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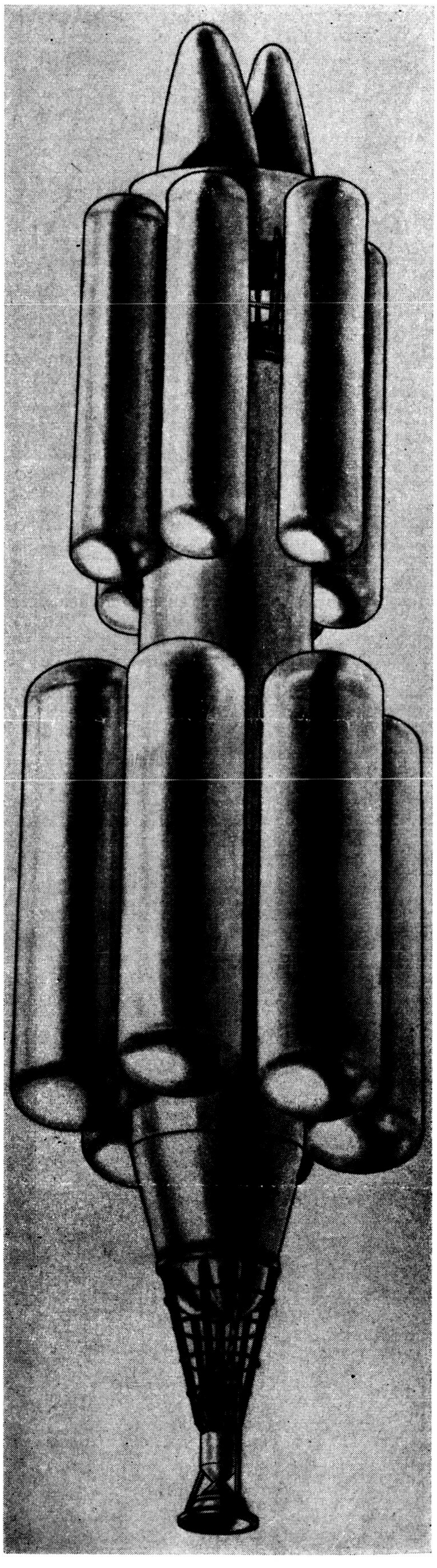

A Study of Manned

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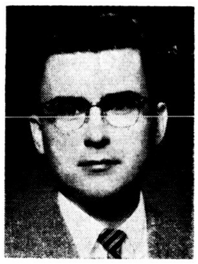

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Mr. Weber received his Bachelor's and Master's degrees in mechanical engineering from Yale University. He joined the Lewis Research Center in 1951, where he has conducted research in the general areas of cycle analysis and mission studies. Since 1957 he has been Head of the Propulsion Section, Mission Analysis Branch. His published papers have included such areas as conventional, nuclear, and supersonic-combustion ram-jets, air turborockets, thermodynamic properties of gases, lunar trajectories, and multistage rocket performance. He is a member of Tau Beta Pi, Sigma Xi, and the ARS, and is an Associate Fellow of the IAS.

Nuclear-Rocket Missions to Mars

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Requisite system weights for round trips to Mars from Earth orbit with a seven-man crew are presented. Within the limitations of current knowledge, the effects of mission duration, radiation environment of space, and the use of atmospheric braking are evaluated.

A MAJOR GOAL of those engaged in astronautics is the exploration of the planets of our solar system. From what is known currently about our solar system, the most promising celestial body for manned exploration, aside from the Moon, is the planet Mars. The problem discussed in this paper is that of determining system requirements for taking a party of men to Mars and returning them to Earth. In the process of examining this problem, it is hoped that the influences of the major factors involved in such an undertaking shall be illuminated—at least as far as current knowledge of the environment of space, human factors, and the characteristics of propulsion systems will permit.

There have been many other studies of the general subject of trips to Mars. For the most part they have been concerned with particular missions, usually those that require a minimum expenditure of propulsive energy. A notable exception is the study made by Ehricke,[1] wherein the requirements for fast, manned, reconnaissance trips to Mars and Venus were studied in rather great detail. A more recent study[2] of a specific fast trip to Mars is that of Konecci *et al.*, in which the problem of shielding against nuclear-engine radiation was treated in detail. This study examines the manned Mars mission in nuclear-rocket vehicles from a more general viewpoint. In the course of the analysis the effects of atmospheric braking, the radiation environment of space, and mission duration will be investigated. In this manner, it is hoped that it will be possible to determine whether a particular trip, or class of trips, offers any significant advantages.

A particular mission profile (Fig. 1), considered to be representative of early manned missions, has been selected as the basis of this study. The mission begins with the vehicle system in an orbit about the Earth. Depending on the weight required for the mission, it can be inferred that the system has been delivered as a unit to the orbit—or that it has been assembled in the orbit from its major constituents. At the proper time, the vehicle containing a crew of seven men is accelerated by a high-thrust nuclear-rocket engine onto the transfer trajectory to Mars. Upon arrival at Mars, the vehicle is decelerated to establish an orbit about that planet. During a specified wait period, a Mars Landing Vehicle containing two men descends to the Martian surface using atmospheric braking to effect the required deceleration. After a period of exploration these men take off from Mars using chemical-rocket power and effect a rendezvous with the orbit party. The return vehicle then accelerates onto the return trajectory; and, upon reaching Earth, an Earth Landing Vehicle separates from the return vehicle and decelerates to return the entire crew to the surface.

From an engineering point of view, we are concerned with the weight and size of vehicles required to perform such missions. Two major factors act to determine these quantities. The first is the velocity increment (ΔV) (*Continued on page 51*)

Missions to Mars (Continued from page 19)

required to perform the necessary maneuvers. In general, the shorter the duration of the mission, the greater is the ΔV requirement. The second factor is the "payload," which comprises not only the men and the scientific equipment they require but also the equipment needed to sustain and protect them in the hostile environment of space. As will be shown later, the weight required to perform these functions tends to increase with trip time. By combining the effects of these two opposing trends, an evaluation of the effect of mission time can be obtained.

The trajectory requirements for these missions are examined first. Because of the magnitudes of the velocity increments required and the limitations of specific impulse currently believed possible with high-thrust propulsion systems, the feasibility of the use of atmospheric braking in the deceleration phases will be examined as a means to decrease the weights required for the mission.

Following this, the radiation hazard in space will be discussed. The nature of the different types of radiation environment that are encountered during a voyage and the shielding required to protect the crew will be considered. Finally, the effects of these factors will be combined to yield complete vehicle systems for missions of varying duration.

