edge nearest the neutralizer tip. Substantial amounts of barium were found on all upstream areas of the insert.

Except for the electrostatic type vector grid, no other thruster component indicated imminent failure at 9715 hr.

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Comparison of Advanced Propulsion Capabilities for Future Planetary Missions

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This paper summarizes unmanned planetary performance (payload and trip time) of Shuttle-based advanced propulsion systems for 1980–90 missions analyzed as part of the recent NASA/AEC Advanced Propulsion Comparisons Studies. Propulsion system designs and condensed results from over 300 propulsion/mission combinations are discussed. Chemical rocket (CRP), solar electric (SEP), nuclear rocket (NRP), and nuclear electric (NEP) propulsion systems are all considered. In terms of missions flown, total flight time, and number of Shuttle launches required, NEP provides the best performance. Relative to NEP, it is shown that NRP, SEP, and CRP degrade mission performance by 20%, 40%, and 50%, respectively, at nominal payloads.

Introduction

DURING 1972–1973, NASA and the AEC conducted a comprehensive series of studies on the subject of Advanced Propulsion Comparisons (APC). The primary purpose of these studies was to determine the most cost effective upper stage propulsion which, when combined with the Space Shuttle, would be capable of meeting space mission demands in the 1980's. All basic propulsion modes were considered. Energy sources included chemical, solar, and nuclear, while flight modes consisted of ballistic and sustained low thrust. In scope, the studies included geocentric, lunar, and planetary missions. To meet the objective of cost effective comparisons, all aspects of propulsion systems development and utilization were analyzed, including technology bases, development schedules, nonrecurring and recurring cost, performance, payload cost benefits, and an economic impact analysis for the period 1980–1990.

The authors' roles in the APC studies were to provide basic planetary mission performance data, in terms of net payload and trip time, for a wide range of propulsion configurations which can be summarized by four classes of propulsion: 1) chemical rocket propulsion (CRP), 2) solar electric propulsion (SEP), 3) nuclear rocket propulsion (NRP), and 4) nuclear electric propulsion (NEP). The planetary mission model used consisted of 21 missions. It included all planets of the solar system as well as several satellites, comets, and asteroids. In total, over 300 combinations of propulsion configurations and missions were analyzed. This broad base of planetary performance data was combined with a concurrent APC study by Jet Propulsion Lab. (JPL) of planetary spacecraft design requirements to formulate propulsion dependent mission definitions for other areas of the APC analysis.

The purpose of this paper is to summarize the planetary performance results generated by the authors for the NASA/AEC Advanced Propulsion Comparisons Studies. The data presented

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Index categories: LV/M Mission Studies and Economics; Spacecraft Mission Studies and Economics; Lunar and Interplanetary Trajectories.

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Table 1 Condensed planetary mission list from APC studies

No.	Launch	Mission	Baseline payloads (kg) CRP	SEP[a]	NRP	NEP[b]
1	1985	0.1 a.u. solar probe	385	385	385	320
2	1984	Mercury orbiter (1.4×1.4)[c]	695	695	695	450
3	1989	Venus large lander	1400	1400	1400	1400
4	1983	Venus radar mapper (1.1×1.1)	715	715	715	715
5	1984	Mars orbiter/rover (1.8×28.4)	2655	2655	2655	2425
6	1994	Phobos/Diemos rendezvous/SSR[d]	1100	1100	1100	1100
7	1990	Mars orbiter/SSR (1.3×1.3)	4100	4100	4100	3920
8	1981/82	Encke rendezvous	645	645	645	550
9	1983	Halley rendezvous	...	...	...	550
10	1985	Vesta rendezvous	745	745	745	565
11	1993	Flora rendezvous/SSR	2145	2100	2145	1965
12	1993	Ceres rendezvous/SSR	2145	2100	2145	1965
13	1985	Jupiter orbiter (4×45.1)	695	695	695	580
14	1990	Jupiter/Pluto swingby	685	685	685	625
15	1990	Ganymede orbiter/lander (1.04×1.04)	1825	1825	1825	1715
16	1984	Saturn orbiter/probe (3×58.7)	860	860	860	670
17	1988	Saturn orbiter/ring probe (1.2×2.3)	985	985	985	830
18	1984	Saturn/Uranus/Neptune swingby probe	710	710	710	650
19	1987	Uranus orbiter/probe (1.2×36.6)	950	950	950	730
20	1989	Neptune orbiter/probe (1.2×41.8)	950	950	950	730

[a] Assuming an autonomous SEP stage.
[b] Assuming an integrated payload/NEP system.
[c] (periapse × apoapse) in planet radii.
[d] SSR, Surface Sample Return.

in the paper have been considerably edited and condensed for brevity and convenience. A complete report is given in Ref. 1.

Mission Analysis

Performance results are presented below for the condensed set of planetary missions given in Table 1. The mission list includes as targets all planets of the solar system (except Earth), satellites, comets, asteroids, and the sun. The mission flight modes considered include flyby, swingby, rendezvous (R), orbit, entry (probe), and surface sample return (SSR). Where the orbit mission mode is applicable, the orbit dimensions are included in parentheses after the mission title. Baseline mission payloads (exclusive of retro propulsion), as provided by JPL, are included in Table 1 for each propulsion class. As can be seen from these data, the payloads do vary somewhat with the class of propulsion considered, the largest difference observed with NEP where the spacecraft was assumed integrated with the propulsion system. Payload and flight time performance were evaluated for specific launch opportunities, as shown in Table 1. However, many of the missions considered are relatively insensitive to launch opportunity so that the performance results need not be restricted to a single mission program plan. Planetary program planning was a task addressed by JPL in the APC studies. Their results are presented in Ref. 2.

