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Applications of Snap-50 Class Powerplants to Selected Unmanned Electric Propulsion Missions

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The mission applications include: solar, Jupiter, and Saturn probes, a Venus satellite, and a continuous lunar logistic supply operation. General results of payload fraction vs mission time for the interplanetary missions are presented in plots containing contours of constant thruster efficiency and specific impulse. These plots may be used with good accuracy to evaluate any thruster with a known variation of efficiency with specific impulse. Performance comparisons are made between electric and solid-core nuclear rocket propulsion for all of the missions. A general conclusion of the study is that a powerplant specific weight of less than 30 lb/kwe must be achieved in order for electric propulsion to compete with nuclear rockets in the 700 to 900 sec range of specific impulse. It is also shown that electric propulsion devices having a specific impulse of less than 3000 sec are not competitive with either nuclear rockets or ion engines even if they can achieve high efficiency. The results for the lunar supply operation show that electric propulsion is superior to nuclear rockets on a performance basis and that the two systems are competitive on a specific cost basis.

Introduction

If a program of unmanned space missions includes missions that differ greatly in degree of difficulty, no one type of propulsion can be expected to be clearly superior for all of them. Since it may be impractical and too expensive to develop specific advanced propulsion vehicles for each mission, the final choice may be a compromise system capable of performing each of the missions well but not necessarily better than other contenders.

Barring any unforeseen breakthroughs in the possible development of gas-core nuclear rockets or nuclear-pulse rockets, the only propulsion systems that could be available in the 1970's for unmanned exploration of the solar system are chemical rockets, solid-core nuclear rockets, and nuclear-electric rockets. For unmanned scientific interplanetary missions, such as probes to the major planets and the sun, chemical rockets are at best only marginally adequate. Another difficult unmanned mission is a lunar logistic supply operation; although the lunar supply operation can be performed with chemically propelled vehicles, it could certainly be done more efficiently and economically using more advanced propulsion.

The aim of this study was to evaluate the potentials of solid-core nuclear rockets and nuclear-electric propulsion as candidates for a compromise propulsion system for five selected unmanned missions that are representative of the range of difficulty of a whole program of unmanned missions. These are a Venus orbiter, Jupiter, Saturn, and solar probes, and a lunar logistic supply operation.

Some of the more significant previous mission studies of electric propulsion are listed as Refs. 1–12. It is believed that the present paper represents a significant contribution to this class of studies for the following reasons: 1) it presents a complete comparison of the respective payload capabilities of solid-core nuclear rockets and nuclear-electric propulsion for the representative missions covering a range of specific impulse and powerplant specific weights, 2) the electric-propulsion performance results for the interplanetary missions can be applied within limits to any launcher and thruster, and 3) the comparative economic results for the lunar logistic supply operation have not been presented before.

Definitions and Ground Rules

Space Vehicles

Electrically propelled vehicles

The initial gross weight of an electrically propelled vehicle is considered to be composed of the following: powerplant (including power conditioning equipment and shielding), thruster, propellant tanks, propellant, payload, and a structural framework to which these components are attached.

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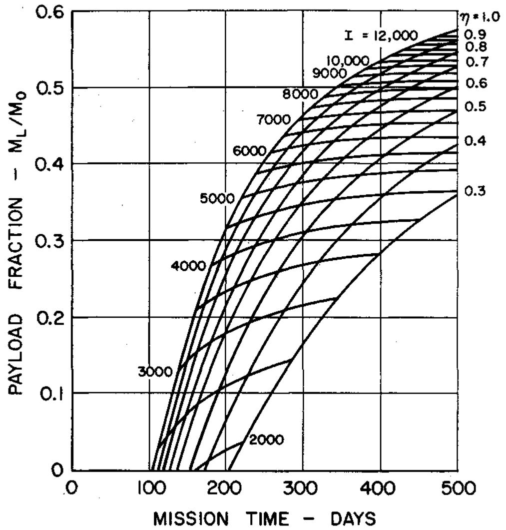

Fig. 1 Venus satellite mission; powerplant specific weight = 20 lb/kwe; powerplant fraction = 0.25.

The weight of the powerplant is treated parametrically in the study and is represented by a specific powerplant weight parameter α defined as the powerplant weight per kilowatt of electric power delivered to the thruster. The weights of power conditioning equipment and shielding are included in the specific powerplant weight. The thruster or thrusters are assumed to weigh 1% of the vehicle gross weight, and the structure frame is assumed to be $3\frac{1}{3}$% of the vehicle gross weight. It is assumed that all of the propellant is usable.

The tankage weight is assumed to be proportional to the ratio of propellant weight to propellant bulk density with a proportionality constant of $\frac{1}{2}$. Rough stress calculations were performed for spherical titanium tanks containing both mercury and cesium with a maximum launch acceleration of 10 g and a safety factor of 1.25. These calculations indicated that the previous assumption would result in conservative tank weight estimates for the range of tank sizes considered.