Trajectory Requirements

Many possible trajectories for accomplishing a round trip to Mars have been studied in some detail by using a three-dimensional model of the solar system and "best fit" ellipses for the orbits of Earth and Mars. Because this study has been limited to high-thrust propulsion systems, impulsive velocity increments have been used throughout with a patched-conic representation of the trajectories. The approximate information thus derived has been checked against precision n-body calculations and has been found to agree to within about 2 percent in the ΔV requirements, which is sufficient accuracy for present purposes.

The minimum total velocity increments have been determined for a range of mission times and are presented in Fig. 2. It is noteworthy that the total velocity increment does not decrease continuously as mission time increases. This would seem to rule out missions of certain durations, at least from a propulsive energy standpoint. For example, missions lasting between 500 and 700 days require velocity increments equal to or greater than that required for a 500-day mission.

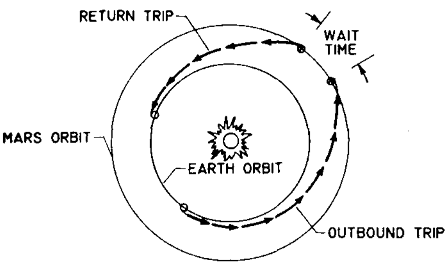

Fig. 1. Mission profile.

The effect of wait time is different for the two regions in this figure. As previously shown in the two-dimensional data,[3] for the left-hand branch, an increase in wait time is accompanied by an increase in total velocity increment. For the right-hand branch, it is possible to decrease the ΔV required by increasing the wait time. For the purposes of this study it was decided to use wait times of 40 days for mission durations up to 800 days. For longer trip times, wait times of up to 450 days were used in order to take advantage of the lower ΔV for such missions.

Regardless of what mission time is selected, the magnitude of the total ΔV required for the round trip from orbit to orbit is quite high. Considering the essentially exponential growth of vehicle weight with propulsive ΔV, it would be most desirable to reduce the propulsive ΔV required. The presence of an atmosphere at Earth and Mars offers the opportunity to reduce the propulsion requirements through the use of atmospheric braking for the deceleration phases of the mission. How much of the total velocity increment can be saved in this manner is indicated in Fig. 3, wherein the previous data are reproduced along with curves indicating the requirements if atmospheric braking is used upon return to Earth alone, as well as for such deceleration at both Mars and at Earth.

It is obvious that using atmospheric braking can reduce substantially the propulsion requirements for Mars round trips. In addition to the general decrease in the level of the propulsive velocity increment required, another important effect of the use of atmospheric braking is the change in the relative ΔV's for the long and the short trips. The greater the use of atmospheric braking, the less we are penalized for making the short trips.

Atmospheric Braking

The potential advantage of the use of atmospheric braking having been established, one must inquire as to the feasibility of such a process for the missions

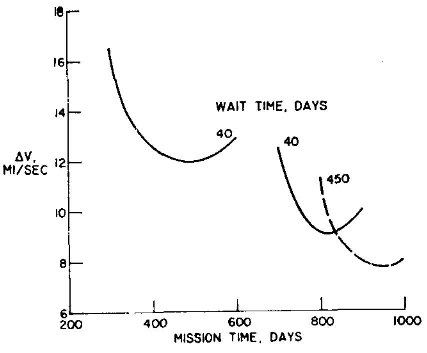

Fig. 2. Effect of mission time; optimum departure dates, all-propulsive case.

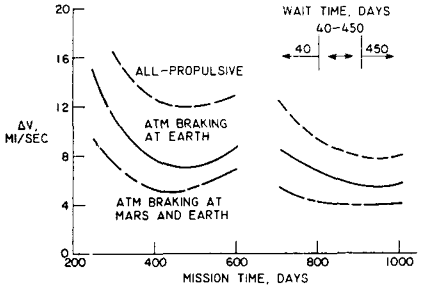

Fig. 3. Benefits of atmospheric braking; optimum departure dates.

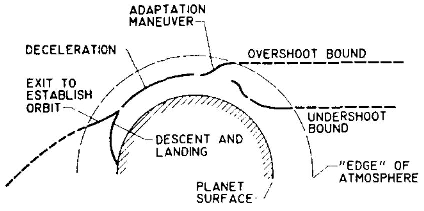

Fig. 4. Atmospheric braking and associated maneuvers.

under consideration. The vehicle entry speeds for interplanetary trips are, of course, hyperbolic. For most of the trip durations considered, however, these speeds do not exceed 10 miles/second. For comparison, the entry speed from a lunar mission is about 7 miles/second.

The first thing to be considered in the application of atmospheric braking is the series of maneuvers required to effect the desired deceleration. The various phases of such an operation (Fig. 4) are indicated and discussed in the order of their occurrence.

Maneuvers

During the extra-atmospheric approach to the planet, the guidance system must guide the vehicle so that it encounters the atmosphere within a certain region which has been termed the aerodynamic "entry corridor."[4] The boundaries of the aerodynamic entry corridor are termed the overshoot and undershoot bounds. Approaches at altitudes greater than that of the overshoot bound will result in the ve-

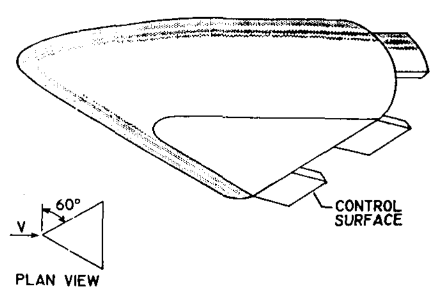

Fig. 5. Atmospheric-entry vehicle.

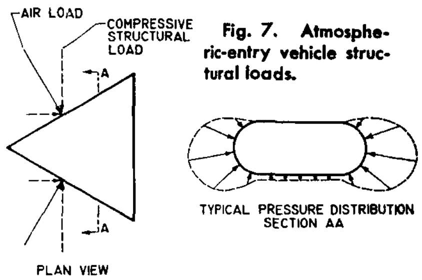

Fig. 7. Atmospheric-entry vehicle structural loads.

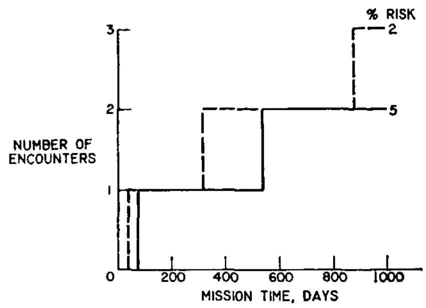

Fig. 9. Predicted number of giant major flares; average frequency, once every 4 years.

