chapter 12
Space Nuclear Propulsion

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Introduction

The usual approach to building a nuclear rocket is to use a nuclear reactor to heat a propellant, which is then exhausted rearward to give useful thrust. A generalization of the performance available by this procedure is shown in Fig. 12-1, where the specific impulse achievable is given as a function of the ratio of hydrogen propellant exhausted to uranium fuel burned. Specific impulse is defined as the ratio of pounds of thrust obtained to pounds per second of fuel and propellant consumed. Thus, a high specific impulse indicates low fuel consumption, an extremely desirable property. If very little hydrogen propellant were used, i.e., if the rocket were run on fissioning uranium only, very high specific impulses (over 1 million sec) would be achieved. Under these circumstances, however, not only is the price of fuel and propellant prohibitively high, but the operating temperatures become hundreds of millions of degrees. The temperature problems at that level are at present unsolved. If the operating temperature is reduced to that value which permits the operation of a solid reactor within the thrust chamber, then the specific impulse tends to be limited to perhaps 1,000 sec. In this region, several million pounds of hydrogen are utilized for every pound of uranium fuel actually burned. Between these two extremes lies a region of great current interest. If we could sustain the nuclear reaction in a gas, we could operate at temperatures beyond the solid-core limitations. If we try to reach specific impulses as high as 10,000 or 20,000 sec, the temperatures required are in the 20,000 to 50,000°F region. Although this is extremely hot, it is still three or more orders of magnitude less than the temperatures required for thermonuclear fusion. It is thus an interesting question as to whether or not we could obtain vastly increased space-engine performance by learning to build gaseous-fission engines, which, although hot, do not involve the extremely high temperatures of fusion.

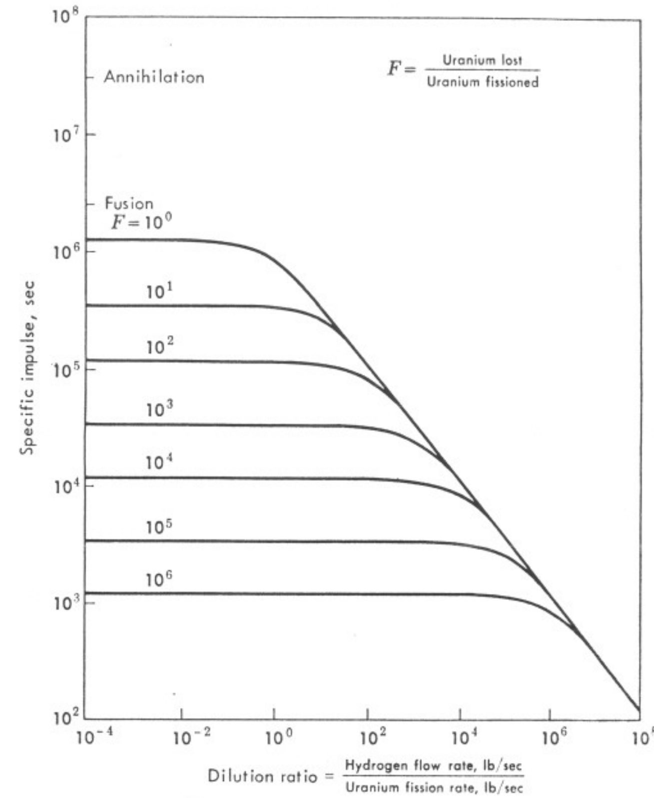

Fig. 12-1 Fission-reactor specific impulse.

It is possible to get an estimate of fuel cost directly from Fig. 12-1. This is done in Fig. 12-2 for different values of uranium containment under the assumption that hydrogen costs 25 cents per pound and uranium $5,000 per pound. It can be seen that, with perfect containment, the cost remains very low until about 10,000 sec specific impulse, after which it rises steeply. Containment is a particularly difficult problem in gaseous-fission systems, however, and separation ratios (the ratio of unburned fuel to propellant in the exhaust) of greater than 10^{-4} cause the cost of fuel plus propellant to increase greatly.