In addition to ion rockets, liquid-hydrogen-fueled arc-jets were briefly examined. For vehicles powered by arc-jets, the tank weight is assumed to be 0.15 of the propellant weight. It is further assumed that, for continuous thrust operation, the arc-jet would use propellant faster than the boiloff rate so that negligible thermal insulation is necessary.

Solid-core nuclear rocket vehicle

The initial gross weight of the nuclear-rocket vehicle is assumed to comprise the following: nuclear-rocket engine, structure frame, propellant tanks, propellant, insulation, and payload. In the study two types of solid-core nuclear-rocket engines are considered: an advanced tungsten-core engine for the interplanetary missions and a graphite-core engine for the lunar logistic supply operation. The weight estimates for these two rocket engines are minimum weights of 1000 and 4500 lb for the tungsten and graphite engines, respectively, plus 1 lb/Mw of reactor power that corresponds roughly to 1 lb for every 45 lb of thrust at a specific impulse of 800 sec. These weight rules of thumb for nuclear rockets were obtained from Ref. 13. An initial thrust-to-weight ratio of 0.5 is assumed for all of the nuclear-rocket vehicles in the study.

The structure frame is assumed to be $3\frac{1}{3}$% of the vehicle gross weight. The liquid hydrogen tank weights were assumed to be 11% and 15% of the total propellant weight for vehicles launched by the Saturn IB and Saturn V, respectively. The higher proportionate tank weights of the Saturn V launched vehicles reflect the assumed meteoroid protection required in the lunar supply operation. It is further assumed that 2% of the propellant is ullage. The insulation and propellant boiloff weights were assumed to be negligible for the lunar supply operation, but these weights were estimated for the Venus orbiter mission by means of an analysis in an earlier mission study performed at the United Aircraft (UAC) Research Laboratories.[10]

Mission Profiles

Interplanetary missions

Both the low-thrust and high-thrust interplanetary missions are assumed to commence in a circular earth orbit of 300-naut-mile alt into which the vehicle has been launched by a Saturn IB launcher. The low-thrust vehicle follows a constant-thrust spiral escape trajectory from earth and then follows a constant-thrust plus optimum-coast heliocentric planetary rendezvous or flyby trajectory as the case may be. For the interplanetary trajectory calculations, the results of optimum $J = \int a^2 dt$ and powered time vs mission time were obtained from Ref. 14 for constant-thrust acceleration plus optimum-coast operation. The fact that minimum J is insensitive to variations in specific impulse except for very low flight times permits the use of a single J curve (calculated on the basis of infinite specific impulse) for a wide range of values of specific impulse. In the case of the orbiter, a capture maneuver takes place at the destination planet. The maneuver is a constant-thrust quasi-circular spiral inward to the final radius, which is assumed to be at 300-naut-mile alt above the surface.

It is assumed that the high-thrust vehicle undergoes an impulsive change in velocity in the initial circular orbit and then follows a ballistic transfer to the target planet where a second impulse is used to establish a circular orbit at 300-naut-mile alt if necessary. For each value of mission time, the ballistic trajectory having minimum ΔV requirements is used.

The planetary orbits are assumed to be circular and coplanar in both the low-thrust and high-thrust trajectory calculations. This assumption is quite reasonable for the planetary orbits considered in this paper.

Lunar logistic supply operation

This operation is envisioned for the support of a post-Apollo manned lunar exploration effort. It would be a continuous operation supplying discrete payloads to a lunar orbit periodically to satisfy a certain required supply rate (lb/day). To maintain the operation, a fleet of spacecraft would shuttle between a 300-naut-mile circular earth orbit and a 20-naut-mile circular lunar orbit, depositing their payloads in the lunar orbit and returning to earth orbit to pick up new payloads and full propellant tanks for the next round trip. The new payloads, tanks, and propellant would be launched into the earth orbit as required for rendezvous with the spacecraft. New tankage is employed for each round trip, since tankage is required to contain the propellant during the launch from the earth's surface.

The payload packages are landed on the lunar surface by means of single-use chemical rockets employing a storable propellant (N_2O_4/Aerozine with a specific impulse of 312 sec) for payloads delivered by electrically propelled vehicles, and a cryogenic propellant (H_2/O_2 with a specific impulse of 420 sec) for payloads delivered by the nuclear-rocket vehicles. The storable propellant provides performance as good as if not better than the cryogenic propellant for payloads delivered by electric propulsion considering the cryogenic storage problem associated with the 100 to 200-day one-way trip times. On the other hand there is no appreciable cryogenic storage problem associated with the nuclear-rocket operation, since the one-way trip time is of the order of a few days. Although the nuclear rocket does enjoy some performance advantage because of the acceptability of cryogenic high-energy propellants, this advantage is limited considerably by the fact that the ΔV requirement for deorbiting the payload at the moon is small (5500 fps). The storable propellant results in a payload fraction (i.e. payload/weight in lunar orbit) of 0.533, whereas the fraction for liquid hydrogen and oxygen is 0.571. Although the nuclear rocket is capable of landing payloads on the lunar surface directly, this possibility is not considered because of radiation hazards and radioactive contamination problems.

