APPLICATION OF NUCLEAR ROCKET PROPULSION TO MANNED MARS SPACECRAFT

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I. Summary

The application of nuclear rocket propulsion to a manned interplanetary spacecraft is examined in terms of vehicle integration problems. Payload and velocity requirements for Mars round trip missions are used to establish the propulsion system constraints. Graphite and metallic reactors are considered, and the effects of engine restart capability upon mission profile and vehicle configuration are included. Vehicle preliminary designs have been prepared to study compatibility with SATURN V launch vehicles and Earth orbital assembly. A reference design is described in detail for the case of a four man crew and 15 months total mission duration.

II. Introduction

Beginning in the next decade, the development of interplanetary exploration is expected to focus upon manned flights to Mars. Beyond the initial reconnaissance phase, any substantial planetary exploration will require propulsion systems of considerably higher performance than the most advanced chemical rockets. This paper presents a summary of recent General Electric studies on the application of heat transfer nuclear rockets to manned Mars missions in the 1970's. Major spacecraft problem areas were examined to identify development requirements, and to aid in the preparation of propulsion system specifications.

III. Mission Requirements

Payload and velocity requirements have been established for typical mission profiles in order to provide a basis for evaluating the relative performance of different propulsion systems. Wherever applicable, the interactions between payload and other vehicle subsystems were examined to indicate possible weight savings attributable to "dual usage"; (for example, the arrangement of propellant tankage for maximum solar flare shielding effect). Although many options are available in setting crew size and overall mission profile, a preliminary selection was made early in the study in order to permit more detailed analysis of vehicle and powerplant design problems. Figure 1 shows some of the more important elements of a mission profile, and outlines a specific case selected for this study. Subsequent discussion of system interactions will concentrate upon the sample case, with variations introduced in certain areas.

Basically, the mission involves launching of multiple SATURN V vehicles into low Earth orbit for final spacecraft assembly. The interplanetary vehicle consists of a central propulsion module surrounded by a cluster of expendable propellant tanks as shown in Figure 2. These tanks are jettisoned as they are emptied, to provide staging as in Figure 3. Payload includes a two deck command module containing the life support systems, living accommodations for a four man crew, communications gear, experimental equipment, and a control center. Solar flare protection consists of a vacuum jacketed capsule projecting downward into the main hydrogen tank. This "storm cellar" is lined with shielding material to augment the annulus of liquid hydrogen which surrounds the capsule. Note that the proposed configuration does not provide an artificial "g" capability. If zero "g" cannot be tolerated for the long duration of an interplanetary mission, a rotating cabin section could be factored into the design. However, this approach would increase the spacecraft gross weight due to structural integration problems.

A chemically propelled landing module, suitable for short duration (5-10 days) Mars surface exploration is attached to the forward end of the spacecraft during Earth-Mars transit. For the case of a four man mission, the landing module would carry two crew members to the planet's surface while the interplanetary vehicle remains in an eccentric parking orbit at Mars. After returning to orbit, the excursion module is abandoned and the crew departs for Earth with the main vehicle in the configuration shown by Figure 4. During the Mars escape maneuver, the last two external tanks are emptied, as is the aft compartment of the main tank. The forward section of the main tank, which surrounds the solar flare shelter, still contains hydrogen throughout the Mars-Earth transfer.

Upon approaching Earth, the two empty tanks are released, and the nuclear rocket engine is used to brake the vehicle into a high altitude parking orbit. The crew will then transfer to a ferry vehicle for Earth re-entry, Alternatively, it would be possible to reduce the velocity increment required of the interplanetary spacecraft by employing direct re-entry from the Mars transfer path. However, this would require that an Earth re-entry vehicle be transported throughout the entire mission, thereby increasing the weight carried on the spacecraft. Since direct re-entry alleviates the need for a large propulsion maneuver at the terminal end of the mission, little or no propellant would be available for solar flare shielding during the return flight coast period. The flare shield weight would then have to be increased to insure crew protection in the "empty" vehicle.

A. Payload

Based upon the mission described here, the weight and other significant parameters of the payload were established. Major systems comprising the payload are as follows:

- Space Radiation Shielding
- Life Support and Crew Facilities
- Electronic Systems
- Auxiliary Power Supply
- Mars Excursion Module

1. Space Radiation Shielding - The long mission duration of manned interplanetary flight exposes the crew to significant space radiation doseage, in addition to that contributed by the nuclear propulsion system. Sources include galactic charged nuclei, particles trapped in planetary magnetic fields (Van Allen), and most important, solar flare protons. The first of these, cosmic radiation, represents a continuing or chronic dose of from 0.5 to 1.0 rem per week for an unshielded man in free space. Since the radiation consists mainly of very high energy heavy nuclei, little can be done to shield against cosmic dose. For example, 80 gm/cm^2 of water shielding will only reduce the dose rate by a factor of two. Consequently, the normal crew quarters cannot be shielded sufficiently to yield any substantial drop in cosmic flux, and on a typical Mars mission of 12 to 15 months duration, the crew will accumulate from 25 to 60 rem from this source alone. Van Allen radiation does not appear to present a severe problem, since the transit through high flux, trapped particle regions is rapid enough with a nuclear rocket to make this dose negligible. However, further knowledge of the environment near Mars is required to determine whether there are radiation belts present, and if so, what effect they will have upon the selection of a Mars parking orbit.

Solar flare radiation presents a most serious problem for long duration space missions. The approximate frequency of solar events is summarized in Table I for the different classes of intensity.

TABLE I

Class of Solar Flare	Average Frequency of Occurance
1	2 hours
2	Daily
3	Monthly
3+	3 to 6 months
Giant	4 years (irregular)

Class 1 and 2 events do not constitute a danger for lightly shielded vehicles (a few grams per square centimeter), because of their low particle energy. The major events, however, involve energies as high as 20 Bev, so that extensive shielding will be required. Figure 5 shows calculated dose per flare as a function of water shield thickness[1] for each of three typical solar events. Note that the giant flare dose cannot readily be reduced below 30 or 40 rem, even with extremely heavy shields. This is due to the fact that the spectrum does not experience a steep decrease in flux until about 1 Bev energy level. Also, most of the protons will undergo nuclear coolisions producing secondary protons, neutrons, and mesons. Fortunately, the probability of encountering even one such flare during a Mars mission is low (~25 %), so that a dose of 50 rem could be tolerated from a giant flare. It should be noted that the curves of Figure 5 are applicable at about 1 AU distance from the sun. In an actual mission, the vehicle trajectory may come as close as 0.5 AU at perihelion. If a giant flare occurred near this point in the flight, crew dose would be as much as four times that shown. Probability of encounter during the portion of the flight at less than 1 AU, is, of course very small.

One of the more attractive concepts for integrating a solar flare shield with the spacecraft is illustrated in Figure 6. Here, the liquid hydrogen propellant surrounds a small volume "storm cellar." The vehicle is arranged so that this particular tank remains full of hydrogen throughout the mission; not to be emptied until the final retro-propulsion maneuvers required for Earth capture. Heat transfer from the flare shelter to the liquid hydrogen is minimized by placing multilayer radiation barriers in the vacuum gap between shelter wall and tank wall. When not in use, the shelter can be sealed off and evacuated of air to further reduce heat load. Additional possibilities for reducing total shield weight are shown in Figure 7. The two position shield, for example, allows the use of a maximum shelter volume for all but the most intense flares. For a giant flare, the two halves of the shield are telescoped together, thereby doubling the wall thickness at the expense of reduced volume during the periods of highest flux. Still another possibility is to apply spot shielding in the form of 30 to 50 gm/cm^2 of water in a plastic envelope which encases the crewmen directly, thereby augmenting the main shield. This water could be transferred from the life support system inventory.

Table II gives the distribution of shield materials used in the sample vehicle design, and summarizes the overall dimensions and weights of the system. For this example, the telescoping shield and spot shielding were not considered, so from that standpoint, the design is conservative.

TABLE II

Shelter Volume	300 ft^3
Capacity	4 crewmen
Thickness of Liquid Hydrogen	17.0 gm/cm^2
Thickness of Aluminum	2.5 gm/cm^2
Thickness of Carbon Shield	15.0 gm/cm^2
Equivalent Water Thickness	50.0 gm/cm^2
Weight of Carbon Shield	9400#
Shield Structure	1100#
Total Weight Penalty of Shield System	10,500#

With this shelter, the integrated dose during a 15 month mission will be about 40 rem, assuming one giant flare. Dose contribution from class 3 and 3+ events will be negligible if the crew enters the shelter during all flares in these categories. Thus, residence time in the shelter should amount to 15 to 20 intervals of a few days duration each.

2. Life Support and Crew Facilities - From data published by D.C. Popma of NASA Langley, life support system weights were determined for a four man crew.[2] The oxygen system is based upon cryogenic storage with a regenerative molecular sieve, and the water system uses passive reclamation plus recovery from the humidity control system. A food allowance of 2.5 pounds per man-day is included, so that the total life support requirement for a 15 month mission comes to 11,500 pounds. Power consumption for atmosphere control, water recovery, and general housekeeping functions will average 3 to 4 KW.

In addition to the life support equipment and materials, crew cabin structure and furnishings mut be considered. A two deck capsule of 22 feet diameter by 15 feet in length was established to provide about 5000 cubic feet of total space. The vehicle control center occupies one deck, with living quarters on the other. Meteoroid protection is afforded by an outer shell of aluminum sandwich construction weighing 2.5 $\#/ft^2$. Aluminum honeycomb is used for all internal structure. An airlock permits extra-vehicular operations during rendezvous. Weights are given in Table III.

TABLE III

	Weight #
Outer Shell	2800
Internal Structure and Fittings	1600
Furnishings	800
Crew and Personal Effects	800
	6000 #

3. Electronic Systems - The weight of electronics gear will vary somewhat because of differences in mission profile and scientific objectives. For this study, it was assumed that the spacecraft would have to be continuously oriented for minimum solar heating of the hydrogen tanks, and this factor was considered in the attitude control system weight. In addition to the basic vehicle requirements, allowance was also made for special survey equipment to be used during the waiting period in Mars orbit. Table IV shows the breakdown of weights for electronic subsystems on the 4 man vehicle. A radiator is included to reject the 3 KW of heat generated in radar and communications transmitters.

TABLE IV

	Weight #
Communications	
Two-3 KW Transmitters (Redundant)	300
Receivers and Recorders	125
Antennas	75
TV Monitors	50
Guidance and Control	
Navigation Equipment (Including Computer)	650
Attitude Control (Sun Oriented System)	1200
Orbital Survey Equipment	
Mapping Radar System	750
Optical Cameras	450
TV Camera	50

TABLE IV (Cont'd)

	Weight #
Scientific Sensors	200
Electronics Cooling Radiator ($200\ ft^2$ Area)	250
TOTAL	4100 #

The orbital survey equipment (1450 pounds) is jettisoned in Mars orbit prior to the return flight. All other equipment constitutes payload which must be carried through the entire round trip.