Figures 12-1 and 12-2 are relatively fundamental and form the basis for much of the rest of the discussion. First solid-core nuclear-rocket possibilities, then the gaseous-fission region will be examined. The missions which can be flown with such engines will be considered, and some speculation on the utility of nuclear electric rockets will be presented. Most of the concepts presented are explained more fully in Refs. 1 to 6.

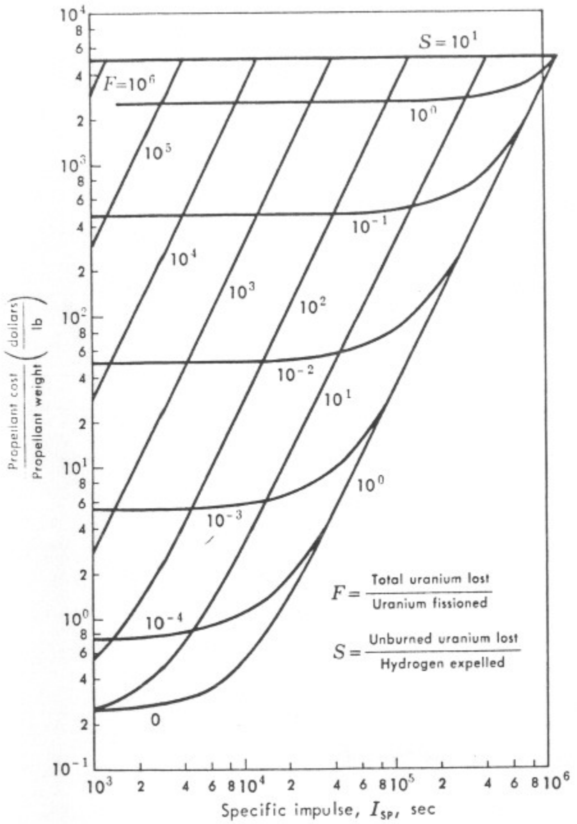

Fig. 12-2 Cost of propellant for fission-reactor rocket.

12-1 Solid-core-fission Rockets

An artist's sketch of a solid-core nuclear rocket is shown in Fig. 12-3. Hydrogen flows into the nozzle wall and through the reflector, cooling both, then through the reactor core, where it is heated, and finally out of the nozzle. A limitation occurs in specific impulse with this system, because of the fact that high temperatures are required to get high specific impulse, and the reactor must be hotter than the propellant in order to transfer heat to it. This limitation cannot be removed by the use of ablating insulation, such as has been used to solve reentry heat problems, for in this case the problem is, not to protect the reactor from heat, but to transfer heat from the reactor to the propellant. Figure 12-4 is an expansion of Fig. 12-1 in the region of interest in reference to solid-core nuclear rockets. If we take the highest solid-material temperature known at the moment, the specific impulse of solid-core rockets could be

improved to somewhat beyond 1,000 sec. Beyond this, we can conceive of a rocket with uranium fuel contained in liquid form as a next step. Note, however, that there is not a great deal of difference between the highest known melting temperature of a solid and the highest known boiling temperature of a liquid. Thus, only a small region of operation

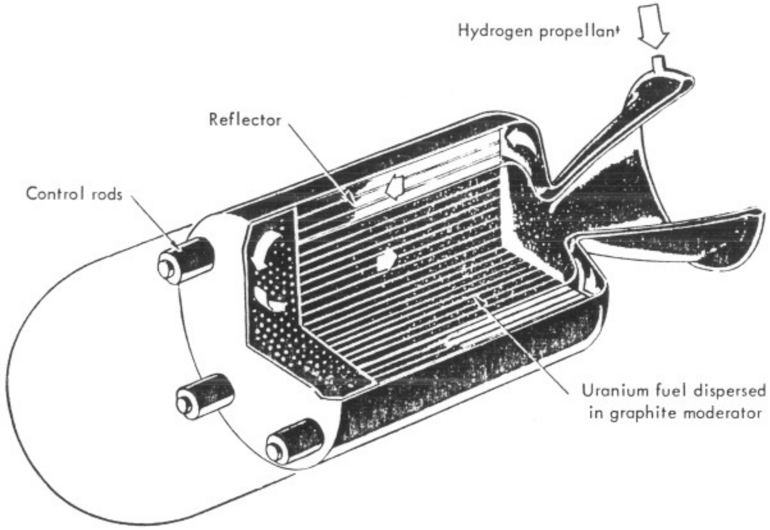

Fig. 12-3 Solid-core reactor. (Graphite.)

