8 p.

N64 23231

Code none

Cat-27

A SUMMARY OF NUCLEAR ROCKET APPLICATIONS

by

PAUL G. JOHNSON
AEC-NASA Space Nuclear Propulsion Office
Washington, D. C.

AIAA Paper
No. 64-388

1st AIAA Annual Meeting

Washington, D. C. June 29 – July 2, 1964

A SUMMARY OF NUCLEAR ROCKET APPLICATIONS

Paul G. Johnson
Reactor Engineer
AEC-NASA Space Nuclear Propulsion Office

Introduction

During the past few years NASA has spent many millions of dollars on studies of advanced missions and vehicles. Many of the studies were concerned with applications of nuclear rockets. Although some of the proposed uses seemed rather improbable, the bulk of the work has contributed to an improved understanding of nuclear rocket utility. Consequently, the time has come to make another summary of the applications picture.

The expected performance of solid-core nuclear rocket engines is reasonably well defined. The principal areas of application are also well known. The uncertainties lie in questions of timing and relative emphasis and level of effort. When will men go to planets? When will nuclear rockets be operational? How extensively will we explore the moon? How many major space flight programs can we afford to pursue at one time? This paper offers no clairvoyance on these subjects. Rather, its purpose is to show how nuclear rockets can be utilized in each of the recognized areas of space flight and to offer a few common-sense thoughts on the missions and their requirements for advanced propulsion systems.

Application Areas

One way of viewing the overall space program is shown in figure 1.[1] Across the top of the matrix are listed the three major regions of space flight: Earth orbit, lunar, and planetary. Arranged vertically are three plateaus of space flight effort: unmanned, early manned, and operational manned flight. Currently-approved programs, such as Apollo and Gemini, are marked with an asterisk.

Those areas of space flight to which nuclear rocket propulsion is applicable tend to lie in the lower righthand part of the chart. Included in these categories are the more difficult space missions, with high energy requirements and heavy payloads. This paper discusses applications of nuclear rockets to (1) manned lunar exploration beyond the initial landings and the subsequent operations using Apollo-type hardware, (2) unmanned planetary missions involving heavy payloads, and (3) manned missions to the near planets. The latter continues to be regarded as the primary justification for nuclear rockets in the future space program.

Manned Planetary Missions

Point of View

A point of view regarding manned planetary exploration is presented in the following paragraphs. First of all, manned flight to Mars and Venus is held to be the major space objective in the 1980's, requiring preparation throughout the 1970's. Manned planetary flight will be, at the time of the early expeditions, a very difficult undertaking. The advancements we make in nuclear propulsion and other technologies in the next ten to fifteen years will make these journeys possible but not easy.

One reason for the difficulty of these missions is the desire to do an adequate job of exploration with a generally acceptable level of risk. This desire will rule out trying early, high-risk, stripped-down missions. An adequate crew with adequate exploration equipment and life support will be insisted upon.

From a review of the many studies of manned planetary expeditions, the diversity of possible and proposed mission modes is apparent. Our point of view is that only reasonable extensions of capability should be counted on in the various areas of technology. Operational nuclear rockets for planetary spaceflight are considered reasonable. Certain other techniques which have been proposed are judged to be less reasonable. For example, there are mission modes which reduce energy requirements and atmosphere-entry speeds significantly but which are of about two and a half years' duration. An unreasonable burden seems to be shifted onto the life-support and human-factors areas and onto the reliability of systems which must operate for most of the trip. Nuclear propulsion brings 400-500-day trips into the reasonable range of initial weight in orbit.

The desire to do an adequate job of exploration with only moderate increases in technical capability will force the manned planetary vehicle toward excessive initial weight. Under such circumstances the performance advantages of nuclear rockets will be very important. The solid-core nuclear rocket is expected to be the most advanced propulsion system operational in the time period of early manned planetary flight. Furthermore, the useful time period for nuclear rocket application will be long enough to warrant the development of a manned planetary flight capability based on this type of propulsion.

These statements regarding nuclear rocket applicability imply certain assumptions about other advanced propulsion systems and concepts. The scope of this paper permits only a brief statement of viewpoint. Nuclear-electric propulsion is regarded as a possible contemporary of solid-core nuclear rockets, particularly if the low-acceleration systems are to be used in conjunction with high-acceleration systems for operations in planetary gravitational fields. Other advanced concepts, including gas-core nuclear rockets and nuclear-pulse rockets, are considered to be in a later time frame. At least, they are not well enough understood to permit scheduling in the same time period as solid-core nuclear rockets.