A number of assumptions and constraints were used in analyzing the mission trajectory characteristics. These are as follows:

1) All earth departures from a 235 nm circular parking orbit.
2) All transfers based on optimum departure dates (no launch period allowances considered in energy requirements).
3) No DLA restrictions or energy allowances accounted for.
4) No guidance/navigation allowances accounted for.
5) All postlaunch CRP and NRP maneuvers considered impulsive.
6) All low-thrust spiral maneuver requirements analyzed based on final circular orbits of a semi-major axis equivalent to desired elliptical orbits.

Two underlying reasons led to the formation of these ground rules. First, as much standardization as possible was desired to justify subsequent relative performance comparisons. Second, a number of simplifying assumptions were necessary in order to complete the analysis of a relatively large number of propulsion/mission combinations (over 300) within a short period of time. Although the resulting performance data are somewhat optimistic as a result of the lack of launch window and navigation allowances, they were quite acceptable for their intended use in formulating relative performance comparisons.

Three trajectory generating computer programs were employed in the analysis: 1) SPARC[3] for planetary swingby transfers, 2) MULIMP[4] for optimum ballistic orbiter and rendezvous trajectories, and 3) CHEBYTOP II[5] for optimum low-thrust (SEP and NEP) transfers. In addition to these computing tools, a number of smaller programs were employed for the analysis of planet-centered maneuver requirements, such as impulsive and spiral capture maneuvers, orbit plane change and phasing maneuvers, and satellite capture maneuvers. Wherever possible, existing transfer data from handbooks[6] and previous studies were incorporated in the analysis to minimize computational costs.

Propulsion Systems

The four classes of propulsion considered in the analysis, as mentioned above, are CRP, SEP, NRP, and NEP. In addition, chemical retro stage scaling laws were employed to size postlaunch impulsive propulsion requirements as needed. Both nominal and growth (advanced technology) propulsion options were considered in the SEP, NRP, and NEP classes, as well as for the scaled retro propulsion (RP). The total set of propulsion subclasses and their periods of application are as follows:

CRP (1980–90)
SEP (1980–90) GSEP (1983–90)
NRP (1980–90) GNRP (1986–90)
NEP (1980–90) GNEP (1986–90)
RP (1980–85) GRP (1986–90)

where the preface "G" on an acronym indicates growth (advanced technology) configurations.

CRP, Chemical Rocket Propulsion

Two principal stage designs were considered for the CRP class of propulsion: 1) the existing Centaur upper stage, and 2) a

Table 2 Chemical Rocket Propulsion (CRP) stage parameters

Design parameters	Stage		
	Centaur	Tug	APC kick[a]
Stage thrust (lbf)	30,000	18,000	3,000
Engine specific impulse (lbf-sec/lbm)	444	470	378
Maximum propellant loading (lbm)	30,000	53,870	5,084
Stage burnout weight (lbm)	4,413	5,860	1,216
Interstage weight (lbm)[b]	...	300	...
Payload adapter weight (lbm)[c]	116	100	20
Shuttle pallet weight (lbm)	2,350	1,390	...

[a] A conceptual FLOX/MMH pressure-fed autonomous stage design.
[b] Applicable to multi-Tug stage configurations.
[c] Adapter to payload or kick stage (if included).

conceptual design of the Space Tug. In addition, a conceptual pressure-fed kick stage was available with either the Centaur or Tug for improved payload performance on high-energy missions. Basic system parameters of these three CRP stages are summarized in Table 2.

The Centaur performance parameters represent the D1-T version of the stage, which has been well defined in previous documents. The comparatively large pallet weight, however, is worth noting. The Centaur does not tax the maximum capability of the Shuttle except with very large payloads (> 20,000 lbm). Hence, it is possible to propose a substantial pallet carriage in the Shuttle bay which effectively reduces modification to the existing stage design.

The Tug parameters reflect an advanced technology cryogenic propulsion unit‡ with a stage mass fraction just under 0.9. The stage has one engine which generates 18,000 lbf thrust at an I_{sp} of 470 sec using a chamber pressure of 1840 psia. The stage

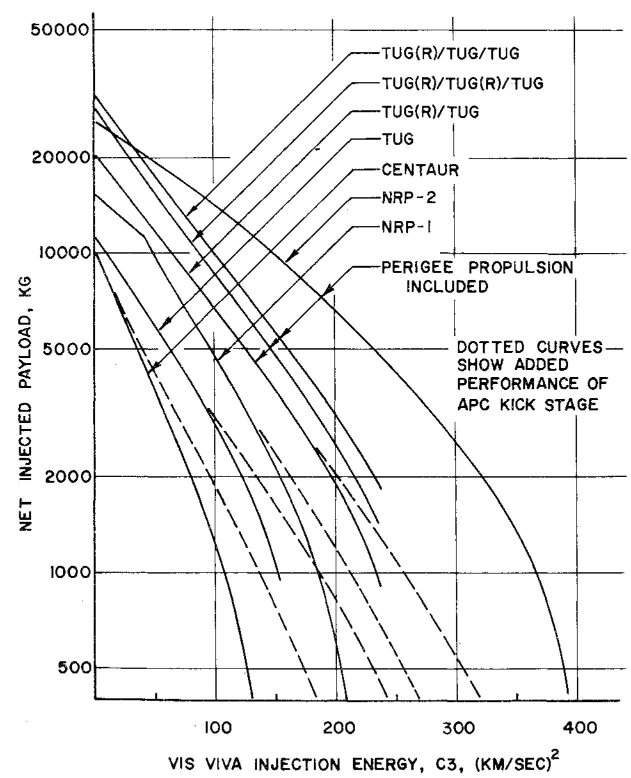

Fig. 1 CRP and NRP earth escape payload performance.