In the analysis of the lunar supply operation, the low-thrust trajectory requirements were obtained from data presented in Ref. 15. For the high-thrust trajectories the method of patched conics was used.

The electrically propelled vehicles engaged in the lunar supply operation would comprise two modules: one containing the propulsion system and a structure framework and the other containing the propellant, tankage, and payload. Each module is launched separately by one or more Saturn V's. The initial gross weight of each vehicle is considered to be the same for every trip. Thus, at the beginning of a particular space vehicle's lifetime the two modules are put into orbit simultaneously. The space vehicle delivers its payload to the lunar orbit and returns to the original earth orbit where it is met by another launch of the payload-propellant module. This operation continues uninterrupted during the useful lifetime of the vehicle. In this study the two values of lifetime assumed are 10,000 and 15,000 hr. Upon reaching the design lifetime, the spent vehicle is left in the lunar satellite orbit. On this last trip no propellant is needed for return, so that relative to the intermediate round trips, a larger payload can be carried.

Unlike the powerplant of the electrically propelled vehicle, the weight of the graphite-core nuclear-rocket engine is a small part of the total vehicle weight. Therefore a slightly different mode of operation is assumed. Initially the whole vehicle is launched by a Saturn V into the standard 300-nautmile circular orbit. For the next lunar trip, another Saturn V would launch the same total weight of payload, propellant, and tankage. However, these elements are launched into an orbit of slightly higher energy because the engine and structure frame are already in orbit. Because of the higher orbit, the space propulsion requirements for the second and succeeding round trips are less than that for the first, and the payloads carried can be somewhat greater. On the final (one-way) trip of the vehicle, a still larger payload can be carried since no propellant would be required for the return trip. Thus, during the life of one vehicle, there would ideally be three different sizes of payload carried.

The performance and specific cost analyses of nuclear propulsion applied to the lunar supply operation indicate that a round trip of 10.5 days gives maximum performance whereas for minimum specific cost the round-trip time should be around 5.2 days. In order to present the best possible results for the nuclear rocket, the optimum value of round-trip time for each criterion was used in the respective performance and specific cost comparisons. It should be stated, however, that the specific cost does not increase appreciably for operation at 10.5 days nor is the performance materially reduced for operation at 5.2 days.

The assumed initial gross weights of the high-thrust and low-thrust vehicles used for comparison in the lunar supply operation are 220,000 and 660,000 lb, respectively. The high-thrust vehicle gross weight corresponds to the Saturn V payload weight in a low-altitude circular earth orbit, whereas the low-thrust vehicle gross weight corresponds to the payloads of three Saturn V launchers. As pointed out previously, it is assumed that one of these would launch the powerplant module, and the other two would launch the payload-propellant module in two elements that would be joined in orbit. As a consequence of the mechanics of the lunar operation, the gross weight of the low-thrust vehicle must be large in order that the value of powerplant fraction be close to optimum. The gross weight of 660,000 lb represents the best of a number of launcher combinations investigated on both a performance and specific cost basis. The required power for this vehicle is about 10 Mwe, which is considerably above the presently anticipated power of the SNAP-50 system (300 kwe to 2 Mwe). Although this power could possibly be supplied by multiple SNAP-50's, in parallel this would present severe packaging problems. Furthermore the cost of such a powerplant would be much greater than that of a single 10 Mwe powerplant.

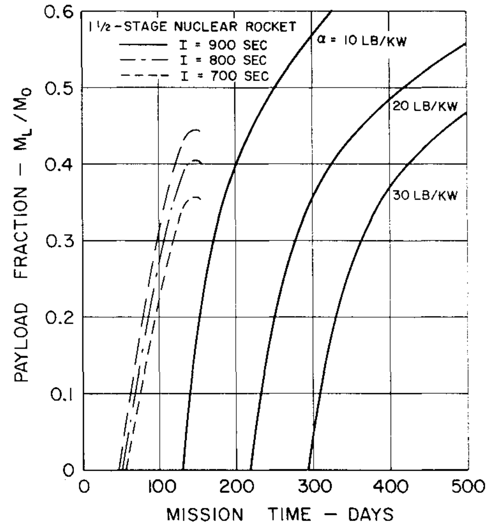

Fig. 2 Venus satellite mission; powerplant specific weight = α; powerplant fraction = 0.25; Saturn IB launcher.

Results for Interplanetary Missions

Venus Satellite Mission

A generally applicable plot of payload fraction as a function of mission time, power conversion efficiency (η), and specific impulse (I) for the Venus satellite mission performed with electric propulsion is presented in Fig. 1. The assumed specific weight of the powerplant is 20 lb/kwe, and the fraction of the total vehicle weight allocated to the powerplant is 0.25.