Reference Mission Mode

While trajectory innovations are still being uncovered, a reference mission mode can be defined for early manned Mars missions. By reference mode is meant that mode which, at the moment, appears to be the first choice and against which all alternate modes should be compared.

The trajectory would be an opposition-class roundtrip, characterized by a relatively short duration of 400 to 500 days. The hazards of the space environment, the psychological problems of long space journeys, and the difficulty of providing high component reliability for years of operation will cause emphasis to be placed on relatively short trip times. Furthermore, crew size tends to be proportioned to trip time. By comparison, the duration of a long-duration (conjunction-class) trip would be 800 to 1000 days.

Propulsion on the reference mission mode would be provided by three nuclear rocket stages: the first to leave Earth orbit, the second to brake into Mars orbit, and the third to leave Mars orbit. No propulsion would be provided at Earth return unless the speed of the spacecraft exceeded a reasonable speed for direct entry into the atmosphere. A reasonable entry speed is judged to be about 50,000 feet per second. Any higher capability would be very advantageous, but our opinion favors 50,000 as a value that can be counted on for the 1980 time period. This is a controversial area and deserves much attention. Chemical rocket braking is assumed because of the small weight of the entry module.

It should be kept in mind that these mode specifications refer to early missions. Later sections will discuss alternatives which may be adopted for operational missions. There is also an interesting possibility in the family of Mars trajectories which swing by Venus on either the way out or the way back. For the purposes of this report the Venus swingby is considered part of the opposition-class trajectory family.

Initial Weight Requirements

For comparisons of Mars-mission propulsion systems, the payload weights can be assumed fixed and the comparison made on the basis of initial weight in Earth orbit. Initial weight is a function of (1) payload weights, (2) mission energy requirements and (3) propulsion and vehicle performance parameters. Even with all the studies conducted in this area, payload weights and velocity increments still vary considerably from study to study. This is normal in an early definition phase. However, by specifying a reference mission mode and adopting the most generally accepted estimates of spacecraft module weights, a satisfactory comparison of propulsion systems can be made.

Build-up of Initial Weight Figure 2 may be instructive in visualizing the build-up of initial weight in Earth orbit. The example is an opposition-class trip in an unfavorable year. Two payloads are shown in the center of the chart: (1) the return payload, consisting of mission module and Earth-entry module, and (2) the exploration payload, made up of one or more Mars excursion modules and any other weight left at Mars. In this instance, the two payload weights are assumed to be nearly equal: 130,000 pounds for the return payload and 100,000 pounds for the exploration payload. However, a payload carried through three propulsion periods is more costly in initial weight than one that goes only to Mars. This is shown by the leverage factors (or multiplication or sensitivity factors, if you prefer). The exploration payload contributes only 400,000 pounds to the initial weight of the nuclear rocket; the return payload accounts for 1,700,000 pounds. For the chemical rocket, in the righthand bar, the contribution of the return payload is even more dominant. The lower specific impulse causes the leverage factors to increase and their ratio to grow in the direction of greater sensitivity to return payload.

The primary purpose of Figure 2 is to illustrate the great importance of the mission-module and entry-module weights in determining initial weight. With this orientation, we can appreciate the significance of a prediction that 6-8 men (rather than 3) are needed in the crew of a 400-500-day Mars mission or a calculation that shows the Earth retro rocket weighing three times as much as the entry module (the penalty for a limited entry speed).

Initial Weight Comparison In Figure 3 the results of several such calculations are plotted in terms of initial weight vs. launch date. The example is for a constant trip time of 420 days with 40 days stay time at Mars. Payloads are the same as in the previous figure. This figure illustrates three things: (1) that nuclear rockets are supeior in initial weight to chemical rockets by factors of 2-3, even in low-energy years, (2) that the effect on initial weight of limiting Earth-atmosphere-entry speed to 50,000 feet per second varies from 20-35% in a bad year to nothing in a good year, and (3) that the effect of a 100-second change in nuclear rocket specific impulse would be about 30% in initial weight, which is important but does not make or break the case for nuclear rockets. Although the shapes of the curves are not accurate, the 17-year cycle is illustrated and shown to be nearly eliminated for nuclear rockets when unlimited entry speed can be handled without retro propulsion. Chemical-rocket performance is much more sensitive to entry speed and energy cycles because this form of propulsion is very marginal in this application.