4. Auxiliary Power System - Total electrical power requirements will average between 5 and 8 KW for this mission. If cryogenic recondensing systems are used to reduce the insulation required for propellant storage, required power increases to about 30 KW. Even at the lower level, the choice of electric generating systems for a long term mission is extremely limited. Fuel cells, for example, would need over 35 pounds of fuel and tankage per kilowatt day, or about 65 tons for a typical mission. Solar power is penalized by the low solar energy density at Mars, which results in large collector area and high specific weight. Even with advanced photovoltaic systems, a panel area of 300 ft^2/KW and a weight of $400\#/KW$ of raw power appear to be minimum. Depending upon the Mars parking orbit selected, the area and weight will increase by a factor of two or more. This problem may be solved for short stay times (5 to 10 days) by using fuel cells to supplement the solar power supply while in Mars orbit. Other solar power plants such as the turboelectric Sunflower could possibly reduce system weight, but collector area will not be appreciably less than the required photovoltaic panel area.

Nuclear Auxiliary powerplants are attractive for this mission, particularly at the higher power level (30 KW or more). Much of the required shielding is provided by the liquid hydrogen propellant, and with proper arrangement of the vehicle systems, some hydrogen will be present until the final Earth capture phase of the mission. The major problem is that of obtaining sufficient life and reliability in the power conversion equipment, particularly with turbogenerator systems. One approach is to use a number of small turbine units with some redundancy to assure at least partial power output. For example, an 8 KW plant might use a SNAP-2 reactor with two operating turbines, and a third idle unit which could be brought into service by actuation of one-shot valves in the event of a failure. Similarly, a SNAP-8 reactor could power eight SNAP-2 turbo-alternators, with one or more standby units available.

For applications up to about 12 KW, a thermoelectric generator can also be considered, with SNAP-8 reactor as the heat source. This type of system has the advantage of no moving parts, and a modular construction that yields a gradual degradation of power rather than catastrophic failure. Despite its greater weight relative to rotating generators, the thermoelectric powerplant has considerable promise for the Mars mission because of inherently high reliability with static conversion. The system would employ two liquid metal (NaK) loops separated by a heat-exchanger generator containing the thermoelectric couples.

As shown in Figure 8, the primary loop supplies NaK to the inside of tubular elements which are cooled by secondary NaK flowing over their outer surfaces. The secondary (shell side) coolant rejects heat to a radiator operating at an average temperature of about $575^{\circ}F$. Pumps are static EM devices of the DC conduction type.

Table V gives the system weights for each of the powerplant types discussed here.

TABLE V

A. 8 KW SNAP-2 with 2 Turbo-Alternators

Reactor and Primary Loop	400 #
Shield	1400 #
Power Conversion Equip.	250 #
Radiator-Condenser (270 ft^2)	550 #
	2600 #

B. 8 KW SNAP-8 with Thermoelectric Generator

Reactor and Primary Loop	700 #
Shield	2000 #
Power Conversion Equip.	1000 #
Radiator (920 ft^2)	1800 #
	5050 #

C. 30 KW SNAP-8 with 8 SNAP-2 Turbo-Alternators

Reactor and Primary Loop	700 #
Shield	2000 #
Power Conversion Equip.	1000 #
Radiator Condenser (1080 ft^2)	2200 #
	5900 #

Figure 9 shows how these systems would be integrated with the nuclear rocket vehicle. For the larger systems, a split clamshell radiator is deployed to obtain the necessary heat rejection area, and to reduce thermal insulation requirements on the aft end of the main propellant tank.

Since the auxiliary power reactor will not have to operate while the nuclear rocket is firing, the nuclear rocket shield will provide adequate neutronic decoupling between the two reactor cores. Crew dose from the nuclear electric system is plotted in Figure 10 for an 8 KW thermoelectric powerplant. Dose rate is seen to be negligible with the hydrogen tank full, but becomes prohibitive with an empty tank. This does not constitute a problem, however, since the auxiliary powerplant can be shut down after Earth capture, and a small photovoltaic or fuel cell supply used to power life support equipment during the final hours of the mission.

5. Mars Excursion Module - Even with the most optimistic projection of nuclear rocket capability, direct Mars landing and takeoff with the interplanetary vehicle will not be competitive with an excursion module approach. This is due not only to the practical difficulties of atmospheric operation with the nuclear stage, but also to the large difference in payload capsule requirements for the interplanetary vehicle compared with the landing craft. In addition, transporting to Mars surface the large quantities of propellant and tankage required for the return flight would cost considerably more propulsion than leaving this material in orbit. For this reason, the chemically propelled separate excursion module concept was selected for this study.

Actual choice of parking orbit for the interplanetary vehicle is a complex tradeoff between the high specific impulse - high gross weight nuclear vehicle and the lighter weight but low specific impulse excursion module. If trapped radiation belts do not present a problem, an eccentric orbit of law periapsis is indicated, since this minimizes propulsion on the heavy interplanetary spacecraft. Moreover, the propulsion penalty imposed upon the excursion vehicle is not as severe as might be indicated by the specific impulse disadvantage of the chemical rocket with respect to nuclear. This is due to the fact that the landing phase reties mainly upon atmospheric braking rather than propulsion, and the heat shield is equivalent to a very high specific impulse propulsion system. Thus, an eccentric orbit only affects the Mars takeoff stage AV requirement, Should radiation belts make the eccentric parking orbit unacceptable, the landing vehicle might be allowed to enter directly during final approach to the planet, and retro propulsion then used to place the interplanetary vehicle in a near circular high altitude orbit. In either case, the Mars takeoff stage of the excursion module will need sufficient propulsion to rendezvous with the interplanetary spacecraft. Depending upon the orbit eccentricity, this amounts to somewhere between 12,000 and 15,000 ft/sec of actual velocity increment, not counting gravity and drag losses.

Before determining the weight of the Mars excursion craft, it is necessary to establish the following design and operational features of the vehicle:

- Entry Body Configuration
- Mars Landing System
- Take-off Stage Propulsion

For the sample mission discussed here, two of the four crewmen constitute the landing party. A two man life support capsule with communications and control equipment must be delivered to the surface and returned to orbit for rendezvous with the nuclear rocket. In addition to the basic capsule of 5200 pounds, a nominal payload of 5000 pounds has been included for scientific gear and portable life support equipment necessary for the 5 day Martian stay. This equipment is abandoned on Mars, and thus, does not affect the propulsion requirements of the take-off stage.

(a) Entry Body Configuration - Selection of a suitable configuration for the round trip excursion module is considerably more difficult than in the case of an unmanned landing capsule which only makes a one-way trip. The entry vehicle must have sufficient payload volume to accommodate not only the crew and supporting equipment, but also the entire propulsion system of the takeoff stage. In the extreme case, this involves the protection of low density cryogenic propellants such as liquid hydrogen and oxygen against the heat of atmospheric entry. Configuration of the landing vehicle is dependent upon the degree of maneuvering desired. Possibilities include conical, winged, or lifting body designs as depicted in Figure 11. Even though the conical design has limited maneuverability, its lower structure and heat shield weight are probably over-riding factors. Moreover, the blunt conical shape affords a low surface to volume ratio for simpler packaging and thermal control of the payload.

Entry from the eccentric parking orbit is accomplished by firing a small braking rocket to bring the excursion module periapsis into the Mars atmosphere at a shallow angle. The relatively low entry velocity (~ 15,000 ft/sec) and gradual atmospheric density gradient result in relatively moderate heating and deceleration loads. Entry capsule analyses carried out under the GE Voyager studies[3] indicate that the heat shield and structure weight will amount to only about 20% of the capsule gross weight for velocities up to 25,000 ft/sec.

(b) Mars Landing System - Parachutes, horizontal glide, and propulsion hovering were all considered as possible landing techniques? However, the thin atmosphere and unprepared landing site will pose major problems for either of the first two methods. For example, 28 parachutes, each 90 ft in diameter would be needed to land the excursion module at 40 ft/sec. impact velocity. Weight of the parachute system is estimated to be 3,300 pounds, not counting any additional structure or shock absorbers for the impact of landing at this velocity. Approach velocity with winged glider vehicles would be several hundred knots, clearly unsuitable for anything but the smoothest prepared runway. Consequently, vertical landing with propulsion hovering, similar to the LEM technique, was chosen for the sample design to achieve maximum flexibility in meeting unforseen surface conditions. The landing sequence, then, involves aerodynamic braking to sub-sonic velocity, deployment of a drogue chute, release of the heat shield, and finally, retro-propulsion for approximately one minute to effect the final cross range maneuvers and braking at touch-down. Vertical landing not only provides last minute site evaluation, but also results in a minimum of preparation for takeoff, since the landing stage can serve as a launch platform. Approximately 1700 pounds of hydrogen-oxygen propellant, plus 350 pounds of tankage and engine weight will be required for the two man landing vehicle.

(c) Takeoff Stage Propulsion - Depending upon the rendezvous orbit altitude and eccentricity, the Mars takeoff vehicle will have to attain between 12,000 and 15,000 ft/sec. of velocity increment. Allowing for gravity and drag losses, the total ΔV requirement will run from 14,000 to 18,000 ft/sec. Single stage propulsion could be used, but for the higher velocities, staging will save appreciable weight. Assuming tank structure and engine weight will amount to 10% of propellant loading, and allowing for 10% boil-off loss while on Mars, a 17,500 ft/sec hydrogen-oxygen propelled Mars takeoff vehicle would weight 24,200 pounds for two stage configurations.

As a secondary function, the chemical propulsion system of the Mars takeoff vehicle provides a considerable degree of abort capability during the initial phases of the mission when the interplanetary spacecraft is being launched from Earth orbit. By removing some of the excursion module equipment, the four man crew could return to Earth orbit in the small chemically propelled vehicle, if abort is required within the first few days after Earth escape. Limited capability of the excursion module life support capsule will, of course, restrict the use of this abort system to near-Earth emergencies.

(d) Excursion Module Sample Designs - Based upon the previously discussed choices, vehicle weights are presented in Table VI. These weights are representative of a conical configuration similar to that shown in Figure 12.

TABLE VI

Landing Stage	
De Orbit Rocket	600
Heat Shield and Structure	4750
Landing Rocket Propellant	1700
Scientific Payload	5000
	12,050 #
Takeoff Stages (ΔV = 17,500 ft/sec.)	
Two Man Capsule, Controls and Power Supply	5200
Structure and Engine	1700
Propellant Loading (10% Boil off)	17,300
	24,200 #
TOTAL	36,250 #

If extensive glide maneuvering is desired, some form of winged vehicle, such as that shown in Figure 13, might be used. Weights have not been estimated for this type of configuration, but a considerable increase in landing stage structure and heat shielding will be necessary.