‡ The Tug stage parameters were provided by MSFC/NASA as a then current baseline design. Since the Tug is still under evaluation, these data have and will continue to change somewhat in the near future.

structure design is optimized for Shuttle applications, which in turn reflects a smaller pallet weight than for the Centaur. Tug options were considered as both single and multiple stage configurations and in recoverable as well as expendable flight modes. Performance data for the various configurations used were generated at LeRC/NASA. No flight performance reserve was assumed for these calculations. All multiple Tug configurations utilize perigee propulsion for escape. Reusable Tug, TUG (R), applications assume a 3 min coast for payload separation between injection and stage retro burn and a total stage time out of parking orbit of 48 hr.

The kick stage, defined specifically for the APC studies by JPL, is an autonomous pressure-fed earth-storable stage with an I_{sp} of 378 sec. The rather heavy stage inert weight of 1216 lbm includes astrionic and power supply allowances necessary for autonomy. The injected payload performance of each of the CRP options considered in the planetary mission performance analysis is plotted in Fig. 1 as a function of escape energy C3. Improved performances with the APC Kick Stage are shown as dotted lines.

Retro propulsion (RP) for mission performance estimates (when applicable) was based on a set of scaling laws generated by JPL for chemical space-storable retro stages. The logic of application of these relationships along with their specific parameters, is outlined in Fig. 2. Note that several engine thrust levels and propellant I_{sp}'s are assumed depending upon the retro stage size, area of application (presence of local gravity field or not), and time period of use. Multiple retro stages were assumed when the magnitude of the total postlaunch impulse exceeded the exhaust velocity of the particular RP option in question. The total impulse was then divided equally between the stages for near-optimum sizing.

NRP, Nuclear Rocket Propulsion

Nuclear rocket stage options were evolved from two basic stage configurations which are illustrated in Fig. 3. The first is a single Shuttle launched propulsion module which is volume-limited (rather than weight limited) by the Shuttle cargo bay. An allowance of 10 ft is included for the kick stage (optional) and payload. The propulsion module can be fitted out with either a nominal LASL-α Small Nuclear Rocket Engine (SNRE) or the growth LASL-γ version. NRP options in this configuration are referred to as —1 (dash one).

The second configuration used is a double Shuttle launched arrangement which adds two propellant modules (with a second launch) to the basic propulsion module just discussed. Again this is a volume limited configuration. The tank modules are carried above the payload to be jettisoned individually upon propellant depletion. Attaching the tank modules alongside the propulsion module would be an alternative approach with less interstage structure, but more complicated earth-orbital assembly. Either the nominal LASL-α or growth LASL-γ versions of the SNRE may be used with this configuration. NRP options of this configuration are referred to as a) —2 for

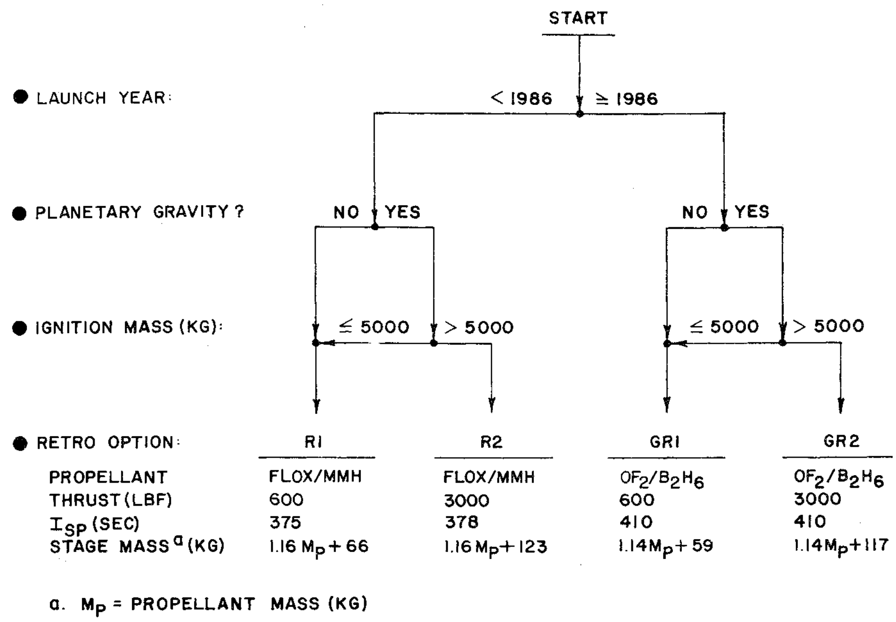

Fig. 2 Retro propulsion Scaling Law logic tree and parameters.

Table 3 Nuclear Rocket Propulsion (NRP) stage parameters

Design parameters	Stage configurations					
	NRP—1	GNRP—1	NRP—2	GNRP—2	NRP—3	GNRP—3
Small Nuclear Rocket Engine (SNRE)	LASL-α	LASL-γ	LASL-α	LASL-γ	LASL-α	LASL-γ
Stage thrust (lbf)	16,312	14,625	16,312	14,625	16,312	14,625
Engine specific impulse (lbf-sec/lbm)	875	975	875	975	875	975
No. of tank modules	0	0	2	2	2	2
Maximum propellant weight (lbm)						
base stage	29,627	29,627	29,627	29,627	27,365	27,365
tank module[a]	...	...	21,663	21,663	21,663	21,663
total	29,627	29,627	72,953	72,953	70,691	70,691
Burnout weight (lbm)						
base stage	10,244[b]	10,644[b]	10,439[b]	10,839[b]	12,555[c]	13,755[c]
tank module[a]	...	...	3,234	3,234	3,234	3,234
Interstage adapter weight (lbm)[d]	...	...	345	345	345	345

[a] Weights given per tank module; each module is jettisoned upon propellant depletion.
[b] Includes 300 lbm propellant for final engine aftercooling.
[c] Includes 300 lbm propellant for final aftercooling and 660 lbm propellant lost as hot gas.
[d] Weight for payload-space trusswork (see Fig. 3); stage-to-stage and stage-to-payload adapter weights are included in burnout weights.