In addition to the obvious comparison of nuclear and chemical rockets for Mars missions, figure 3 shows the magnitude of initial weight which will result from the requirements of adequate exploration and only reasonable extensions of technology. The overall range is 1.5 to 3 million pounds in Earth orbit. A nominal value of 2 million pounds appears to be a good target for a flight in the early 1980's, using values of specific impulse and entry speed considered consistent with the time period. These initial weights are compatible with projected launch vehicles, whereas corresponding chemical-rocket weights are marginal at best. In later launch opportunities, when the desired payloads will have increased, the same basic vehicle will be applicable. Energy requirements will be lower in the mid and late 1980's and system performance will have been improved through experience.

Weight Reduction Techniques Of course, the picture is never this simple and straightforward. Many techniques can be considered for reducing the weight in orbit. The most important ones are listed in the following table along with approximate weight-reductions. The percentages apply to nuclear rocket vehicles; corresponding percentages for chemical rockets would be greater.

Initial Weight Reduction

Technique	% Weight Reduction
Bi-planet encounter	to 40 (?)
Perihelion braking	+ or -
Mars atmosphere braking	to 30
Mars elliptic capture	20 - 30 (ideal)
Hyperbolic rendezvous	5 - 25
Multi-vehicle modes	less than 10

With the exception of the bi-planet encounter, these weight-reduction techniques are either unattractive or unlikely to be usable in early missions. In a bi-planet encounter (known also as Venus kick, Venus swingby or Venus flyby), the spacecraft flies close by Venus on either the outbound or return trajectory of an opposition-class Mars round trip.[2] The gravitational deflection of Venus is planned to modify the basic path in such a way that the Earth atmosphere-entry velocity is kept below 40,000 feet per second. The total trip time is extended by no more than a few months at the most. The payoff from this technique varies from one opportunity to another and is not yet fully analyzed, but weight reductions up to 40% seem possible. Furthermore, there is reason to believe that worthwhile trajectories of this type can be found in every launch opportunity. This technique appears (at the time this paper is written) to be a candidate for the reference mission mode, especially because of the opportunity to get data on two planets with one trip and one vehicle.

The only other technique which rates special attention is Mars atmospheric braking. The payoff is large; the initial-weight reduction for use of Mars aerobraking in a nuclear-rocket spacecraft may be as high as 20-30%.[3] However, this capability will be hard to develop because of the inaccessibility of the Mars atmosphere and the telemetry problems of probe experiments. A reasonable assumption is that early flights will use propulsive braking into Mars capture orbit and that aerobraking will be introduced later. The necessary experiments in the Mars atmosphere could be among the accomplishments of the early manned missions.

The other techniques are of lesser interest. Perihelion braking, involving propulsive braking at the highest velocity point in the return trajectory, may result in either a reduction or an increase in initial weight. The outcome depends upon the nominal Earth-approach speed and the ratio of weights to be decelerated at perihelion and at atmosphere entry. Elliptic capture at Mars offers, in theory, a significant reduction in initial weight, typically 20-35%.[3] However, this advantage may largely vanish when launch-delay penalties and increased landing-system requirements are accounted for. Elliptic capture orbits seem like an interim mode at best, perhaps reserved as a last resort in case the vehicle falls short of its design performance.

Use of more than one vehicle, as proposed in the last two listed techniques, complicates the mission without much gain. In hyperbolic rendezvous the Mars craft is met in the vicinity of Earth by a shuttle vehicle.[4] Consequently, the Earth-entry module need not be carried to Mars, and the initial weight of the interplanetary vehicle is reduced by 5 to 25%. However, the total weight of the two vehicles may not be reduced. Sending two vehicles to Mars, an unmanned one by a low-energy trajectory and a manned one by a high-energy path, would lower the total initial weight by no more than 10%.[3] A convoy, with two or more vehicles traveling together, would not be intended to reduce initial weight. Its purpose would be to increase mission success probability through use of interchangeable modules.[4] Some of these multi-vehicle modes may find application in later operational phases of space transportation.

Operational Requirements

Simplified analysis, such as those used in this paper to illustrate basic relationships, generally ignore the inevitable practical considerations. Operational capability demands launch windows and allowances for launch delays in orbit; auxiliary propulsion systems must be added for mid-course velocity increments; non-optimum components and systems will be used in order to take advantage of previous developments and to minimize the number of new items needed.