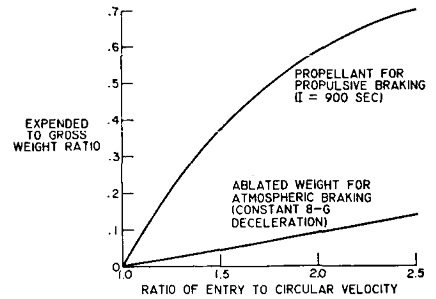

Fig. 6. Weight expended for decelerating at Earth.

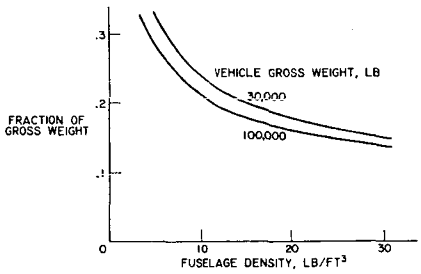

Fig. 8. Structural plus secondary heat protection weight (G_{max} = 10).

hicle not being able to acquire the desired deceleration flight path. Approaches on paths lower than that of the undershoot bound result in flight paths that exceed the vehicle load limit, heating limit, or both. In general, deep entry corridors are desirable in that they act to reduce the burden on the approach guidance system.

Following the entry operation, the vehicle is maneuvered to place it on the desired deceleration flight path. A desirable deceleration flight path is one that results in a minimum total heat input to the vehicle. Several deceleration paths have been considered[5] and, for the paths considered, the one that yields the lowest heat load is a constant-*g* path with the total acceleration held at the vehicle limit value. Such paths have about one half the heat load as constant-angle-of-attack paths of the same maximum *g*-load.

After the specified decrease in vehicle speed is achieved, the vehicle either descends and lands or is maneuvered so that it leaves the atmosphere and ascends to establish an orbit. For the latter purpose, a rocket impulse must be provided at the apogee of the exit orbit in order to raise the perigee altitude above the atmosphere.

Entry Vehicle Performance and Design

Selection of the best vehicle design for an atmospheric braking operation is a process of compromise among the conflicting requirements of the various phases of the operation. Each phase has been studied and its characteristics determined. From the results of these investigations, suitable compromises have been made.

The aerodynamic entry corridors were studied as functions of approach velocity, vehicle maximum L/D, deceleration level, and mode of operation during entry.[6] Guidance corridors were evaluated using the technique described by D. Harry and Friedlander.[7] Subject to the constraints imposed by heating during the deceleration, it was found that guidance and aerodynamic corridors are compatible for entries at speeds up to 10 miles/second, for vehicles with a maximum L/D of unity. This range of entry speeds encompasses those corresponding to almost all the mission times considered in the study.

A vehicle configuration which can perform such an entry is shown in Fig. 5. It has the following characteristics: delta planform with 60° of leading-edge sweep, a hemicylindrical leading edge having a radius of one half the vehicle thickness, and a thickness-to-maximum-chord ratio of about 0.4. Such a vehicle has a maximum lift-drag ratio of 1.0. For a 30,000-lb gross weight, the vehicle would have a span of about 22 ft, a root chord of about 19 ft, and would be about 8 ft thick.

The heat protection for such vehicles was studied[5] for different entry speeds and deceleration flight paths. There are two modes of heat input to the vehicle: convective heating and that resulting from hot-gas radiation. Hot-gas radiation can generally be avoided, even at the flight speeds considered, if the vehicle surface angles are limited to 20 or 30 deg. with respect to the free-stream direction. With such low surface angles, the gas static temperature does not become high enough to give rise to appreciable radiative heat transfer. Such angle limits have been observed in the present calculations.

The heat protection system used comprises two parts—the primary and secondary systems. The primary system protects against the major portion of the potential total heat load. The secondary system serves to maintain the desired temperature for the payload and structure of the vehicle. A number of primary systems can be considered. Of these, ablation appears most attractive in terms of weight required for the hyperbolic speeds characteristic of entry from interplanetary missions. For this primary system, the secondary system selected consists of some insulation and a water boiling system.

A parameter of primary interest for the braking operation is the ratio of the weight of material ablated during this process to the gross weight of the vehicle. This ratio for deceleration at Earth to circular speed is shown as a function of the entry speed in Fig. 6 for a vehicle typical of those investigated during the study. This vehicle has an $(L/D)_{max}$ of 1, operates at a maximum of 8 *g*'s, and has an initial weight of 30,000 lb. For a constant fuselage density, the weight ratios indicated by the curve were found to be insensitive to variations in gross weight.

For entry speeds below twice circular at Earth, the weight of the primary heat protection material ablated to decelerate to circular velocity is less than 10 percent of the gross weight. If the same deceleration were to be performed by propulsive means, much larger weights would be expended. To illustrate this, the second curve of Fig. 6 shows the propellant fraction required to perform the same operation with a nuclear rocket with a specific impulse of 900 sec. For the case of entry at twice circular speed, the rocket braking requires about six times as much weight expended as does the atmospheric braking. For this comparison, it has been assumed, for both cases, that deceleration from circular to suborbital speed, as is required for landing, is accomplished by atmospheric braking, the generally accepted method. For a direct descent to the surface from twice circular speed, the total ablated weight is 13 percent

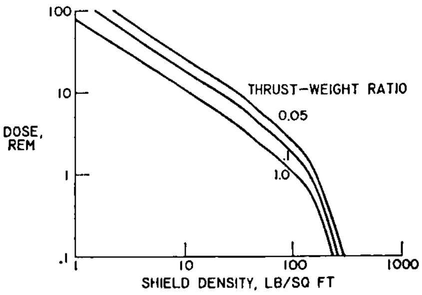

Fig. 10. Shielding density for Van Allen belt radiation; parabolic escape from 150-nm circ. orbit.

of the initial weight as compared with the 9 percent required to decelerate only to circular speed.

Structure—In order to discuss the weight of the structure of the atmospheric braking vehicle, the types and nature of the loads it encounters must be considered. A sketch (Fig. 7) indicates the loads for the vehicle described previously. During entry, the vehicle operates at low angles of attack. This type of operation results in the principal air loads being on the sides of the vehicle, which places the top and bottom surfaces in compression. These air loads are approximately proportional to the g-loads, or the deceleration rate. In contrast to the case of more conventional aircraft where the structure is designed by the normal acceleration loads, in the present case the structure is designed by the axial acceleration loads.