Earth-escape propulsion only, and b) —3 for Earth-escape and retro maneuvers. The —3 version requires a higher inert weight for long term propellant storage between maneuvers.

NRP stage parameters for six options (two technology levels times three configurations) are summarized in Table 3. NRP options reflect nominal technology (LASL-α SNRE) prior to 1986, while GNRP options indicated advanced technology (LASL-γ SNRE). Note that aftercooling propellant, necessary to prevent SNRE melting, is included in the inert weight along with payload adapters. Shuttle pallet weights are not given since all of these options are volume-limited. Reduced propellant capacity and increased inert weight of the base (propulsion) stage of the —3 versions are evidenced by the data given in Table 3.

Performance data for the NRP options were generated by G. A. Hazelrigg of Princeton University.[7] NRP performance data, also presented in Fig. 1, are limited to the NRP—1 and NRP—2 options. Note that the NRP—2 curve utilizes perigee propulsion to reduce gravity losses over the long burning period of the stage (≈ one hr). A more complete set of performance data, including that of the longlife —3 version, is presented in Ref. 7 along with the specific groundrules assumed in making the

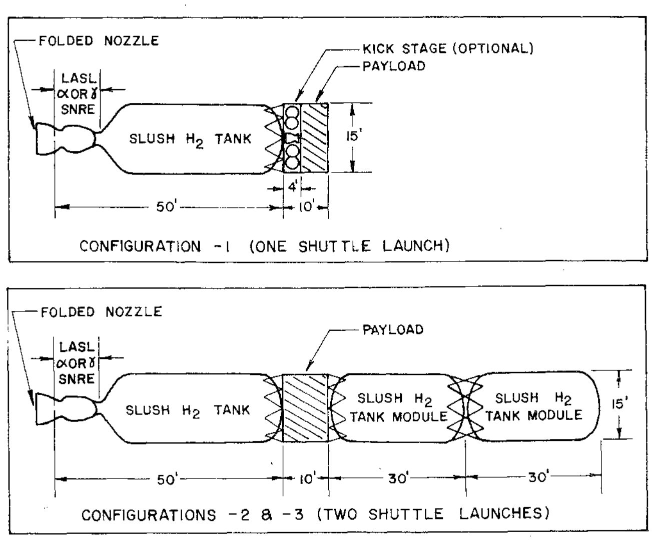

Fig. 3 NRP stage definitions.

performance computations. A comparison between NRP and CRP options in Fig. 1, shows that the NRP—1 option has somewhat better escape payload performance than the expendable Tug, while the NRP—2 option is better than all other options presented including the triple Tug (R)/Tug/Tug configuration.

SEP, Solar Electric Propulsion

The two basic options considered for the SEP class of propulsion are a 20 kw nominal system and a 40 kw growth version. Electrical power designation refers to the design input power (at 1 a.u.) to the power conditioning subsystem. It is important to note that either SEP option operates in conjunction with a high-thrust injection stage (CRP or NRP) which provides sufficient excess velocity to escape Earth prior to low-thrust startup. Also, the SEP system is configured as an autonomous stage having minimal (mechanical) interface with the payload spacecraft and providing general applicability to all planetary missions. Table 4 summarizes the basic performance parameters of the SEP stage options.

Propulsion system specific impulse and over-all efficiency are fixed for all missions at the values of 3000 sec and 67%, respectively. The thruster type considered is a 30 cm mercury electron bombardment ion engine. Specific mass of the (dry) propulsion system includes the solar cell array, power conditioning units, thrusters, thrust vector control mechanisms, and associated integrating structure. Nominal and growth systems are assumed to have a specific mass of 30 and 20 kg/kw, respectively; this difference is mainly due to anticipated advances in solar array technology. Propellant tankage is 3% of the mercury propellant loading which is, of course, mission-dependent.

One major impact on performance of the stage concept is the allocation of a stage subsystem weight (300 kg) applied uniformly to all missions. This nonpropulsion, nonpayload related subsystem required for stage autonomy includes such engineering support elements as communications, data handling, computer and sequencer, attitude and thermal control, mechanisms, and associated structure. An auxiliary power requirement of 400 w (independent of solar distance) is specified for operation of these stage subsystems. Although this power drain is a small fraction of SEP power at 1 a.u. it can have a significant effect on payload capability for certain missions (comet and asteroid rendezvous) where thruster operation is required for solar distances out to 3.5 a.u.

Specific thruster on-time constraints are not considered in the analysis. However, this is treated indirectly by limiting the total propulsion time to 350 days for outer planet missions. In effect, this tends to ensure that realistic constraints on minimum power and thruster lifetime are satisfied. Payload performance degradation due to the 350-day limit is of small consequence for outer

Table 4 SEP stage parameters

Stage parameters	SEP	GSEP
Power input at 1 a.u., kw	20	40
Specific impulse, sec	3,000	3,000
Propulsion efficiency, %	67	67
Propulsion system mass, kg	600	800
Propellant tankage fraction, %	3	3
Stage subsystem mass, kg	300	300
Auxiliary power, watts	400	400
Solar power, P/P_0	G(r)[a]	G(r)[a]
Propulsion time constraint	none[b]	none[b]
Thrust direction program	optimum	optimum

[a] $G(r) = \begin{cases} 1.4382/r^2 - 0.2235/r^3 - 0.2147/r^4 & \text{for } r > 0.68 \text{ a.u.} \\ 1.3952 & \text{for } r < 0.68 \text{ a.u.} \end{cases}$

This relationship was adopted from the JPL studies of the Solar Electric Multi-Mission Spacecraft (SEMMS).

