant field accelerator, which includes Hall current accelerators; the pulsed plasma gun, either coaxial or rail; the radial pinch engine; the oscillating-electron engine; the traveling wave accelerator; and the magnetic expansion thrustor.

Constant field accelerators, frequently studied as an "afterburner" for arc jet engines, are relatively simple in design and operation, but still not well understood theoretically. Optimism with respect to their efficiencies decreased somewhat when it turned out that earlier mass-flow data were masked by erosion and outgassing. The radial pinch engine, under development at Republic Aviation Corporation, reportedly achieved an efficiency of about 30% after correction for erosion and outgassing effects. This engine is pulsed at a rate of a few pulses per second.

Efficiencies of plasma engines are still confined to the region below about 40%. If viewed against the first experiments with plasma engines six or eight years ago when efficiencies were on the order of 5%, the present status reflects a remarkable progress. Present efficiencies may even be sufficient for some secondary missions such as attitude or orbital control, provided that the electric power source is needed anyway for the operation of the payload mission during non-propulsive periods. However, an electric propulsion system should have an efficiency of at least 60% to be of potential use for space missions in which the electric system must provide a major part of the propulsion. An efficiency of 80 to 85% is very desirable.

Processes of heat transfer, energy conversion, interaction of local electrostatic fields, momentum transport between electrons and ions, energy losses, and erosion are obviously quite complex and not fully understood. Consequently, most of the work in the plasma engine field aims at these basic problems rather than at engine models which can be tested in space.

A considerable part of present plasma work is concentrated at the NASA-Lewis Research Center. In particular, Hall current accelerators, traveling wave accelerators, and magnetic expansion thrustors are under investigation in-house, and in a number of contracts with General Technology Corporation, United Aircraft, AVCO, and others. The Air Force, although with somewhat decreased emphasis, pursues plasma thrustor development both in-house and by contract. Among the companies active in plasma research are Electro-Optical Systems, RCA, and General Electric.

A very interesting and successful application of electromagnetic acceleration was reported by Electro-Optical Systems [8]. In a configuration called a "modified Hall current accelerator," a central cathode is surrounded by a cylindrical anode. The discharge d-c current is almost radial. A coil produces a strong magnetic field with a radial as well as an axial component in the region of the arc (Fig. 3). The radial current and the axial field component produce a circular current component around the axis; this current, together with the radial field component, produces a force in the axial direction. So far, exhaust velocities up to 80 000 m sec^{-1} ($I_{sp} \approx 8000$ sec) at power efficiencies of about 50% were measured with hydrogen as propellant. Other gases also performed satisfactorily.

Space Testing and Mission Planning

Space testing of electric thrustors in short ballistic flights, originally planned for 1962, is now expected during this year. It will include the WADD-sponsored EOS ion engine (tungsten-cesium, contact ionization) which had to undergo a redesign of the power supply. Flight testing of two NASA-sponsored ion engines, the Kaufman engine developed at NASA-Lewis Research Center (ion bombardment, mercury), and the Hughes engine (tungsten-cesium, contact ionization), will have been preceded by a period of very extensive ground testing at the Lewis laboratories.

The first mission application of electric thrustors, high-precision control of a synchronous communication satellite with tungsten-cesium ion engines and a solar power supply, may occur within the next few years. The system is presently under development at Hughes Research Laboratories under NASA contract. The next mission may be a solar-powered, ion propelled probe which spirals slowly through the Van Allen Belts and maps the radiation intensities in fine detail. Design studies of this project are under way at various places.

Unmanned missions with electric propulsion systems have found much attention recently [9, 10, 11]. As pointed out by Moeckel [10], one single type of an electrically propelled space vehicle, launched by a two-stage Saturn IB, and powered by a SNAP-50 type electric source, can accomplish most of the unmanned exploration missions within our solar system. If we were restricted to chemical and nuclear propulsion for the spacecraft, even Saturn V or a more advanced vehicle as booster could accomplish only a fraction of the desired missions (Table I).

Studies of specific missions for electric propulsion systems were still parametric; however, the assumed figures for power levels, specific power, efficiencies, and systems lifetimes were chosen according to estimates which appear realistic at the present time.

Fimple and Edelbaum of United Aircraft Corporation [12] studied planetary probe flights and lunar supply missions, based on SNAP-50 power supplies, and Saturn IB and Saturn V carrier vehicles. Of particular interest in these studies is the comparison of transportation capabilities of electrically propelled vehicles with transportation capabilities of nuclear rockets. This comparison, of course, must take into consideration such figures as specific power of the electric and specific impulse of the nuclear systems. It will give the payload ratio as a function of the total mission time for the planetary probes; in the case of the lunar supply mission, the number of Saturn V launches per month was calculated as a function of the supply rate in tons per month.