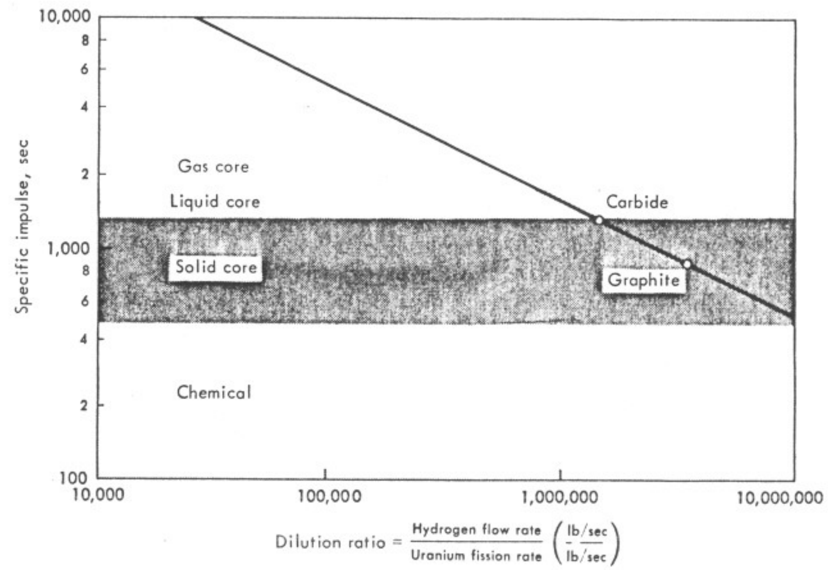

Fig. 12-4 Fission-reactor specific impulse.

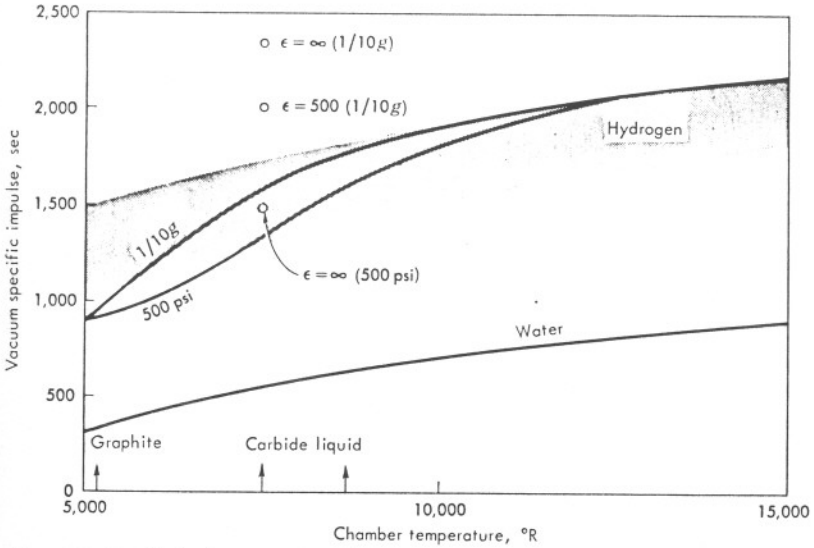

Fig. 12-5 Variation of I_{sp} with temperature. (Nozzle-expansion ratio $\epsilon = 40$.)

is available to the liquid-core system that would not be available to solid cores. Beyond the liquid region, it is necessary to have the reacting fuel in the gaseous state, and it would seem that research should be concentrated there rather than in the liquid region. Although this is a logical conclusion, it could be invalidated if it should turn out to be difficult to make solid reactors in the higher temperature region. Then the liquid system might obtain almost a doubling of performance over the current solid systems, which might be worthwhile for some missions.