[b] A fixed propulsion time of 350 days was used for all outer planet missions.

Table 5 NEP system parameters

System parameters	NEP	GNEP
Power input to PCU, kw	120	240
Specific impulse, sec	4,000–7,000	4,000–7,000
Propulsion efficiency, %	variable[a]	variable[a]
Propulsion system mass, kg	3,600	6,000
Propellant tankage fraction, %	3	3
Stage subsystems mass, kg		
nonjettison mode	0	0
jettison mode	300	300
Propulsion time constraint, hrs	20,000	30,000
Thrust direction program	optimum	optimum

[a] Over-all propulsion system efficiency is given by the following function of specific impulse: $\eta = \{0.853/[1+(1810/I_{sp})^2]\}$.

planet missions. The SEP stage is jettisoned for most planet orbiter missions with orbit insertion accomplished by a chemical retro. Possible exceptions to this ground rule are sample return missions which can utilize the SEP stage for spiral capture and escape maneuvers at the target planet. Finally, net payload is defined for both SEP and NEP options as the injected initial mass less the sum of the propulsion system, propellant and tankage, stage subsystems, and chemical retro masses.

NEP, Nuclear Electric Propulsion

Table 5 lists the performance parameters and basic definition of the nominal 120 kw and growth 240 kw NEP systems. The nominal NEP always operates in conjunction with a CRP injection stage (Centaur or Tug) for Earth-escape maneuvers. In contrast, the GNEP employs the Earth-escape spiral mode starting from a 235 naut mile circular orbit where it is assumed to be placed by a single Shuttle launch. A typical escape spiral maneuver requires about 250 days. The basic configurations of the NEP and GNEP assume an integrated propulsion system/payload spacecraft design operating in a nonjettison mode; i.e., the nuclear electric powerplant is carried throughout the mission and employs a spiral maneuver rather than a chemical retro for target orbit capture. An alternative jettison mode is also considered for certain outer planet orbiter missions. While not necessarily recommended, it has been found that this mode of operation often allows a reduced trip time at a given payload level by eliminating the time required for spiral capture. When the jettison mode is specified, 300 kg of stage subsystems are allocated as in the case of the SEP stage definition.

The NEP system design is based on the nuclear reactor-thermionic conversion concept which has received considerable study attention. Propulsion system components include the thermionic reactor, heat rejection radiators, neutron shield, power conditioning units, thrusters, thrust vector control mechanisms, and associated integrating structure. Total propulsion system mass is 3600 kg (at 30 kg/kw) for the 120 kw option and 6000 kg (at 25 kg/kw) for the growth 240 kw option. The auxiliary power requirement is presumed to be accounted for in the power system design (drained off before PCU input). It is not specifically treated as in the case of SEP where the auxiliary power effect on performance is much more significant. A propulsion time constraint of 20,000 hr for the nominal NEP system and 30,000 hr for the growth version is specified. In terms of the method of analysis, performance data are obtained for several discrete values of specific impulse over the range 4000–7000 sec. Near-optimum values of specific impulse for a particular mission are obtained by graphical interpolation of payload curves and propulsion time constraint curves.

Performance Results

Planetary mission performance was measured, for the most part, in terms of payload capability and flight time with each propulsion option studied. A total of over 300 payload/flight time curves were generated in the analysis and are contained in

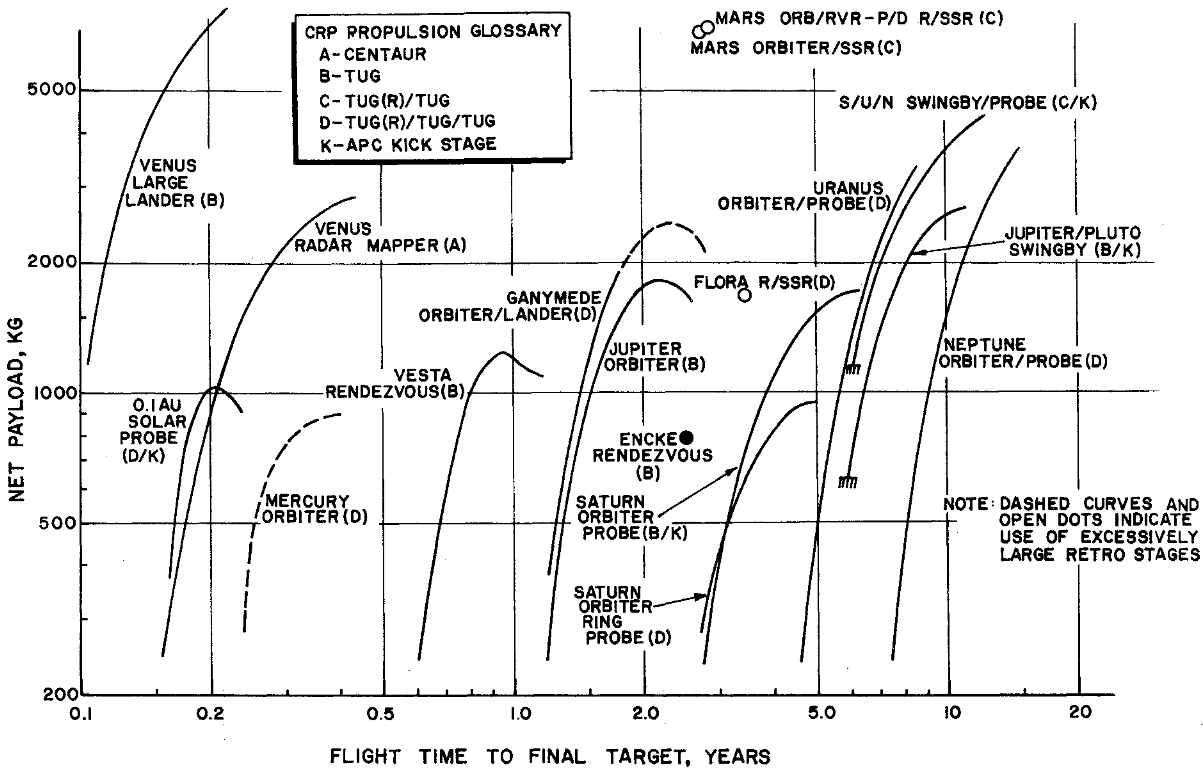

Fig. 4 Performance summary of CRP scenario.