As a typical example of a planetary probe mission, a Jupiter probe was selected. Figure 4 shows the payload ratio as a function of the mission time. A bombardment-type ion thrustor (Kaufman engine) with constant specific impulse and 80% efficiency, and an unpowered coasting period during transfer, were assumed for the electric vehicle; for the nuclear rocket, a $1\frac{1}{2}$-stage advanced tungsten-core system was chosen. The comparison shows that the capabilities of electric systems are superior, except for very small payload ratios at which the nuclear systems may be a little faster.

Transportation capabilities of lunar ferry vehicles are shown in Fig. 5. Round trip time for nuclear vehicles is 10.5 days; electric vehicles, as studied here, require 100 to 200 days for a round trip. In both cases, landing on the moon is accomplished by a chemical propulsion system. Again, it is noted that the transportation capability of electric vehicles is superior, under the assumption that transfer time is not of particular concern in a well-organized supply operation.

The manned mission to Mars, which had been the subject of several studies in the past, found new attention in the light of radiation shielding requirements, and of recent trajectory work [10, 13]. Moeckel's results [10] are shown in Fig. 6, which compares nuclear and electric vehicles on the basis of equal initial masses in the earth orbit. The diagram shows that electric systems will be superior to nuclear vehicles even at short flight times, provided that an electric power source of suffiently high specific power can be developed. With power sources that appear feasible on the basis of present technologies, a manned Mars flight would require a total travel time of about 450 days.

As shown in the classical paper by Irving [14], a variable-thrust system is capable of larger payloads than a constant-thrust system at equal initial masses and flight times. However, this advantage is considerable only on vehicles designed for small payloads and shortest possible travel times. With medium payloads, the difference in capabilities is small (Fig. 7) [15]. In view of this fact, and of the very complex power supply system required for a variable-thrust engine, most of the recent studies assumed constant thrust periods, and unpowered coasting periods in between.

Power Sources

Hardly any paper on electric propulsion which was published recently failed to point out the desperate need for electric space power sources. Electric thrustor development trails that of large booster vehicles by several years; electric space power supplies will trail electric thrustors again by several years.

The SNAP-8, originally planned for 30 kwe and a specific power of about 0.03 kw kg^{-1} (specific mass of 30 kg kw^{-1}), had to undergo a thorough redesign. It will be valuable as a testbed for new technologies, but it will be too heavy for practical space missions. Solar-electric power sources up to about 30 kwe found considerable interest and design effort recently. Rankine cycle, Brayton cycle, and thermionic converters are considered as heat-to-electricity converters. The great hope of developers of electric thrustors, and of planners of space missions, is the SNAP-50, which will provide electric power of 300 to 1000 kw. This system is under design and development in a combined effort of the Air Force and the AEC. A number of the basic problems of energy conversion, among them high-temperature operation of turbines, bearings, valves and seals, and corrosion of liquid-metal cycle components, are not yet satisfactorily solved.

One continuous source of difficulties appears to be the mercury loop. Mercury has a poor lubricating quality, and very high corrosiveness at high temperatures. A single-loop system with an alkali such as rubidium offers far better prospects of success.

Thermionic conversion, which avoids moving parts, still faces unsolved problems of erosion, high temperature gradients, and lifetimes. One problem encountered by power supply systems operating in space met with some relief during the past year: it appears now that the abundance of meteoroids in space is far less than previously assumed, and that the mass of heat radiators need not be as large as expected so far. Besides generation of electric power, conversion of the primary voltage and current into the voltages and currents needed by the propulsion system is an immediate requirement. Technologies for this conversion are not yet fully developed for space use; existing systems are still heavy and complicated. Efforts are under way at various places to develop better and lighter systems. It is possible that the requirement for an elaborate power conversion system will be one of the decisive factors in the selection of electric propulsion systems for flight missions.

Electric power sources in the multi-kilowatt range will be required by at least three independent consumers in our space program: electric propulsion, manned spacecraft, and lunar base operations. As pointed out in recent studies on power supplies [16, 17, 18], these requirements can be met by four classes of power systems: 1 to 3 kwe, 30 to 60 kwe, 300 to 1000 kwe, and 4 to 10 Mwe. A vigorous effort to develop power supplies in these classes is not only a natural, but also an indispensable part of our national space program.

Conclusions

A few summarizing statements will recapitulate briefly the status of electric propulsion as of summer 1964.

1. Further progress in efficiency and lifetime was achieved by all types of electric thrustors. Efficiencies of electro-thermal and electrodynamic systems reached 40 to 50%, those of electrostatic systems 80 to 85%.

2. Lewis Research Center advanced the Kaufman ion engine to a high degree of perfection. With a cesium arc as ion source (EOS), this engine will combine high efficiency with long lifetime.