Figure 12-5 shows the variation of specific impulse as a function of chamber temperature up to 15,000°R. In addition to simply increasing the temperature, other possibilities for improving performance are indicated. The $\frac{1}{10}$ g corresponds to low chamber pressures at which the situation can be improved if dissociation of hydrogen occurs. Energy is delivered to the propellant stream by dissociating molecules rather than by raising the temperature. If no recombination occurs in the nozzle, the improvement comes about because of the lower molecular weight of the dissociated hydrogen stream. If, furthermore, the nozzle design is such that recombination occurs, energy is released in the nozzle with consequent improvement in the performance in a manner similar to afterburning in a jet engine. This may require long nozzles, with consequent weight penalties. Dissociation is a function both of temperature and of operating pressure, increasing with higher temperatures and lower pressures.

Obviously, development of reactor and nozzle designs to promote dissociation and recombination is a fertile field for investigation. It can be seen that if the carbide reactor could be developed, and in addition some of the previously mentioned improvements obtained, specific impulses in the vicinity of 1,300 to 2,000 would be feasible.

12-2 Gaseous-fission Rockets

Figure 12-6 is an artist's sketch of a possible gaseous-fission reactor. Until recently it was difficult to envision a means for making such reactors feasible. Most of the earlier ideas for ways of utilizing gaseous-fission cavity reactors for propulsion involved diffusion of the propellant through the gaseous fuel so that heating occurred by direct conduction and convection. It was then necessary to separate the two gases and, it was hoped, retain virtually all the fuel on board while all the propellant was exhausted. Hydrogen was normally assumed as propellant, since low temperatures are always comforting, even in non-temperature-limited cases. Schemes such as magnetic-field containment or the use of centrifugal separation in some form of vortex were considered. The weight of the magnetic equipment was, as always, a problem, and the details of vortex stability and containment with any substantial diffusion rate have remained vexing.

Another family of systems has originated from these investigations. Although deceptively similar in appearance, they operate on a basically different principle. These are systems which heat the propellant by means of radiation from the fissioning plasma, rather than by direct intermixing. The containment problem, therefore, is not one of separa-

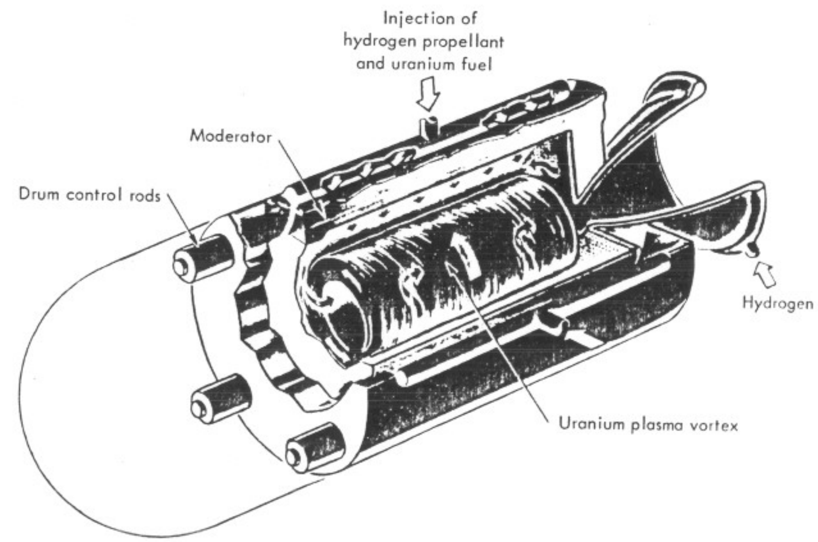

Fig. 12-6 Cavity reactor. (Gaseous vortex.)