Ref. 1. In order to summarize these results here it was necessary to find a means of editing this large amount of data. The method chosen was to develop four mission/propulsion scenarios, one for each class of propulsion; i.e., CRP, SEP, NRP, and NEP. Several groundrules used in developing these scenarios were as follows.

1) Centaur, an existing stage, could be used in all scenarios for initial low-energy missions.

2) Propulsion option selections should favor nominal rather than growth designs, whenever possible.

3) Maximum payload performance should exceed baseline payload requirements (Table 1) by a factor of 2–3 so that payload effects could be demonstrated.

4) Maximum flight time should be kept below 6 yr, if possible, due to adverse cost effects of long life spacecraft design.

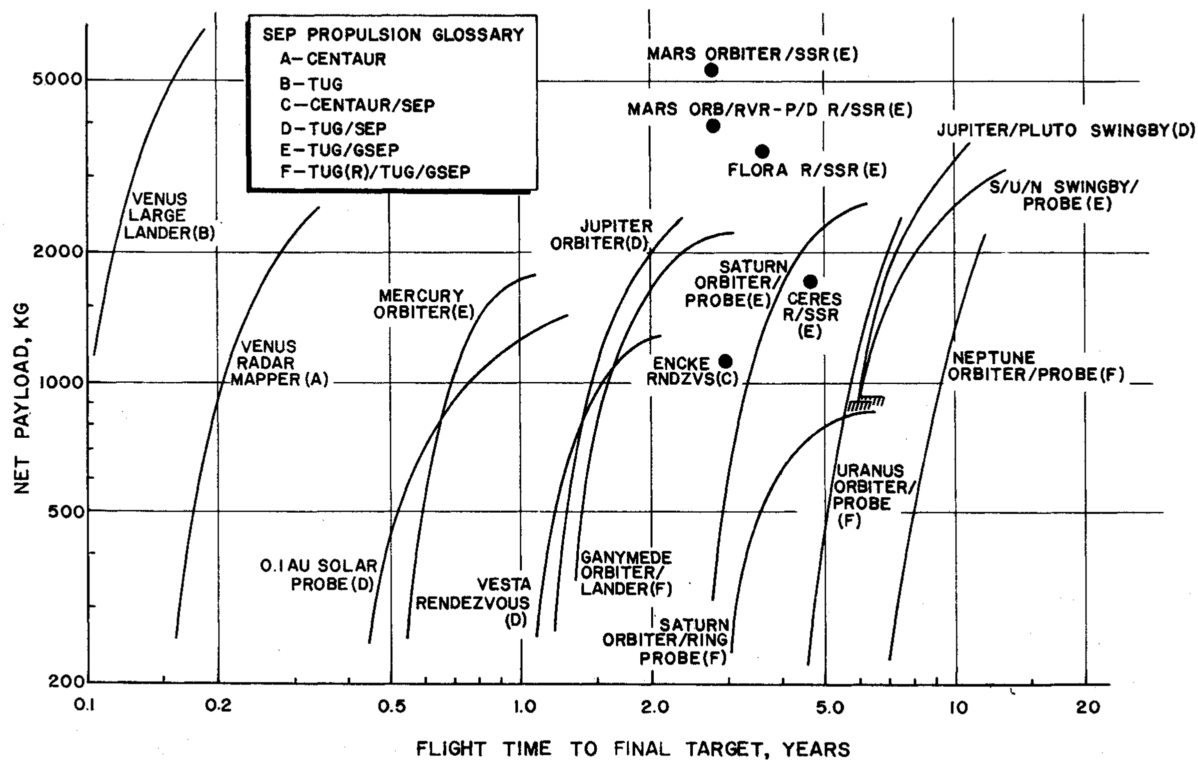

Fig. 5 Performance summary of SEP scenario.

