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A PROPULSION ORIENTED STUDY OF MISSION MODES FOR MANNED MARS LANDING

By G. R. WOODCOCK
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NASA

George C. Marshall Space Flight Center, Huntsville, Alabama

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ABSTRACT

Results are given of a systems analysis of manned Mars landing missions for a variety of mission modes, including chemical, nuclear, and electric propulsion, and aerodynamic braking. Consistent ground rules and assumptions were used. The baseline mission requires 450 days for execution and places 4 men on the surface of Mars for 20 days. Advanced missions are discussed. Each mission was analyzed assuming first Saturn V, and then a large reusable Post-Saturn vehicle to be available, in order to provide a comparison. A cost advantage was found for the reusable Post-Saturn for all but very minimal planetary programs. Of the available technologies, the graphite nuclear rocket was found generally preferable to other systems for mission propulsion. Advanced nuclear propulsion, such as ORION, was found to have great potential for advanced missions.

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SUMMARY

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INTRODUCTION

Continuation of a vigorous national space program will incorporate planetary exploration as part of the overall exploration and exploitation of space. At the present time, the state of space technology limits planetary exploration to unmanned probes. It is our present belief, however, as exemplified by the vigorous manned space flight programs of NASA, that the objectives of space exploration can be fully met only by manned missions. Manned missions to the planets, and especially to Mars, have been the subject of a number of advanced systems studies. The intent of these studies has been to bring into focus the general character and scope of such missions, to identify the technical advancements needed to make the missions practical, and to ascertain roughly the cost of the missions and in what time frame they might be feasible.

With respect to Mars, three general categories of missions have been identified. The first category, manned flyby missions, appears to be attainable with modest extensions of Saturn/Apollo technology. The second category, early manned stopover and landing missions, requires substantial technological advance. The third category, which could be considered large-scale exploration, can be only roughly forecast with today's state of knowledge. Many analyses of manned stopover missions in the second class have been performed by a number of organizations, using a wide variety of groundrules, assumptions, and mission modes. Several propulsion concepts have been incorporated into these studies.

Not surprisingly, conflicting conclusions and recommendations have resulted. Since one of the principal benefits of advanced systems studies is to identify those areas of technology that will have maximum payoff in terms of extension of our space capability, it is felt that although manned stopover planetary missions are many years in the future, the time is ripe for attempts to resolve some of the conflicts by analysis of a variety of mission modes with a common set of groundrules and assumptions.

This study was initiated in November 1964 to respond to a request from Dr. Mueller, Associate Administrator for Manned Space Flight, to Dr. Koelle, MSFC Future Projects Office Director.

The purpose of this paper is twofold: First, with common groundrules and assumptions, to compare, on the basis of cost and performance criteria, manned planetary stopover missions utilizing several modes and propulsion systems. The second purpose is to present an assessment of the present state of the art and probable development requirements of the propulsion systems considered.

The author wishes to express appreciation to those who contributed to the analysis which formed the basis of this report. This especially includes Robert G. Voss and Terry H. Sharpe of the Future Projects Office, who did the cost analysis, and Ronald J. Harris of the Propulsion and Vehicle Engineering Laboratory (Nuclear Group), who provided basic space vehicle mass data. Sincere appreciation is also extended to Dr. Harry O. Ruppe who provided analysis of the electric propulsion systems, and to Dr. H. Hermann Koelle, who provided overall direction of the effort.

PRINCIPAL CONCLUSIONS

1. A fully reusable Post-Saturn class vehicle in the one million pound payload class will be sound investment toward an efficient manned planetary exploration program, regardless of which mission modes are selected. This requirement approaches the "mandatory" status, in case several manned planetary orbiting and landing missions are planned, using chemical propulsion only.

2. Continued development of the solid-core nuclear rocket engine in the NERVA II class is, at this time, identified as the best strategy toward providing

reasonably effective propulsion for manned planetary missions. It appears as good if not better than, from a cost and reliability standpoint, all competitors with the exception of high-thrust advanced nuclear propulsion concepts. The future of the latter concepts cannot be clearly predicted at present.

3. An initial Mars landing attempt could be performed with Saturn V as the launch vehicle, provided that a nuclear rocket system or its performance equivalent is available, and that extensive orbital operations are used, and that the launch of one Mars ship per launch window is considered satisfactory.

4. Orbital launch operations will be required for a reasonable planetary program, regardless of concepts selected. The single exception in this report, Post-Saturn/ORION, appears true only for the baseline mission, and definitely not for more ambitious missions.

5. Research toward feasibility of advanced nuclear propulsion should be continued aggressively and broadened in scope to include all concepts that presently appear promising. These include, but are not necessarily limited to: nuclear pulse; gaseous core fission; MHD-electric; and fusion-based systems. If such systems can be developed, they will provide a major breakthrough in space transportation systems, with the nuclear pulse systems being the leading contender at this point in time.

These conclusions are discussed further in the "Results" section of this report.

SUMMARY OF GROUND RULES AND CRITERIA

Major categories of cost for the cost analysis were as follows:

1. RDT&E costs for spacecraft

2. Manufacturing costs for spacecraft

3. Cost of transportation of spacecraft to Earth orbit

4. Earth orbit operations burden rate

5. Share of launch vehicle RDT&E cost assignable to the planetary program: For Saturn V uprating, 50%; and for Post-Saturn development, 50% (if applicable)

6. Development of orbital launch operations, if required

In order to demonstrate the change in relative importance of R & D cost and operational cost with the magnitude of planetary mission activity, costs were determined for three cases; 1 mission, 10 missions, and 100 missions. Although it is certainly not reasonable that a large number of identical missions to the same destination would be flown, this assumption was made to simplify the analysis. It was assumed that the baseline mission could be correlated with a variety of different planetary missions in a direct manner. Validity of this assumption is briefly discussed in a later section of this report.

Analysis of orbital operations requirements was based on a rapid estimate, parametric method originated by Dr. H. H. Koelle.[1] Each mission type was analyzed for two cases: first, assuming the use of uprated Saturn V vehicles, 160 tons payload into 485 km orbit; and then the use of large, reusable, Post-Saturn[2] vehicles, 455 tons payload into 485 km orbit, for delivering space vehicle systems to Earth orbit. Orbital operations cargo flights were assumed to use uprated Saturn V's, and passenger flights were assumed to use a reusable orbital transport.[3] The following cost and reliability assumptions were employed:

1. Mission reliability of Saturn V: 0.92

2. Mission reliability of Post-Saturn: 0.95

3. Mission reliability of reusable orbital transport: 0.98

4. Orbital assembly and preparation: take place over a 90-day period prior to orbital launch.

5. Space vehicle reliability: 0.98 per propulsive stage; 0.95 per aerodynamic braking maneuver.

6. Probability of meeting orbit launch window = $(\text{Orbital operations reliability})^{0.25}$

7. Orbital operations reliability versus number of missions in program:

	1 Mission	10 Missions	100 Missions
per assembly operation	0.96	0.97	0.98
per tanking operation	0.98	0.985	0.99

8. Unreliability of orbital operations is assumed largely compensated for by additional launches and is considered in orbital mass and cost burdens. However, per assumption 6, this is assumed to be not entirely successful.

9. Reliability of spacecraft life support and of Mars Excursion Module (MEM) is not considered, as these items enter identically into every mission.

10. Launch facilities cost burden: Some of the missions analyzed required very high launch rates in order to complete the orbital assembly and launch within 90 days. Cost burdens for extra launch facilities was added to launch vehicle RDT&E as follows:

a. For each Saturn V launch in excess of 8 in 90 days: $ 25 million

b. For each Post-Saturn launch in excess of 4 in 90 days: $ 75 million

11. RDT&E and production costs are given in Table III.

Mission modes were compared on the basis of several criteria. The criteria were typical of those used in value analysis and were intended to indicate relative attractiveness of the various mission modes. The most important criteria were:

1. Total cost per man-day on Mars

2. Total R&D expenditure up to and including first mission attempt

BASELINE MISSION AND MISSION MODES

Baseline Mission. The baseline mission was chosen to exemplify a first-generation stopover at Mars. Although other planets might have been considered, Mars was chosen on the basis of its being of principal interest at the present time, and because it was desired to restrict the analysis to a single target planet for the sake of brevity. Characteristics of the baseline mission are given in Table I.