Although each of these factors makes only a small addition to initial weight, the cumulative effect will not be negligible. Consequently, the weights shown in comparisons like figure 3 should be regarded as somewhat optimistic, at least for early flights. Mid-course ΔV's are in the hundreds of feet per second; launch-delay ΔV's may be in the thousands. The overall effect is likely to be a 10-20% increase in initial weight.

An example of launch delay from Mars orbit is illustrated in figure 4.[5] The problem is that an orbit about the planet, circular or elliptic, will regress at a rate of several degrees per day. If the departure date is known at the time of arrival, an initial orbit plane can be selected so that it will regress to the proper orientation for departure. However, any delay in the scheduled departure will cause the orbit inclination to be wrong for the desired heliocentric transfer orbit. Thus, a plane-change ΔV is shown in figure 4. Furthermore, if a constant Earth-return date is to be maintained, each day of delay shortens the return flight time accordingly. An additional ΔV is shown for this provision. Since the nominal Mars-departure ΔV will be in the range of 15,000-20,000 feet per second, each additional 1000 allowed for launch delay will add over 5% to the vehicle initial weight in Earth orbit.

Propulsion System Requirements

The propulsion of a 2-million-pound space vehicle requires a nuclear rocket engine of several thousand megawatts. Figure 5 shows the major factors which, from a parametric performance standpoint, influence the choice of engine power. Initial weight in Earth orbit is plotted against total reactor power in the Earth-departure propulsion system. A 1983 opposition-class trip is used for illustration. The important fact is that the initial weight is about 2 million pounds.

The curves of constant unit-engine power show the penalty associated with clustering large numbers of low-power engines. Optimum first-stage thrust-weight ratio is almost constant at about 0.2. Therefore, the number of engines is nearly inversely proportional to unit thrust. Use of four 2000-Mw engines rather than two 5000-Mw engines results in an increase in vehicle weight of about 15%. The difference between 3500 and 5000 Mw is less than 5%.

The selection of unit-engine power cannot be made solely on the basis of Earth-orbit propulsion. The usefulness of the thrust level in later stages of the Mars craft must also be considered, as well as in other possible applications in the space program. Figure 5 is based on single engines of the indicated power being used in the Mars-arrival and Mars-departure stages. Since Mars departure gross weight is about 300,000 pounds, a power of 3500 or 5000 Mw is well above optimum. However, in the interest of minimizing the amount of engine development, use of a single engine power may be desirable. The data in Figure 5 include the penalties for using only a single engine power in all stages, thereby narrowing the differences in resultant initial weight. From these considerations, it is likely that the nuclear rocket propulsion systems for a manned Mars expedition would consist of (1) a cluster of 2 or 3 engines in the Earth-departure stage, with unit-engine power being about 5000 Mw (250,000 pounds thrust), (2) a single engine of the same rating for Mars-orbit attainment, and (3) another single engine for Mars departure. Because of the weight penalties and operational problems associated with aftercooling and restart, the reuse of an engine is not included in the reference mission mode. Clustering, however, appears to be a necessity, although close clustering may not be required. In fact, the clustering of tanks, each with its own engine, may be the most easily developed and utilitarian configuration.

Manned Lunar Missions

Point of View

Our point of view regarding manned lunar missions is given in this section. This is a hard area to clarify because we must deal with the real worldof existing stages and approved-program schedules. The objective of this paper is to show how nuclear rockets could be used to advantage in several possible phases of post-Apollo lunar exploration.

Manned lunar operations can be divided into four phases: (1) initial landings and orbital reconnaissance, (2) early exploration, (3) extended exploration, and (4) exploitation of lunar resources. The first two will be carried out using Apollo-type hardware. Personnel transport would be by means of the Apollo/LEM. Logistic support would use a cargo version of LEM or, possibly, a new direct cargo lander sized to go on the all-chemical Saturn V. If the initial exploration shows reason to expand the scale of lunar operations, a period of extended exploration would be entered, calling for larger accumulations of equipment on the moon and higher launch rates. Later, again depending upon what is found on the moon, a phase of exploitation may be entered. The fondest hope is that a source of rocket propellants will be found.