6. Payload Summary - Based upon the subsystems discussed in the previous sections, a payload weight for the sample vehicle design is given in Table VII. These weights are then used to develop propulsion system comparisons.

TABLE VII

8 KW Nuclear APU (Thermoelectric)	5,050
2.5 KW Photovoltaic APU (Backup)	1,000
Life Support	11,500
Crew Cabin and Furnishings	6,000
Electronics	4,100
Mars-Excursion Module	36,250
Radiation Shelter	10,500
TOTAL	74,400 #

Out of this total, approximately 39,750 pounds will be jettisoned at Mars or en-route to Mars so that the return payload is only 34,650 pounds.

There are, of course, many possibilities for dividing payloads among multiple vehicles, some of which travel only one way as logistics carriers. In this study, however, the assumption is made that every vehicle will be self sufficient, even if the mission involves more than a single spacecraft.

B. Characteristic Velocities

Depending upon the launch year, trip time, waiting time at Mars, and orbit conditions at both Mars and Earth, the mission characteristic velocity will vary over a wide range. Figure 14 shows the relationship between minimum ΔV and total round trip time for the highly favorable 1971 opposition.[5] Departure is from a 300 mile initial Earth orbit, and eccentric orbits are assumed at Mars, and upon return to Earth. It is significant that the ΔV requirement drops rapidly with increasing trip time until about 15 months total mission duration (assuming a short stay of about 5 days at Mars). For longer missions, the return flight becomes more difficult and requires close approach to the sun at perihelion (less than 0.5 AU), so that missions from 15 to 24 months in length do not appear attractive. Thus, a Mars trip must either be performed within the restricted time limits of a single opposition, or a long stay time at Mars will be necessary. Typical waiting times of 130 to 450 days are indicated, but the greater flexibility in choosing return launch dates results in a lower overall velocity requirement for missions from about 24 to 30 months total duration.

For purposes of determining propulsion requirements, it is assumed that initial flights would have to be performed within a single opposition, staying in the vicinity of Mars for only a few days. Subsequent flights may be programmed for a long stay at Mars to take advantage of lower energy requirements. The large propulsion savings thus made possible should prove a powerful incentive towards early construction of a Martian bsse if any continuing exploration is planned. The savings for long waiting time are particularly attractive during difficult oppositions such as in the late 1970's, since the energy requirement for a very long mission (over 2 years) does not increase as rapidly as in the case of 12-15 month trips. One particularly important factor in establishing a Mars mission profile is the selection of trajectories such that the outbound segment of the flight requires less ΔV than the return leg. This results in lower vehicle gross weight, since the return payload is less than half that initially carried to Mars.

In order to illustrate the effect of departure year upon ΔV, the curve of Figure 15 is presented for 15 month duration trips from 1969 to 1984. From a minimum of 47,000 ft/sec. in 1971, the requirement increases to a peak of about 70,000 ft/sec. in 1979. Detailed studies were carried out on the basis of a 1973 mission, with variations introduced to cover later opportunities. It is recognized, of course, that 1973 represents an optimistic date for Mars landing missions in view of the current status of nuclear propulsion development, and the lack of any national committment to post-Apollo manned space exploration. Nevertheless, the concepts presented here should be applicable to any time period with appropriate adjustment in vehicle weight and engine size.

IV. Propulsion System Analysis

Having selected a consistent set of .payloads and mission characteristic velocities, it is possible to define the general features of a propulsion system suitable for performing the manned Mars mission. In the case of hydrogen cooled solid core nuclear rockets, the propulsion system consists of two major subsystems:

- Engine and Shielding
- Propellant Tankage and Insulation

A. Engine and Shielding Characteristics

1. Solid Core Nuclear Rockets - Two different types of solid core nuclear rocket were considered in this study; the graphite moderated thermal reactor system and the refractory metal fast spectrum reactor. Although the graphite engine is the only one under full scale development at this time, the metallic core design offers considerable promise for the manned Mars mission due to its inherently high re-start capability, and resistance to fuel clad erosion for long burning times. Furthermore, the smaller size of a fast reactor provides an advantage in lower engine and shield weight. As shown in Figure 16, the size and weight advantage of a fast metallic core is particularly significant in the lower thrust ratings. These values were estimated on the basis of equal I_{sp} (830 seconds), Should it be necessary to restrict the graphite core to lower exhaust temperatures in order to attain sufficient lifetime, the lower I_{sp} will result in an even greater difference.

Small core frontal area is a particularly significant factor since it permits the clustering of several low thrust engines without incurring excessive shield weight penalties. For example, a typical manned vehicle weighing 1,000,000 pounds in Earth orbit could use three or four 40 or 50,000 pound thrust metallic engines. Similar clustering of graphite reactors is not attractive, however, because the engine weight and core frontal area for graphite engine of 50,000 pounds thrust is very little less than for a 150,000 pound thrust unit. Clustering capability, together with long life potential for each reactor, will afford high overall reliability without imposing unrealistic reliability goals upon the individual engines.

2. Shielding - The approximate shielding requirements for nuclear rocket engines can be established by relating the reactor power level, operating time, allowable crew dose, and propellant tank length. Figure 17 shows unshielded neutron and gamma doses for a typical range of conditions. These dose levels are based upon a linear decrease in hydrogen column length throughout the specified firing interval, and include decay dose to infinite time.[6] A sample case is indicated for a 2600 MW reactor operating 60 minutes with a 75 foot hydrogen tank separating the reactor from the crew quarters. Total unshielded neutron and gamma doses would be 2×10^6 and 3×10^6 REM respectively. Since the crew absorbs considerable solar and cosmic radiation dose during a mission (-100 REM), the reactor contribution must be kept low. Figure 18 relates the neutron and gamma attenuation to shadow shield weight in #/ft^2 of frontal area.

In order to obtain nearly equal dose from neutron and gamma sources, a mixed shield of 88% tungsten and **12%** lithium hydride is indicated. A total thickness of 770 #/ft^2 will restrict crew dose to less than 35 REM for the case previously discussed (2600 MW for 60 minutes). If the crew remains in the solar flare shelter during all firing intervals, shield weight could be reduced by about 100 #/ft^2 due to the shelter wall material.

3. Selection of Engine Size - Three factors are of major importance in the selection of engine size for a particular vehicle design.

- The tradeoff of gravity losses (low T/W) against engine weight (high T/W)
- Engine operating life
- Thrust rating vs. availability and development lead time

Optimum thrust to weight ratios have been determined by Harris and Austin of NASA Marshall 7 for maximum payload fraction from orbital launch to injunction at various hyperbolic excess velocities. These studies show that the ideal thrust to vehicle initial weight should be from 0.25 to 0.35 for typical values of I_{sp} and structure fraction. However, the curve is fairly flat, and dropping T/W to 0.15 will only increase the vehicle orbital weight requirement by 1.7% weight for a given payload injected to 14,000 ft./sec. hyperbolic excess velocity. Thus, considerations must be given to other factors before a selection of engine thrust level can be made.

Engine life expectancy will, of course, be a serious constraint in early applications, and thus may force the use of high thrust ratings (T/W from .30 to .50). On the other hand, smaller engines are more likely to be available at any given date, so that a low initial thrust is desirable (T/W .15 to .20). In this study, a value of .15 was selected for cases involving the fast metallic engine, since this type of core should provide considerably longer life than the most severe Mars mission requirement. Slightly higher thrust to weight ratios were used for graphite engines, however, so that engine operating time was restricted to less than 30 minutes for any stage. In addition, ratings of less than 50,000 pound thrust were not considered for graphite core engines.

Based upon the initial T/W of 0.15 for metallic core powerplants, typical stage thrust requirements range between 120,000 and 400,000 pounds for orbital launch propulsion. For a typical case (1973 launch) the requirement is 130,000 pounds thrust, and the weight of a single engine for this example can be obtained by combining the basic engine weight and frontal area data (Figure 16) with the shield curves (Figure 18). The resultant weight, 10,100 pounds, represents an engine thrust to weight ratio of 13 to 1. A graphite engine system of similar rating would weight 20,900 pounds, for a thrust to weight of 6.25 to 1. If the desired rating of 130,000 pounds were to be obtained by clustering 3 smaller engines, the thrust to weight ratios for the engine and shield clusters were found to be 7.8 for the metallic system and 2.2 for graphite. The influence of engine thrust to weight ratio upon required spacecraft gross weight in orbit is shown in Figure 19 for a typical set of Mars mission parameters. This curve is based upon a vehicle with staged tankage, but no staging of engines. Note that the vehicle *gross* weight increases sharply if engine thrust to weight drops below 5 to 1.

B. Propellant Storage

Although nuclear rockets offer a specific impulse advantage of about 2 to 1 over advanced chemical rockets, the performance margin is partially offset by the inherently higher dead weight fraction of a nuclear stage. This is due not only to the weight of engine and shielding, but also to the structure and insulation required for launching and long term storage of low density liquid hydrogen propellant. In order to examine the tankage problem in some detail, a vehicle configuration was developed, and structural and thermal control studies carried out for the particular tank sizes and shapes required for this vehicle. Figure 20 shows assembly of the configuration which consists of a central payload and propulsion module surrounded by multiple drop-off tanks. The central module is launched by a single SATURN V booster, and includes a large hydrogen tank of 112,000 pounds capacity. Each of the drop-off tanks holds 45,000 pounds of hydrogen, and these are launched in sets of four on additional SATURN V tanker vehicles. The actual number of drop-off will vary, depending upon the Δ V required, but for a 1973 mission, twelve are needed. Figure 21 shows the proposed launch arrangement of the tanker vehicles, with the four loaded tanks mounted in tandem within a structural shell that also serves as the aerodynamic shroud. By employing external structure to reinforce the tanks during booster acceleration, the weight of tankage can be minimized. After installation on the nuclear rocket vehicle, the light weight tanks will be exposed to only moderate acceleration (less than 1 g), rather than the 7 or 8 g experienced in attaining initial orbit.

1. Structural Requirements - A comprehensive analysis of nuclear rocket propellant tank structure requires definition of a vehicle configuration, and the study of booster and environmental loading conditions experienced throughout the launch, assembly, and interplanetary phases of typical mission. Tank designs must be based upon consideration of aerodynamic, acceleration, maneuvering, and pressurization loads, as well as the limitations of booster shroud dimensions. Furthermore, the tankage arrangement should be compatible with earth orbital assembly techniques, since multiple launching will be necessary for manned interplanetary missions using SATURN V booster.