TABLE I

BASELINE MISSION CHARACTERISTICS

Mission Duration	450 days (approx.)
Mission Year	1984 opposition
Crew Size	8 men
Crew to Mars Surface	4 men
Surface Staytime	20 days
Mission Module Mass	38.6 tons
Mars Excursion Module Mass	36.4 tons
Earth Reentry Module Mass	6 tons
Life Support Expendables	22.8 kg/day
Midcourse Corrections: Outbound	100 m/sec
Inbound	100 m/sec
Maximum Allowed Earth Entry Speed	15 km/sec
Mars Excursion Module Mass Breakdown:	
Initial Mass	36.4 tons
Descent Propellant	6.68 tons
Dry Weight for Descent	5.46 tons
Life Support and Equipment	6.82 tons
Ascent Liftoff Mass	17.4 tons
Ascent Propellant	12.57 tons
Dry Mass	4.19 tons
Payload Returned to Earth Return Vehicle	0.68 tons

System performance assumptions were as follows:

1. Cryogenic (LOX/LH_2) chemical propulsion: C=4300 m/sec (exhaust velocity)

2. Solid-core nuclear propulsion: C=7850 m/sec

3. Nuclear pulse (ORION) propulsion: C=18,150 m/sec

4. Storable chemical propulsion: C=3240 m/sec

5. Electric propulsion: 9 kg (mass)/kW jet; exhaust velocity variable in optimum manner.

6. Aerodynamic braking at Mars: Aero-braking structure and heat shield mass 18.5% of useful mass delivered to Mars orbit, 100 m/sec velocity requirement for circularization maneuver on storable chemicals following aerobraking.

(NOTE: It is assumed that Earth approach speed in excess of 15.2 km/sec is reduced to that speed by propulsive braking with storable chemical propellants at an Isp of 3,240 m/sec, except for mission modes employing electric or ORION propulsion, in which case the primary propulsion system is used.)

Excursions on the baseline mission included:

1. Missions for year of opposition 1993, a representative "difficult" year.

2. Return to a synchronous Earth orbit rather than direct entry (considered only for electric and nuclear propulsion systems).

3. The use of multiples of this baseline mission to represent more ambitious space programs.

The baseline mission profile consists of four interplanetary transfer maneuvers plus Mars excursion and auxiliary maneuvers which occur as noted below. Data were obtained from several sources.[4, 5, 6, 7]

Earth Departure. For high-thrust systems, a single propulsive period to transfer injection is employed. After one-third of the transfer coast period has been completed, expended life support stores and the Earth escape stages are jettisoned and a storable propellant midcourse correction is executed; a 100 m/sec velocity increment is provided. Low-thrust systems (electric propulsion) are boosted by a high-thrust system (cryogenic chemical or solid core nuclear) to local (geocentric) parabolic velocity, following which low-thrust propulsion is initiated. Although a very elementary analysis will show that high-thrust boost to local parabolic velocity is not optimal with regard to minimizing initial mass, the deviation from optimum is in general small.[8] Low-thrust systems are assumed to make course corrections continuously; no penalty is assumed.

Mars Arrival. Mars arrival may employ a high-thrust propulsive maneuver to attain a circular orbit about Mars, a low-thrust spiraling into circular orbit, or aerodynamic braking in Mars atmosphere. In the last case, the vehicle trajectory enters the Mars atmosphere in such a way that it will skip out again after having achieved the required velocity decrement. After skip-out, storable liquid rocket propulsion is used to circularize into a stable orbit about Mars. Aerodynamic lift modulation and sophisticated onboard guidance are implied.

Mars Surface Excursion. The Mars Excursion Module separates from the interplanetary spacecraft, with the descent crew aboard; retrofire (storable or solid propellants) initiates controlled entry. Atmospheric braking is followed by a parachute descent, with rocket landing similar to that of the LEM. The descent crew is not provided with surface mobility other than walking in a pressure suit. Limited capability is assumed for carrying instruments down and samples up. The MEM serves as a shelter during surface stay; it is assumed to consist of a descent stage (which incorporates the aerodynamic heat shield, parachutes, and landing rockets), and two ascent stages employing storable propellants. The ascent velocity increment of roughly 5 km/sec nearly precludes single-stage ascent with storable propellants. The ascent terminates with rendezvous with the interplanetary spacecraft and transfer of the crew and samples.

Mars Departure. Prior to Mars departure, spent life-support stores, spent stages, and the spent MEM stage are jettisoned. For high-thrust systems, Mars departure takes place with a single propulsive maneuver to transfer injection. Low-thrust systems must spiral out to escape and transfer. One-third to one-half of the way along the transfer, provision is made for a 100 m/sec midcourse correction with storable propellants. If Venus swingby is to be employed, the initial injection is to Venus transfer. One hundred m/sec course corrections are provided between Mars and Venus and between Venus and Earth. The swingby itself is non propulsive. No scientific payloads (probes, etc.) were assumed for the Venus encounter.

Earth Arrival. For those modes incorporating direct entry into Earth's atmosphere, propulsive braking was assumed as needed to reduce entry velocity to no greater than 15.2 km/sec. If nuclear pulse propulsion or electric propulsion were used for previous maneuvers, they were assumed available for Earth braking. Other mission modes used storable chemical propulsion for the braking maneuver. Electric propulsion and nuclear pulse missions, which entered a high-altitude (24-hour) orbit upon Earth arrival, were also studied. Electric propulsion spiraled into this orbit, whereas the (high-thrust) nuclear pulse used a three-impulse maneuver. [9]

Selected Mission Modes. Table II summarizes the baseline vehicle concepts and mission modes with which this study was begun. Modes 13 and 14 were not analyzed in detail because a means of even roughly optimizing the mission

TABLE II

MISSION/PROPULSION SYSTEM MATRIX

		Earth Orbit Departure				Mars Orbit Capture					Mars Orbit Departure					Earth Return				
																Direct Entry (50,000 ft/sec Max. Speed)			Capture in 24 hr. Orbit	
																Power Braking				
Mission Mode	Venus Swingby	Chemical (H_2-O_2)	Solid Core Nuclear	Nuclear-Electric	Nuclear-Pulse	Aero-Braking	Chemical (H_2-O_2)	Solid Core Nuclear	Nuclear-Electric	Nuclear-Pulse	Chemical (Storable)	Chemical (H_2-O_2)	Solid Core Nuclear	Nuclear-Electric	Nuclear-Pulse	Chem. (Stor.)	Nucl. Elec.	Nucl. Pulse	Nuclear Electric	Nuclear Pulse
1		X					X					X				X				
2	X	X					X					X								
3	X	X				X					X									
4		X		X					X					X			X			
5		X		X					X					X					X	
6			X	X					X					X			X			
7			X	X					X					X					X	
8			X					X				X				X				
9	X		X					X				X								
10			X					X					X			X				
11	X		X					X					X							
12	X		X			X					X									
13			X					X						X			X			
14			X					X						X					X	
15					X					X					X			X		
16					X					X					X					X

was not available in time. Modes 1, 3, 6, 10, 12 and 15 were selected for presentation in this summary report. At a later date, a detailed report will be issued covering all modes and giving additional details. Most of the "difficult" year mission details are excluded from this report because Venus swingby data were not available for the selected "difficult" year.

METHODS OF ANALYSIS (SUMMARY)

Hardware and propellant mass requirements for Modes 1, 3, 10 and 12 (Table II) were determined by an initial mass optimization computer program developed by STL[5]. This program, developed primarily for solid-core nuclear propulsion mission analysis, is also capable of analyzing chemical propulsion and aerodynamic braking modes. Propulsion performance data were assumed as previously noted, and parametric relations were used to compute inert weights for propulsion modules. These relations were based on conceptual design studies, but assumed tank sizes to be tailored to each application in order to minimize inerts. Chemical rocket engines were assumed to be available at whatever thrust level was desired. Solid-core nuclear engines were assumed as the NERVA II concept (230, 000 lbs of thrust). The optimum number of these engines for each phase of a mission was selected by the program. Typical propellant fractions, (mass of propellant)/(initial mass of stage, including engines), are:

1. Cryogenic chemical (LOX/LH_2): 90%

2. Nuclear (LH_2): 77.4%

3. Storable chemical (N_2O_4/Hydrazine blend): 89.5%

Propellant fraction was dependent on stage size which explains why the cryogenic example (slightly larger than an S-II) is better than the storable example (slightly larger than a Titan II). Allowances for insulation and meteoroid protection were included.