For lunar missions beyond early exploration, there may be a demand for development of a new Saturn V third stage or an orbit-launch stage. In this time period, probably the mid to late 1970's, the Saturn V could be an uprated version. For extended exploration a logical set of new developments would be an expendable nuclear-rocket-propelled third stage on Saturn V and appropriate lunar logistic stages. A further enlarged rate of traffic, as in an exploitation phase, would call for reduction of transportation costs. Reusable surface-to-orbit shuttles would be used at each terminal, and a reusable ferry would carry both men and materials between Earth and lunar orbits. The ferry could use nuclear rocket propulsion, provided the necessary engine lifetime and restart characteristics can be achieved.

As we consider and determine the course of lunar operations, we must also consider the desire to gain experience with an advanced propulsion system. Since nuclear rocket propulsion will be used later, it makes sense to introduce it early--as early as the capability becomes available at no penalty in cost effectiveness. On this basis, it is reasonable to expect that all new major stages will be nuclear rocket stages. Of course, this statement applies primarily to large space stages and probably not to stages which operate to or from a surface, have small payloads (under about 100,000 lb), or start much below orbit. Such a recommendation is similar to that which led to the introduction of hydrogen-oxygen upper stages in the Saturn family. Early experience would have a large payoff in reliability and solution of operational problems.

Lunar Payloads

The pertinent payload numbers for various Saturn V third stages are shown on figure 6. Weight landed on the moon is plotted against vehicle weight at third-stage cutoff, i.e., weight boosted to lunar transfer velocity. Nuclear propulsion is used only up to lunar transfer velocity. The upper line shows landed payload for an unmanned logistic carrier. The next lower line is for a manned craft with return capability. The landed payload is less because of added shielding and instrumentation weights in the manned nuclear stage.

The amount of landed payload required to give 3-man return capability is shown by the dashed line, leaving the shaded area to represent the excess capability of the higher performance systems. It should be borne in mind that this excess payload may not be useful in just any way desired. Volume restrictions or other configuration constraints may make this payload capability less effective than a comparable weight on a strictly cargo flight However, most of the weight should be useful for supplying provisions for the crew, thereby providing added staytime. A landed weight of 650 pounds per 3 man days of staytime extension has been estimated for this situation.[6]

Along the abscissa are spotted the payloads of several versions of a three-stage Saturn V. The basic, all-chemical vehicle is shown to put about 95,000 pounds onto a lunar trajectory, resulting in 28,000 pounds of cargo landed. Substituting a nuclear orbit-launch third stage, labeled S-NA, for the S-IVB stage raises the cargo weight to 37,000 pounds. With the standard first and second stages of Saturn V and sub-orbit start of the nuclear rocket stage, labeled S-NB, the landed cargo weight is 47,000 pounds. Another combination is an uprated Saturn V and appropriate nuclear rocket third stages. If the all-chemical uprated vehicle is capable of injecting 134,000 pounds onto lunar transfer orbit, a landed payload of 45,000 pounds may be attained. Almost any other amount of uprating could have been assumed, but the corresponding landed weights would also lie along the performance line in figure 6. With a nuclear orbit-launch stage the uprated Saturn V is estimated to put down 60,000 pounds of lunar cargo.

For the Apollo/Direct mode, wherein three men are landed directly on the surface without lunar orbit rendezvous, the S-NA on a standard Saturn V has only marginal capability. The S-NB on the standard lower stages would do about as well as an all-chemical uprated Saturn V. Substitution of an orbit-launch nuclear stage for the modified S-IVB stage (assumed to have an enlarged tank and new engine) would provide very large cargo weights to be landed on the same flight as the crew. The total number of launches to support a lunar station should be decreased accordingly.

Propulsion System Requirements

All of the manned lunar missions are assumed to involve nuclear-stage weights of 250,000-650,000 pounds, i.e. one or two low-orbit payloads of a standard or uprated Saturn V. Consequently, the desired thrust of the nuclear-rocket engine is 50,000-200,000 pounds (reactor power, 1000-4000Mw). For propulsion of a spacecraft which is a single Saturn V payload the desired thrust is 50,000-125,000 pounds (1000-2500 Mw).

Comparing these numbers with the requirements for manned planetary missions, we see that the two ranges of desired thrust are centered on different optimum values. A middle range exists in which a single thrust level could serve either application, but both systems would be somewhat off-optimum. A 3500-5000 Mw engine would be heavier than optimum for a Saturn V third stage, especially for a standard Saturn; a 1000-2500 Mw engine would require more clustering than is desirable for a manned Mars mission, especially if our estimates of initial weight turn out to be low. The weight penalties associated with excess engine weight or low thrust would reduce the nuclear payload advantage, but such a compromise cannot be ruled out. The final selection of engine power(s) will depend on many factors in addition to payload optimization.

