GPO PRICE $ ______

CFSTI PRICE(S) $ ______

Hard copy (HC) 2.00

Microfiche (MF) .50

ff 653 July 65

A CRITICAL DISCUSSION ON THE PLANNING CONSIDERATIONS OF ELECTRIC PROPULSION FOR SPACE MISSIONS

Prepared for the AIAA Third Biennial

Aerospace Power Systems Conference

Sept. 1-4 - Philadelphia, Pennsylvania

FACILITY FORM 602

N66-15320 (ACCESSION NUMBER)	(THRU)
32 (PAGES)	1 (CODE)
CR 60750 (NASA CR OR TMX OR AD NUMBER)	28 (CATEGORY)

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ABSTRACT

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A discussion of a paper by B. Pinkel[1] was requested and is given here. It is to be noted that consideration of electric propulsion development cannot be taken out of the context of the development of the entire field of advanced propulsion. This discussion therefore provides a comparison of the present major technology programs in advanced propulsion on the basis both of technological state-of-the-art and of missions capability. Some additional features concerning the economic problems of deep space exploration are provided which may seriously limit the application of any form of advanced propulsion to unmanned exploration. For manned planetary exploration, it is entirely possible that a combined system utilizing both nuclear and electric propulsion systems is more reasonable than either propulsion system alone.

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A CRITICAL DISCUSSION ON THE PLANNING CONSIDERATIONS OF ELECTRIC PROPULSION FOR SPACE MISSIONS

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I. INTRODUCTION

The conclusions in the paper by B. Pinkel[1] deal with very large and advanced space missions using multimegawatt electric-propulsion systems launched by Saturn V or larger boosters. Large programs of this nature are quite a distance into the future and certainly require additional technology development before they become realistic. The needs of our NASA planetary space program have not yet progressed to the point where we are pressing for "early initiation of a full-scale development effort on a large space propulsion system." This also includes chemical and nuclear rockets as well as electric propulsion.

On the other hand, we are becoming increasingly aware of a rapidly growing technology in smaller electric propulsion systems with performance potential far beyond the current nuclear rocket technology. We are equally aware that the next advancement of nuclear-electric propulsion technology promises to be far superior to the next advancement of nuclear rocket technology. Dr. Pinkel's comparison of current electric propulsion technology with the advanced nuclear rocket technology may be somewhat misleading in that the nuclear rocket systems specified are not being developed under the ROVER/NERVA program but require an advanced development program. This places nuclear rocket development on the same time and cost basis as electric propulsion development. Any comparison of these systems, therefore, must rest on a comparison of technology and of mission performance. It then becomes evident that the development of electric propulsion is at least as important as the development of nuclear rockets.

II. NUCLEAR ROCKET TECHNOLOGY

The main nuclear rocket technology programs in existence today are summarized in Table I. In addition to these, research has been started on such systems as liquid-core reactors, fusion reactors, and nuclear-impulse rockets (Orion).

The ROVER/NERVA program is based on a graphite-core reactor using hydrogen temperatures of at least 2600°F. Total system weight, including rocket motor, shielding, feed system, structure, etc., will probably lie between 20,000 and 30,000 lb. Design goals require a specific impulse of at least 700 sec at a power level in excess of 1000MW_H. According to reports, flight tests, under a continuing program, are not anticipated before the early to mid-1970's (baring any further major development problems).

In addition to the ROVER program, thought is now being given to the development of a fast-spectrum, metal-ceramic reactor which will provide a specific impulse up to 830 sec. As may be recognized, such a system will require a considerable extension of the technological state of the art. Hydrogen temperature to achieve the specific impulse may be as high as 4500°F. For large thrust-to-weight ratios necessary to nuclear-rocket missions, propellant flow rates per unit cross section of the core must also be large. This implies large heat fluxes which, in turn, will probably necessitate fuel temperatures significantly higher than the hydrogen temperature. Since UO_2, the most refractory of reactor fuels, has a melting temperature of about 5000°F, considerable problem may be encountered in the development of light weight cores. Some improvement may be achieved by UO_2-ceramic fuel, but this also decreases fuel density causing core size to grow. Other undesirable features of UO_2 are poor heat-transfer characteristics and high vaporization rates.

With the high-temperature hydrogen, greater dependence will undoubtedly be placed on tungsten technology which shows a melting temperature of $6100^{\circ}F$. Fabrication of large tungsten components has yet to be achieved, and the problems of integrating these components into propulsion systems cannot be adequately estimated at this time.

The next step in achieving higher propellant temperatures appears to be the theoretically less-temperature-limited, gaseous-core reactor. Ultimate potential increase of specific impulse may be a factor of 2 or 3 over solid-fuel reactor systems. These very highly advanced systems are, however, faced with problems of nuclear containment and criticality, and radiant heat transfer. The hydrogen propellant in a gas-core reactor is nearly radiation transparent at temperatures between 3000K and 10,000K (4940F and 17,540F). Since this range is exactly the proposed range of operation of the gaseous-core reactor, some solution to the problem of absorptivity in hydrogen is necessary to the achievement of high flow rates, and hence high thrust.

At present there is a small study effort on fluid physics, but the work is not heavily funded. Qualitative feasibility studies generally involve non-thermal, non-nuclear experiments. At best, such systems might possibly become available within the next 20 to 30 years.

It appears, therefore, that until major advances are made in the technological state of the art, nuclear rocket systems are limited to specific impulses below about 800 seconds. Even at this specific impulse a high thrust per unit weight level is not yet assured. If and when a fast metal-ceramic reactor system is developed, system weight might be reduced, but temperature limitations will still predominate.

III. NUCLEAR-ELECTRIC PROPULSION TECHNOLOGY

Major electric propulsion technology programs suitable for deep space exploration are summarized in Table II. The current program provides for a SNAP-50 powerplant development by the AEC and an ion motor development exemplified by the recent SERT-1 flight test. Electric rocket systems have now been flown twice, and the latest test was successful. In ground test, we may note that the combined total impulse produced by all electric thrustor tests to date still exceeds the total impulse produced by nuclear rocket tests. Ion-motor reliability is now approaching 1000 hours.

The SNAP-8 reactor has passed its first 1000 hours of full-power operation. Lithium-columbium component technology at the AEC/CANEL facility has been proven to 10,000 hours at 2000^{o}F or over. Preliminary SNAP-50 reactor fuel studies are approaching their required test times, although it may be two more years before final fuel selection is made. The present SNAP-50 reactor design work is now nearing completion of its first year.

The main difference between the nuclear rocket and electric propulsion technologies is programmatic. There is a growing need to coordinate and integrate electric propulsion with its power source, a feature which already exists in nuclear rocket development. In all respects the current electric propulsion program is fully as active as the nuclear rocket program, and, if properly coordinated, can commence prototype subsystem flight testing in the early 1970's.