This plot can be employed to determine the payload fraction of the space vehicle as a function of mission time for an electric propulsion system using any thruster and booster within limits. The limit on booster size is the limitation on the validity of the various space vehicle component weight fraction assumptions when the gross weight differs greatly from what may be launched by a Saturn IB. The limitations on choice of thruster lies in the fact that different thrusters use different propellants, the density of which may require different tank weights than those used in the present calculations. However, since the tank weight of an electrically propelled vehicle is a very small part of the total weight, not much accuracy is sacrificed in using Fig. 1 for thrusters employing propellants with densities as diverse even as liquid hydrogen and mercury. The assumed tank weights for Fig. 1 are based on a propellant with a density equal to the average between those of mercury and cesium.

For any specific thruster the variation of power conversion efficiency with specific impulse can be plotted on Fig. 1 to determine the payload fraction as a function of mission time. If these curves were to be superimposed on similar plots for other values of powerplant fraction, the resulting envelope of such curves would give maximum payload with optimum values of powerplant fraction and specific impulse for every value of mission time. For most missions and mission times,

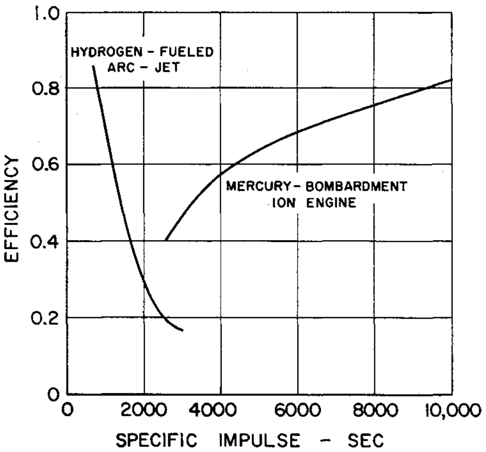

Fig. 3 Estimated variation of efficiency with specific impulse.

however, the optimum value of powerplant fraction turns out to be in the neighborhood of 0.25. Therefore, plots for only this one powerplant fraction are presented here. Lack of space prevents inclusion of similar general plots for other values of powerplant specific weights.

For specific comparison with the nuclear rocket, a mercury-bombardment ion thruster was selected. The payload comparison is shown in Fig. 2 for specific powerplant weights of 10, 20, and 30 lb/kwe and assumed values of specific impulse for the nuclear rocket of 700, 800, and 900 sec. The assumed efficiency-specific impulse curve for the mercury-bombardment ion engine[16] is shown in Fig. 3. The booster for the nuclear-rocket vehicle is assumed to be a Saturn IB. It is seen in Fig. 2 that for the Venus satellite mission the use of electric propulsion is not particularly attractive, because when comparing the nuclear rocket with the lowest of the assumed values of specific impulse (700 sec) with the electric propulsion system having the lowest value of powerplant specific weight (10 lb/kwe), the nuclear rocket can deliver larger payloads in less time.

Jupiter Probe Mission

For electric propulsion systems, general plots of payload fraction as a function of mission time for given values of power conversion efficiency and specific impulse for the Jupiter probe mission are presented in Figs. 4 and 5 for an assumed powerplant fraction of 0.25 and values of powerplant specific weight of 20 and 30 lb/kwe, respectively. Again, any variation of efficiency with specific impulse may be plotted on the grids of these figures (4 and 5) to obtain payload fraction as a function of mission time for the given thruster.

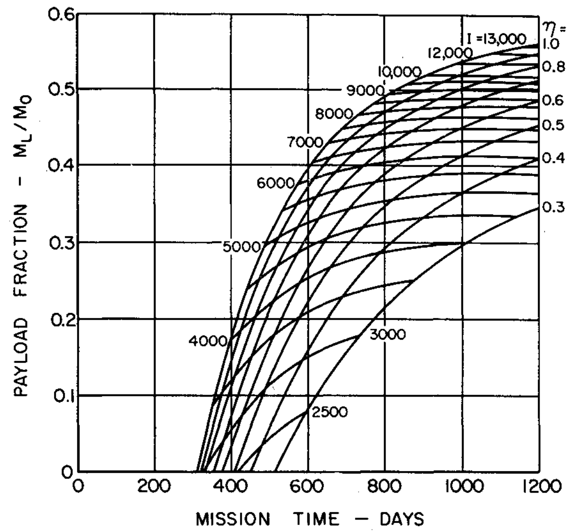

Fig. 5 Jupiter probe mission; powerplant specific weight = 30 lb/kwe; powerplant fraction = 0.25.

The efficiency vs specific impulse variation of Fig. 3 for the mercury-bombardment ion engine was used to obtain the electric propulsion curves of Fig. 6. For comparison, curves for the nuclear rocket for values of specific impulse of 700, 800, and 900 sec are also shown in Fig. 6. It is seen that electrical propulsion would appear attractive, even at a specific weight as high as 30 lb/kwe, if mission times longer than about 600 days were acceptable. Current estimates indicate a reasonable system lifetime of about 10,000 hr (417 days). At this value of mission time, Fig. 6 shows that specific weight of electric propulsion units must be below the 15 to 20 lb/kwe range in order for electric propulsion to be superior to the solid-core nuclear rocket for this mission.