3. Electro-Optical Systems and Hughes Research Laboratories advanced the tungsten-cesium ion engine to a point where high efficiencies are assured and long lifetimes appear obtainable.

4. Two new thrustors with good prospects for high efficiencies and satisfactory lifetimes were developed: the thermo-ionic thrustor (Giannini Scientific Corporation), and the annular Hall current accelerator (Electro-Optical Systems). Both have high thrust densities and exhaust velocities on the order of 80 000 to 100 000 m sec^{-1} ($I_{sp} \approx$ 8000 to 10 000 sec).

5. Mission studies further demonstrated the great usefulness of electric propulsion for unmanned and manned space missions. An electric probe, boosted by a Saturn IB carrier (2-stage), could carry out a large number of desirable unmanned missions in planetary space, many of them not feasible with chemical or nuclear propulsion.

6. Development of a solar-electric power supply on the order of 30 kwe, and of nuclear-electric supplies of 300 to 1000 kwe and of 4 to 10 MWe, is of great urgency. Systems to convert electric power efficiently should be developed with equal priority. Electric space power will be needed independently for electric propulsion, for unmanned and manned spacecraft, and for a lunar base.

7. Electric propulsion development should not wait until a specific "requirement" arises. Electric propulsion should be developed vigorously as a new technology which will extend our space exploration capability decisively beyond the capabilities of chemical and nuclear systems.

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TABLE I

PAYLOAD FOR SPACE PROBE MISSIONS (kg)

(From data of ref 26)

Mission	Payload kg			
	Electric propulsion 11 000 kg initial mass		Nuclear rocket 35 000 kg initial mass "advanced reactor"	Chemical rocket 140 000 kg initial mass H_2-O_2 propellant
	α =0 16 kw kg^{-1}	0 08 kw kg^{-1}		
Mars Orbiter (800 km orbit)	5 200 (250 days)	3 400 (250 days)	8 300 (230 days)	13 500 (230 days)
Jupiter Flyby	5 500 (500 days)	3 700 (500 days)	6,800 (500 days)	17 000 (700 days)
30 Out-of-Ecliptic	3 900 (232 days)	2 000 (232 days)	1 800 (232 days)	No mission
Solar Probe (0 094 A U)	4,100 (200 days)	2 100 (200 days)	No mission	No mission
Pluto Flyby	2 800 (1100 days)	900 (1100 days)	No mission	No mission
Saturn Orbiter (3200 km orbit)	2 300 (1000 days)	500 (1000 days)	No mission	No mission
Jupiter Orbiter (3200 km orbit)	1,800 (900 days)	135 (900 days)	No mission	No mission

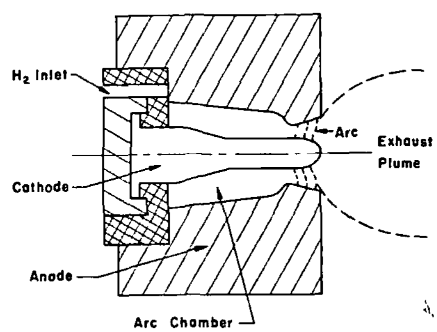

Fig I Thermo-Ionic Accelerator (Giannini [4])

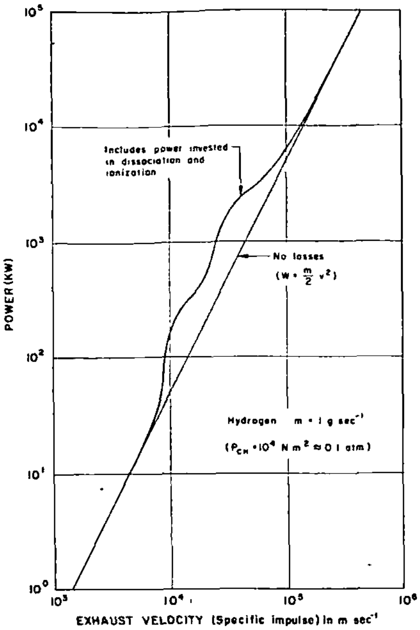

Fig 2 Power vs Exhaust Velocity For Hydrogen Flow of 1 g sec^{-1} [4]

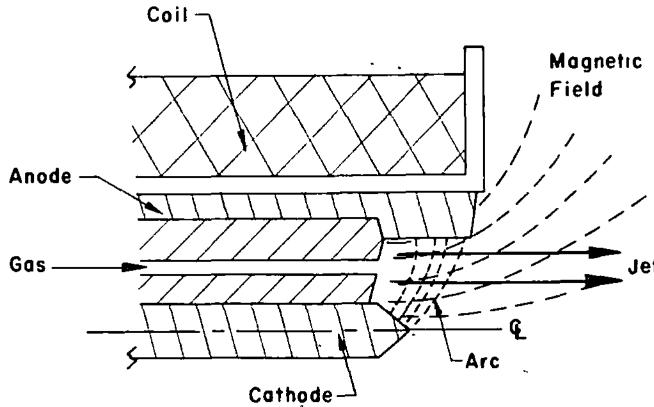

Fig 3 Schematic Diagram of Annular Hall Current Accelerator (EOS [8])