Advanced electric propulsion systems as shown in Table II should be able to take advantage of the remarkable strides forward evidenced recently by in-pile thermionic reactor studies. Thermionic converter lifetimes are rapidly approaching 2000 hours at temperature and under nuclear irradiation. Reactor design work has now been initiated and a healthy and competitive atmosphere exists.

Even aside from the reactor technology, many problem areas still exist, but the major materials problems are being met and the temperature levels needed for advanced systems are still low enough to eliminate requirements for further technological breakthrough. In this respect, it begins to appear that the advanced electric-propulsion systems may be well ahead of the advanced nuclear rocket systems.

There is no doubt that the long lifetime requirements of nuclear-electric systems represents a new dimension in reliability. But we are faced with this requirement already in deep space exploration. The Mariner spacecraft being launched toward Mars this year will require 5000 hours of operation for a successful mission. Nuclear rocket systems, though not operating during missions with extended flight times, must operate after extended storage in a hostile environment. Such procedure is often as difficult a reliability problem as constant operation throughout the mission.

The advanced thrustor systems for electric propulsion should see the development of useful MHD thrustors to augment the capabilities of advanced ion propulsion systems. The high-impulse arcjet, receiving excited attention at the Electric Propulsion Conference this week, appears to be one of the contenders in this field.

The next step in nuclear-electric system development appears to be a further advancement of thermionic reactor technology. Alternatively, it is possible an appropriate breakthrough may occur in MHD power generation. In addition, novel improvements may be anticipated in all aspects of the field of nuclear-electric power generation and electric propulsion. This includes improvements in nuclear shielding and radiators for higher temperatures, electric thrustors and their power conditioning and control, propellant feed systems, structures, and instrumentation. As Dr. Pinkel has stated, we want to be in a position to take fullest possible advantage of these future improvements.

IV. MISSIONS COMPARISONS

The summary comparison of technologies in the previous sections indicates that, column for column, Tables I and II are on an equivalent development basis. In this section we will attempt to partially summarize the mission capabilities for these systems. The discussion is limited to the current systems and advanced systems, however, because column 3 is so nebulous as to leave extreme doubt as to its validity.

Three missions only will be considered here: planetary orbits about Mars, Jupiter, and Mercury. These are selected because they are the most favorable for nuclear rockets on a flight time basis and also because they were the main comparisons in the paper under discussion here. Planetary flyby probes can all be accomplished by chemical systems with reasonable payloads, except that flight times to most of the outer planets could become excessive (on the order of 5 years or longer). Spacecraft probes close in to the sun and high out of the ecliptic plane can be provided by modest electric propulsion systems[(2)], but extremely large boosters and multiple upper stages are needed for chemical or nuclear rocket systems. Some thought is being given to the use of Jupiter to provide a cometary trajectory for such flights, but this is extremely complicated and requires extended flight times.

Figure 1[3] of this paper is a plot of required velocity increment, ΔV, from an initial Earth orbit to the planets of interest. These curves are for planetary impulsive transfer missions typified by chemical and nuclear rockets. Dates shown are for best encounter over the next several years, and do not necessarily represent proposed flights. For each mission, the Hohmann transfer requirements are represented by the lowest velocity increment and the longest flight time. As flight time is decreased, the required velocity increment increases.

Since electric propulsion missions are not accomplished by impulsive transfer, a different set of criteria apply. An equivalent velocity increment may be selected, but it is much higher for low-thrust trajectories than for impulsive transfer trajectories. A more convenient format for electric propulsion mission is illustrated in Fig. 2, where mission dependence on specific power level is clearly indicated. These curves show the terminal mass delivery capabilities at the destination planet for a small range of power levels and flight times. Notice the great increase of delivered mass for a small increase of flight time. Specific power level, P_o^*, is defined as watts of electrical power delivered to the propulsion system per kilogram of spacecraft initial mass in Earth orbit. Thus, we have normalized the curves to make them useful for any booster we may wish to select.

Only the power level has been defined in these curves. Terminal mass includes the entire weight of spacecraft delivered at the destination, including the powerplant, propulsion, structure, tankage, etc. A somewhat comparable terminal-mass curve for chemical and nuclear rocket spacecraft is shown in Fig. 3. This is a curve of spacecraft weight as a function of the velocity increment, ΔV, defined previously in Fig. 1. The electric propulsion curve plotted here shows the terminal mass fraction that can be delivered to the same planetary destination; but as was explained, this requires a higher equivalent ΔV. The curves for the chemical rockets assume a structural factor of 0.85, and thus represent actual payload fractions. The two-stage system assumes that the first stage is used for heliocentric injection from low Earth orbit, and that the second stage is fired at the destination planet to achieve planetary orbit. For Mars and Venus orbital missions, the chemical systems can yield payload fractions of 0.1 and 0.2. For the major planet probes, the two-stage chemical system can deliver payload fractions of 0.05 and 0.1. Payload fractions for the major planet orbital missions are extremely marginal.

The performance capabilities of the nuclear-rocket systems with I_s of 700 and 800 sec are also shown in Fig. 3. Tankage (including propellant reserves and residuals) was optimistically estimated at 10 per cent of the propellant weight, since cryogenic hydrogen is required. But the weight of the nuclear-rocket system has not yet been subtracted from the terminal mass. The $1\frac{1}{2}$-stage system drops empty propellant tanks after heliocentric injection from low Earth orbit, but the nuclear rocket is reused at the destination planet to provide the second velocity increment needed to go into planetary orbit. These curves also assume a true impulsive transfer. If thrust acceleration for nuclear rockets is less than approximately 0.5 g, additional losses would be involved. For Mars and Venus orbital missions, the nuclear rocket may deliver a terminal mass fraction of 0.40 to 0.45; a terminal mass fraction of 0.2 to 0.3 may be delivered at a major planet flyby. For major planet orbital missions, the nuclear rocket may deliver a terminal mass fraction of 0.1.

After the weight of the rocket system has been subtracted from the terminal mass, even the nuclear system appears marginal for the major planet orbital missions. A Mercury orbiter is beyond the capability of current nuclear rocket technology if standard orbital transfers are considered.

A graphical display of terminal mass delivered to the planets, however, is not a true representation of the payload capabilities of the three systems shown. The propulsion system of the nuclear-rocket vehicle is included in the terminal mass but is no longer useful; and the nuclear-electric supply of the nuclear-electric propulsion system, though operating, must also be charged mainly against the propulsion system.