Solar Probe Mission

The general plot of payload fraction vs mission time for electric propulsion applied to the solar probe mission is presented

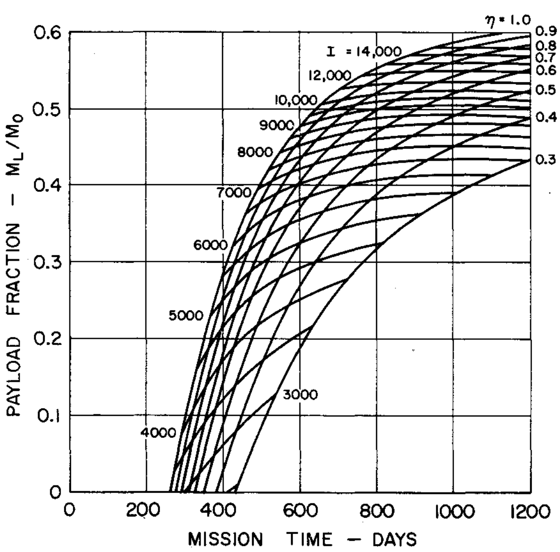

Fig. 4 Jupiter probe mission; powerplant specific weight = 20 lb/kwe; powerplant fraction = 0.25.

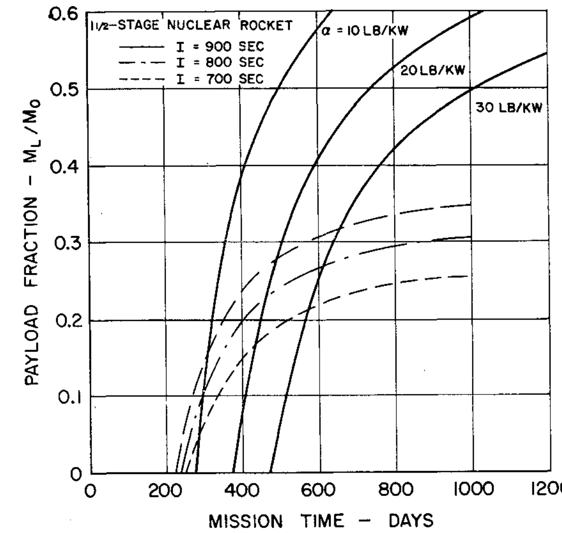

Fig. 6 Jupiter probe mission; powerplant specific weight = α; powerplant fraction = 0.25; Saturn IB launcher.

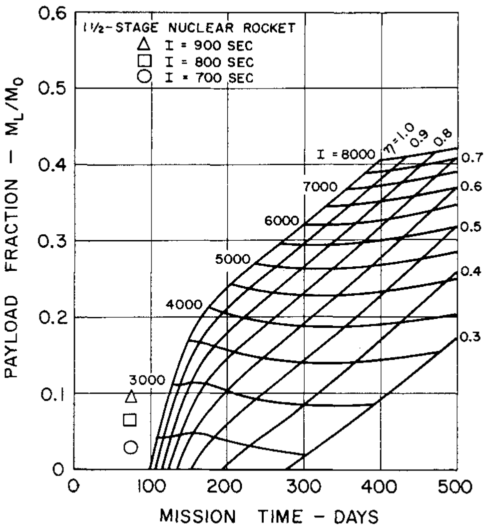

Fig. 7 Solar probe mission; powerplant specific weight = 20 lb/kwe; powerplant fraction = 0.25; Saturn IB launcher for nuclear rocket.

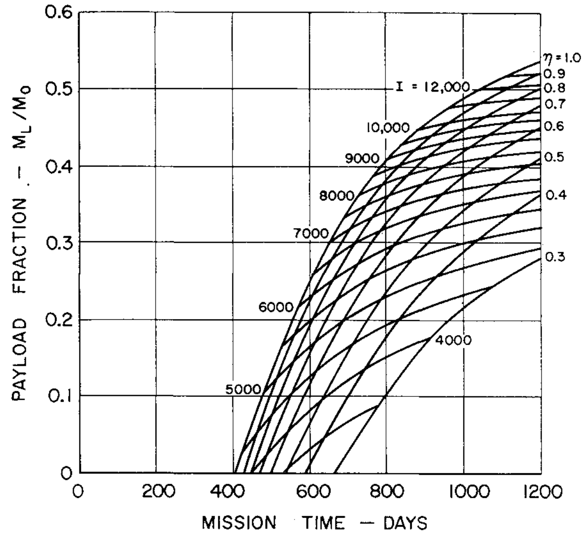

Fig. 9 Saturn probe mission; powerplant specific weight = 20 lb/kwe; powerplant fraction = 0.25.

in Fig. 7 for a specific powerplant weight of 20 lb/kwe and for a powerplant fraction of 0.25. Also shown on this plot are payload fractions for the nuclear rocket launched by the Saturn IB. Since the payload fractions achievable with nuclear rockets are very small, only the peak values, corresponding to minimum-energy transfer, are shown for assumed values of specific impulse of 700, 800, and 900 sec.