One of the advantages of the heat protection system chosen is that it yields a "cool" structure, which permits the use of conventional aircraft materials such as aluminum. The estimated values of the weight of vehicle structure and secondary cooling system are shown as a function of vehicle density (Fig. 8). For both Mars and Earth, the necessary corridor depths call for an entry maneuver at about 10 *g*. As this determines the design stress level, the curves shown are applicable for entry at either planet.

For the 30,000-lb vehicle previously described (typical of those considered for the return to Earth), the vehicle density is about 15 lb/cu ft; the corresponding structural weight fraction is about 20 percent. The structural weight fraction shows a significant rise with decreasing fuselage density. This is a very important characteristic when the use of atmospheric braking is considered for establishing an orbit about Mars where the entry vehicle must contain the propellant for the return propulsion from Mars. For nuclear-rocket engines this propellant is hydrogen, which is very tenuous and results in vehicle densities of the order of 5 lb/cu ft. The structural weight for such vehicles is quite high. This, in addition to the fact that the structural weight is not used again for a descent operation, can reduce considerably the possible advantages of atmospheric braking for this phase of the mission. For entry vehicles which do not contain hydrogen, the density is higher and the structural weights are much lower.

Summarizing the foregoing sections, it has been established that the aerodynamic entry corridors and the guidance corridors are compatible for entries at speeds up to about twice circular speed at Earth. For the heat-transfer characteristics assumed, the weight expended during a braking maneuver using an ablation heat protection system is much less than that required for propulsive braking. All told, the use of atmospheric braking seems quite promising, except possibly when the vehicle must contain large quantities of hydrogen for subsequent propulsion periods.

Radiation Hazards

Thus far, the factors which have been considered do not depend on the type of payload, animate or inanimate, except possibly through the *g*-load selected. When the missions are manned, the equipment and supplies necessary to sustain and protect the crew must be considered. These factors determine size and weight of "payload" and interact with propulsion requirements to determine the absolute size of the vehicle system.

At the outset of this study, it soon became apparent that the crew's exposure to radiation from various sources and the consequent biological shielding requirements constitute one of the major factors affecting the results. Although current knowledge of radiation hazards is, to say the least, not completely satisfactory, shielding estimates based on available information have been incorporated into this study in order to form some idea of the magnitudes involved.

Radiation Sources

Van Allen—A belt of trapped radiation surrounds the Earth except in the polar regions. Two zones of intense radiation exist within the belt. The inner zone contains many electrons, but more importantly, a large number of protons,

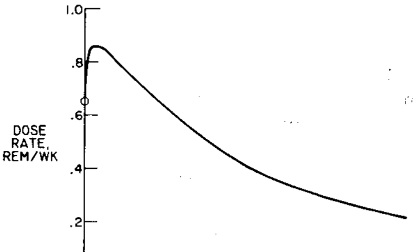

Fig. 11. Shielding density for cosmic radiation; carbon shield.

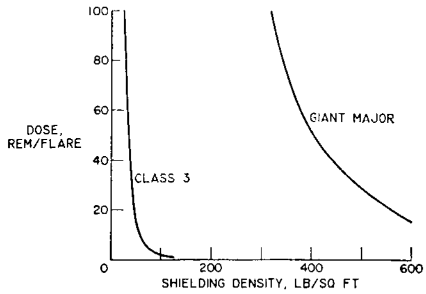

Fig. 12. Shielding density for solar flares; carbon shield.

those of energies over 30 mev being confined to altitudes between about 400 and 5,000 nautical miles. The outer zone extends over a much wider range of altitudes but is mostly composed of electrons, which are easily stopped by a thin sheet of metal. In the calculations, it was assumed that the radiation flux is constant for all altitudes within the inner zone and is equal to the value at the middle of the zone.

Cosmic Ray—Cosmic radiation consists of very energetic atomic nuclei, over 90 percent of which are protons. However, the heavier particles comprise more than 30 percent of the total by weight and also have far more deleterious effects on man. Cosmic-ray fluxes exhibit a significant variation with time which is related to solar activity. In this paper, however, the cosmic radiation rate was taken as a constant, equivalent to 0.65 rem/week.

Solar Flares—At irregular intervals the Sun emits bursts of radiation which are classified according to the area of the visible disturbance on the Sun's surface. Class 1 and 2 flares occur almost continuously, but their accompanying radiation is believed to be sufficiently low in energy that it is stopped by even thin walls and is, therefore, ignored herein. Class 3 flares, which occur on the average of about once a month, emit mostly protons (of energies up to 500 mev) with possibly 10-percent alpha particles.

At rare intervals there occur giant major flares. These are large flares of the class 3 category which may emit up to 10,000 times the usual intensity radiation with particle energies as high as 20 bev. In the 20 years during which facilities for observing these flares have been available, six have been recorded, occurring on the average about once every 4 years. Not much confidence can be placed in the accuracy of an average based on only six points; that is, the next six flares may very well have an appreciably different average period.

Even if the *average* interval between giant flares were accurately known, we must recognize that the flares do not occur exactly on schedule. If their occurrence is treated as random, the probability of encountering various numbers of flares during a trip of specified duration can be readily calculated. The results of such a calculation are shown (Fig. 9) for two particular levels of risk (i.e., there is a certain probability that more than the predicted number will actually be encountered). What constitutes an acceptable risk level depends on the seriousness of permitting the crew to suffer more than the design dose for the trip. This point is discussed in a later section.

Engine—Nuclear engines will subject the crew to some radiation in addition to the external sources already mentioned. This radiation is composed of gamma rays and neutrons, which are of fairly low energy in comparison with the other particles that have been described.

Shielding Density

Curves have been drawn to show the variation in dose rate with shielding

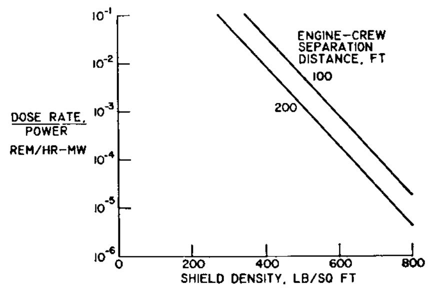

Fig. 13. Shielding density for nuclear-engine radiation; propellant tank length, 100 ft.

density (in lb/sq ft of surface area) for each of the radiation sources described. The uncertainties in this area are threefold: (1) the quantitative characteristics of the radiation in space are poorly known (i.e., number of particles, energy spectrum, isotropy, etc.), (2) the interactions of high-energy particles with various shield materials are in doubt, and (3) the effects of the particles of different energy on human tissue (i.e., the relative biological effectiveness) are largely unknown. The data selected for use in this paper have been taken principally from the results of Wallner and Kaufman.[8,9] The Noyes and Brown data[10] were also used for the Van Allen radiation calculations.