Figures 4, 5, and 6 are the result of combining the information in Tables I and II with that of Figures 1, 2, and 3. Figure 4 illustrates the Mars orbiter mission payloads at about 1.5 radii. For nuclear rocket missions, the lowest curve represents a graphite reactor at 700 sec specific impulse, while the upper curve represents the best refractory-metal reactor at 830 sec specific impulse. The electric rocket missions are plotted for a range of specific weights for the propulsion system, assuming that the propulsion weight is 25% of the initial spacecraft weight. It is interesting to note that the electric propulsion curves essentially form a natural extension of the nuclear rocket curves for any given level of technology.

Figure 5 is a similar comparison of a Jupiter orbit mission at approximately 15 planetary radii. For this mission, flight time superiority, as well as payload superiority, becomes evident for electric propulsion systems. This trend toward ever-increasing flight-time difference continues out to Pluto, and is graphically illustrated in Fig. 7.

Figure 6 illustrates the Mercury orbit mission at 1.8 planetary radii. For this mission, the nuclear rocket systems show a marginal capability, although flight times are inherently short. As pointed out in Ref. (3), use of a SNAP-50 system on a Saturn I-B launch vehicle would provide a competitive payload to nuclear rockets on a Saturn V but not, of course, at the same flight time.

The electric propulsion missions curves of this discussion are more conservative than those of Reference (1). Dr. Pinkel has assumed an optimum variable-thrust propulsion system, initially proposed by Irving[4]. Further work by Melbourne[5], at JPL, Zimmerman[6], at NASA-Lewis, and others have taken into account the practical characteristics of electric-propulsion thrustors. The mission curves discussed here utilize the constant-thrust, optimum-coast trajectories, thus incurring some loss of payload at comparable flight times.

In spite of this conservatism, however, it becomes apparent that electric propulsion spacecraft systems are not only competitive on a technology basis but are considerably superior on the basis of mission comparison.

One other point should be considered here before we go on to discuss briefly the economic aspects of space exploration. It is a basic oversimplification to consider that electric propulsion is competing with nuclear-rocket propulsion. Each has merit in its own right for different applications. Neither is a panacea for space exploration.

As an example of this feature, the manned Mars mission has been shown by Dr. Pinkel as a competitive mission. On the other hand, at the First Annual AIAA meeting, MacKay[7] presented a mission analysis which shows a combined system far superior to either system alone. His curves are reproduced here as Figures 8 and 9. Doubtless, there are other planetary missions which can take advantage of a similar combination.

Economics of Space Exploration

This nation has embarked on a scientific program of space exploration which must, of necessity, be limited in total available national resources. The return on these expenditures is quite varied and includes such intangibles as scientific data, national prestige, support of the economy, personal and organizational publicity, political opportunity, etc. The present distribution of available resources attempts to maximize the total returns on our investment. Any relationship of these returns to a cost per pound of payload would be difficult to determine. Taking a page out of the JPL book, 4000 pictures from a 750-lb Ranger spacecraft were considered an excellent return on a $260 million investment. This amounts to $350,000 per pound of payload. The cost per picture averages to $6500.

Because of the present distribution of resources, the total NASA FY 1964 budget for lunar and planetary exploration was $271 million. Such a budget currently precludes multi-billion dollar development programs. In fact, it is not presently reasonable to consider Saturn I-B launch vehicles, let alone Saturn V. The largest launch vehicle currently programmed for unmanned planetary exploration is the Atlas-Centaur with an ultimate Earth-orbit payload on the order of 10,000 lb.

Another economic factor of significance in unmanned planetary exploration is the payload cost. Exclusive of development, a spacecraft currently costs close to $30,000 per pound. Total spacecraft costs are greater than the launch vehicles on which they ride. Their sole purpose is to proceed to a destination and return vital, new scientific data to Earth. Under the present economic conditions, large payloads ultimately offered by the Saturn V are only a pleasant dream.

Unmanned space exploration planning has recently been expanded to introduce the Saturn I-B if it can be done at modest cost. It is in this context that electric propulsion presently gains its greatest impetus. If nuclear-electric propulsion technology can be proven at an early date, and if it could economically provide a few hundred pounds of scientific instrumentation at the far reaches of our solar system, it would be one of the most useful and versatile tools yet proposed for space exploration. Even then it would require a sizeable expansion of resources for planetary exploration. As may be seen from Fig. 10, nuclear rocket systems lack any potential capability for this application.

Acknowledgement

This paper was prepared upon the strong recommendation of the AIAA Technical Committee for Electric Propulsion, whose support has made it possible for presentation at this time. I also wish to thank E. Stuhlinger, W. E. Moeckel, T. N. Edelbaum, R. S. H. Toms, T. F. Widmer, and Mrs. E. W. Speiser for their cooperation and timely comments in the preparation of this discussion.

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Table I

Nuclear Rocket Technology

	Program		
	Current	Advanced	Research
Specific Impulse	$\gg$ 700 sec	$\ll$ 830 sec	$>$ 2000 sec
Hydrogen Temp.	$\gg 2600^{\circ}F$	$\ll 4500^{\circ}F$	$> 15{,}000^{\circ}F$
Reactor Weight	15,000 lb	5000-10,000 lb	?
System Weight	20,000-30,000 lb	10,000-20,000 lb	?
Reactor Fuel (assumed)	UC-C	UO_2 - Ceramic	?
Fuel Melting Temp.	$\sim 4800^{\circ}F$	$\sim 5050^{\circ}F$	N.A.
Program Status	Funded	R&D	Basic Research
Proposed Booster	Saturn V	Saturn V & Over	Saturn V & Over

Table II

Nuclear-Electric Propulsion Technology

	Program		
	Current	Advanced	Research
A. Power System			
Power Level	300-1000 Kwe	1-5 Mwe	5-40 Mwe
Specific Weight*	15-30 lb/Kwe	10-20 lb/Kwe	5-10 lb/Kwe
Type	Rankine	Thermionic	Thermionic
Reactor Temp.	2200°F	3200°F	3800°F
Reactor Fuel	UC	UC + ?	?
Fuel Containment	Cb-1%Zr	Tungsten	?
Lifetime	10,000-20,000 Hr	20,000-30,000 hr	30,000-50,000 hr
Status	Funded	R&D	R&D
B. Propulsion System (less power)			
Specific Impulse	5,000-15,000 sec	4,000-20,000 sec	3,000-50,000 sec
Specific Weight	10-20 lb/Kwe	5-15 lb/Kwe	5-10 lb/Kwe
Type	Ion	Ion/MHD	MHD/Ion
Lifetime	10,000-20,000 hr	20,000-30,000 hr	30,000-50,000 hr
Status	Partially Funded	R&D	R&D
Proposed Booster	Saturn I-Saturn V	Saturn V	Saturn V & Over