Unmanned Interplanetary Missions

Figure 7 presents a summary of interplanetary payload performance of Saturn V based systems.[7] These are for one-way trips, either into a planetary orbit or a flyby to the cited location. The solid bars show the capability of an all-chemical Saturn V with a cryogenic fourth stage; the striped bars extend to the payload of a Saturn V with a nuclear third stage. Nuclear payloads are shown to be 50-80% higher than all-chemical payloads. This increased capability could also be converted into shorter trip times.

Three questions need to be answered with regard to the use of nuclear-rocket propulsion in unmanned missions: (1) How much interest is there in such high-payload missions? (2) Will there by an operational nuclear-rocket stage in the time period when these high-payload missions should be flown? (3) Are there more appropriate advanced propulsion systems for such missions? Obviously, the answers are yet to be worked out. The area of unmanned support of manned planetary missions may be the most likely place to find a requirement for 50,000 pounds in Mars orbit. Such a spacecraft could do a good job of determining atmosphere characteristics, surface conditions and the space environment. Adequate power (e.g. SNAP 8) and communications equipment could be carried to send back useful quantities of engineering data. Long-lived automated laboratories could be placed on the planetary surface.

The question of availability implies that a nuclear-rocket stage would not be specially developed for unmanned missions; the question regarding alternate propulsion concepts implies that something like a nuclear-electric stage would be developed solely to perform unmanned missions. The close relationship between manned lunar missions and these unmanned interplanetary missions is obvious: they both involve upper stages on the Saturn V. Therefore, a nuclear-rocket third stage developed for lunar exploration could also serve to send many extra tons of instruments to these other interesting places. So could a nuclear-electric-propelled spacecraft. In fact, the combination of high-and low-acceleration nuclear systems would be superior to either used separately.

The answers to these questions will probably lie in the success we have in perfecting nuclear propulsion systems. Planning of the overall space program, including both manned and unmanned missions to the moon and planets and in interplanetary space, will eventually lead to recommendations regarding development of new stages and advanced propulsion systems. In the meantime performance estimates, such as figures 6 and 7 and corresponding data for other systems, must be kept up to date. Such data, supported by demonstrations of the validity of the predictions, will help program planners to phase in new systems and evolve a versatile stable of vehicles.

Concluding Remarks

This summary of nuclear rocket applications should make clear the following points:

(1) Nuclear rocket propulsion offers significant advantages in the next round of space exploration missions beyond Apollo.

(2) Early introduction of nuclear rocket propulsion, as soon as the capability becomes available, will be a great aid to later applications. Therefore, nuclear rockets should be seriously considered for all new major vehicle stages, except possibly those for surface use, small payloads (under 100,000 lb.) or very high thrust (second stage boost).

(3) Manned planetary missions, in particular, will benefit from nuclear rockets. The missions will be very difficult in any case; without nuclear rockets the initial weights in Earth orbit would be unreasonable.

(4) Manned lunar missions also stand to benefit from nuclear rockets, at least in later phases of exploration or exploitation. The number of launches can be reduced, and direct landing flights can carry both men and cargo.

(5) A propulsion system and stage developed for manned lunar flight could also serve in (a) unmanned interplanetary missions, wherever very large payloads are needed, and (b) the planetary-departure phase of a manned Mars or Venus mission, with some refinements made to handle long term space storage.

The exact way in which nuclear rockets are used in the space program will depend upon many factors other than payload comparisons. One of the primary influences will be the relative emphasis placed on manned operations in Earth orbit, on the moon, and to the near planets. An equally strong influence will be the experience in the nuclear rocket development program, which will indicate the possible characteristics and availability dates of the propulsion systems. Mission studies will continue to explore the various application areas. Capabilities and modes of operation will become more clearly defined. In the meantime, our point of view is that solid core nuclear rockets will power the next generation of manned space vehicles.

References

1. Hyatt, Abraham: Planning for the Future Goals of NASA. Proceedings of NASA-University Conference on the Science and Technology of Space Exploration, Chicago, Ill., November 1-3, 1962, v.1, p. 15, NASA SP-11.

2. Sohn, Robert L: Summary of Manned Mars Mission Study for NASA Ames Research Center. Presented at Manned Mars Mission Symposium, Marshall Space Flight Center, Huntsville, Ala., Jan. 28-30, 1964, TRW Space Technology Laboratories Report 8572-6009-RU-000.