Based upon these factors, the previously described configuration of staged tanks has been developed. Weights were derived for cylindrical tanks using various construction materials. Figure 22 summarizes structural weight as a function of propellant loading for aluminum monocoque, aluminum honeycomb, and beryllium monocoque tanks. Preliminary Saturn loading conditions were assumed, and two particular cases are identified as representative of the main tanks and drop-off tanks required for the reference vehicle design. The range of values (shaded area) indicated for aluminum monocoque tanks represents studies of a nuclear manned maneuverable vehicle (Titan III booster) and a RIFT type vehicle (SATURN V booster).

In order to develop overall vehicle weight tabulations, two points were selected from the curve for aluminum honeycomb. These values are probably conservative, since the actual in-space weight of the tanks might be reduced further by jettisoning part of the structure after attaining orbit. The tankers, for example, can be designed with an independent structure so that no bending loads will be imposed upon the tank shell. Weight of the redundant structure placed-into orbit is not nearly as critical as the weight represented by the tank wall, since the latter must be accelerated through the velocity increment of the interplanetary flight, as well as the launch to orbit phase.

2. Propellant Thermal Control - In addition to the tank shell required for structural reasons, the propellant storage system weight includes a substantial penalty for thermal control of liquid hydrogen over the long periods typical of interplanetary flight. Various combinations of vented, non-vented, and refrigerated storage techniques have been considered.[8] Refrigeration (recondensation) appears to have some weight advantage for storage times in excess of about 250 days, based upon presently available technology in helium cycle refrigeration machinery. Figure 23 shows a diagram of such a system, and gives the relationship between re-liquification capacity, weight, and electrical power input. A capacity of 42 pounds LH_2 per day was found to be suitable for the 112,000 pound main tank in the sample vehicle design, and this system would require 22.8 KW electrical input power to yield 100 watts of net refrigeration. Taking SNAP-8 as the electrical power source, an incremental weight of about 150#/KW must be charged against the refrigeration plant. Thus, the 100 watt refrigerator (22.8 KW input) will cost 3420 pounds of power supply. plus 1165 pounds of helium cycle machinery, for an overall specific weight of 46#/watt. Advances in powerplant and refrigeration technology could produce a saving of about 2 to 1 compared to these weights, so that for planning purposes, a range of specific weights between 25 and 50 pounds per watt of refrigeration should be considered.

Detailed studies were performed to determine heating loads experienced during each phase of the Mars mission. Figure 24 presents the results of these studies for the two previously identified propellant loadings (45,000 and 112,000 pounds LH_2). Vented and non-vented designs were considered. and in the case of the main tank, the weight of a combined insulation-refrigeration system has been plotted for reference. Typical storage times are identified for each group of tanks. In all cases, the tanks weights used in vehicle performance analyses were taken from the non-vented curve, and refrigeration was not assumed. The saving attainable with refrigeration (about 1000 pounds for the main tank) is not sufficient to offset the reliability problems associated with long term operation of the refrigeration equipment. Insulation weights are based upon the vehicle being oriented to within $\pm 1^{\circ}$ for minimum solar view factor during coast phases of the mission. The heating load due to a 600°F radiator (nuclear APU) was also factored into the thermal balance on the main tank.

3. Meteoroid Protection - Due to their large surface area, the hydrogen tanks of a nuclear rocket spacecraft will be exposed to possible meteoroid puncture on long duration missions. The actual distribution of meteoroid flux in deep space is, of course, poorly defined at this time. However, the meteoroid problem can be explored by assuming that the near-earth conditions will prevail throughout a Mars mission. On this conservative basis, the staged tank configuration has been examined in detail to arrive at overall survival probabilities. As shown in Figure 25, the inherent protection afforded by the tank shell and thermal insulation is equivalent to about .264 inches of aluminum. Based upon the criteria proposed by Loeffler, Lieblein and Clough of NASA Lewis, the relationship between vulnerable area, survival probability, exposure time, and material thickness can be derived. Figure 26 presents these parameters for aluminum arranged in a "bumper" configuration which is representative of the tank wall and its separate insulation material.

In each phase of the mission, the effective tank exposure area changes due to the varying number of tanks present, and their relative positions with respect to each other. Table VIII summarizes the area and survival probability data developed for a sample mission.

TABLE VIII

MISSION PHASE	EARTH ORBIT ASSEMBLY		EARTH TO MARS COAST		MARS ORBIT WAIT		MARS TO EARTH COAST		OVERALL
EXPOSURE TIME PER PHASE	1000 HRS.		5000 HRS.		100 HRS.		5000 HRS.		11, 100 HRS.
TANKS	A_S	P_1	A_S	P_1	A_S	P_1	A_S	P_1	P_2
DROP-OFF									
• GROUP I	14,400	.950	0	1 0	0	1.0	0	1.0	.950
• GROUP II (4 TANKS)	9600	.970	7900	985	0	1.0	0	1.0	.955
• GROUP III (2 TANKS)	4800	.995	2400	.995	3900	999	0	1.0	.989
MAIN									
AFT	2400	.999	1600	999	2400	999	0	1.0	.997
FORWARD COMPARTMENT	2400	.999	500	999	1200	999	1600	.995	.991
P_3		.915		.978		.997		995	P_0=.888

A_S= TOTAL EXPOSED SURFACE OF TANKS IN A GIVEN GROUP (FT^2)

In this table, P_1 represents the nonpuncture probability of a given tank group during one mission phase and P_2, the probability for that group throughout the entire mission. Similarly, the probability for all tanks within a single mission phase is P_3, and for all tanks throughout the mission P_o - Note that even with the pessimistic meteoroid population data, P_o is 0.888, and if this is divided by P_3 for the Earth orbit assembly phase, the tanks have 0.97 probability of non-puncture for all phases after checkout and launching from Earth orbit. Therefore, unless the meteoroid environment of interplanetary space proves unexpectedly high, no protection other than the inherent structure and insulation should be necessary for liquid hydrogen tanks.

4. Tankage Weight Summary - By combining the previously determined structure and insulation weights, representative stage propellant fractions were established. Although the fraction varies somewhat, depending upon the tank group being considered, a value of 0.82 was found to be typical. Total propellant loading, then, is 82% of stage weight, without considering the nuclear engine or the payload. Part of the total propellant will be lost due to residuals in the tankage, and the requirement for decay heat removal after each firing interval. These losses total about 6% of the propellant, and this can be treated as part of the staging weight in the analysis of vehicle performance.

V. VEHICLE PERFORMANCE EVALUATION

Payload, propulsion, and mission profile factors were combined to arrive at overall vehicle performance characteristics. Major emphasis was placed upon the restartable engine - staged tank configuration, but a vehicle with staged engines and tankage was also considered in order to illustrate the capability of a first generation engine, assuming that only limited restart capability may be attained.

1. Restartable Engine - Staged Tankage Configuration - The major system parameters of a vehicle are presented in Table IX for the case of a 1973 mission. This case represents the basic design described throughout the paper.

TABLE IX

Propulsion System Weight	
130 K Fast Metallic Engine (2600 MW)	7,000 #
Engine Shield (10 REM Dose)	3,100 #
Tankage and Insulation ($\lambda = .82$)	143,500 #
Propellant Loss	40,000 #
Useful Propellant	612,000 #
	805,600 #
Payload Weight	
Return Payload	34,650 #
Payload Jettisoned at Mars	39,750 #
	74,400
Spacecraft Performance, (Gross Weight = 880,000#)	
Total ΔV Capability	54,900 ft/sec
Total Burning Time	65 minutes

Figure 27 shows the variation in performance of the vehicle for different payloads. The effect of jettisoning a major portion of the total payload at Mars arrival can be seen from these curves.

The effect of varying launch year upon vehicle weight is an extremely important factor in evaluating the Mars mission capability of nuclear rockets. Table X shows that the weight required in orbit increases from a minimum of 790,000# in 1971 to a peak of 2,500,000# in 1979, then gradually decreases through the early 1980's. The staged tank configuration is attractive up to about 60,000 ft/sec total ΔV, but for the higher energy missions, there is an advantage in staging the engines as well. For this reason, the 1977, '79, and '82 missions are shown to use a large orbital launch stage in combination with the same basic vehicle configuration proposed for the less difficult missions. Thus, a vehicle designed for the 1973 or '75 mission might be adapted to later missions by adding a separate stage of appropriate ΔV. Thrust requirement for this stage ranges from about 250,000# to 400,000#.

TABLE X

LAUNCH YEAR	ΔV ORBIT TO ORBIT AND RETURN (FT/SEC)	CONFIGURATION	GROSS WEIGHT IN EARTH ORBIT (#)	MINIMUM STAGE THRUST REQUIREMENT (#)	NO. OF SATURN V BOOSTERS
1971	47,000	SINGLE STAGE ENGINE (S) WITH DROP-OFF TANKS	790,000	120,000	4
1973	50,000	"			
1975	57,000		1,180,000	170,000	5
1977	67,000	SAME PLUS EARTH ORBITAL LAUNCH STAGE	1,850,000	100,000 a 280,000	8
1979	70,000		2,500,000	100,000 a 400,000	11
1982	61,000	"	1,700,000	100,000 a 250,000	7
1984	53,000	SINGLE STAGE ENGINE (S) WITH DROP-OFF TANKS	980,000	150,000	4

- ECCENTRIC MARS ORBIT
- 300 MILE INTIAL EARTH ORBIT
- I_{sp} = 830 SEC
- ENGINE THRUST TO WEIGHT = 13
- 5000 FT/SEC ADDITIONAL ΔV REQ'D FOR:
  - GRAVITY LOSSES
  - MID COURSE CORRECTION
  - MARGIN
- 2 MAN CHEMICALLY PROPELLED LANDING VEHICLE CARRIED

2. Staged Engine Configuration - The 1973 mission was used to illustrate the combined effect of limited life and restart, and higher engine weight, upon vehicle characteristics. Table XI shows the major parameters of such a vehicle having three nuclear propulsion stages. Total burning time per stage is limited to 30 minutes. One restart is required for the second stage engine. This engine is used for Mars capture, and is then restarted after only 5 days shutdown, for the Mars departure maneuvers. Engine and shield weights are based upon the curves of Figure 16 for graphite core systems. Spacecraft gross weight of 1,350,000# requires 6 SATURN V boosters, as compared with 4 boosters for the staged tank configuration.

TABLE XI

Propulsion System Weight	
Engine and Shield	
Stage One - 250 K	23,000 #
Stage Two - 200 K	21,000 #
Stage Three - 50 K	18,500 #
Tankage and Insulation ($\lambda = .82$)	217,100 #
Propellant Loss	60,000 #
Useful Propellant	936,000 #
	1,275,600 #

TABLE XI. (Cont'd)

- Payload Weight

Return Payload	34,650 #
Payload Jettisoned at Mars	39,750 #
	74,400 #

- Spacecraft Performance (Gross Weight 1,350,000 #)

Total ΔV Capability	54,900 ft/sec
Burning Time	
-Engine #1	30 Min.
-Engine #2	
Mars Capture	12 Min.
Mars Departure	14 Min.
-Engine S3	18 Min.