Nuclear pulse data were adapted from a study by General Dynamics/General Atomic. Assumed nuclear pulse Isp was previously given; other data are as follows:[7]

1. Engine Diameter: 10 meters

2. Engine Thrust: 355 tons

3. Engine Mass: 91 tons

The nuclear pulse system was not staged. This propulsion module was used for all nuclear pulse mission modes. Pulse propellant units were assumed carried in magazines which were 6% of the mass of pulse units they carried, and which were jettisoned when emptied.

Electric propulsion data were adopted from a Rand study [4]. Assumed performance, as noted, was 9kg/kW jet with Isp varied in optimal manner. The specific mass figure includes electric powerplant mass, power conditioning mass, thrustor mass, support structure mass, power losses in power conditioning, and thrustor inefficiency. Propellant and propellant tankage mass were considered separately. Data for parabolic injection stages were hand-calculated using the parametric analysis from the STL study. [5]

Ground rules and assumptions for cost and reliability analysis were previously given. Development costs for uprated Saturn V and Post-Saturn launch vehicles were taken from detailed cost studies of these vehicles. [10, 16] Other development costs were taken from prior MSFC in-house studies of advanced missions [11, 12] or estimated by the Program Analysis and Control Group of the MSFC Future Projects Office. Principal cost data are given in Table III.

TABLE III

MAJOR COST ITEMS
(Millions of Dollars)

	R&D COST	FIRST UNIT	LEARNING CURVE	PRIOR UNITS
Uprated Saturn V	2000	109	90%	72
Post-Saturn	9000	70/Shot	-	-
Standard Nuclear Module	4700	35	100%	-
Nuclear Electric Propulsion Module	4500	300	100%	-
ORION	8000	70	100%	-
Spacecraft	9000	315	100%	-
Aerobraking Module	1000	100	100%	-

Cost and orbital operations analysis required generating specific conceptual interplanetary vehicle designs. Major hardware items making up these designs are given in the next section of this report, along with principal mass data. The orbital operations analysis utilized a simple computer routine based on a parametric analysis by Dr. H. H. Koelle. Total orbital mass, cost, and crew size required for orbital operations were calculated, and the number of additional launch vehicle flights to place this mass in orbit was determined. Launch and mission crew transport were assumed to use a reusable transport vehicle. Figure 1 is a logic chart summarizing the process of analysis.

Final outputs of the analysis were the data required by the principal criteria. Total expenditure to first mission attempt was taken as the calculated total cost for one mission. The baseline mission, successfully executed, would provide 80 man-days on Mars. Total cost per man-day on Mars was determined from:

$$C_m = \frac{C_T}{80\ N\bar{R}_m}$$

where:

C_T is total cost

C_m is total cost per man-day on Mars

N is number of baseline missions attempted

$\bar{R}_m$ is the estimated mission reliability

RESULTS OF ANALYSIS

Results of the analysis and calculations are summarized in Tables IV through VI for the mission modes of principal interest. Orbital operations data and mission reliability were dependent on the number of missions attempted. These tables give these data only for the case of 10 mission attempts. Final outputs of total operating cost per man-day on Mars are given for 1, 10, and 100 missions. For all cases, orbital operations mass burden was assumed to be handled by Saturn V's. Although this may seem inconsistent where Post-Saturn vehicles are used in the same mission, it is likely that the nature of orbital operations will be such that launches for its support will preferably be divided into small units (more nearly of Saturn V payload size) and launched at intervals during the period of orbital preparation of the space vehicle.

Total costs for one mission for each of the principal modes are shown in Figure 2. Saturn V and Post-Saturn cases are given. As previously noted, R& D costs include 50% of the estimated costs of uprating Saturn V, or 50% of the estimated costs of developing Post-Saturn. All of the estimated costs for space-

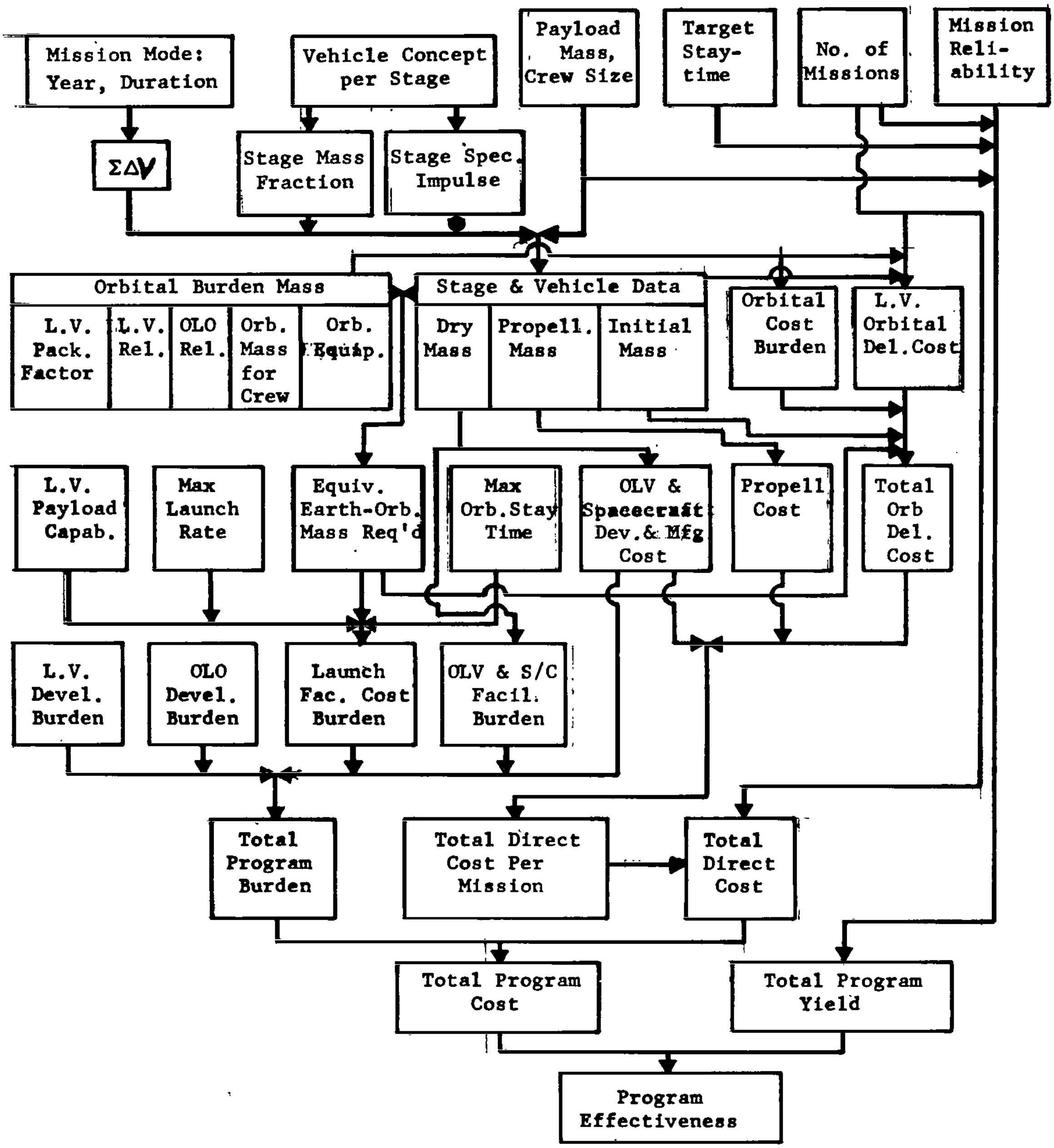

FIGURE 1. LOGIC CHART FOR METHOD OF ANALYSIS

TABLE IV

HARDWARE TYPE AND MASS RESULTS SUMMARY

MISSION MODE	1:CCC (All-Chemical)	3:CAS (Chemical Aerobraking V. Swingby)	6:NEE (Nuclear, Electric)	10:NNN (All Nuclear)	12:NAS (Nuclear, Aerobraking V. Swingby)	15: ORION
IMIEO,* TONS	2350	1059	724	834	746	377
MASS IN** MARS ORBIT TONS	315	290	Not Calculated	210	290	120
TOTAL DRY** MASS	289	177	258	194	195	121.5
EARTH DEPARTURE STAGE	3 Mod. S-II	1 Mod. S-II	3 SNM + (2 eng)	3 SNM (2 eng)	3 SNM (2 eng)	10-meter ORION
MARS ARRIVAL STAGE	1 MOD S-II	Aerobraking Aero-shell Mass=52 T.	Electric	2 SNM (1 eng)	Aerobraking Aero-shell Mass=52 T.	10-meter ORION
MARS DEPARTURE STAGE	1 MOD S-IVB	Storable Chemical	Electric	1 SNM	Storable Chemical	10-meter ORION
EARTH ARRIVAL	Storable brake, dir. entry	Storable brake, dir. entry	Electric brake, dir. entry	Storable brake, dir. entry	Storable brake, dir. entry	ORION brake, dir. entry

* Initial mass in Earth orbit.