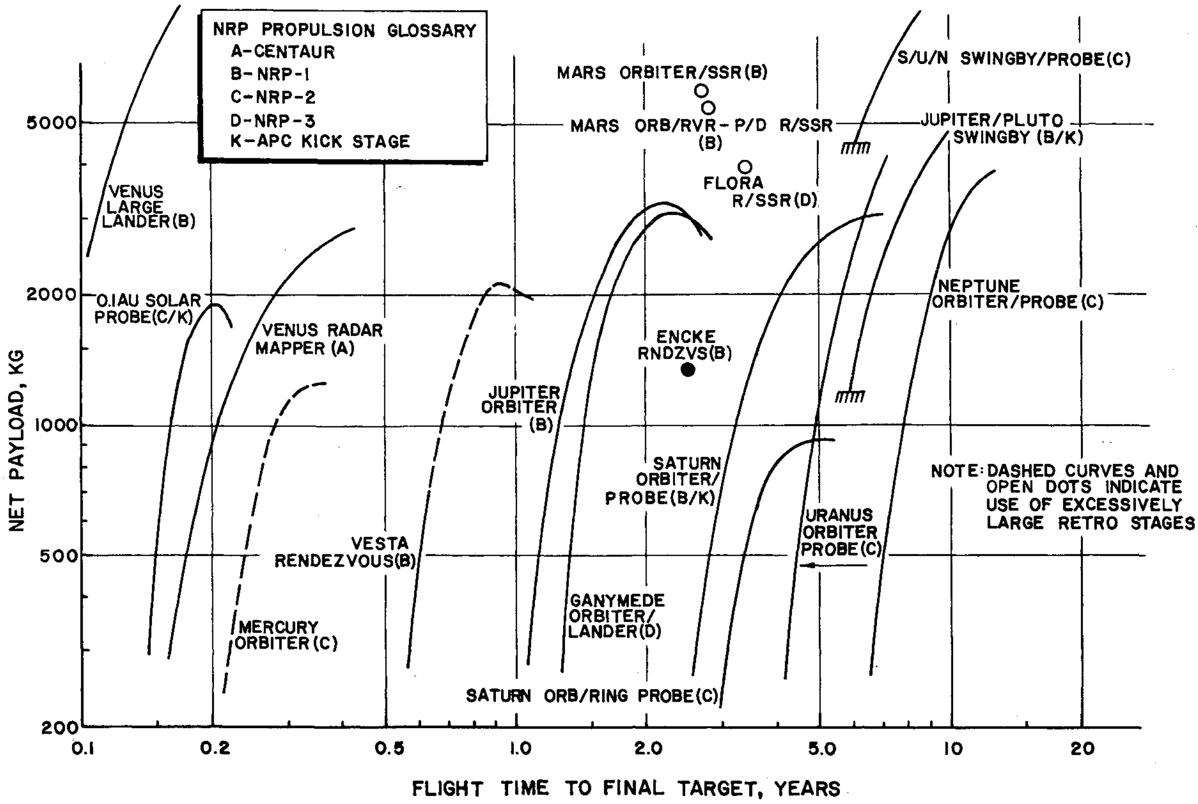

Fig. 6 Performance summary of NRP scenario.

Performance results for the CRP scenario are presented in Fig. 4 in terms of net payload and flight time. Of the 19 missions given in Table 1, 17 are possible with the considered CRP options. Halley rendezvous (No. 9) and Ceres Rendezvous/SSR (No. 12) are not possible. Single points rather than performance curves are presented for SSR flight modes, as well as Encke Rendezvous (No. 8), since staytime and/or arrival date are more meaningful parameters than total trip time for these missions. The curves are dashed (or points left open) in those cases where retro stage requirements are excessive and probably impractical.

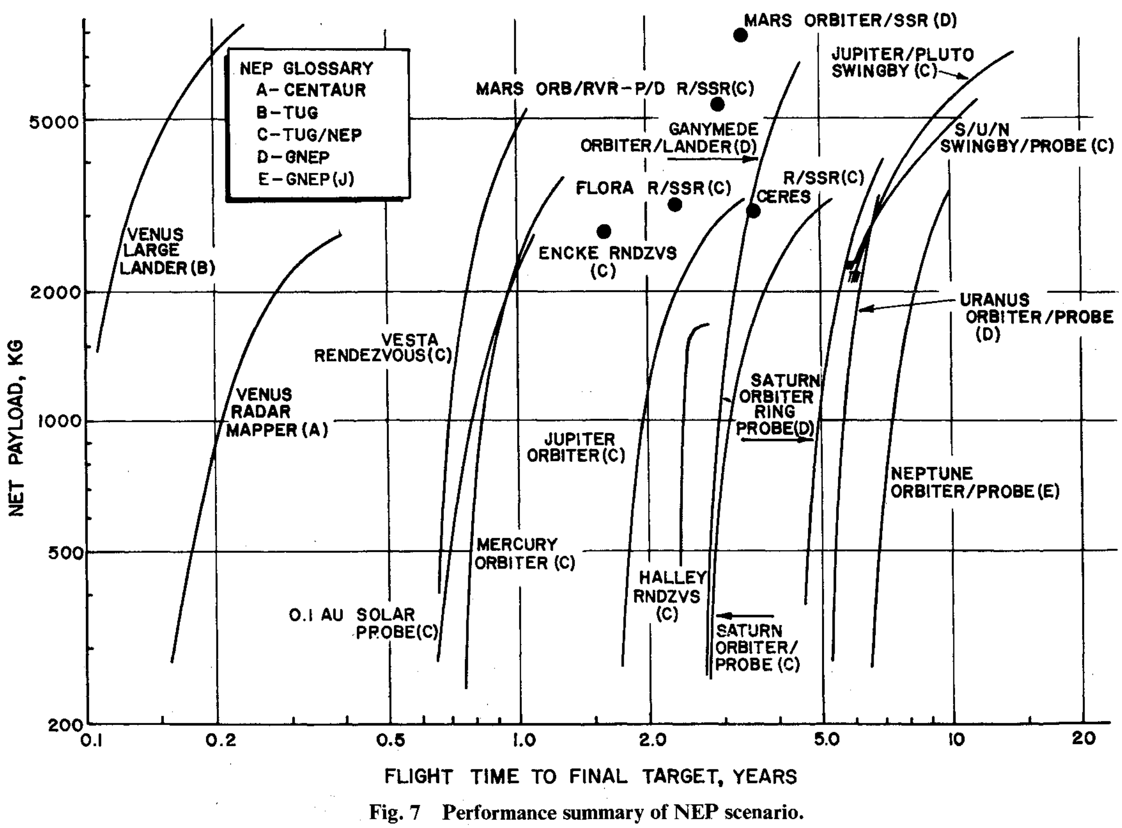

Fig. 7 Performance summary of NEP scenario.