3. All-Chemical Configuration - In order to illustrate the performance advantage of nuclear rocket propulsion over chemical, a four stage hydrogen-oxygen vehicle was sized for the 4 man 1973 Mars landing mission of 15 months duration. With 450 seconds I_{sp} and a useful propellant fraction of 0.90, this vehicle would weight 6,700,000 pounds in orbit, or equivalent to about 28 Saturn V boosters. Thus, unless a system for refueling en-route or at Mars can be developed, prohibitive launch weight will restrict chemical propulsion to less difficult missions such as the flyby, or possibly a 2 1/2 year trip involving long stay time at Mars.

VI. CONCLUSIONS

Presently conceived solid core nuclear rocket propulsion systems should be capable of performing "fast" Mars missions in the 1970's with reasonable launch vehicle requirements. Even without the development of a NOVA class booster, the combination of SATURN V and simple Earth orbit assembly operations can provide sufficiently large spacecraft (400 to 1500 tons) to permit Mars landings throughout the next decade. However, the high energy requirements of the late 70's will make it extremely attractive to develop nuclear propulsion so that it can be available by 1975, at the latest. Total thrust ratings of 120,000 to 170,000 pounds will be adequate for the period through 1975, but higher thrust (up to 400,000 pounds) will have to be developed for later missions.

Although this paper has been limited to high thrust systems, it is recognized that electric propulsion offers considerable promise in further reducing the vehicle weights for manned Mars missions. Studies are currently in progress at GE to assess various combinations of low and high thrust nuclear propulsion applied to Mars flight.

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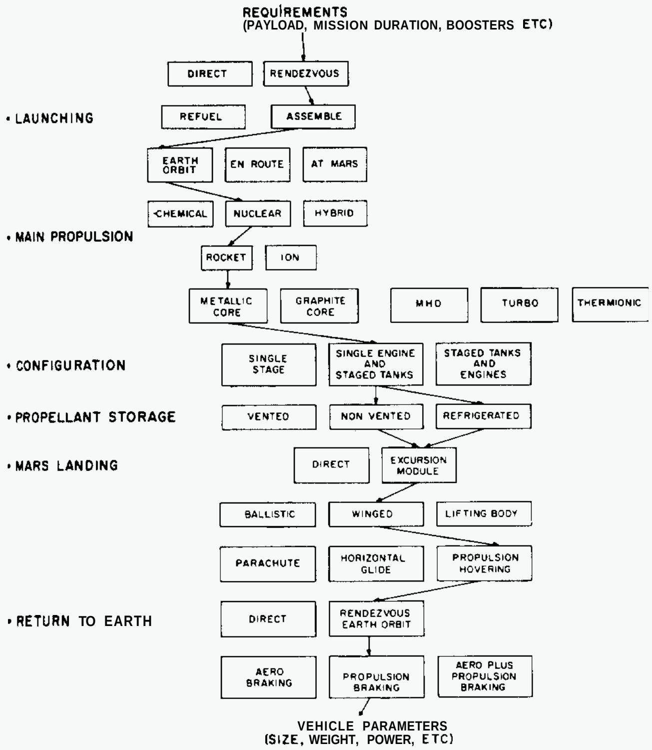