3. Luidens, Roger W.: Manned Mars Landing Mission via High-Thrust Rockets. Mission Examples. NASA TM X-50123, July 1963.

4. Ehricke, Krafft A.: A Study of Early Manned Interplanetary Missions. General Dynamics/Astronautics Report GD/A-AOK64-002, 1/28/64.

5. Bird, John D., Thomas, David F. and Collins, Robert L.: An Investigation of a Manned Mission to Mars. Trajectories and Mission Analysis. NASA TM X-50122, July, 1963.

6. Nuclear Lunar Logistics Study. Final Report Vol. II, Lockheed Report LMSC-B061600 NSP 63-129, December 1, 1963 (Confidential, Restricted Data).

7. Nuclear Interplanetary Mission Study. Summary Report. Lockheed Report NSP 64-34, March 31, 1964.

SOME SPACE EXPLORATION POSSIBILITIES

MISSIONS	REGION		
	EARTH ORBIT	LUNAR	PLANETARY
UNMANNED	1 1958 — UNMANNED SATELLITES SCIENTIFIC SATELLITES * Small Special Purposes * Orbiting Observatories APPLICATION SATELLITES * Communication * Meteorology * Navigation * Engineering Research	2 1962 - 68 LUNAR PROBES * Ranger * Surveyor INTERMEDIATE SPACE PROBES	3 1962 — DEEP SPACE PROBES * Mariner Voyager Search for Extraterrestrial Life Out of Ecliptic Gravitational Experiment Outer Planets and Their Satellites Leave Solar System
MANNED DEVELOPMENTAL	4 1962 - 68 MANNED SATELLITES * Mercury * Gemini Maneuvering Reentry Interim Orbital Labs.	5 Before 1970 MANNED LANDING * Apollo Lunar Logistic System # # - Unmanned	6 After 1975 MANNED EXPEDITIONS Mars Landing Venus Reconnaissance Search for Life on Planets No Authorized Programs Yet
MANNED OPERATIONAL	7 After 1968 ORBITAL OPERATIONS MANNED ORBITING LABS. Operational Ferry Vehicle Recoverable Boosters Engineering Experiment and Development No Authorized Programs Yet	8 After 1970 LUNAR STATION Scientific Observations Lunar Explorations No Authorized Programs Yet	9 After 1980 PLANETARY OPERATIONS MARS STATION ADVANCED MANNED EXPEDITIONS Jupiter Satellites Mercury Others No Authorized Programs Yet

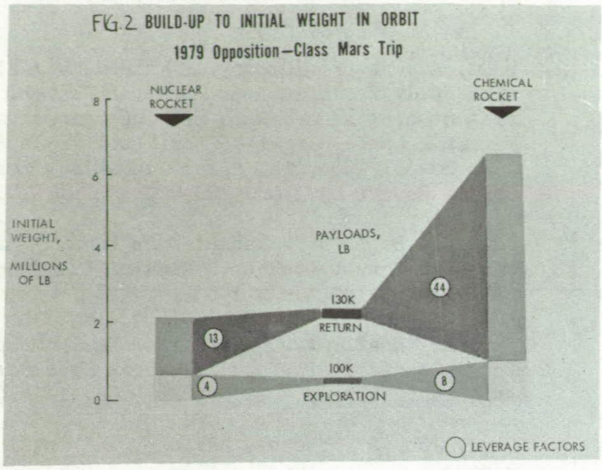


FIG.2 BUILD-UP TO INITIAL WEIGHT IN ORBIT
1979 Opposition—Class Mars Trip

FIG.3 INITIAL WEIGHTS FOR MARS MISSIONS
420-DAY ROUNDTRIP, 40-DAY STOPOVER

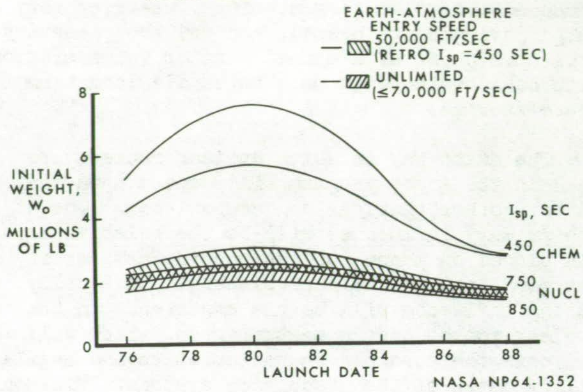


FIG.4 PENALTY FOR LAUNCH DELAY
From Circular Mars Capture Orbit
ALTITUDE, 1000 N. MI.; INCLINATION, 30°