tion but rather one of the prevention of mixing—a fundamentally different containment problem. Vortex-stabilization criteria will certainly differ when hydrogen is not diffusing through the core. If magnetic forces are used in any of the schemes, they become of the intensity required to prevent boundary-layer mixing, almost obviously a much smaller field requirement than that for containment of a fission plasma in the face of hydrogen diffusion. A coaxial-flow reactor has been suggested where a central, slow-moving stream of fission fuel heats an annular, fast-moving stream of hydrogen solely by radiation, with separation obtained by velocity differential (Ref. 7). A scheme has even been suggested, in the form of the "glow-plug" reactor, whereby the fission plasma is contained in a quartz bottle (or one of similar material) (Ref. 8). It is possible to cool the bottle to reasonable temperatures while the propellant is still heated to high temperature by radiation, if the bottle transmits most of the radiant energy. This system would yield perfect containment and the idea is hence a very exciting one. New proposals are numerous in the gaseous-fission-reactor field, and an attempt will be made to catalogue the limitations and capabilities of some of these schemes.

Even if perfect containment, either with some new vortex configuration or with a glow-plug concept, were obtained, the problem of handling the energies involved still would tend to limit engine performance seriously, as previously indicated. The portions of the rocket system which must remain solid would have to be cooled in some way, even though no fission-energy release takes place there. These thermal balances have been treated in generalized fashion by Meghreblian (Refs. 9, 10). The thermal loads are intense and are determined by whether or not any fissioning takes place in the solid material, by the fraction of energy appearing in nuclear radiations which will heat the solid surfaces, and by the question of whether the hydrogen propellant is transparent or opaque to thermal radiation. If the solid surfaces are cooled by regenerative cooling only, there is a limit to the amount of cooling capacity in the propellant at the temperature of the solid elements and therefore there is a limit to the amount of specific impulse achievable with regenerative cooling. Even if no thermal load is radiated to the structure from the gas stream, the nuclear radiation tends to limit the specific impulse to about three times that obtainable with a solid core reactor. The actual limit is shown in Fig. 12-7 related to the fraction of energy released (ζ) which must be regeneratively removed by propellant. In Meghreblian's work this fraction is assumed to be 10 per cent, but it is a strong factor, and the possibility of going to smaller fractions by means of thinner reflectors and/or relatively gamma-transparent shells should not be neglected. Only the neutrons must be reflected, and hence their energy absorbed in the reflector, in order to contribute to reactor criticality. It is logical to balance the neutron-reflective properties of materials with their relative gamma transparencies and thermal cooling properties to give optimum reflectors for these applications. The use of a deliberately thinner reflector would require a modest increase in nuclear-fuel inventory for criticality but would yield both higher thrust-weight ratio and higher basic performance. Any fission products which escape in the exhaust no longer contribute their share of nuclear radiation, again yielding increases in performance. As indicated previously, this is very likely to occur in gaseous-fission systems. These latter two effects are illustrated in Fig. 12-8.

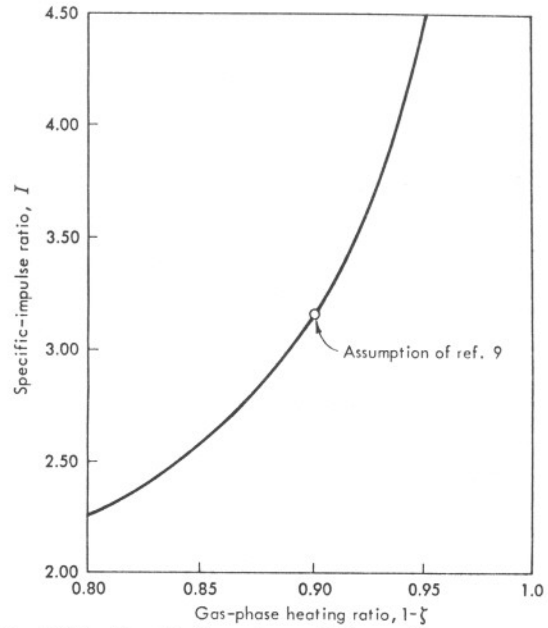

Fig. 12-7 Specific-impulse variation with regenerative cooling only.