Figure 1. Mission Profile Options

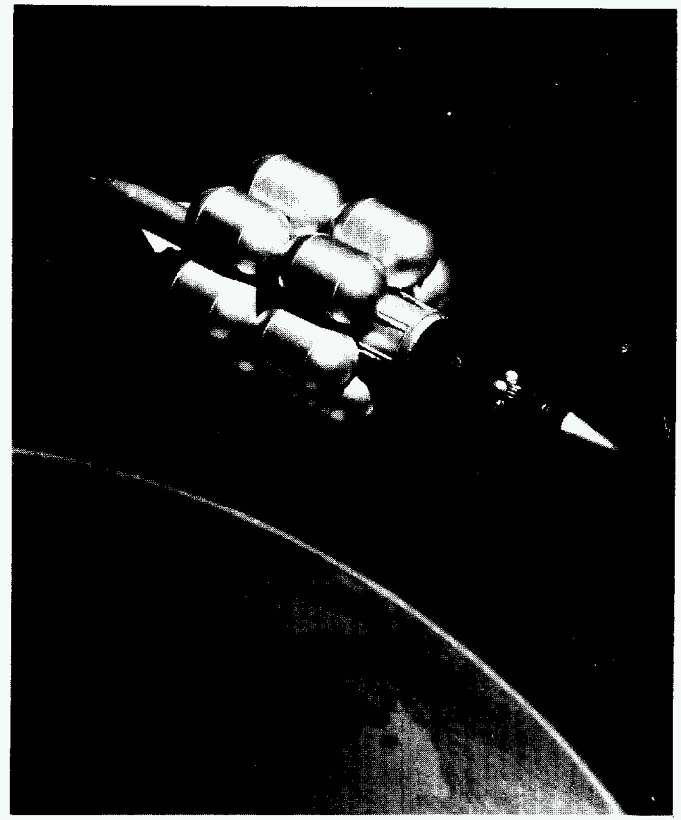


Figure 2. Manned Mars Vehicle Reference Configuration

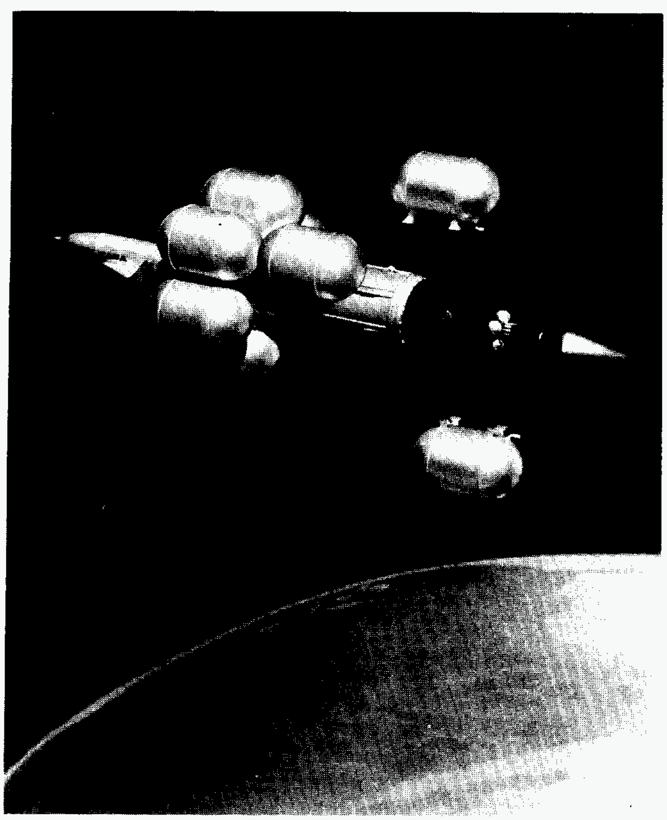


Figure 3. Staging of Propellant Tanks During Earth Escape

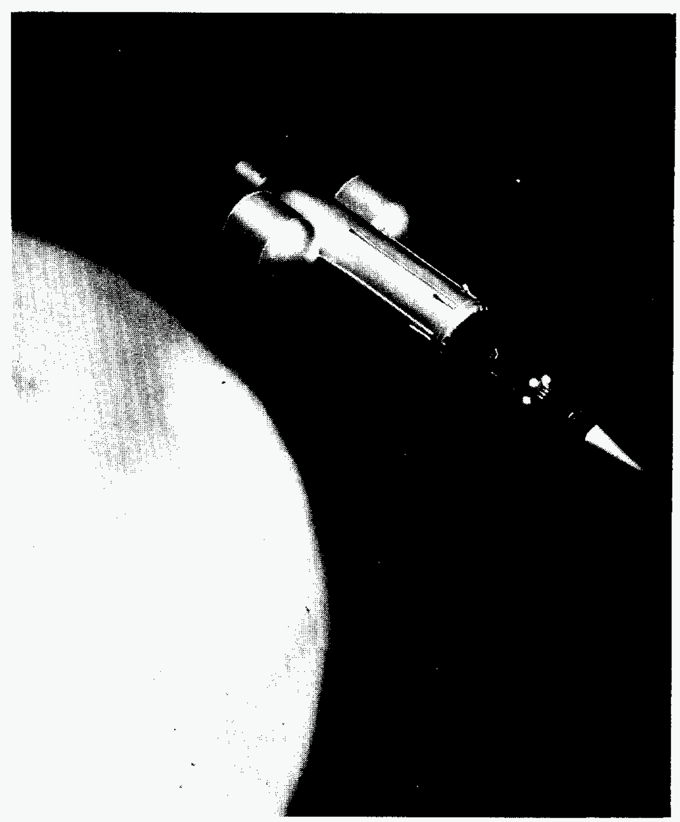


Figure 4. Nuclear Vehicle Configuration During Mars Orbit Escape

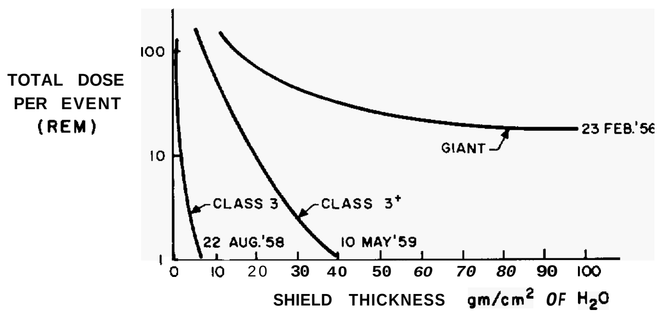


Figure 5. Solar Flare Shielding Requirements

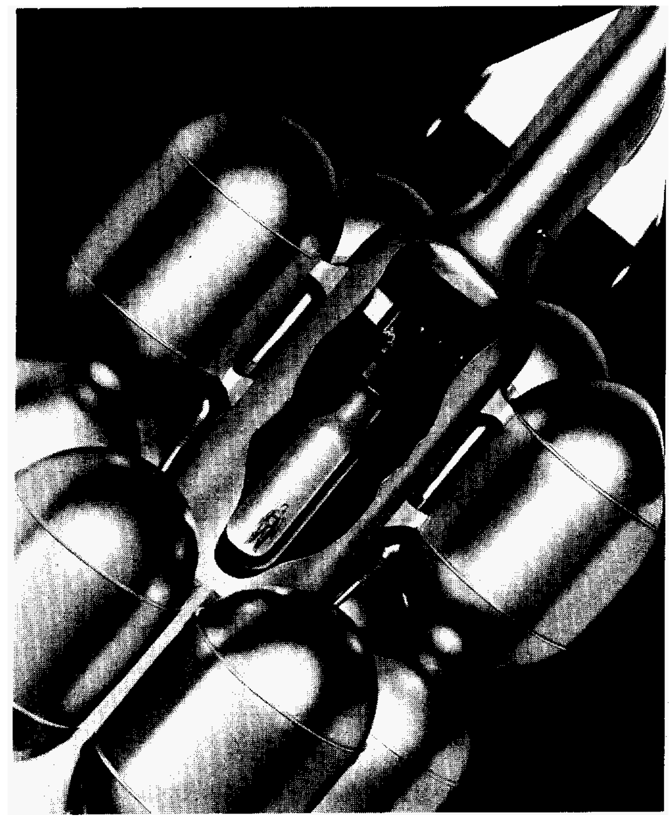


Figure 6. Solar Flare Shelter Arrangement

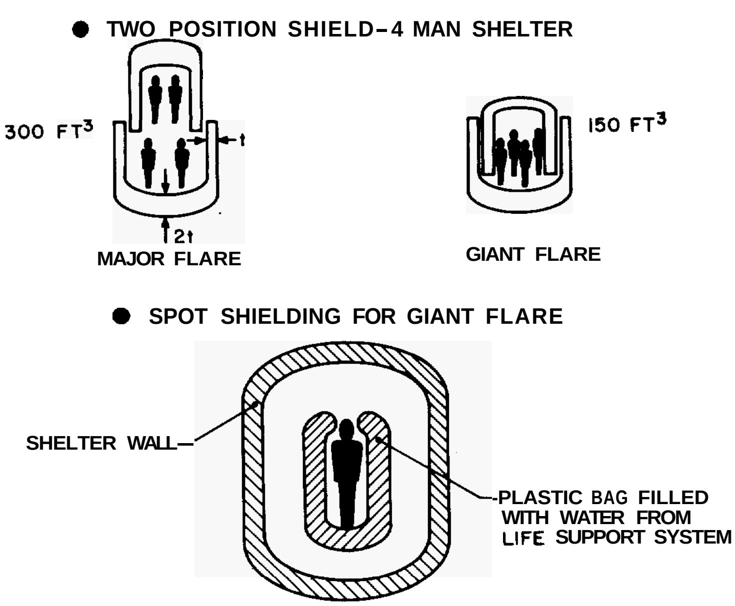


Figure 7. Concepts for Reduced Solar Flare Shield Weight