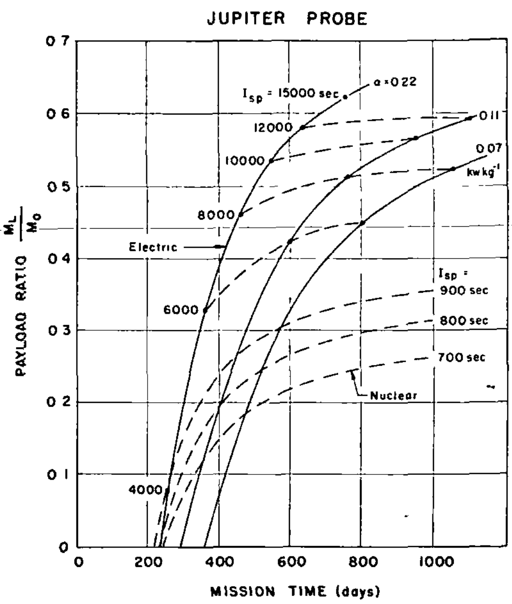

Fig 4 Payload Ratio of Jupiter Probe Vehicles as a Function of Mission Time [12]

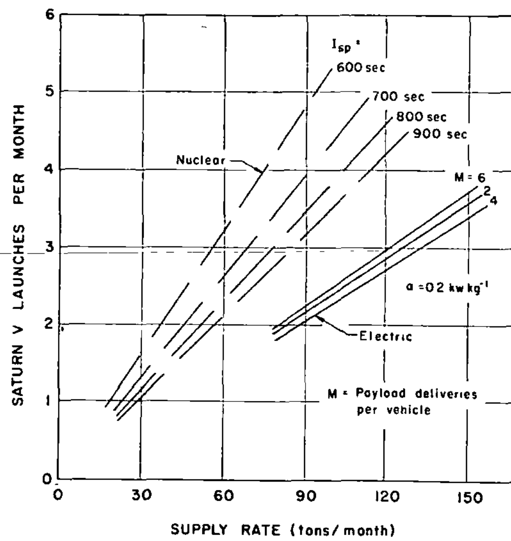

Fig 5 Rate of Saturn V Launches as a Function of Desired Supply Rate of Lunar Station [12]

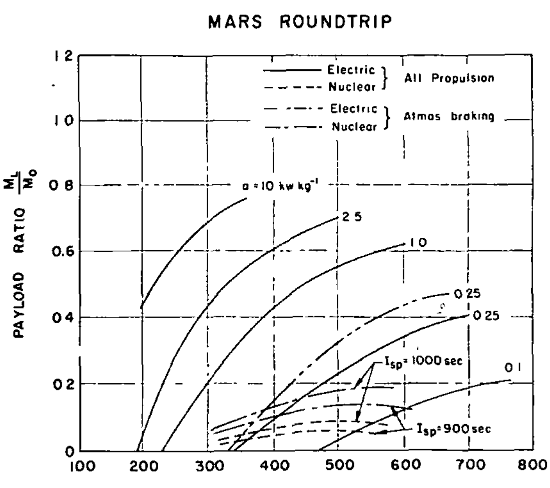

Fig 6 Payload Ratio of Mars Round-Trip Vehicle as a Function of Total Round Trip Time [10]

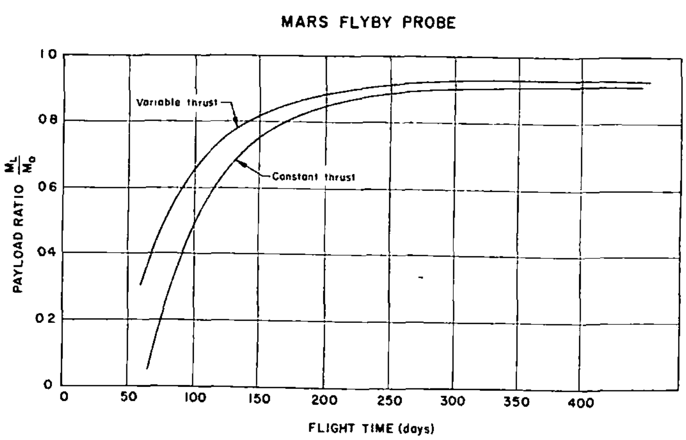

Fig 7 Payload Ratios vs Travel Time of Constant-Thrust and Variable-Thrust Systems [15]