*Includes shield and power conditioning

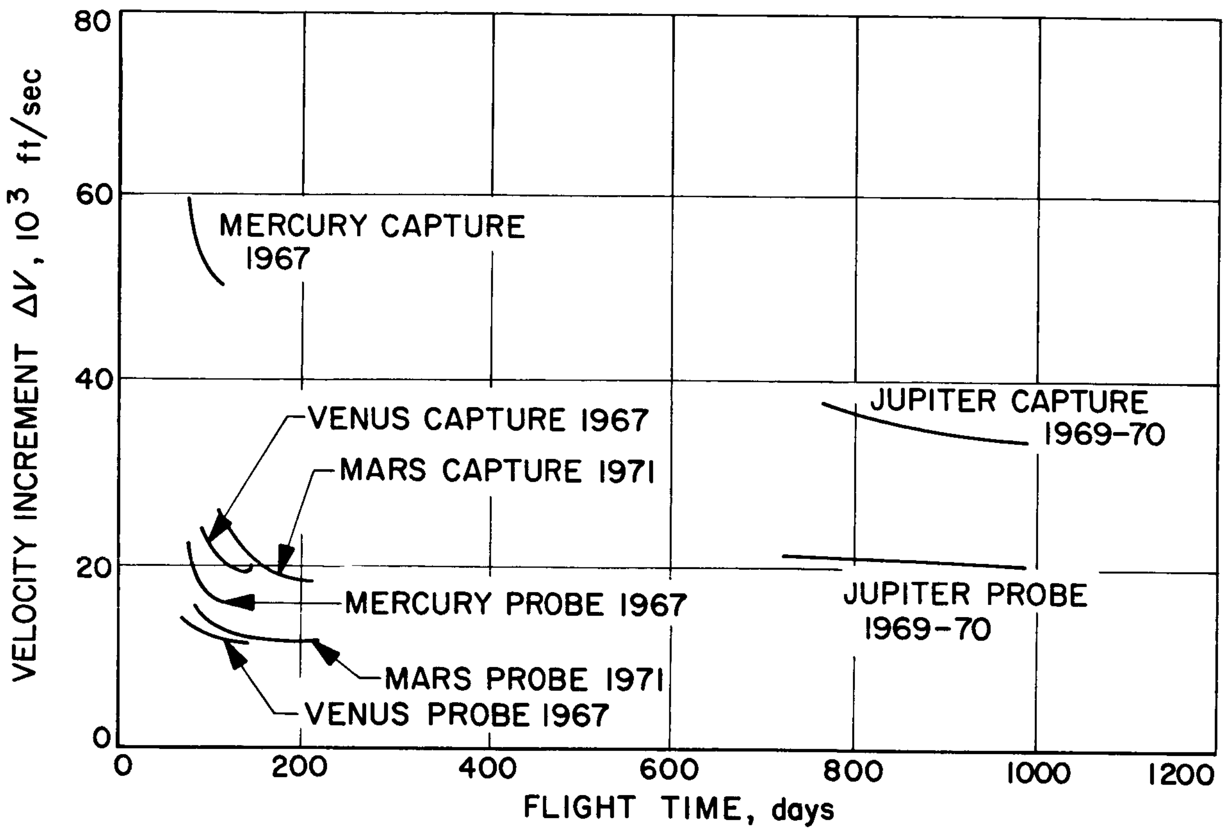

Fig. 1. Velocity requirements for planetary impulsive-transfer mission (best encounter)