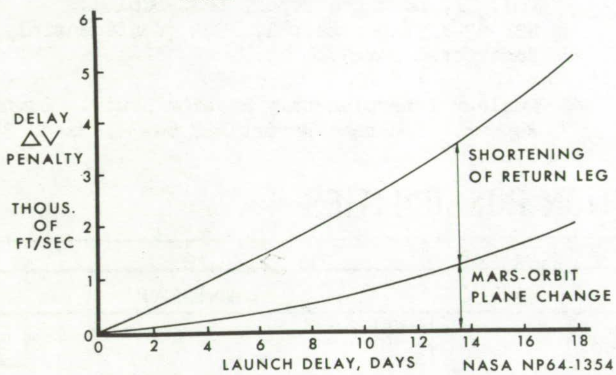


FIG.5 EFFECT OF EARTH-DEPARTURE ENGINE CLUSTERING

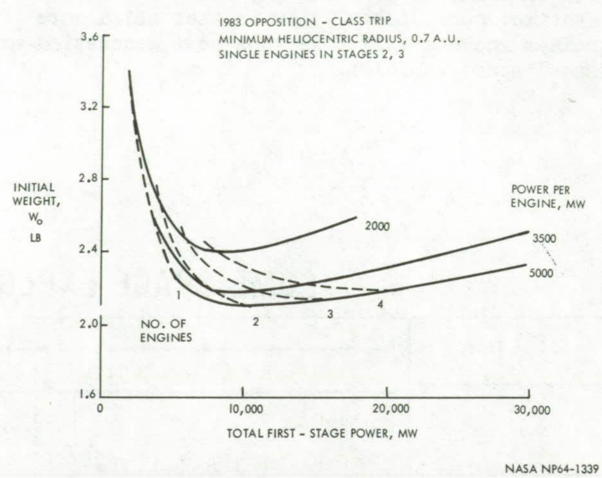
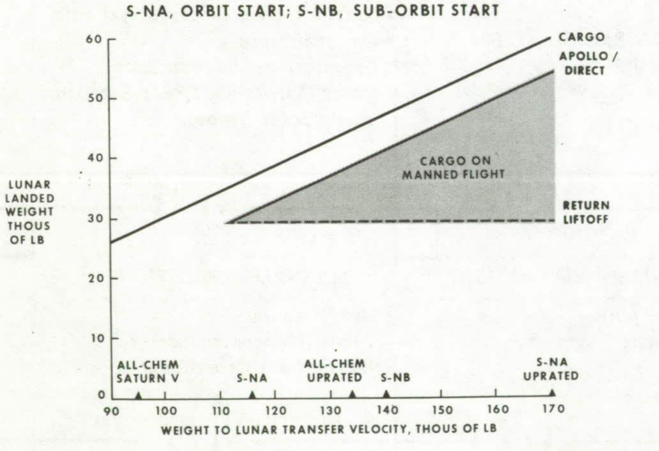


FIG.6 LUNAR LANDED PAYLOADS
S-NA, ORBIT START; S-NB, SUB-ORBIT START



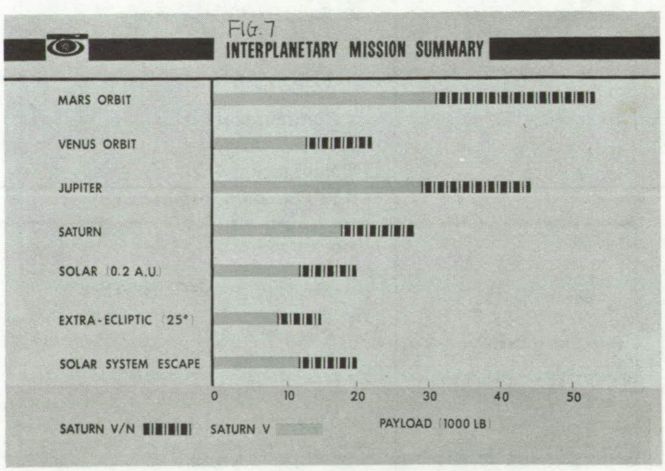


FIG.7 INTERPLANETARY MISSION SUMMARY