** Does not include crew living module or Earth entry module.

\+ Standard nuclear module: 33-ft diameter LH_2 tank, with or without NERVA II engine.

TABLE V

ORBITAL OPERATIONS RESULTS SUMMARY

MISSION CODE	1:CCC (All-Chemical)	3:CAS (Chemical Aerobraking V. Swingby	6:NEE (Nuclear Electric)	10:NNN (All Nuclear)	12:NAS (Nuclear Aerobraking	15: ORION
ORBITAL * Ops Mass burden, tons	SAT V 932 / P/S 514	SAT V 332 / P/S 214	SAT V 266 / P/S 169	SAT V 301 / P/S 232	SAT V 256 / P/S 180	SAT V 151 / P/S 0
SAT V * Launches to support Orb. Ops.	SAT V 10 / P/S 5	SAT V 4 / P/S 3	SAT V 3 / P/S 2	SAT V 4 / P/S 3	SAT V 3 / P/S 2	SAT V 2 / P/S 0
ORB. OPS. COST, $ 10^6 per Mission	SAT V 858 / P/S 505	SAT V 423 / P/S 304	SAT V 377 / P/S 281	SAT V 389 / P/S 314	SAT V 358 / P/S 282	SAT V 298 / P/S 0
LAUNCHES to place ISV** in Orbit	SAT V 22 / P/S 8	SAT V 10 / P/S 3	SAT V 7 / P/S 2	SAT V 7 / P/S 3	SAT V 6 / P/S 2	SAT V 4 / P/S 1
MISSION RELI-ABILITY	SAT V .76 / P/S .84	SAT V .811 / P/S .836	SAT V .843 / P/S .837	SAT V .818 / P/S .837	SAT V .801 / P/S .822	SAT V .822 / P/S .904

* For the 10 mission case.

** Interplanetary space vehicle.

TABLE VI

PROGRAM EFFECTIVENESS SUMMARY *

		Mode 1		3		6		10		12		15	
T.O.C. per Man-day on MARS	1	309	310	272	308	383	415	308	356	336	380	328	368
	10	67	49	46	42.5	54.5	53.6	45.5	46.3	48.4	48.7	42.6	41.5
	100	36	17.2	19	11.9	17.9	13.4	15.5	11.5	15.5	11.9	11.2	7.77

* Total operating cost in millions of dollars per manday. Values are shown for 1, 10, and 100 missions, and for Saturn V and Post-Saturn Launch systems (left-hand and right-hand numbers, respectively).

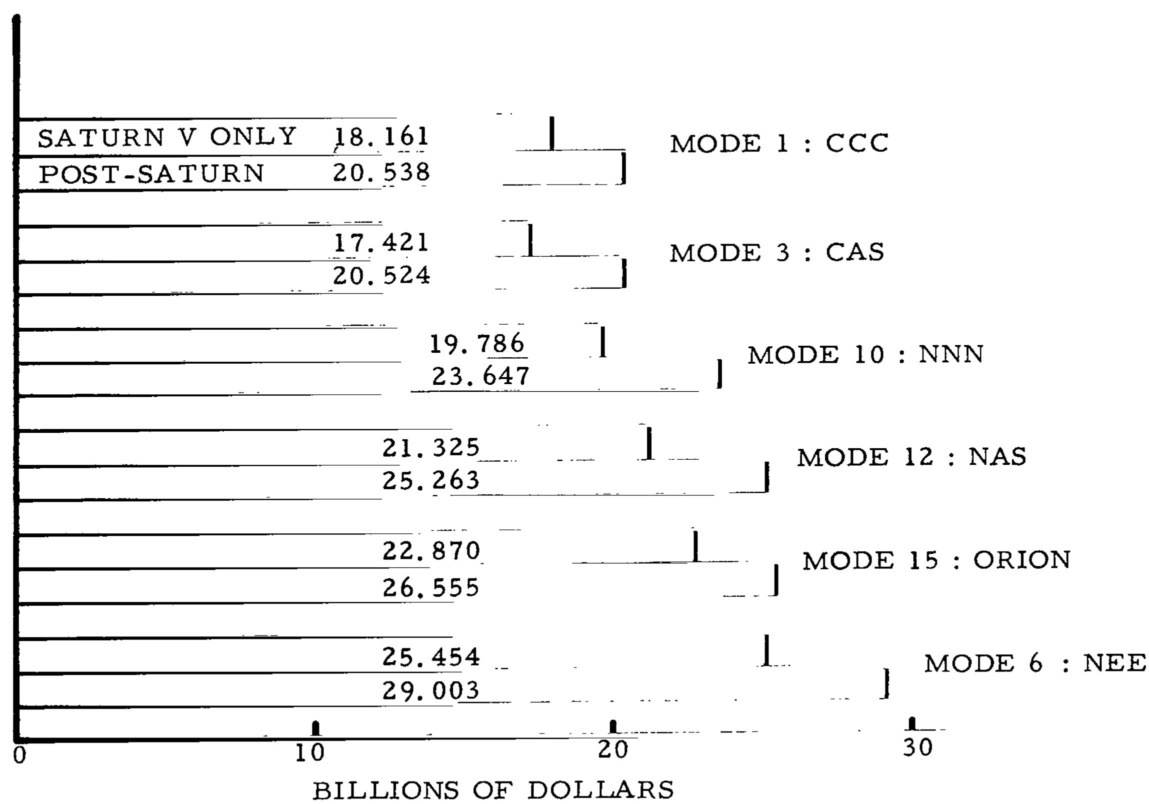

FIGURE 2. TOTAL COST FOR FIRST MISSION ATTEMPT

craft R& D are included. No costs are included for preliminary missions such as Voyager-type probes, or manned planetary flybys. Test flights included in the R& D costs do not include extensive operations involving trial flights to Mars, but do include a nominal allowance (several Saturn V flights) for development of orbital operations, and Earth orbital test flights of space vehicle hardware and systems.

Of those mission modes considered, some of the more efficient appear to be feasible using uprated Saturn V's if one ship per attempt is acceptable. Mode 12, including orbital operations, is estimated to require 9 Saturn V's, Mode 15, 6 Saturn V's, Mode 6 and Mode 10, 11 Saturn V's per space vehicle leaving Earth orbit. These requirements imply stockpiling of launch vehicles and payloads, and a launch every 8 to 15 days in order to limit orbital preparation to 90 days. Unreliability of launch vehicles and additional launches thereby are taken into account in the orbital operations analysis. Additional launch facilities would undoubtedly be required; this matter needs further study. Mode 3, requiring 15 Saturn V's seems marginal, and Mode 1, requiring 34, is quite unattractive and might not be feasible.

Total program costs are shown in Figure 3. These are approximate curves for purposes of illustration. The most expensive program, Mode 1 with Saturn V's, is $ 226 billion. The least expensive program, ORION with Post-Saturn, is $ 56 billion. Costs for the additional mission modes not reported in detail herein fall within this range.

Additional plots used Mode 10 as a baseline. Figure 4 shows cost per man-day on Mars as a function of number of missions, for Mode 10 executed with Saturn V and Post-Saturn. Figure 5 shows the same data replotted with the Post-Saturn mode expressed in percent of the Saturn V mode.