The dose the crew incurs during each traversal of the Van Allen belt (in the more intense equatorial region) is shown in Fig. 10. Low initial thrust-weight ratios are undesirable in this regard, since they prolong belt exposure time. The data presented are for parabolic escape missions; the higher energy trajectories associated with Mars missions tend to reduce the dose, but the effect is very slight.

Fig. 11 shows the dose rate due to cosmic radiation. As a result of the interaction of high-energy particles with the shield to create secondary radiation, it is seen that a thin shield is worse than none at all.

Fig. 12 shows the dose imparted by a single flare of either the ordinary class 3 or the giant major variety. These data are conservative in that they are based on the more violent flares of each type; the average flare is considerably less potent. On the other hand, they are optimistic in that the production of secondary radiation has been ignored.

Fig. 13 shows the very approximate chart used to calculate the dose from the engines. This included the benefit of interposing the propellant tank between the engine and the crew. More accurate estimates were unnecessary, as the engine radiation contributes a fairly small portion of the total dose received by the crew for the configurations used.

Shield Weight

The aforementioned data were combined to find the shield weight required to give the crew various total doses during their journey. Detailed layouts of the living quarters led to a design having a shielded volume of 4,200 cu ft and a surface area of 1,600 sq ft. Within this volume is a small, heavily shielded vault, inside which the entire crew is assumed to remain during all class 3 and giant major flares. This presumes the presence of a suitable monitoring system to warn the men of the start of a flare. The duration of high-intensity radiation from a flare is from a few minutes to 8 hr (although the duration of increased radiation activity may be considerably longer). A cylindrical vault of 500-sq ft surface area was found to provide sufficient volume (about 615 cu ft) to contain the 7 men fairly comfortably for these short periods. Shadow shielding was not employed since, as far as is now known, little of the external radiation is directional in nature.

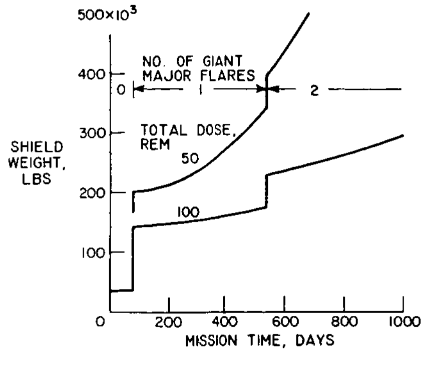

Fig. 14. Shield weight; vault surface area, 500 sq ft.

Fig. 14 shows the calculated shield weight required to limit the total crew dose to either 50 or 100 rem during the trip. The giant major flares were predicted at the 5-percent risk level. The crew was assumed to be in the vault whenever the engines were operating or a Van Allen belt was being penetrated. During these periods the resultant crew dose is generally quite small. When returning to Earth, the crew will be in an aerodynamic reentry vehicle while passing through the belt. The protection afforded by structure and ablation material will result in a dose of about 5 rem from this passage. Throughout the trip it is assumed that the crew members sleep in the vault, thus offering further protection against cosmic radiation during one third of the time.

Except for flight times so short that no giant major flares are to be expected, the presented shield weights are enormous, even for a 100-rem dose. The greatest part of this weight is required to protect against the giant flares. As a result, in most cases it was found best to place all the shielding around the vault, with none around the living quarters.

If it should become possible to predict long periods in which no flares are expected, major reductions can be made (e.g., a 420-day trip would require only a 47,000-lb shield for a 100-rem dose). Since, however, such periods would have to coincide with favorable launch dates from an energy standpoint, the number of suitable launch periods might be unduly restricted.

The approximately 30,000 lb of supplies in the vehicle (for a 420-day mission) provide some shielding effect. They are probably most useful if employed as shadow shielding against the engine radiation, although little attention was given to this point.

Tolerable Doses

To all the uncertainties that entered into the shielding calculations must be added that of exactly what doses the crew should be permitted to take. Currently recommended industrial dose rates are very small, in the order of 5 rem per year, with a host of qualifications and emergency dose values that depend on the rates and duration of exposure. However, doses of 100 rem within a few hours will not cause radiation sickness in most individuals; and the other detrimental results such as shortened life, increased propensity to leukemia, etc., are so slight as to be measurable only statistically in a large population.

As pointed out by several authorities,[11] it does not appear reasonable to restrict the participants in such a hazardous endeavor as space flight to such conservative doses as the recommended industrial value. For the dual reasons of simplicity and insufficient knowledge of anything better, the approach of the present paper is to limit the crew dose to a constant value of either 50 or 100 rem for all trip times.

A reasonable alternative to this approach would be to vary the dose with time (e.g., 50 rem for 1 year, 100 rem for 2 years, etc.). A somewhat different approach would be to assign the allowable dose (either constant or time-dependent) on the basis only of the more-or-less steady background radiation (i.e., cosmic rays and class 3 flares). The intermittent short-time doses (Van Allen belts and giant major flares) would be allowed over and above the regular dosage, with the proviso that the dose from each single incident not exceed a limiting value. Because of the virulence of giant major flares, this approach is not especially helpful for the shorter trips. For example, if the short-term dose were limited to 100 rem, the vault weight can be no lower than 145,000 lb. However, if in addition a background dose of 100 rem is permitted, no increase need be made in shield weight until trips last well over 1,000 days.

In any event, no recommendation of an allowable dose or dose rate is to be inferred from this paper. The values used are to be considered as illustrative and are intended only to serve to point out the severity of the problem and the effects on the mission requirements.

Results and Discussion

Some of the more important factors considered in the present study have been described in the foregoing. Among the other factors that have been incorporated in the analysis are: ecological system requirements, propulsion system characteristics, and propulsion system structures. The assumptions made concerning these items are described in the Appendix of this article.

In subsequent sections, the individual effects of the major factors on the requirements for the missions will be examined first. Then the combined effects will be examined as functions of the mission time.

Vehicle Configuration

A vehicle system (Fig. 15), typical of those derived for the round trips to Mars, is shown. The vehicle consists of a long central tank surrounded by two clusters of smaller tanks. These are surmounted by the cabin, or living quarters, which contains the shielded vault. Above this is a pair of atmospheric entry vehicles. A single nuclear-rocket engine, with a specific impulse of 900 sec, is attached to the bottom of the central tank and is used for all the main propulsion periods. This implies a restartable engine, which seems quite feasible since the period between firings is no less than 40 days.