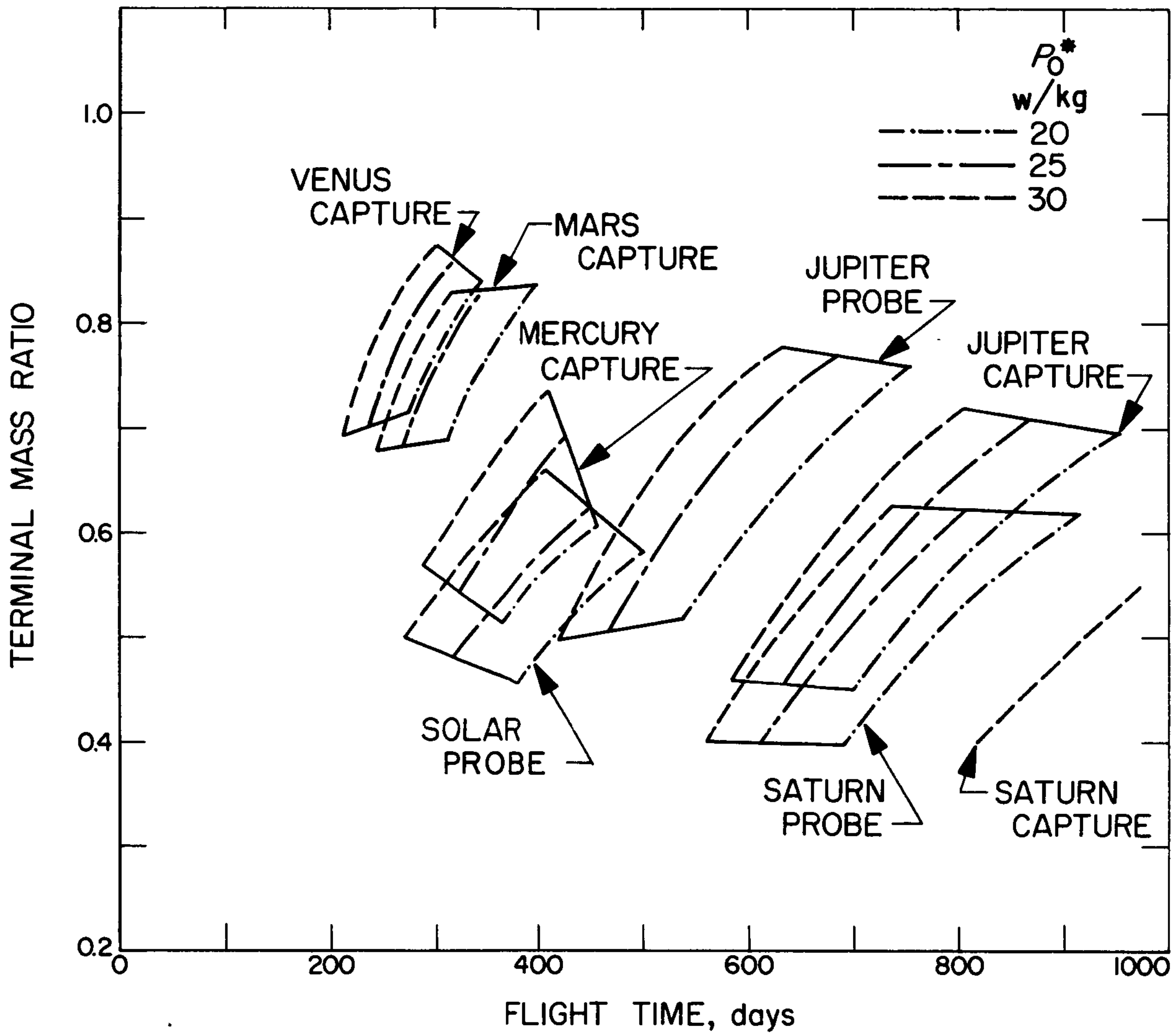

Fig. 2. Performance capabilities of electrically propelled spacecraft

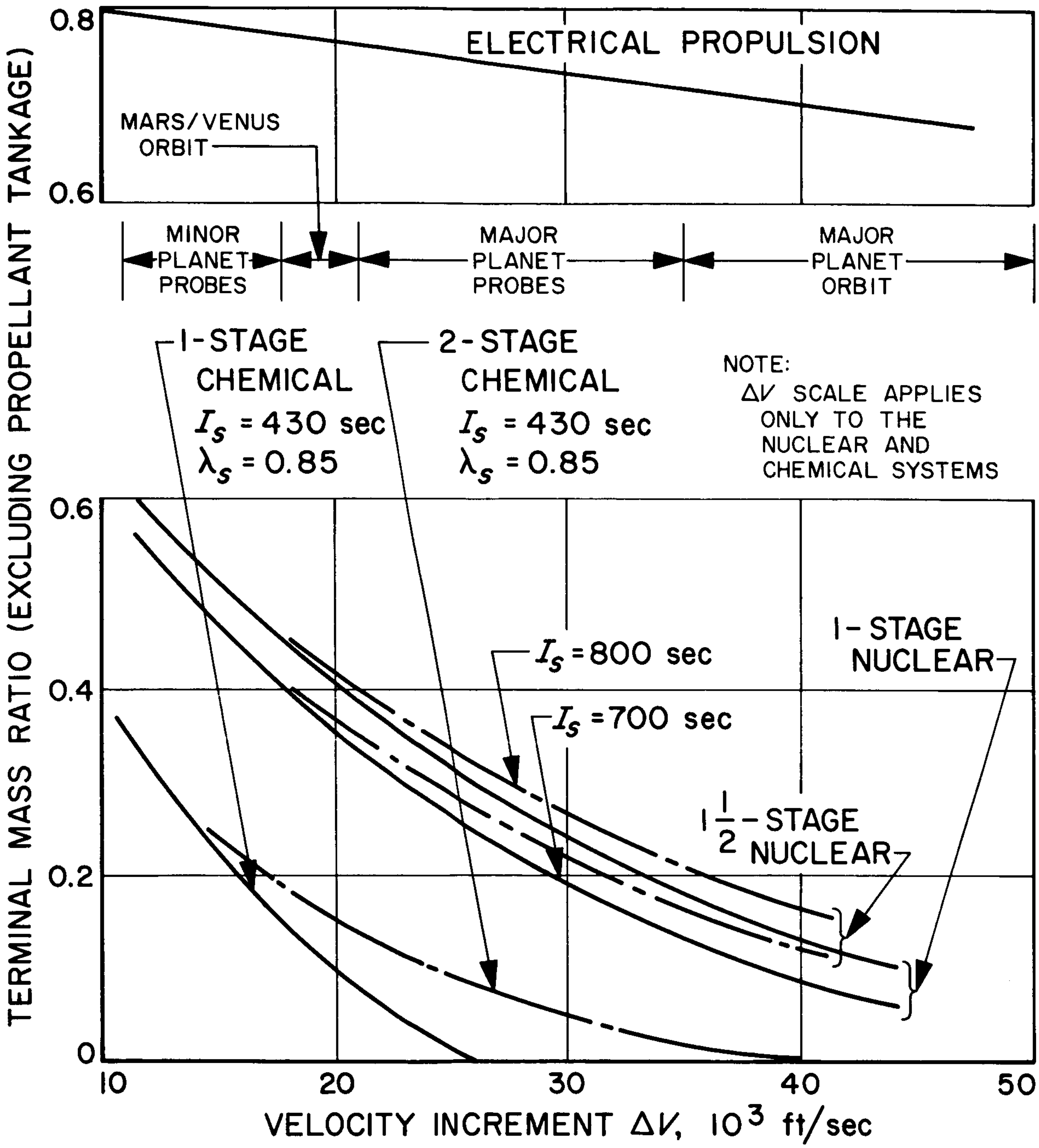

Fig. 3. Terminal mass fractions for planetary missions from Earth orbit