Mode comparisons are given in the format of Figure 5. Figure 6 compares the modes in terms of Mode 10, cost per man-day, for Saturn V; Figure 7 does the same thing for Post-Saturn. In general, it appears that the all-nuclear mode is about as good as any of the others and better than some, with the single exception of the advanced nuclear propulsion, represented by ORION. If another form of advanced nuclear propulsion, such as the gas-core rocket, were developed to approximately the performance level used in this analysis for ORION, similar cost results would be expected. When one compares the development status of the nuclear rocket with the status of competitive systems, one is led to conclude that at this point in time the nuclear rocket is the best choice of these systems for early manned Mars landing missions. At a later date, we may presume that some form of advanced nuclear propulsion will be developed for planetary missions.

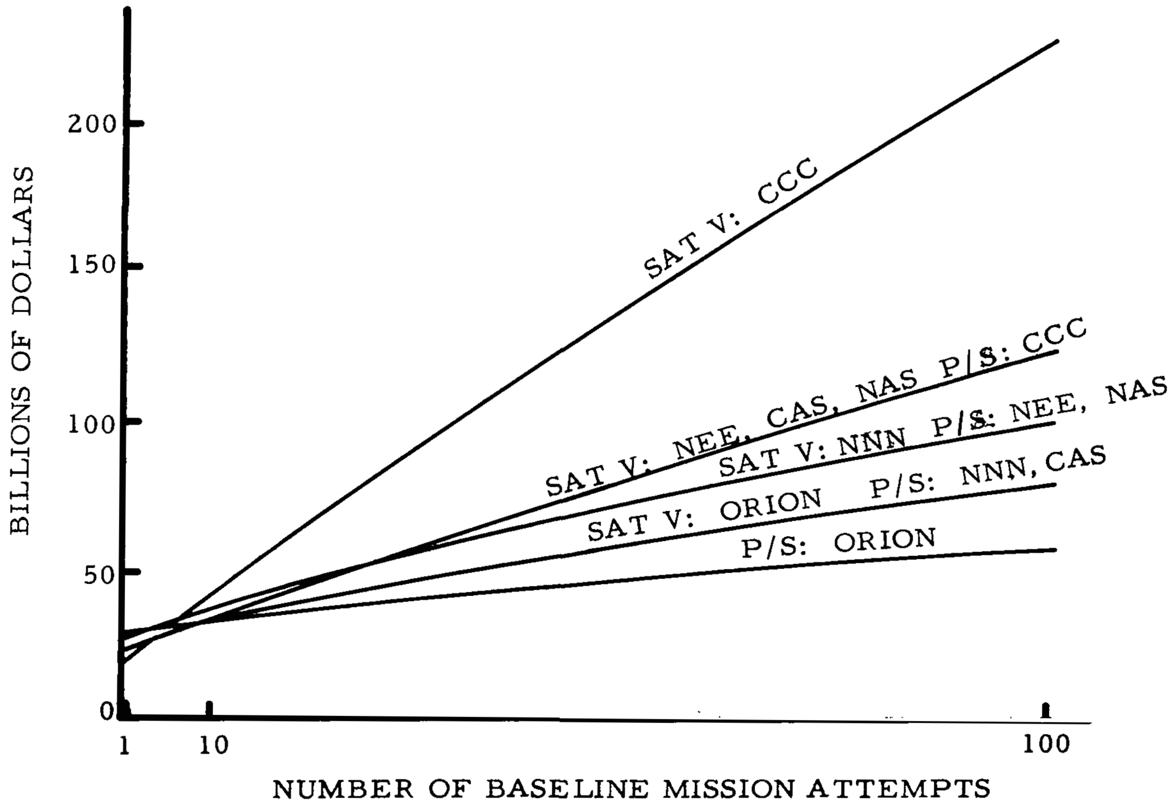

FIGURE 3. TOTAL PROGRAM COST

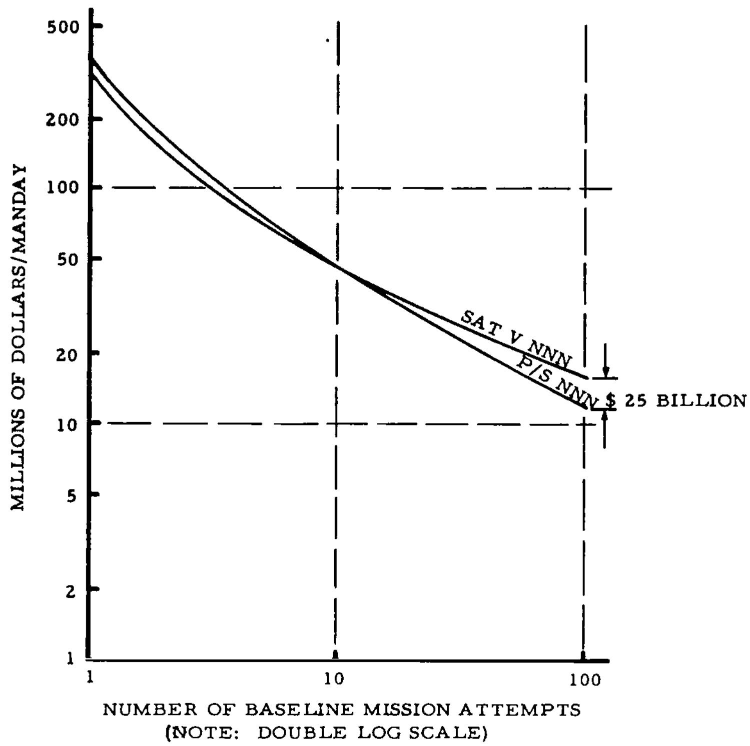

FIGURE 4. TOTAL PROGRAM COST PER NET MAN-DAY ON MARS

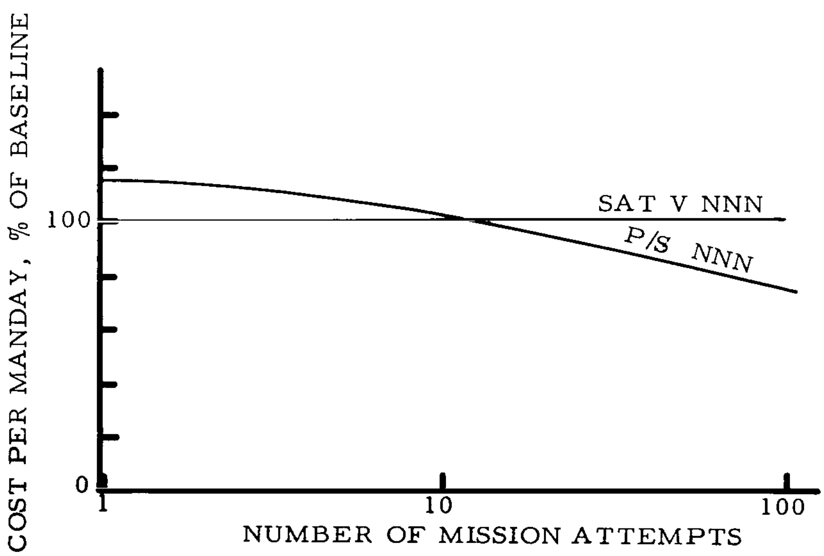

FIGURE 5. NORMALIZED COST PER MAN-DAY (SATURN V vs POST-SATURN).

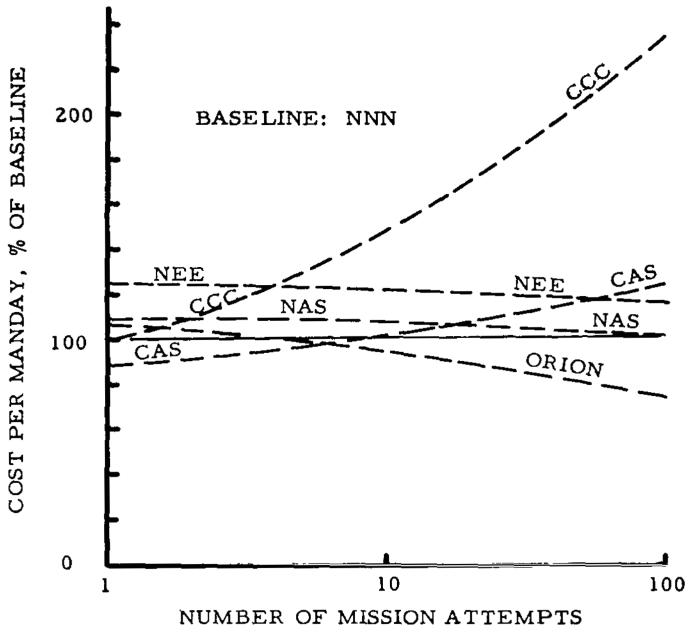

FIGURE 6. NORMALIZED COST PER MAN-DAY(SATURN V).