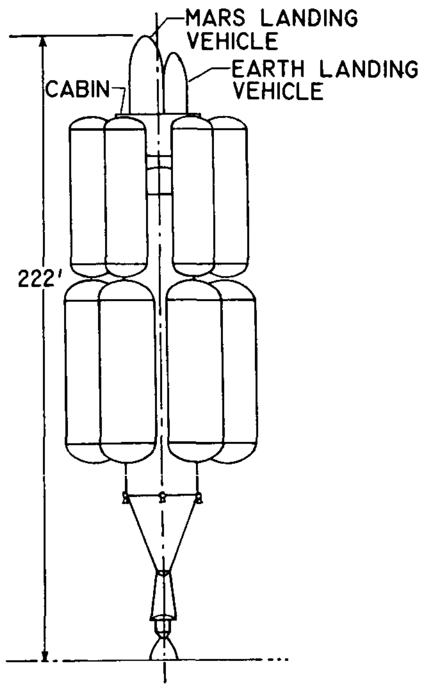

Fig. 15. Typical Mars-mission vehicle.

The vehicle operates in the following manner: Starting from an orbit about Earth at an altitude of about 300 miles, the first propulsion period accelerates the vehicle until it has acquired the transfer energy required. The weight of propellant required for this operation is contained in the lower cluster of tanks. The actual flow of propellant into the engine is from the central tank, which is replenished from the lower cluster during the firing by a suitable propellant transfer system. This flow scheme has been adopted in order to reduce the weight required for the engine-radiation thermal shield. Upon completion of this propulsion period, the lower cluster of tanks is jettisoned. The remainder of the system then coasts to Mars and is decelerated, by the second engine firing, to establish a low circular orbit about Mars. The contents of the upper cluster of tanks are used to replenish the central tank for this firing. Prior to the final propulsion period for the return to Earth, these tanks are discarded, leaving just the central tank and payload.

After the orbit is established at Mars, two of the crew enter the Mars Landing Vehicle and descend to the surface using atmospheric braking for deceleration. This vehicle (Fig. 16) is shown as it appears after a landing. Its initial weight is 60,000 lb and it contains the propellants and propulsion systems required for the terminal landing maneuvers, the ascent to orbit, and the rendezvous with the orbit party. The vehicle separates into two parts with the bottom of the vehicle serving as the launching pad for the ascent. A small tracked vehicle is included as a part of the system to permit the two men to explore more than a limited area of the surface.

After the period of exploration, the men enter the ascent vehicle, taking with them a few hundred pounds of records and specimens (all other data having been transmitted to and recorded by the orbit party). The ascent vehicle is abandoned after rendezvous.

With the crew now reässembled, the engine is fired for the final time, emptying the central tank, which is then jettisoned. The living quarters and the Earth Landing Vehicle then coast back to Earth. Shortly before arrival, the crew transfer to the entry vehicle (initial weight of about 30,000 lb), which enters the atmosphere, decelerates, and lands, completing the mission.

Effects of Atmospheric Braking

The vehicle configuration and mode of operation just described have been used as the basis for most of the comparisons that follow. Selection of this mode of operation can be explained with the aid of Fig. 17 which shows the relative initial weights in orbit about the Earth for three different degrees of utilization of atmospheric braking. This comparison is made for a 420-day round trip. The weight for the case of propulsive braking for all decelerations has been used as the normalizing factor. For this mode of operation it has been assumed that the propulsive deceleration at Earth is used just to establish an orbit and that the rest of the descent is accomplished by an entry vehicle.

The use of full atmospheric braking at Earth reduces the initial weight required to 63.5 percent of that required for the all-propulsive case, a very attractive reduction. For atmospheric braking at both Earth and Mars the initial weight required is 57 percent of the all-propulsive case. This is a reduction of about 10 percent from the weight required for a system employing atmospheric braking at Earth alone, a relatively modest improvement.

The principal reason for the lack of a more striking improvement in the last case is the low fuselage density which results for a Mars entry vehicle that must contain the propellant for the return trip. As noted earlier, this causes a large structural weight. The structural design of atmospheric entry vehicles containing hydrogen involves many uncertainties. Because the present approach to the problem has resulted in only a small improvement in initial weight, the use of atmospheric braking at Earth alone has been adopted as the

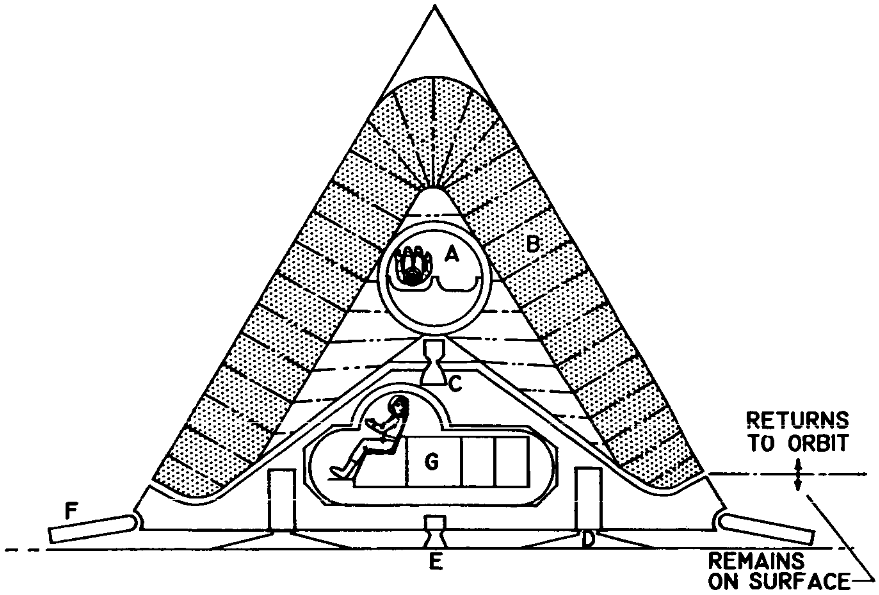

Fig. 16. Mars landing vehicle.
***A*, control capsule; *B*, propellants (storables); *C*, boost rocket motor; *D*, landing gear ("shocks" compressed); *E*, landing retrorocket motor; *F*, aerodynamic control surfaces (in landing position); *G*, surface exploration vehicle.**

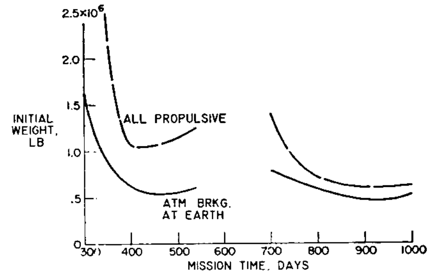

Fig. 19. Benefit of atmospheric braking at Earth; no shielding.