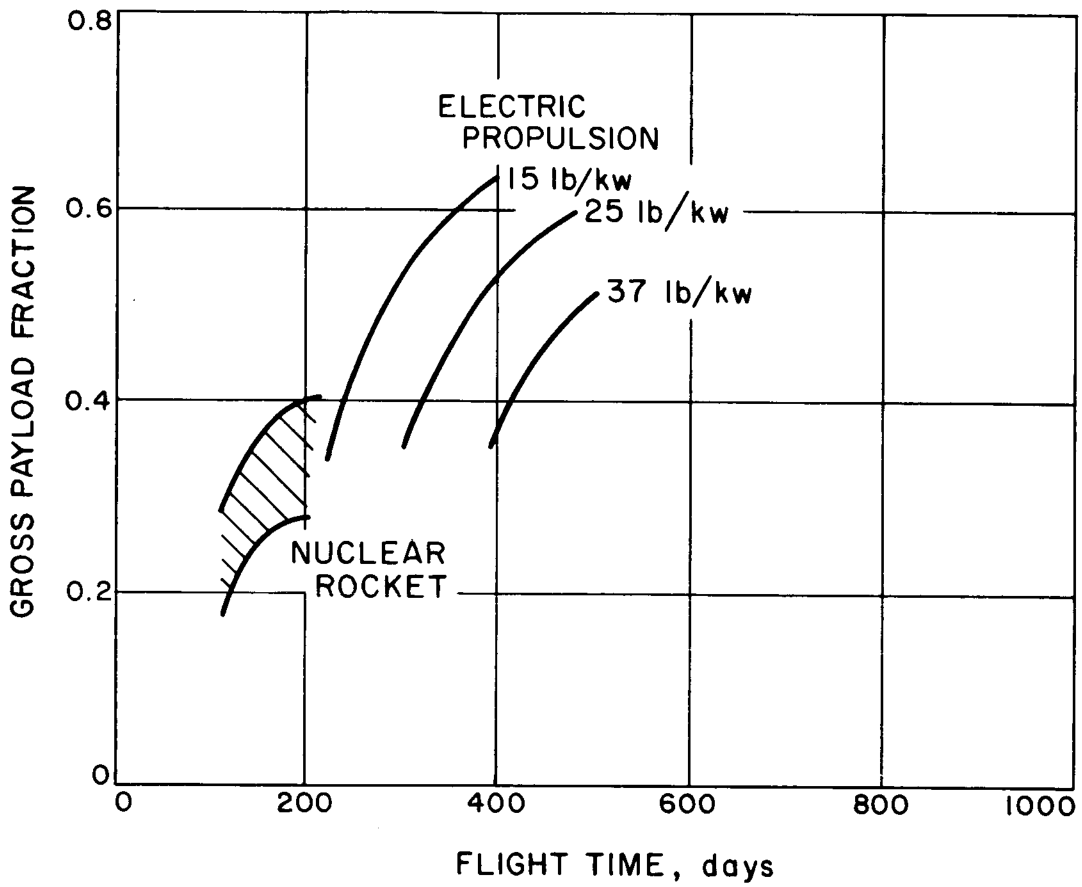

Fig. 4. Mission payload comparisons, Mars orbiter

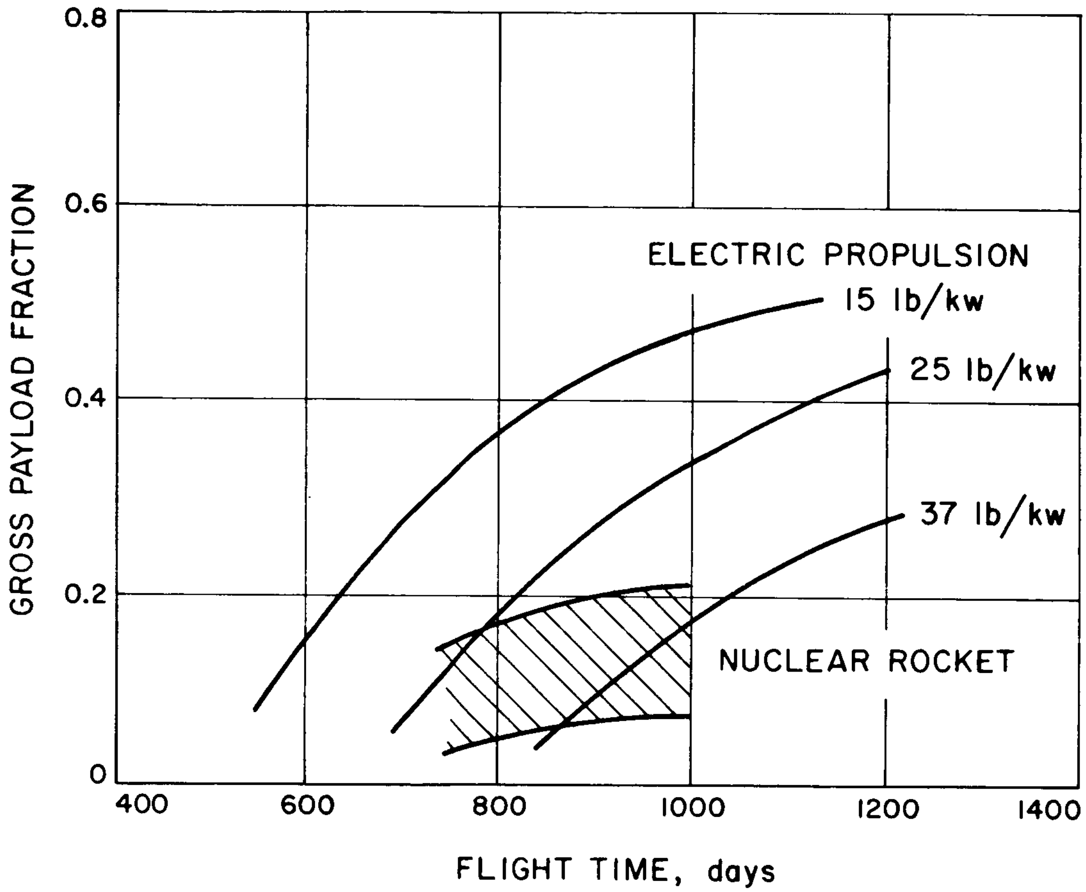

Fig. 5. Mission payload comparisons, Jupiter orbiter

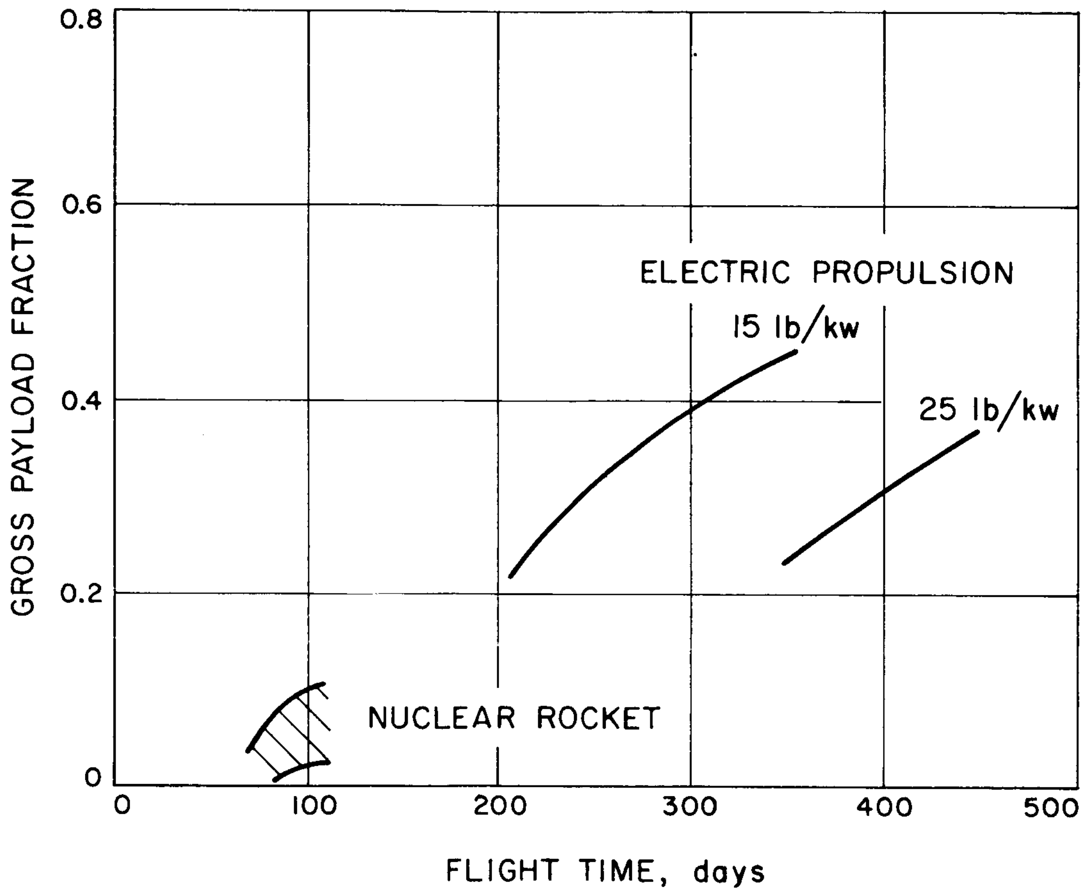