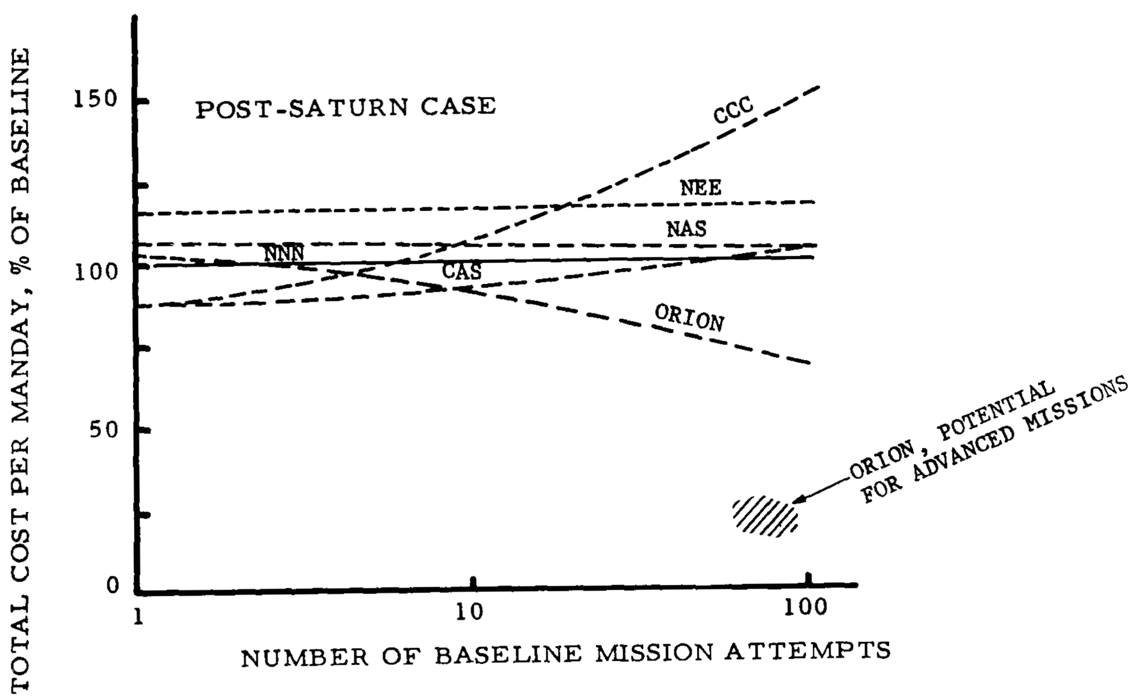

FIGURE 7. NORMALIZED COST PER MAN-DAY(POST-SATURN).

Research on known concepts for advanced nuclear propulsion should be encouraged.

Principal advantages of the Post-Saturn indicated by this and other studies are three in number. First, the full reusability of this vehicle concept, coupled with its large size, results in a predicted delivery-to-orbit cost of about 1/3 that of Saturn V, in terms of dollars per pound of payload. Second, the large payload mass and volume allow a much greater degree of pre-assembly of the space vehicle before launch, and thereby reduce greatly the amount and cost of orbital operations required to accomplish a mission attempt. Thirdly, the number of separate launch operations to be performed during the period of mission preparation is greatly reduced. Thus, the results of this study agree generally with other results [2, 11], which show that the reusable Post-Saturn vehicle will provide large savings in an extensive space program.

The nuclear rocket results, showing it in a favorable light, are dependent on developing means (either effective insulation or hydrogen reliquefaction systems) for handling the liquid hydrogen propellant for long durations (up to roughly 14 months) in the space environment. If insulation were the selected method, it further assumes that the usable volume of a 33-foot outside-diameter tank would not be severely reduced by insulation volume. Nuclear rocket specific impulse for this study was assumed to be 7850 m/sec. This is conservative; it is probable that an operational graphite engine will attain about 8300 m/sec, in which case mission costs would be reduced by a few percent.

The requirement to develop orbital operations for planetary missions seems to be almost inevitable. Alternate courses of action are the development of a very large (roughly 1000 tons payload in orbit) Post-Saturn or development of advanced nuclear propulsion, in time for initial manned planetary missions. These alternates are both extremely unlikely.

Use of uprated Saturn V for initial planetary missions, as noted, implies stock-piling of vehicles. Either additional launch facilities beyond those presently planned, or a stretchout of the orbital assembly period to more than 90 days will also be required. Stochastic studies of the capabilities of Saturn V launch operations for "rapid-fire" launch should be conducted. (Means for doing this are under development by the MSFC Aero-Astrodynamics Laboratory).

Advanced nuclear propulsion, such as ORION, would increase flexibility and capability for planetary missions to a considerable degree. The true potential of advanced nuclear propulsion is not revealed by the baseline mission analysis. Further discussion of this matter is given in the "Excursions from Baseline Mission" section of this report. Several schemes for advanced nuclear propulsion have been proposed. Electric propulsion with an advanced lightweight power supply could be considered advanced nuclear propulsion, even though the 9kg/jkW system was at best only competitive with the all nuclear system in this analysis. Thermionic reactors or reactor-driven MHD systems may ultimately reach the 4-5kg/jkW level. It should be further noted that the electric systems show up to best advantage for high ΔV, long duration missions. Although high I_{sp} at high thrust is preferable to low thrust, proposed high-thrust high-I_{sp} systems have problems (see the "Development Implications" section of this report) not shared by electric propulsion. Other important advanced nuclear systems concepts known to the writer besides ORION and lightweight electric are the gas core nuclear rocket, MHD hybrids, (13) and thermonuclear concepts. Basic thermonuclear research (not propulsion-oriented) is being actively pursued by AEC. Of the other advanced concepts, research on only the gas-core is being funded at anything like an adequate level.

EXCURSIONS FROM BASELINE MISSION

Three excursions from the baseline mission were investigated briefly:

1. Mission in a "difficult" year; 1993 was selected as representative.

2. Return to high altitude Earth orbit instead of direct entry.

3. Ambitious missions.

Items 1 and 2 above were also combined; item 3 included 1 and 2 plus greater payload, longer stopover duration, and more rapid transfers.

The "difficult" year mission, if executed with three impulsive propulsion maneuvers initiated in planetary orbit, requires substantially more delta V than the "easy" year mission, principally because of the eccentricity of Mars' orbit. With continuous propulsion, as provided by the low thrust electric system, the variation from "good" to "bad" years is substantially reduced. In the absence of detailed analytical data, it was assumed for this study that the electric systems would incur a 15% initial mass penalty in a "difficult" year.

In most "difficult" years, the delta V requirements may be reduced to essentially the "easy" year case by employing an unpowered Venus swingby. [14] A modest (20 to 50 day) trip time penalty is usually required if proper advantage of the swingby is to be gained. Consistent data on swingby modes were not available for inclusion in this report. It is expected that they will be generated prior to preparation of the detailed report on this study. Some relaxation of delta V requirements in "difficult" years can also be obtained by use of additional propulsive maneuvers. To the writer's knowledge this situation has not yet been adequately analyzed. Additional maneuvers would present serious problems if required of graphite nuclear rocket stages, but should be routine for storable chemical or nuclear pulse propulsion. The Venus swingby does represent an added complexity to the mission, and would probably not be used where the propulsion system could accommodate the high delta V without severe penalty. Initial masses in Earth orbit for the 1993 missions without swingby are shown in Figure 8, for nuclear rocket and nuclear pulse. Return to Earth orbit requirements are also shown for the nuclear pulse in 1993 and electric propulsion in 1984. Of particular significance is the relative insensitivity of the nuclear pulse system to added mission requirements. The Earth orbit return in 1993 was not analyzed for other systems because of the large penalties involved. (A relaxtion of the mission duration limit would make this mission practical with electric propulsion.)

The performance data for the nuclear pulse system used in this report are believed to be conservative. Initial mass calculations using more optimistic (classified) performance data [15] indicated that all of the missions studied, including return to Earth orbit in 1993, could be accomplished by direct injection (no orbital operations) using a Post-Saturn to place the vehicle in orbit.