For specific comparison with the nuclear rocket, the variation of efficiency with specific impulse of the mercury-bombardment ion engine shown in Fig. 3 was used to produce the curves of Fig. 8. This figure shows that electric propulsion can match the payload capability of the nuclear rocket on a 10,000-hr mission with a specific weight as high as 30 lb/kwe and that substantially greater payloads may be carried in less than 10,000-hr mission time with powerplant specific weights less than 30 lb/kwe.

Saturn Probe Mission

For electric propulsion systems, payload fraction as a function of mission time for given values of power conversion efficiency and specific impulse for the Saturn probe mission is presented in Fig. 9. Results for the mercury-bombardment ion engine are compared with the nuclear-rocket results in Fig. 10. The electric-propulsion system is completely superior for this difficult mission, providing greater payloads in shorter times for all three of the powerplant specific weights shown. However, because of the extreme difficulty of the mission, it would be necessary to achieve a powerplant lifetime in the neighborhood of 15,000 to 20,000 hr in order to realize the advantage.

At this point it would be well to state a general result for all of the interplanetary missions. It will be noticed on the general plots (Figs. 1, 4, 5, 7, and 9) that in all of the areas showing clear superiority of electric propulsion over nuclear rockets, the electric-propulsion specific impulse is greater than at least 3000 sec. Therefore electric thrusters that are limited to values of specific impulse less than 3000 sec are not very attractive for interplanetary missions.

Results for the Lunar Logistic Supply Operation

Performance Analysis

The performance of advanced propulsion systems as applied to the lunar supply operation is illustrated by the Saturn V launch rate (number of launches per month) necessary to maintain a given supply rate (lb/day) to the lunar surface.

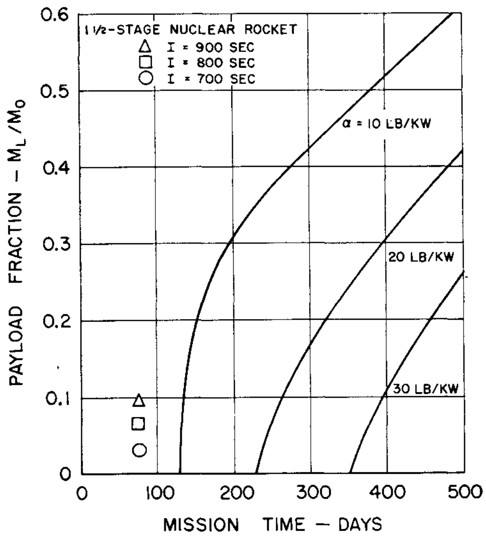

Fig. 8 Solar probe mission; powerplant specific weight = α; powerplant fraction = 0.25; Saturn IB launcher.

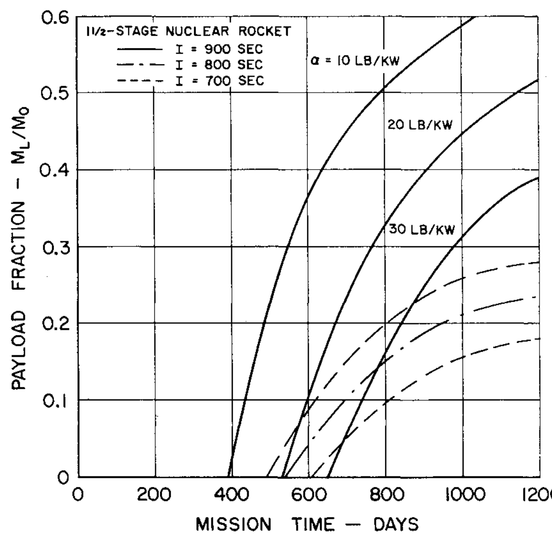

Fig. 10 Saturn probe mission; powerplant specific weight = α; powerplant fraction = 0.25; Saturn IB launcher.

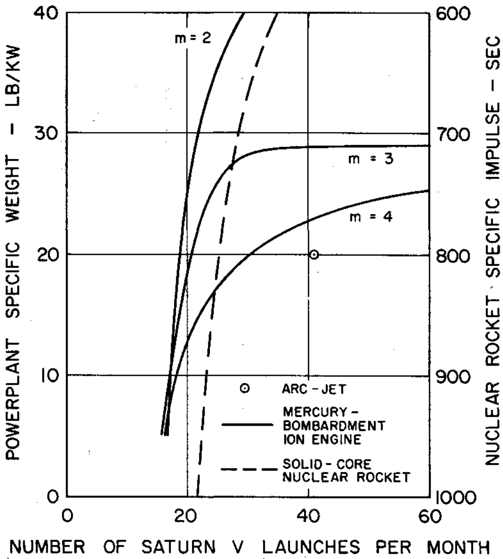

Fig. 11 Lunar logistic supply operation; Saturn V launch rate; supply rate = 50,000 lb/day; *m* - number of payload deliveries per vehicle.