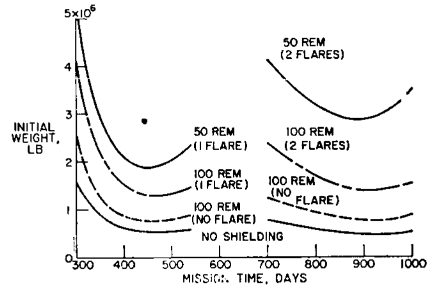

Fig. 20. Effect of radiation assumptions.

standard procedure for the rest of the study.

Effect of Shield Weight

The shielding required for the protection of the crew against the various forms of radiation has a great effect on the vehicle gross weight. The magnitude of this effect is illustrated in Fig. 18, where the ratio of the initial weight required for various amounts of shielding to that for no shielding (other than that afforded by structure and equipment) is presented for a 420-day trip.

The various shield weights can be interpreted as different allowable doses and/or different degrees of optimism concerning the ability to avoid giant major flares. If it is possible to avoid giant major flares and a total dose of 100 rem is permissible, the required shield weight is 47,000 lb. The initial weight in orbit for this case is 40 percent greater than that for no shielding. If the total dose permitted is kept at 100 rem but the one giant major flare anticipated for trips of this duration cannot be avoided, the shield weight jumps to 164,000 lb. The corresponding initial weight is 2.4 times the unshielded value. Finally, if the allowable dose is reduced to the more "conservative" value of 50 rem (again with provision for one major flare), the shield weight becomes 280,000 lb which corresponds to an intial weight ratio of 3.4.

These data serve to underscore previous remarks as to the severity of the effects of shielding and the importance of determining more precisely the nature and virulence of the radiation in space.

Effects of Mission Time

The individual effects of some of the more important factors affecting the system requirements were examined for a specific mission duration in the foregoing sections. The combined effects of these factors are discussed as a function of the mission time in the following sections.

Fig. 19 shows the effects of mission time and mode of operation for vehicles with unshielded crew compartments. Weights indicated may, therefore, be considered to represent the minimum requirements for these missions within the context of the assumptions of this analysis. The uppermost pair of curves represents the case of all-propulsive maneuvers. For this mode of operation it is seen that the fast trips of the left branch require considerably more initial weight than the slower trips of the right branch. Comparing the minimums of the two branches, a 425-day trip has an initial weight of 1.05 million lb, whereas a 900-day trip requires only 0.6 million lb. The fast-trip vehicle is, therefore, 75 percent heavier than the slower trip.

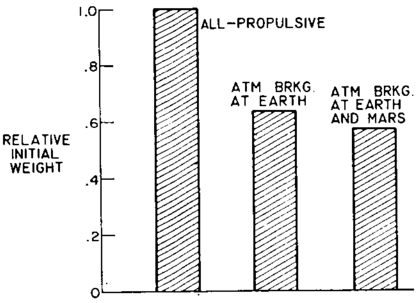

Fig. 17. Effect of atmospheric braking on initial weight; 420-day trip, 100-rem dose.

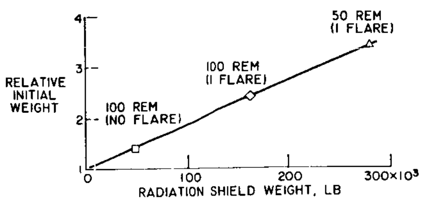

Fig. 18. Effect of shield weight; 420-day trip.

When atmospheric braking is used at Earth, a marked change in the results occurs, as indicated by the lower pair of curves. Not only does the general level of weights decrease as was noted earlier, but the relative position of the two classes of missions changes radically. For this mode of operation there is very little difference in the weights for the long and the short trips. The best slow trip, a 925-day mission, requires 475,000 lb, while the corresponding fast trip, about 460 days, requires 525,000 lb. This is an increase of only about 10 percent for reducing the mission time by a factor of 2.

When the effects of atmospheric braking at Earth and shielding for different dose levels are combined with those of mission time, the Fig. 20 curves result. The lowest set of curves is a repetition of those for no shielding from the preceding figure and serves as a datum.

The second lowest set assumes a permissible dose of 100 rem and the ability to avoid giant major flares. In addition to the increased weights for both branches, the effect of this degree of shielding is to shift the minimum-weight mission from the long-trip branch to that of the fast trips.

The third set, when compared with the set just described, shows the effects of providing shielding for giant major flares at the same dose level. For the left branch, protection is provided for one flare; for the right branch, two flares are to be expected.

The uppermost set of curves is for the most conservative dose assumptions considered in this study (although this may be by no means "conservative"), 50 rem, including the effects of giant major flares. For this case the advantage of the fast trips has increased markedly; the difference in the best weights of both classes is now about 1 million lb.

Some general observations can be made on the results presented in this figure: Regardless of trip time, the weight required in orbit for even moderate amounts of shielding is very great, about 1.3 million lb for the 100-rem dose. For the 50-rem dose, the minimum weight is about 1.9 million lb. The effect of shielding as the allowable dose is decreased is more pronounced for the slow trips than for the fast trips as is evident by comparing the upper two sets of curves. Finally, if we attempt to go to very short trip times, the initial weight increases very rapidly. Typically, going from a trip time of 420 days to one of 350 days increases the initial weight by 55 percent.

Concluding Remarks

An effort has been made in this paper to evaluate the effects of a number of important parameters on the requisite initial weight in orbit for nuclear-rocket manned Martian trips. Realistic trajectory velocity increments and timing data were incorporated in the study.

Atmospheric braking from hyperbolic speeds seems feasible. The use of this technique for the deceleration at Earth reduces substantially the propulsion and initial weight requirements. Although some further weight reduction seems possible, atmospheric braking was not used to establish an orbit at Mars because of the uncertainties involved in the design of entry vehicles containing large quantities of hydrogen.

The shielding necessary to protect the crew from the radiation environment of space has a major effect on the results. Unfortunately, completely satisfactory information concerning these radiation hazards is not available. Based on the available information, very large shield weights result even for what may be considered to be large radiation doses for the crew.