A representative ambitious mission is tabulated in Table VII. This mission is felt to be indicative of the scope of operations required to begin true exploration of Mars (as contrasted to an excursion).

TABLE VII

ADVANCED MISSION CHARACTERISTICS

Crew Size		20 men
Mars Surface Crew Size		20 men
Mars Stopover Duration		120 days
Earth Departure Date		J. D. 2448820
	100 day transfer	
Mars Arrival Date		J. D. 2448920
	120 day stay on Mars	
Mars Departure Date		J. D. 2449040
	100 day transfer	
Earth Arrival Date		J. D. 2449140

TABLE VII (Cont'd)

Velocity Increments: Total	72.3 km/sec
Maneuver No. 1	16 km/sec
Maneuver No. 2	21.5 km/sec
Maneuver No. 3	18.8 km/sec
Maneuver No. 4 (to 24-hr Earth orbit)	16 km/sec
Crew Living Module Mass (including life support)	70 tons
All-up Mass of Mars Excursion Modules (Hydrogen-Oxygen Propellants)	2 @ 70 tons each
Mass of Exploration Hardware: Total	100 tons
Inflatable Shelters	2 @ 11.5 tons each
Roving Vehicles	4 @ 4.5 tons each
Life Support Stores	22 tons
Roving Vehicle Propellant and Spares	4.5 tons
Scientific Laboratory and Equipment	9 tons
Nuclear Reactor Power Supply	9 tons
Roving Vehicle Propellant Reverter	9 tons
Packaging	5.5 tons
Mass of Mars Landers Required to Land Exploration Hardware	90 tons

This mission can be accomplished with the indicated transfer times using high performance [15] nuclear pulse propulsion, with two 10-meter vehicles in Earth orbit with an all-up initial mass of 3,750 tons. As specified, this mission is out of the question for any other of the propulsion systems studied. It can, however, be accomplished with graphite nuclear rockets by making several compromises:

1. Mars staytime reduced to 90 days.

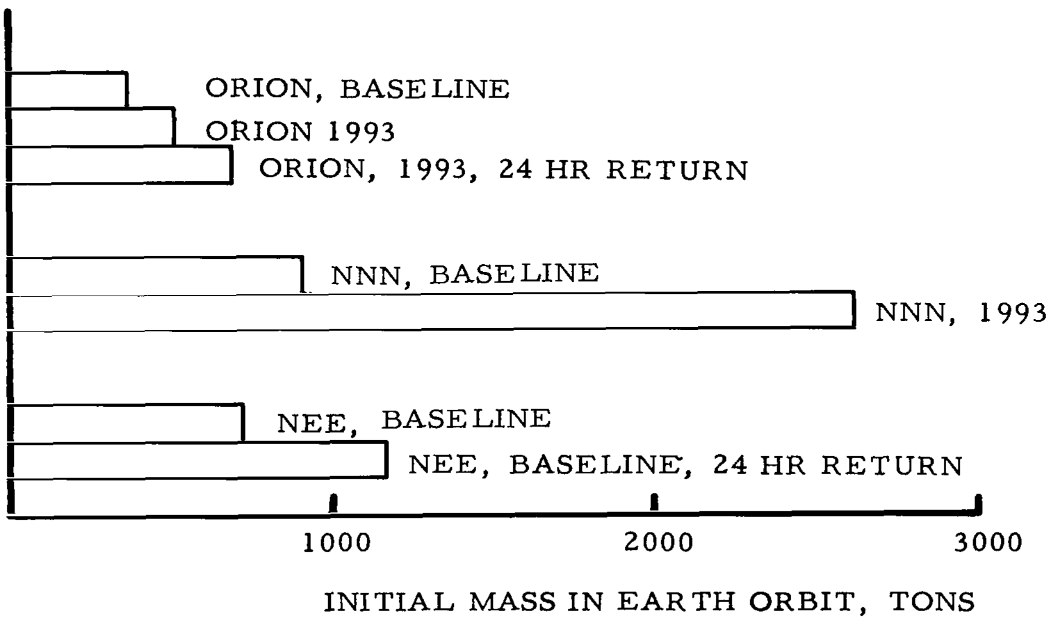

FIGURE 8. INITIAL MASS COMPARISON FOR EXCURSIONS ON BASELINE MISSION

2. Transfer times extended to 220-250 days each way.

3. Return to Earth via direct entry rather than Earth orbit.

4. Scientific payload returned to Earth reduced to a minimum.

Required initial mass in 5000-6000 tons. In terms of initial mass, this mission is equal to roughly 7 baseline missions. In terms of man-days on Mars, it is equivalent to about 30 baseline missions. In terms of knowledge to be gained by the extended exploration, there is no meaningful comparison because of the great advantage given to the ambitious mission by landing the 100 tons of mission support and exploration equipment.

The baseline mission is probably roughly equivalent to a manned Jupiter flyby, with a substantially longer mission duration. The equivalence will be dependent on the mission mode; much more favorable for high specific impulse systems. It also may be equivalent to a Venus landing mission, but this mission would require a vastly superior excursion module; the re-orbit delta V requirement will be about 3 km/sec higher than for Mars. The greater density of Venus atmosphere is a further burden. Mission equivalences will be investigated in further detail for the detail report on this study.

DEVELOPMENT IMPLICATIONS

Uprating of the payload capacity of the two-stages Saturn V to 160 tons in a 485-km orbit was assumed for this study, representing roughly a 60 percent increase in payload capacity. Saturn V improvement studies have indicated a variety of ways that this could be accomplished. A typical example, and not necessarily the best choice, would be lengthening the S-IC stage for increased propellant capacity, strapping on 120-inch solid booster motors, lengthening the S-II stage for increased propellant capacity, and using higher thrust, higher-performance engines on the S-II stage. In general, for planetary missions, it appears that the larger the Saturn V payload mass and envelope, the better.

Employment of Saturn stages, such as the S-II or S-IVB, as orbit launch vehicles for manned planetary landing missions, appears to be an unattractive approach. Such a development effort would be quite substantial, particularly in developing cryogenic insulation techniques and orbital operations. It is, however, quite possible that Saturn stages as orbit launch vehicles would be very attractive for less demanding missions, such as Mars flyby or Venus orbiter. Insulation and orbital operations technology will, of course, be indirectly applicable to other systems.

As noted, development of a large Post-Saturn recoverable launch vehicle will be a very good investment for any manned planetary exploration program of meaningful magnitude. There seems at the present time to be no particular urgency about the initiation of such a vehicle development. Initiation in the early 1970's would presumably be soon enough. Meanwhile, development of ballistic recovery technology for launch vehicles should be pursued vigorously. Recovery of both stages is essential for realization of the projected economic benefits of a large Post-Saturn vehicle. Much more basic technology data must be developed in this area before credible designs of recovery systems for such vehicles can be initiated. Improvements in propulsion technology in the area of rocket engine performance reliability and reusability will also be great benefit to such a vehicle.

Electric propulsion was not reflected in this report as superior to the graphite nuclear rocket. Because of the great difference in characteristics between electric propulsion and the graphite rocket, this conclusion must be treated with considerable caution. The following specific comments apply:

1. Attractiveness of electric propulsion is very sensitive to the specific power assumed, and also to mission duration. Relatively modest increases in mission duration can result in substantial reductions in required initial mass.

2. Electric propulsion will be particularly attractive for long-range, high-velocity, one-way probe missions.

3. If mission duration limits are not too strict, the comparison of electric propulsion with the graphite rocket for higher velocity manned missions will be much more favorable.

It should further be noted that development of nuclear-electric power supplies up to the several hundred kilowatt range will be required for establishment of extraterrestrial (i.e., lunar and planetary) bases whether or not electric power is used for propulsion.