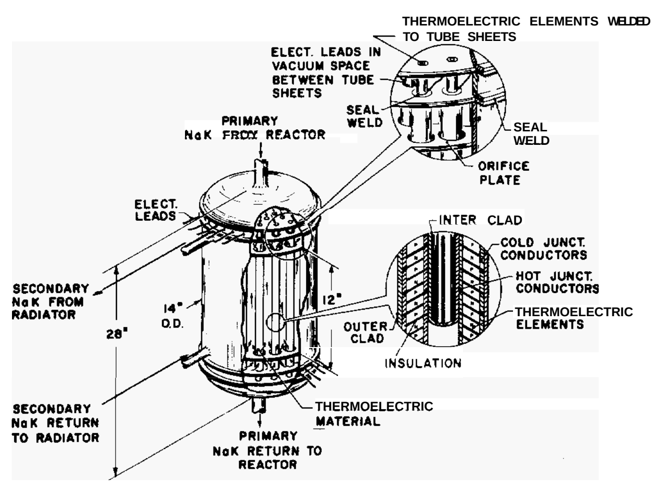


Figure 8. Thermoelectric Generator Arrangement

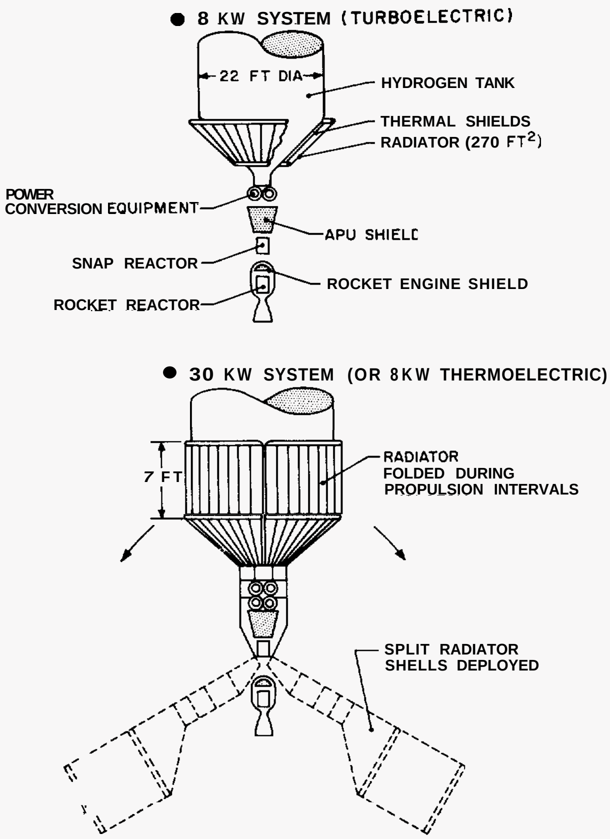


Figure 9. Integration of Nuclear Electric APU

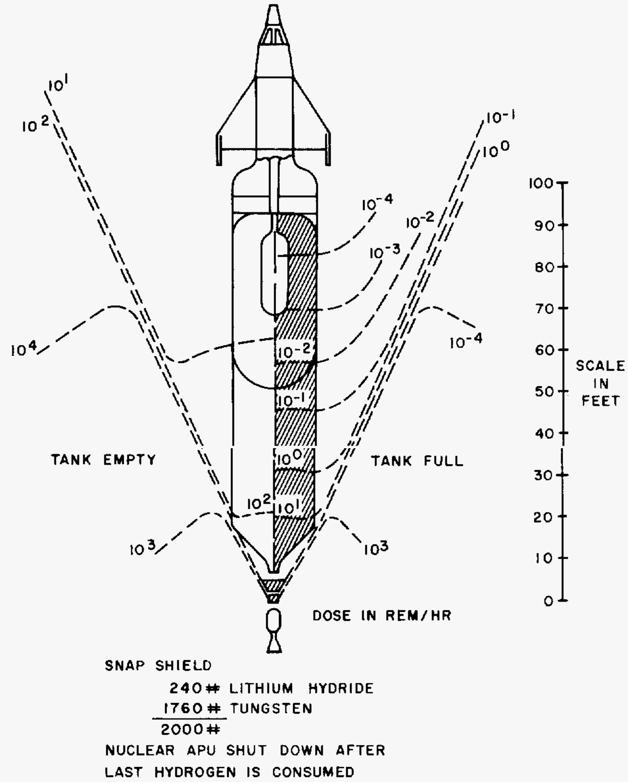


Figure 10. Isodose Plot for 8Kw Nuclear Thermo-electric APU

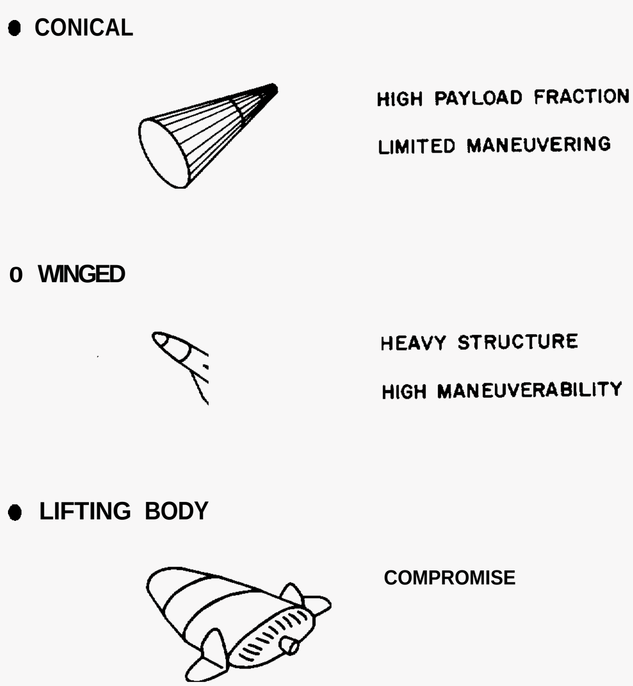


Figure 11. Mars Excursion Module Concepts

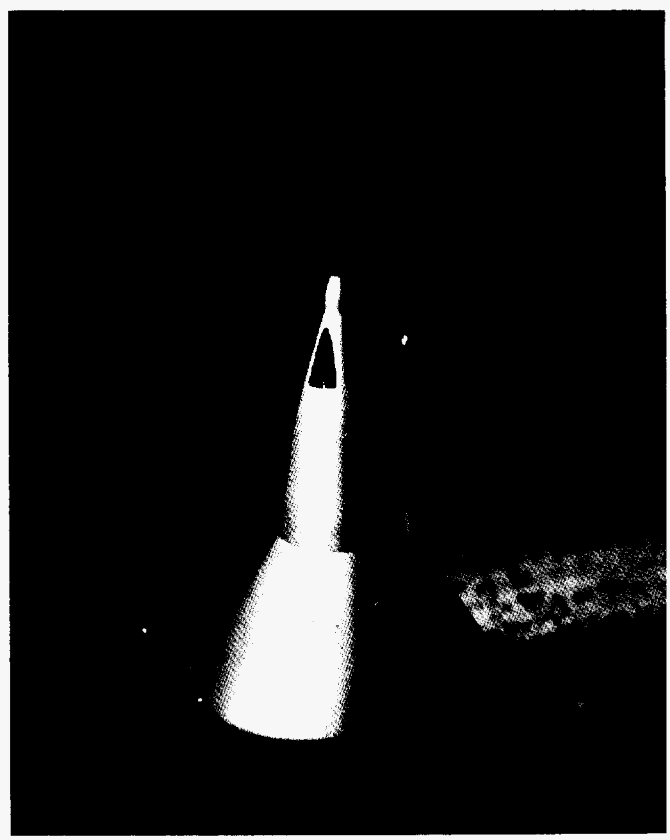


Figure 12. Conical Configuration Mars Excursion Module

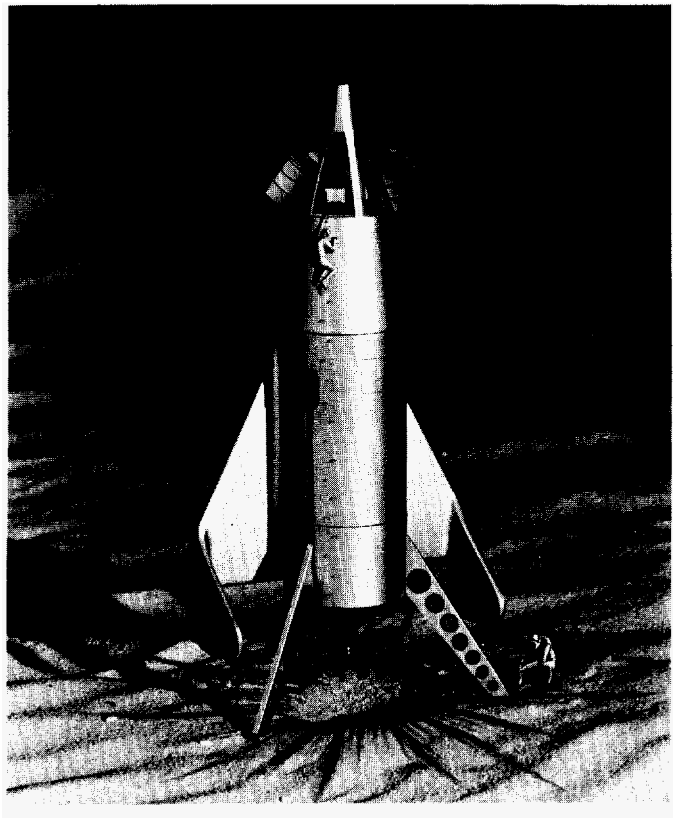


Figure **13**. Winged Configuration Mars Excursion Module

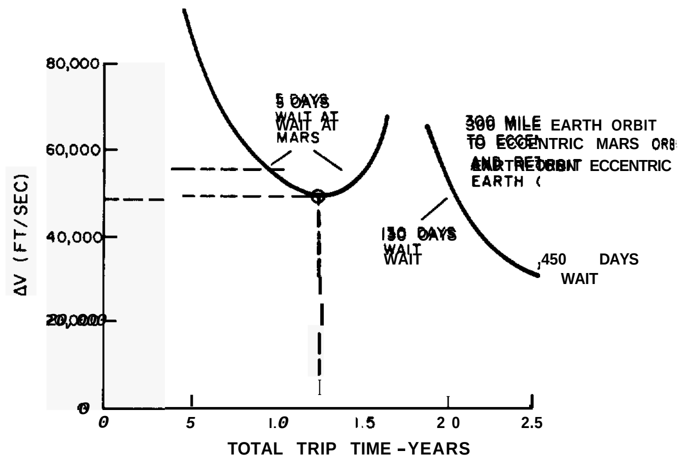
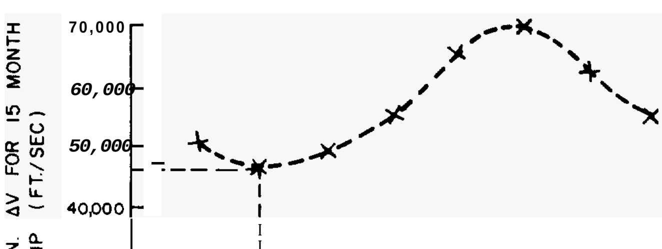