When all the elements of the analysis are combined, it is found that short trips are as, or more, economical, in terms of initial weight, than long-duration missions. Based on the assumptions of the present analysis, a vehicle for a 420-day mission (with 40 days at Mars) would have an initial weight of about 1.35 million lb for a design crew radiation dose of 100 rem. The optimum initial thrust-weight ratio for this system is about 0.2, which corresponds to a reactor power of 6,000 mw. These values are for an Earth Landing Vehicle weighing 30,000 lb. The initial system weight is not very sensitive to variations in the weight of the Earth Landing Vehicle. For example, if the weight of the landing vehicle were reduced to 15,000 lb, the initial weight would be reduced by only 5 percent.

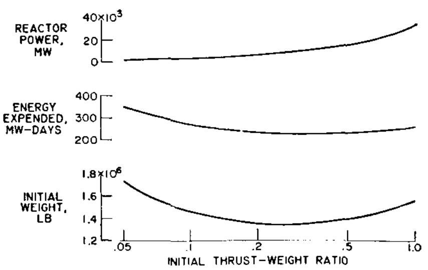

Fig. 22. Effect of thrust-weight ratio; 420-day trip, 100-rem dose.

In summary, the weight penalties caused by the need for radiation shielding are mitigated by the combination of efficient engines, atmospheric braking, and fast trips. While the resulting systems are of great size and a manned Mars mission will be a tremendous undertaking, it cannot be called technically unfeasible. Furthermore, advances in many areas (beyond the state assumed in this study) may be expected to reduce the magnitude of the task.

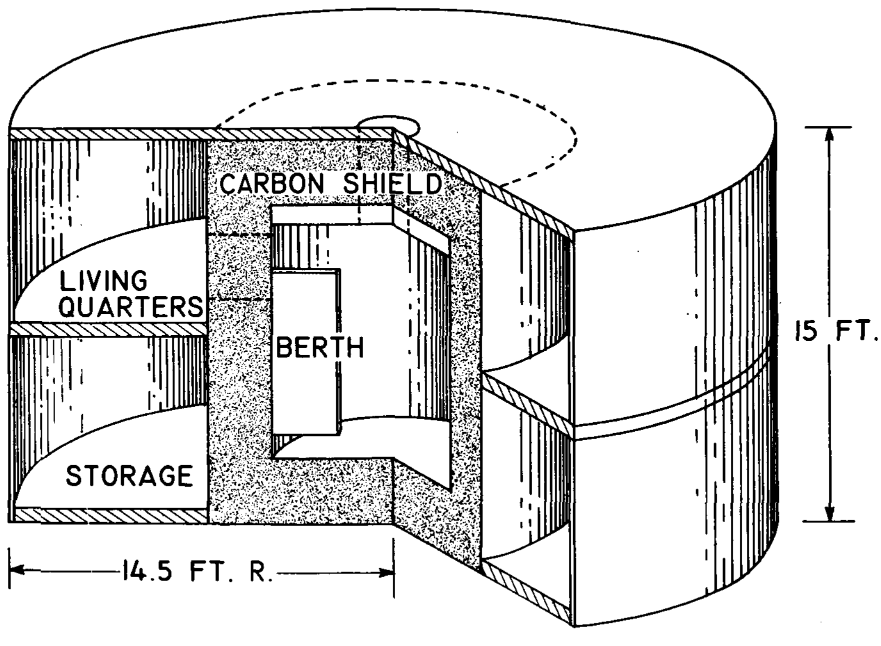

Fig. 21. Seven-man cabin.

Appendix

Vehicle Design Assumptions

Many assumptions are required in an analysis covering as many fields as this study. The assumptions not previously described in the body of this paper are given in the following paragraphs. These assumptions are based, where possible, on a consensus of more detailed studies of the particular topic that have appeared in the literature.

Living Quarters or Cabin—The Mars mission vehicle cabin (Fig. 21) is a circular cylinder 29 ft in diameter and 15 ft high. In the center of the cabin is the shielded vault. The upper level contains the living and working space. The lower level is used for storage of the expendable supplies. The crew is assumed to sleep in the vault and to operate on 8-hour shifts. About 50 sq ft of floor space have been provided per man on the upper level. The space provided falls between that provided for chief petty officers and commissioned officers on submarines.[12] The structure and equipment weight, for this cabin, exclusive of the weight of the shielded vault, was estimated to be about 30,000 lb.

Ecological Considerations—The ecological system assumed comprises an open oxygen system and a closed water cycle. Oxygen consumption was taken at 3 lb/man-day and is supplied by a liquid-oxygen converter.[13] An initial supply of 20 gallons of water per man was assumed; and, since there is a net water production when the reconditioning of the atmosphere is taken into account, this supply[14,15] should be sufficient. Food consumption was taken at 3 lb/man-day. All told, the supplies are expended at the rate of about 12 lb/man-day. The difference between this figure and the sum of the oxygen and food rates comprises CO_2 absorber, odor absorber, and contingency allowances.

Propulsion Systems—The performance assumed herein for the nuclear-rocket engines represents a fairly advanced solid-core design. The gas temperature for such a reactor would be about 4,500°F, producing a vacuum specific impulse of 900 seconds at a nozzle area ratio of 50:1. The weight of the complete engine has been taken as 0.03 times the thrust. Doubling the engine weight produces only an 8-percent increase in initial vehicle weight for a typical case.

Additional weight is allowed for shadow shielding of the engine for both thermal and biological protection. In a typical case, this amounts to less than 1 percent of the vehicle gross weight.

Chemical rocket propulsion is used for the Mars Landing Vehicle. A storable-propellant combination, N_2O_4-N_2H_4, is assumed, with a specific impulse of 300 sec.

Propulsion System Structure—The propellant tanks are assumed to be of the pressure-stabilized type. The tanks were assumed to weigh 0.08 times the weight of propellant contained. The thrust-sensitive structure was taken as 0.01 times the engine thrust.

Propellant Thermal Protection—To minimize the propellant lost due to heating by the thermal radiation from the Sun and the planets, the multiple-reflective-foil protection technique[16] was used. Calculations of optimum thermal protection system designs resulted in a 1½-percent increase in initial weight for a typical short trip and a 3-percent increase for a typical long trip.

Optimization of Thrust-Weight Ratio

The data in the body of the report are for an initial thrust-weight ratio of 0.5. Fig. 22 shows the variations of initial weight, energy expended, and reactor power with the thrust-to-weight ratio at the beginning of the trip for a 420-day mission. The minimum-weight vehicle is achieved at a thrust-weight ratio of about 0.25 for the assumptions of this study. The corresponding power level is 8,000 mw. For a very slight penalty in initial weight, about 10,000 lb, the power level required can be reduced to 6,000 mw by using a thrust-weight ratio of 0.2.

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