Figures 9 and 10 show a predicted development schedule and cost for a one megawatt nuclear-electric power plant of the Rankine cycle type (similar to SNAP-50). These data are abstracted from the previously noted RAND study.(4) In the last year or two, prospects for development of thermionic nuclear-electric systems have brightened considerably. Thermionic systems offer the potential advantage of very high order redundancy, simplicity, and low weight. If these

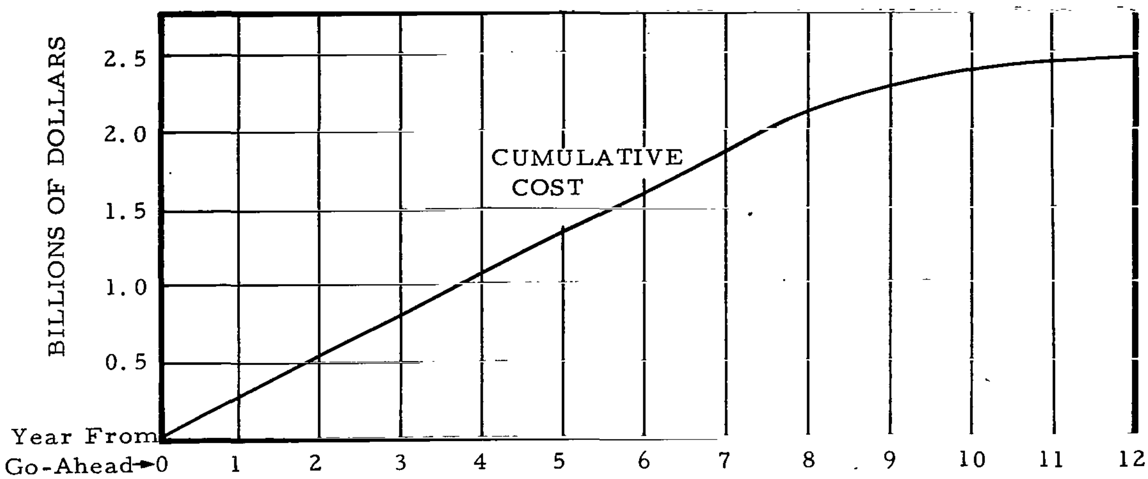

FIGURE 9. DEVELOPMENT COST FOR 1 MWe RANKINE CYCLE NUCLEAR-ELECTRIC POWER SOURCE (EXCLUDES FLIGHT TESTING)

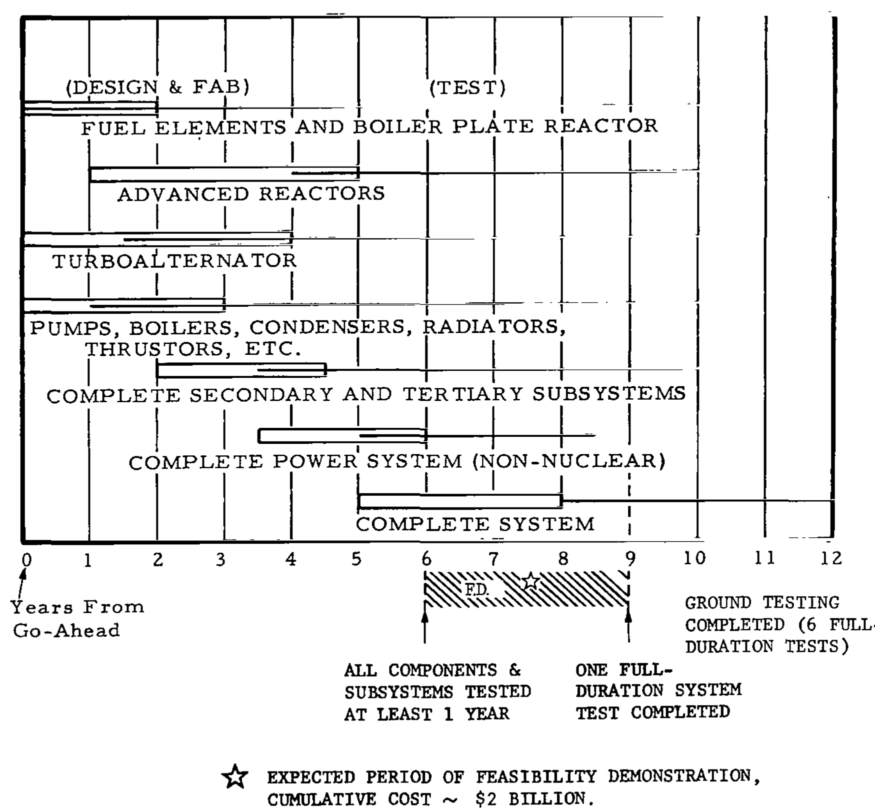

FIGURE 10. PREDICTED DEVELOPMENT SCHEDULE FOR 1 MWe NUCLEAR-ELECTRIC POWER SOURCE.

advantages can be substantiated in practice, shorter duration and less expensive development schedules may be possible. At the present time, funding for thermionic systems is at a very low level. Increased activity in this area to the extent of providing an early proof of feasibility is strongly recommended.

The graphite nuclear rocket presently appears to be the best bet for early manned planetary landing missions. Detailed forecasts of development schedules are classified. However, it appears that the total investment required to develop a flight-rated engine in the 5000 megawatt class, together with a propulsion stage module, is on the order of 5 billion dollars. Because the nuclear rocket technology program is already well underway, there would appear to be little doubt that this propulsion type can be available for the subject missions. Great emphasis should be placed on the development of insulation and reliquefaction techniques to make practical long-term storage of hydrogen propellant in the space environment. This effort should include an adequate orbital experiments program. If manned planetary flyby missions are conducted, it would be extremely desirable, if possible, to use a nuclear rocket propulsion system to execute these missions in order to gain operations experience prior to attempting a manned planetary landing. This would require substantially earlier availability of a nuclear rocket and would imply an increase in present annual expenditures for nuclear rocket development in the near future. Development of a 1000 megawatt (Nerva I) class flight engine is not recommended. Identifiable missions for the nuclear rocket definitely require the 5000 megawatt (Nerva II) class engine.

Aerodynamic braking at Mars was not found to be superior to the all-propulsive mode. It does, however, present, to some extent, an alternative to long-term cryogenic storage in space. Storage duration requirements are reduced from about 14 months to about 3 to 6 months. Aerodynamic braking for landing of the Mars excursion module is, of course, essential. It would appear that aerodynamic braking of the interplanetary spacecraft would require a much more accurate knowledge of the Mars atmosphere than would aerodynamic braking of the Mars excursion module.

Development costs for an advanced nuclear propulsion system such as ORION are rather uncertain at this time as compared to development predictions for a more near-term system such as the nuclear rocket. The figure of 8 billion dollars used in this report is much higher than the 3 billion dollar cost estimate from General Atomics. Specific development problems, objectives, techniques and proposed schedules are classified. The real problem at the present time with advanced nuclear propulsion is not systems development but research and technology efforts directed at proving feasibility and providing performance estimates. Adequate research funding now, which would be a minute fraction of development funds presently being expended in the propulsion area, would be an extremely valuable investment in advanced planetary mission capability for the future. This comment is applicable to all forms of advanced nuclear propulsion. Both the nuclear pulse concept and the gas core concept pose feasibility problems. There is little doubt of the technical feasibility of ORION, but this concept is at present politically in disfavor. Technical feasibility of the gas core nuclear rocket in a useful form is at least questionable.

PRINCIPAL RECOMMENDATIONS FOR FURTHER STUDY

Following are recommendations for further study:

1. Extension of this study to refine data which were assumptional or estimated, to determine effects of varying performance data, which were in most cases nominal, and to include additional data and analysis, such as for manned planetary flybys, larger scale missions, and further treatment of missions to planets other than Mars.

2. Investigation of all operational aspects of using the solid core rocket as a planetary mission propulsion system.

   a. Experimental investigations of cryogenics in the space environment for extended periods.

   b. Study of space operations with reactor engines.

3. Investigation of reliability, safety, and abort aspects of manned planetary missions.

4. Continuation of programmatic analysis of reasonable future space program alternatives, as is being carried out by the MSFC Future Projects Office, to predict probable funding schedules which could be allocated to manned planetary activity. This in turn will make possible laying out of reasonable technology programs aimed at this objective.

5. Study of unconventional manned planetary mission modes not represented in this report. If probable extraterrestrial exploration experience to be gained on the Moon is properly taken into account, it is possible that such mission modes could, at insignificant or modest increase in risk, provide manyfold increases in planetary program effectiveness.

6. Detailed analysis of the problems associated with stockpiling and launching, at short intervals, of 5 to 15 Saturn V's as required to initiate a manned Mars landing mission.

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June 29, 1965 APPROVAL TM X-53265

A PROPULSION-ORIENTED STUDY OF
MISSION MODES FOR MANNED MARS LANDING

By G. R. Woodcock

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

H. H. Koelle

H. H. KOELLE
Director, Future Projects Office

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