Figure 14. Mars Velocity Requirements for 1971 Departure



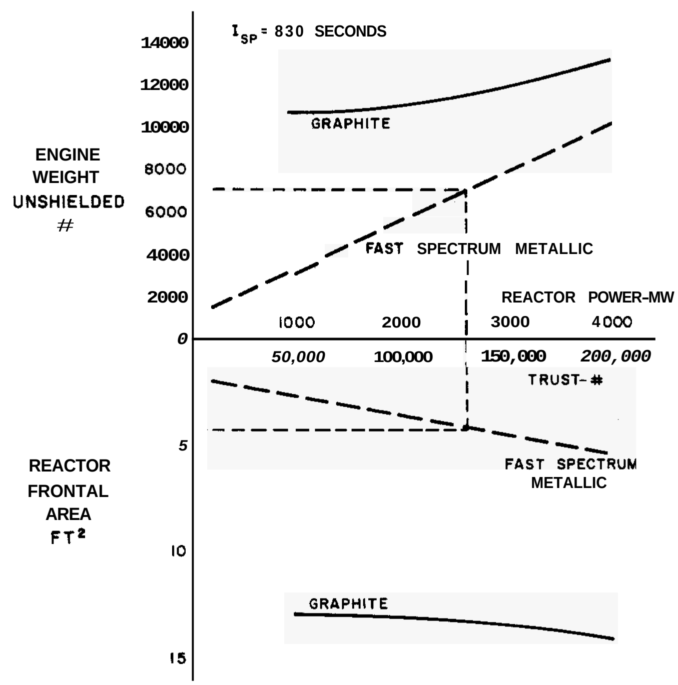


Figure 16. Nuclear Rocket Engine Size and Weight

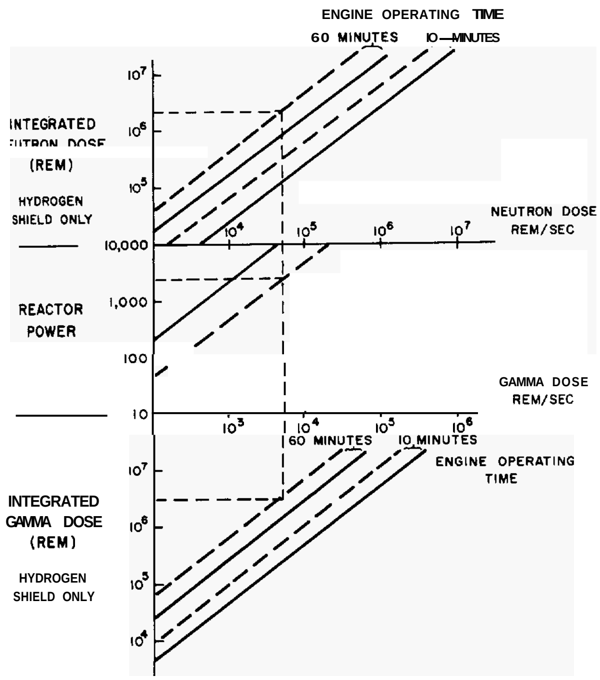


Figure **17.** Neutron and Gamma Dose - Hydrogen Propellant Shield **Only**

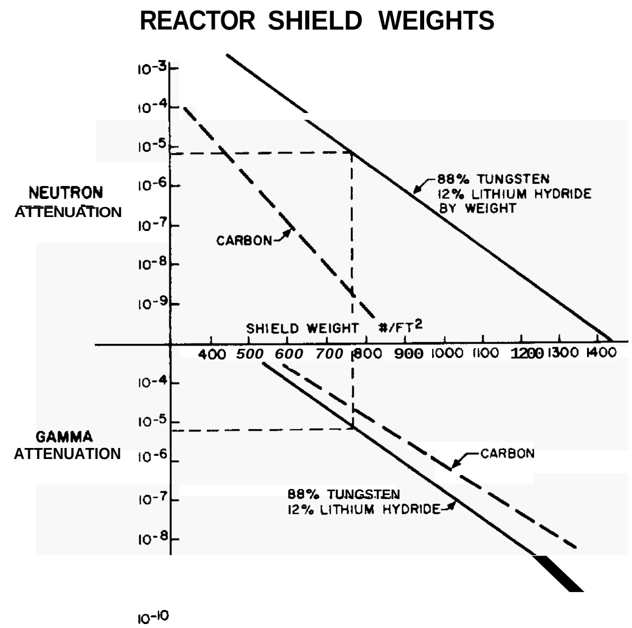


Figure 18. Nuclear Rocket Shield Weights

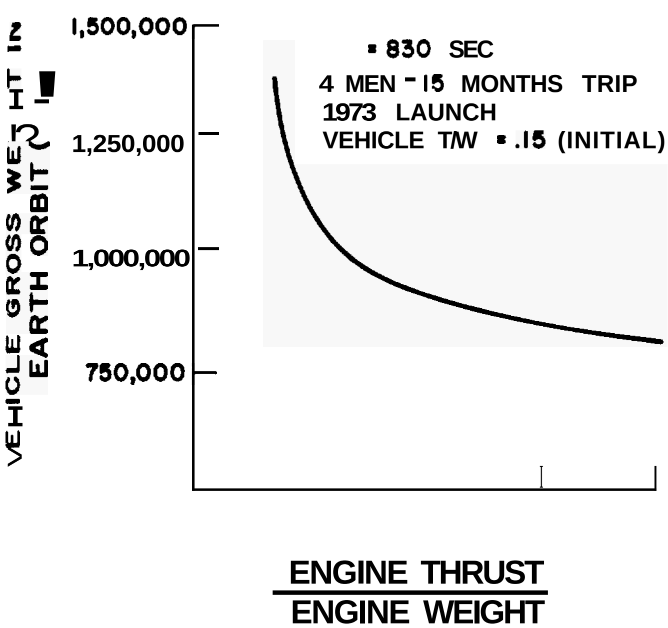


Figure 19. Effect of Engine Weight on Vehicle Weight Requirement

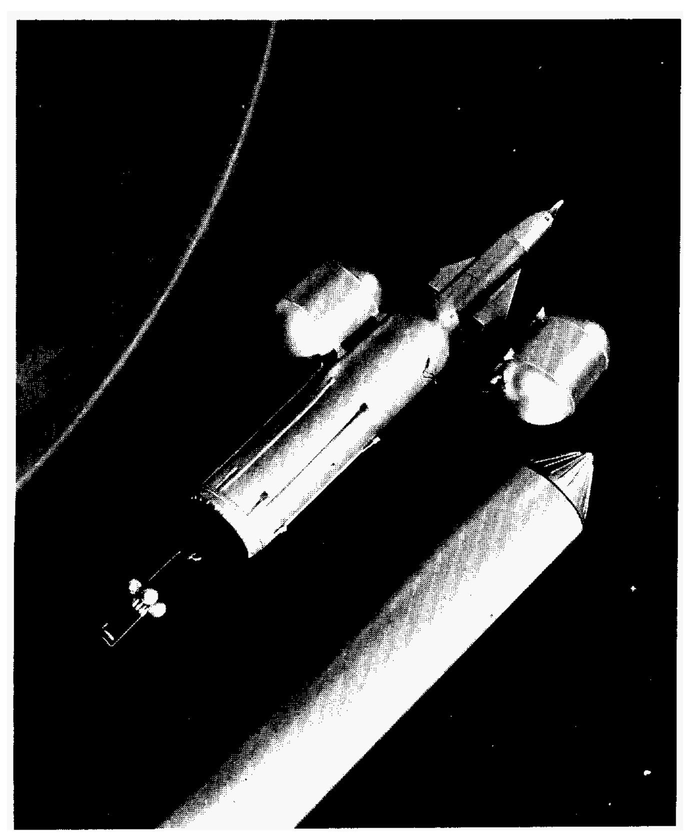


Figure **20.** Assembly of the Nuclear Spacecraft **Tanks** In Earth Orbit

Figure **21.** Launch Configuration of Tanker Vehicle

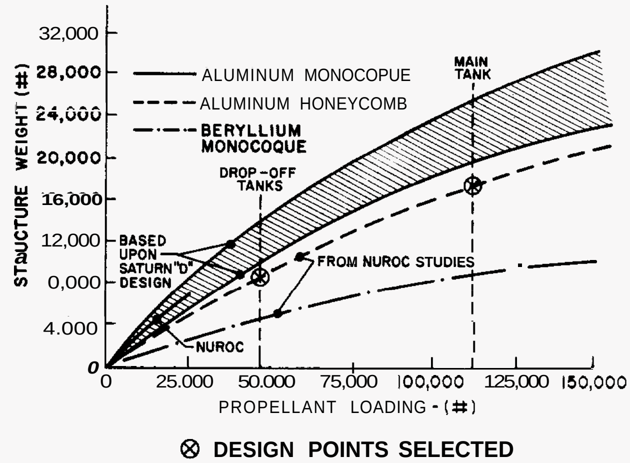


Figure **22.** Structural Weights for Liquid Hydrogen Tanks

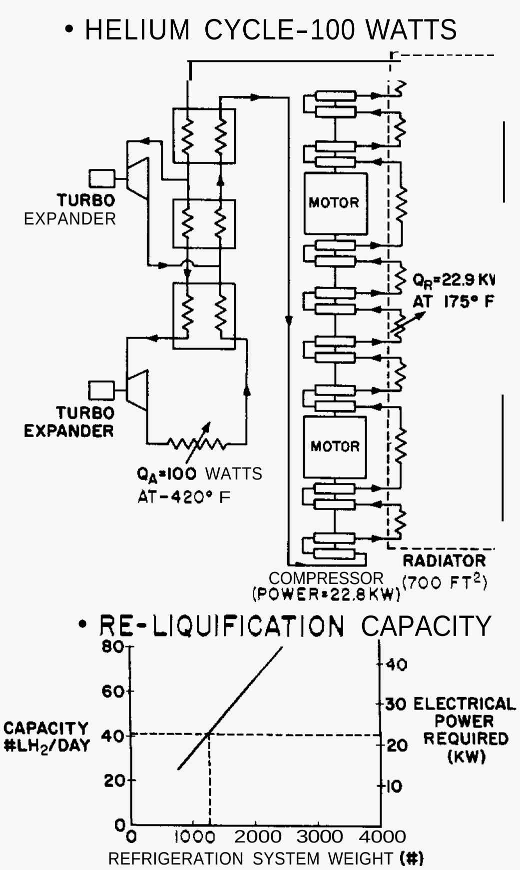


Figure **23.** Cryogenic Refrigeration **System** Characteristics

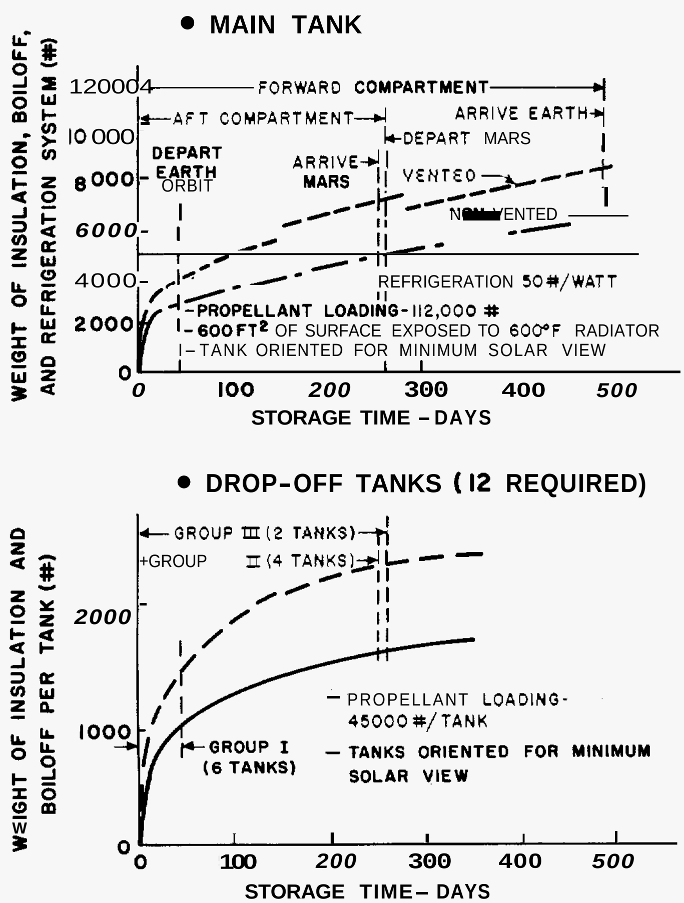


Figure 24. Weight of Insulation and Storage Systems for Liquid Hydrogen Propellant

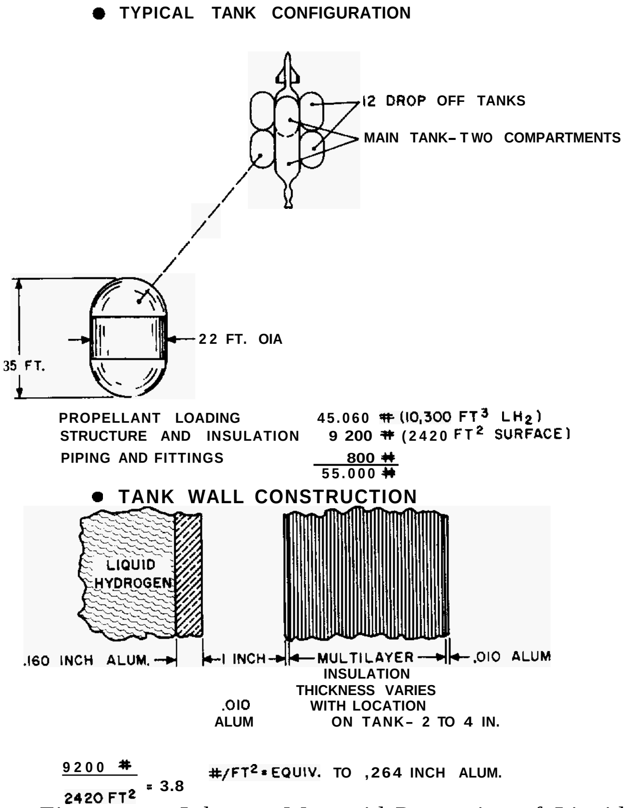
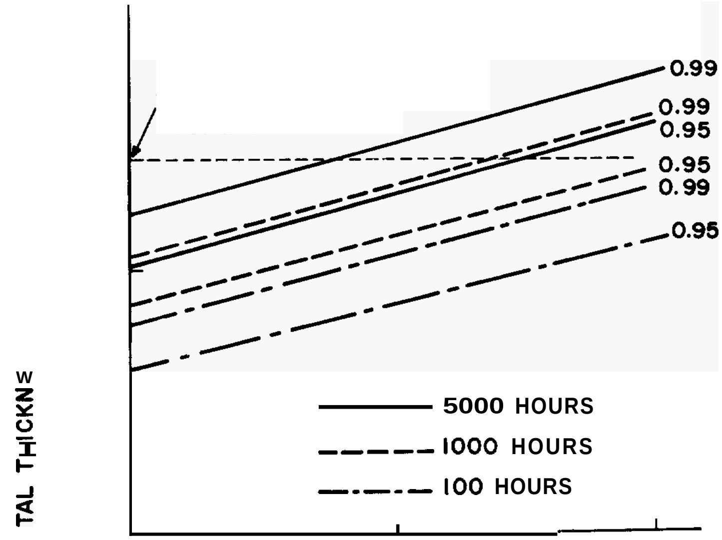


Figure 25. Inherent Meteroid Protection of Liquid Hydrogen Tanks



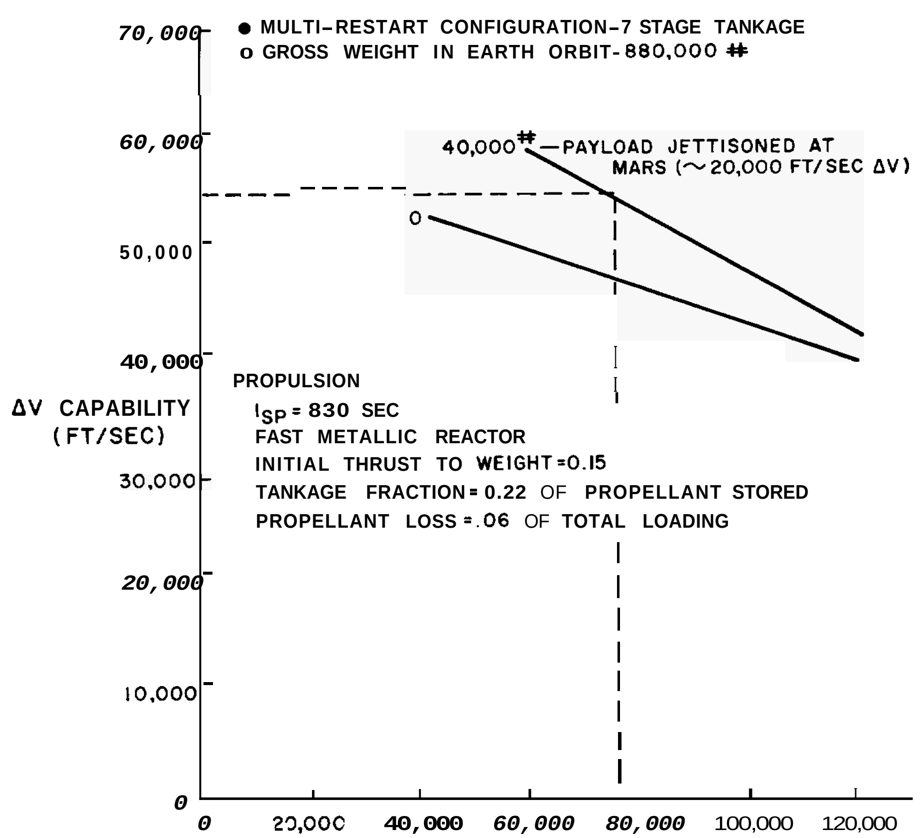


Figure 27. Performance vs. Payload for Multi-Restart Nuclear Rocket Vehicle