Since the analysis of the lunar supply operation is more complicated than that of the interplanetary missions, it is not feasible to present generally applicable results for electric-propulsion systems. Instead, results are presented for both the mercury-bombardment ion engine and a hydrogen-fueled arc-jet. The variations of efficiency with specific impulse for these engines are given in Fig. 3.

Figure 11 shows the electric-propulsion powerplant specific weight and the nuclear-rocket specific impulse plotted against the Saturn V launch rate for an assumed supply rate of 50,000 lb/day. Only one value of the supply rate is included in the figure, since the Saturn V launch rate is simply proportional to the supply rate. With the abscissa as the dependent variable, the figure may be interpreted as indicating the Saturn V launch rate necessary to maintain the given supply rate with either electric-propulsion systems of a given powerplant specific weight or nuclear rockets of a given specific impulse. The results for electric propulsion are for 2, 3, and 4 payload deliveries per vehicle within an assumed powerplant lifetime of 10,000 hr. The corresponding values of trip time and specific impulse for the electric-propulsion systems presented in this figure are shown in Table I. Since the performance of the nuclear rocket does not change significantly with number of payload deliveries per vehicle, this information is not given in Fig. 11.

It is seen in Fig. 11 that the ion-engine system for 2 payload deliveries per vehicle requires a lower Saturn V launch rate throughout the range of powerplant specific weights and nuclear-rocket specific impulses shown. From the single point shown for the arc-jet system at 2 payload deliveries per vehicle and 20 lb/kwe powerplant specific weight it is evident that it cannot compete with the nuclear rocket. For a larger number of payload deliveries per vehicle the arc-jet showing is even worse; therefore, it is not considered further.

Three payload deliveries with the ion-propelled vehicle in 10,000 hr results in a lower Saturn V launch rate than the nuclear rocket only up to a powerplant specific weight of about 28 lb/kwe. For four payload deliveries in 10,000 hr, electric propulsion is superior for powerplant specific weights of less than about 17 lb/kwe. As would be expected, the decrease in allowable trip time associated with an increase in the number of payload deliveries acts to penalize the performance capabilities of electric propulsion, and the crossover point moves to lower values of powerplant specific weight.

Figure 11 has two unrelated ordinate scales that are used to show a general comparison between nuclear and electric propulsion. Specific comparisons are possible only with the knowledge of comparable values so that no significance should be attached to the apparent correspondence between the two. However, general comparisons in terms of relative Saturn V launch rate or crossover values of specific weight or specific impulse for a given launch rate are possible because the curves for the nuclear-rocket vehicles are so nearly vertical that comparisons would be relatively unaffected regardless of the placement of the specific impulse scale.

Economic Analysis

Some selected results of a detailed cost analysis are presented in Fig. 12. An itemized list of assumed initial specific costs, upon which this analysis is based, appears in Table II. In Fig. 12 are shown curves of average specific cost (dollars per pound of payload delivered to the lunar surface) as a function of supply rate for a range of values of supply rate from 10^3 to 10^5 lb/day. The assumed time span of the operation is 10 yr; therefore the average specific cost is the total cost of

Table 1 Electric propulsion trip times and specific impulse for the lunar logistic supply operation

Thruster	Powerplant specific wt, lb/kwe	Powerplant lifetime, hr	Payload deliveries per vehicle	Specific impulse, sec	Round-trip time, days	Outbound time, days
Arc-jet	20	10,000	2	1,750	249	168
Mercury-bombardment ion engine	10	10,000	2	22,700	240	177
			3	12,900	154	109
			4	8,000	114	75
			5	5 400	90	57
	10	15,000	2	>25,000	356	269
			3	21,800	229	167
			4	14,700	170	115
			5	10,400	132	97
			6	7,700	110	75
	20	10,000	2	9,400	240	176
			3	4,600	154	109
			4	2,700	114	75
	20	15,000	2	16,700	356	269
			3	8,800	229	167
			4	5,500	170	115
			5	3,500	132	97
	30	10,000	2	5,300	246	171
	30	15,000	2	9,800	365	260
			3	5,000	232	161

the operation divided by the total payload delivered in 10 yr. The characteristic of decreasing specific cost with increasing supply rate reflects an 85% learning curve assumed for both launch and space vehicle costs. Results for the mercury-bombardment ion engine are shown for a powerplant specific weight of 20 lb/kwe, for three payload deliveries in a single powerplant lifetime of 10,000 hr, and for four payload deliveries in a lifetime of 15,000 hr. These two numbers of payload deliveries (3 and 4) provide the minimum specific cost of all possible integral numbers for the 10,000 and 15,000 hr lifetimes, respectively. The powerplant is assumed to be a single module with an output of about 10 Mwe. As the powerplant cost is comparable to the launch cost (roughly one-quarter to one-third), it is obvious that the powerplant lifetime significantly affects the specific cost as is shown.