Fig. 6. Mission payload comparisons, Mercury orbiter

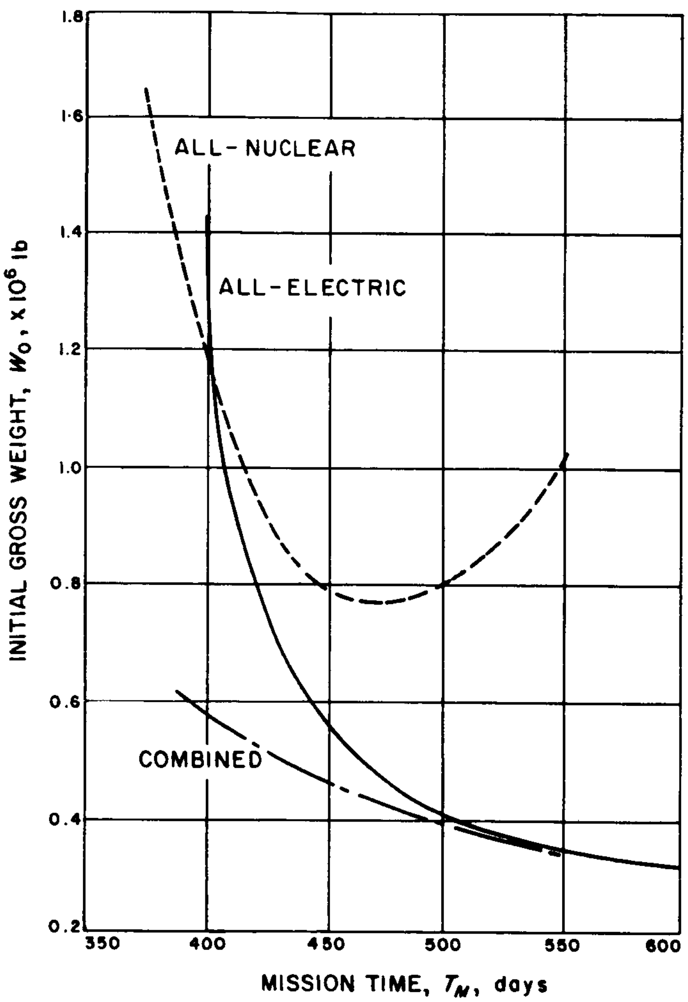

Fig. 7. Weight comparisons, manned Mars missions (Ref. 7)

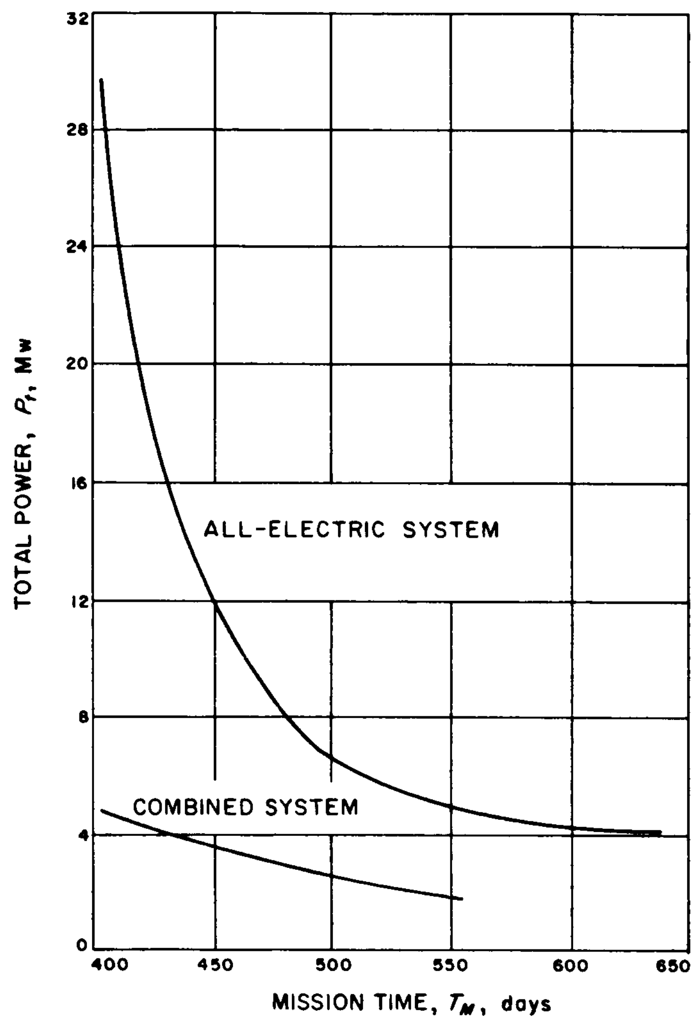

Fig. 8. Power comparisons, manned Mars missions (Ref. 7)