This situation is assumed to exist when the total ignition mass at the first retro burn exceeds 10,000 kg. Even with the largest option chosen (D), it can be seen that several missions have little or no payload reserve above their baseline values given in Table 1. Also, performance of the Neptune Orbiter mission requires more than 7 yr.

The SEP propulsion scenario is presented in Fig. 5. Only the Halley rendezvous mission is not possible. SEP is not necessary for Venus missions. The Mercury Orbiter (No. 2) curve illustrates much better payload performance than in the CRP scenario. Note that there are no situations of excessive retro stage sizes in the data. The Ganymede Orbiter/Lander (No. 15) still has little performance margin, even assuming a dual Tug earth-escape configuration. Also, performance of the Neptune Orbiter mission still requires more than 6 yr at the baseline payload (950 kg).

The NRP propulsion scenario is plotted in Fig. 6. The Halley rendezvous mission (launched after 1980) still is not possible. Compared to the first two scenarios most missions can be accomplished with more payload in shorter time (curves are displaced upward and to the left). However, as with CRP, excessive retro stage sizes are a problem in some cases. Performance of the Neptune Orbiter mission continues to require more than 6 years at baseline payload.

The NEP propulsion scenario is presented in Fig. 7. All 19 missions are possible at baseline or larger payloads. Only performance of the Neptune Orbiter mission persists in requiring more than 6 yr. It is also apparent from the steepness of the performance curves that considerable payload growth margin is possible with little sacrifice in trip time. It might also be noted that in no case is more than one Shuttle launch needed per mission with NEP.

Summary

No conclusions are drawn here regarding propulsion effectiveness and program plans. To do so requires a broader consideration of such factors as budgets, spacecraft design/cost relationships, mission priorities, technical feasibility, propulsion development/recurring costs, etc. However, it may be helpful to further condense the summary performance data presented to observe trends, particularly with respect to increased payload, i.e., lower cost spacecraft design.

To this end, a brief comparative performance analysis was done on each propulsion scenario. The NEP scenario performing the mission model at baseline payloads (see Table 1) was selected as the performance reference. Against this reference each propulsion scenario's ability to perform the mission model was evaluated at four payload levels: baseline, 1.5 × baseline, 2 × baseline, and 2.5 × baseline. The reference case was known to have the best over-all performance, measured in terms of a) number of missions flown, b) total flight time, and c) number of Shuttle launches required. All other cases exhibited, by comparison, a performance degradation. This degradation was measured as a percentage of the reference performance, weighting a) missions not flown twice as heavily as total flight time b) number of required Shuttle launches 0.1 times as important as flight time. This weighting is arbitrary but hopefully indicative of the relative importance of the three questions: 1) Can a mission be done? 2) How long does it take? 3) How many Shuttle launches are required?

Results of this analysis are presented as a bar chart in Fig. 8. Percent performance degradation is exhibited as a function of payload level for each propulsion scenario presented in Figs. 4–7. As expected NEP performs best, followed by NRP, SEP, and CRP, with the last two changing order at the 2:1 payload level. Neither NEP nor NRP shows much change in performance degradation until the 2:1 ratio is reached. CRP, on the other hand, exhibits just the opposite effect, tending to level off above the 1.5:1 ratio. SEP has an almost linear performance degradation with increasing payload.

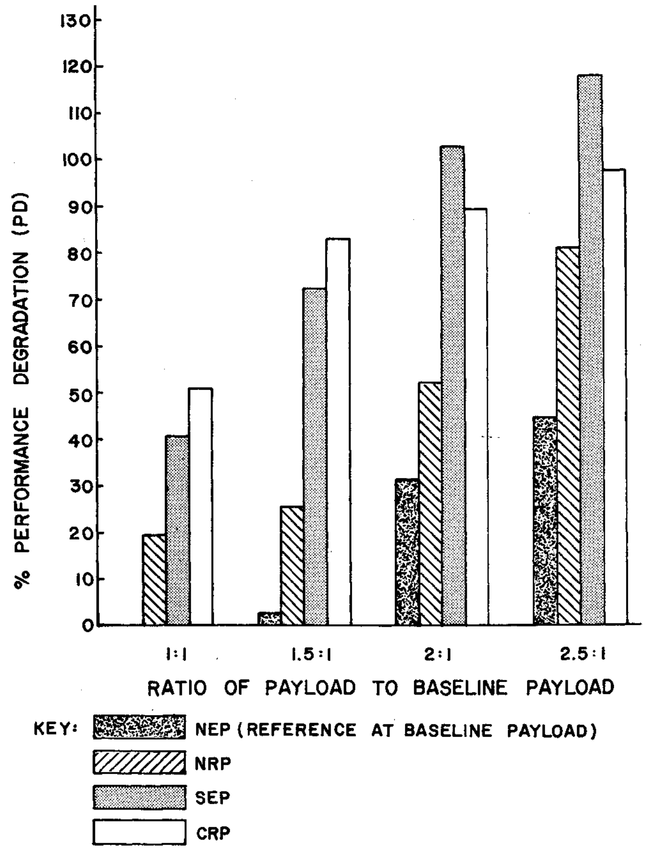

Fig. 8 Over-all planetary performance summary.

Although new performance data are included in the results presented here and in Ref. 1, it is felt that a more important contribution of the analysis has been the presentation of planetary mission performance results for a wide spectrum of advanced propulsion systems generated from a common base of ground-rules, assumptions, and design data which were developed and assembled by the APC Committee. It should be re-emphasized that it is neither the purpose nor the intent of this paper to suggest a preferred class of advanced propulsion for Shuttle-based planetary missions in the 1980's. Rather it is hoped that this publication will contribute to an improved state of awareness of the NASA advanced planning for Shuttle planetary applications.

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