The middle two curves in Fig. 12 represent a nuclear rocket with a specific impulse of 800 sec for 2, 10, and 50 payload deliveries per vehicle. Since the cost of the nuclear-rocket engine is much smaller in comparison to the launch cost than is the powerplant cost of the electrically-propelled vehicle, the number of reuses of the nuclear rocket does not have nearly the effect on specific cost as does the powerplant lifetime of the electric rocket. Comparing the two sets of curves in Fig. 12, it is seen that for a powerplant lifetime of 10,000 hr the electric-propulsion system is more expensive than the nuclear-rocket system. For a powerplant lifetime of 15,000 hr, however, the specific cost of the electric-propulsion system is considerably below that of the nuclear rocket.

In spite of the performance superiority of electric propulsion, it is not surprising that a similar superiority is not shown in specific cost. The high cost of the powerplant partially offsets the lower specific launch cost (lower Saturn V launch rate) associated with the electric-propulsion system. Nevertheless, if the lifetime of the powerplant could be extended to 15,000 or 20,000 hr or the specific weight decreased, significant savings could be realized in specific cost.

General Conclusions

From the comparison made in this study between the two advanced propulsion systems as applied to the specific missions, a number of general conclusions can be drawn. It should be remembered that the comparisons that form the basis of these conclusions are based upon the use of the SNAP-50 with a mercury-bombardment ion engine and upon the use of an advanced metallic-core nuclear rocket for the interplanetary missions. For the lunar logistic supply operation, the comparisons are based on the use of the mercury-bombardment ion engine with multiple SNAP-50's (about 10 Mwe) and a graphite-core nuclear rocket.

Any final conclusions regarding a choice between nuclear-rockets and electric-propulsion systems for a program of unmanned space missions must of course be based upon

Table 2 Assumed initial specific costs

Item	Initial specific cost, dollars per pound
Saturn V launch	200
Powerplant[a]	360
Thruster	360
Structure and tankage	$-410 \log_{10}$ (stage deadweight) $+2190$
Mercury	2.54
Nuclear-rocket engine[b]	1223
Liquid hydrogen	0.20
U235	5500

[a] Does not include cost of U235 reactor fuel; a SNAP-50 class reactor contains about 290 lb of U235.
[b] Includes cost of uranium oxide at $8000/lb.

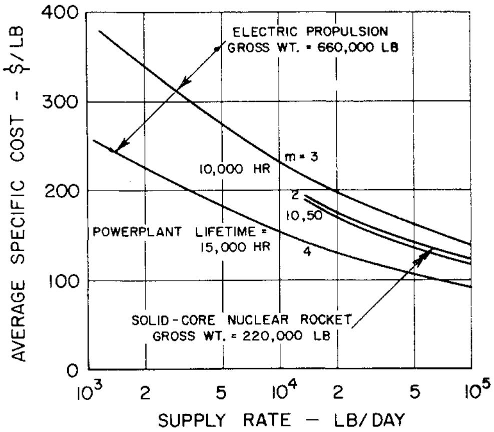

Fig. 12 Lunar logistic supply operation; average specific cost; time span of operation = 10 yr; m = number of payload deliveries per vehicle.

complete knowledge of the values of nuclear-rocket specific impulse, powerplant specific weight, and lifetime that can actually be achieved. Since this knowledge is at present unobtainable, any conclusions that can be made are necessarily qualified. This study was designed to include a wide range of qualifications.

Under these qualifications the following general conclusions can be drawn:

1) In general, a powerplant specific weight of 30 lb/kwe or less must be achieved in order for electric propulsion to compete with nuclear rockets in the 700- to 900-sec range of specific impulse. It should be kept in mind that this specific weight includes not only the basic powerplant but also power conditioning and shielding.

2) For unmanned scientific missions to the nearer planets, nuclear-rocket propulsion is superior to electric propulsion even for a powerplant specific weight as low as 10 lb/kwe.

3) With increasing severity of mission requirements, electric rockets at reasonable values of specific weight (20 to 30 lb/kwe) become increasingly competitive with or superior to nuclear propulsion. However, to realize the ideal performance superiority of electric propulsion for missions to the major planets or to the outer reaches of the solar system, a powerplant lifetime of at least 15,000 hr and preferably longer, must be achieved.

4) Because missions near the sun (solar and Mercury probes) also have difficult propulsion requirements but are of relatively short duration, electric propulsion for these missions enjoys unqualified superiority.

5) Electric-propulsion devices having a specific impulse of less than 3000 sec (arc-jets, resisto-jets, plasma devices) are not competitive with either nuclear rockets or ion engines even if they can achieve high efficiency. These devices would be of interest, if their specific impulse could be increased to more than 3000 sec with no appreciable loss in efficiency.

6) If there exists a need for a large-volume supply of cargo to the moon, the need for which can be scheduled adequately in advance, electric propulsion offers better performance, in terms of payload delivered per unit weight in earth orbit, than nuclear rockets.

7) With an operating lifetime greater than about 12,000 hr, electric propulsion can perform the lunar logistic supply operation more economically than nuclear rockets. In order to achieve this advantage, large vehicles must be employed. These vehicles must have an initial gross weight in earth orbit on the order of 500,000 to 1,000,000 lbs with a power supply having a power output of 10 Mwe or greater.

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