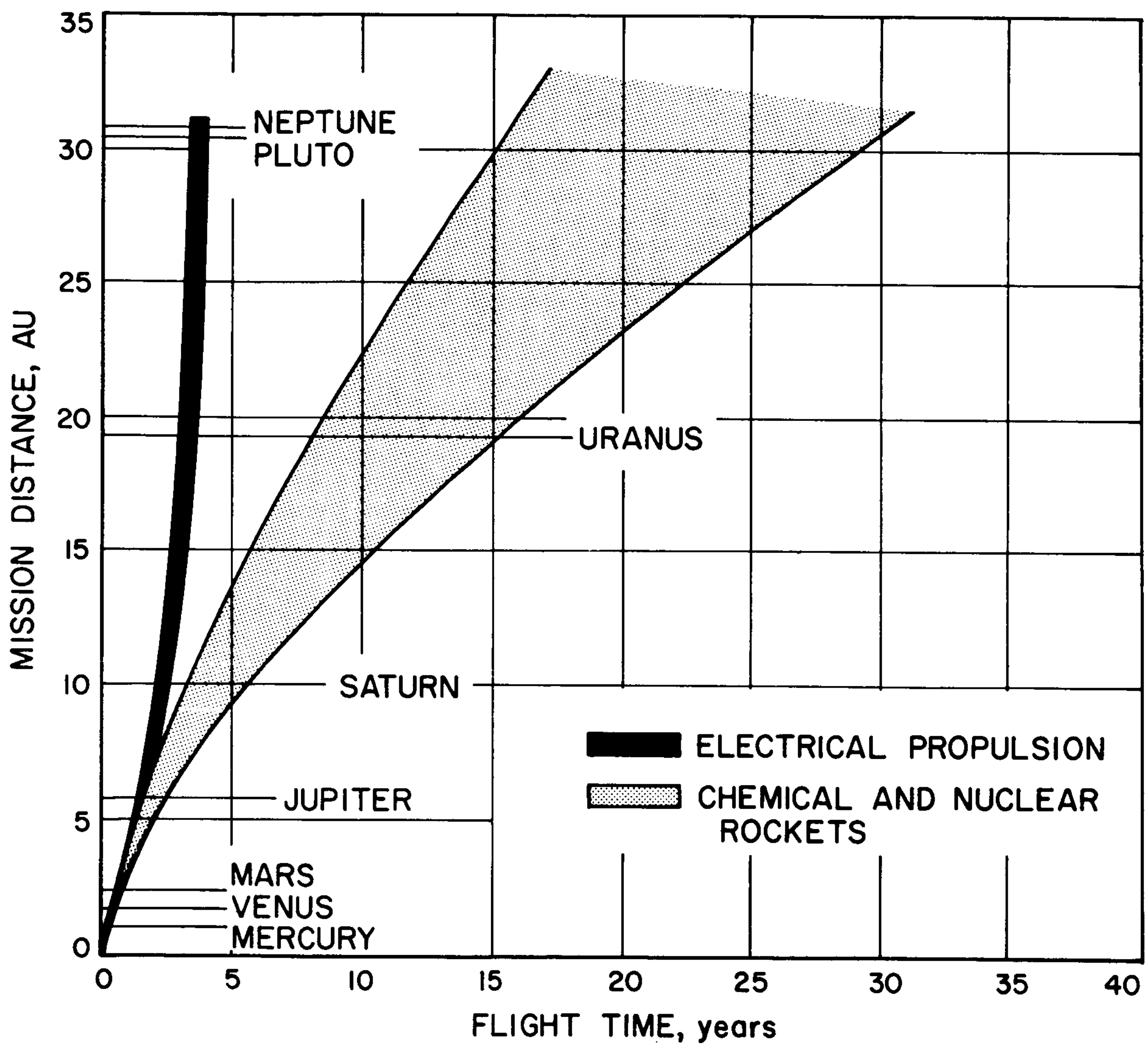

Fig. 9. Flight time requirements for planetary missions

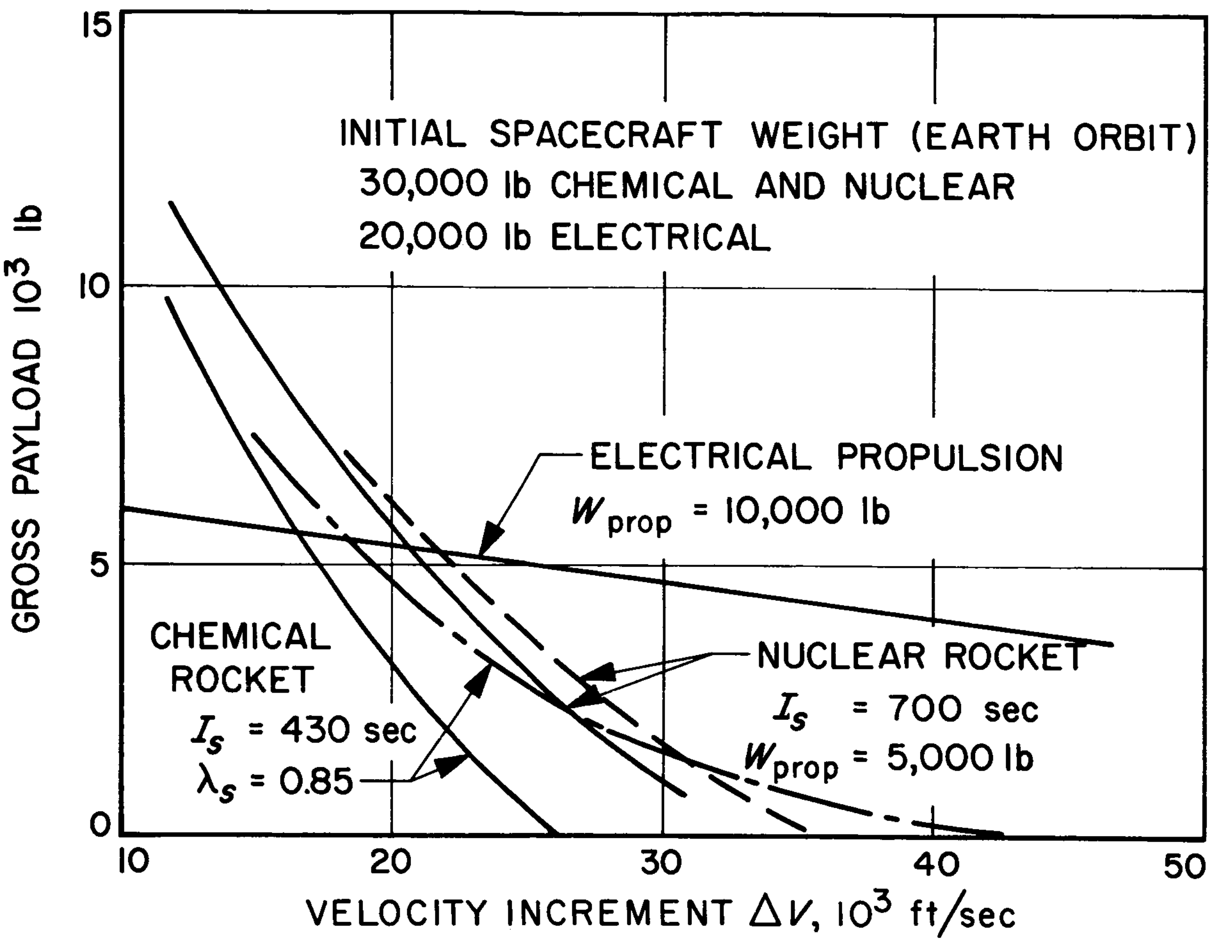

Fig. 10. Payload capabilities for chemical, nuclear, and electric systems with Saturn I-B launch