If engine performance is extended beyond the values achievable with regenerative cooling, a radiator must be added to reject the excess heat and the thrust-weight ratio immediately suffers as a result of radiator weight. This tends to be a severe penalty, since the desirable specific impulse is of the order of 10 or 20 times that achievable in a solid-core reactor, and, consequently, huge quantities of energy must be rejected through the radiator system. The specific-impulse ratio achievable for the case of all fission occurring in the gaseous phase ($f = 0$), both for the case of no thermal radiation from the gas ($\beta_s = 0$) and for the case of representative values of thermal radiation from an opaque gas, is shown in Fig. 12-9, reproduced from Ref. 10. The opaque-gas assumption is appropriate to systems where the propellant is heated by radiation rather

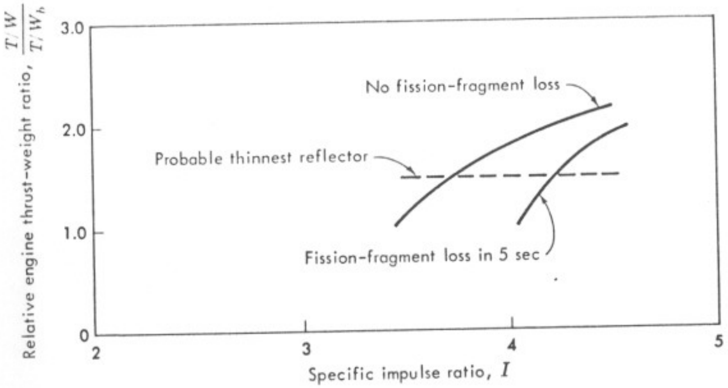

Fig. 12-8 Effect of reflector thickness on reactor performance.

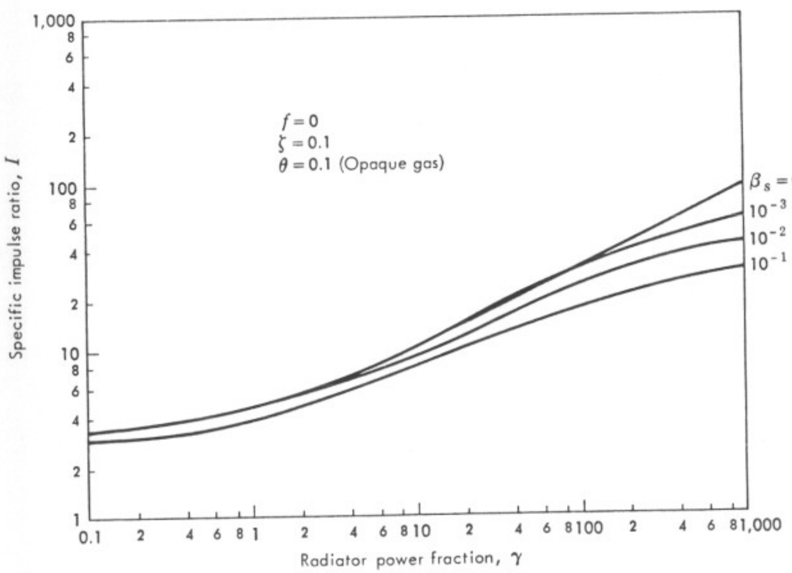

Fig. 12-9 Influence of radiator power on specific impulse.

than diffusion. The radiator power fraction (γ) is the energy which must be rejected from the radiator system compared with that handled by regenerative cooling. It can be seen that this value must be 10 to 100 if specific-impulse improvements of 10 to 20 times that achievable in solid-core reactors are to be generated.

Attempts have been made by various authors to estimate the thrust-weight ratio achievable with such systems. A typical example is shown in Fig. 12-10 as the lowest curve (Ref. 9). Specific-impulse ratio is defined as the ratio of specific impulse to that achieved if the propellant

operated at the temperature assumed for the solid portions of the reactor. This curve clearly indicates that very low thrust-weight ratios are to be expected. Two justified changes in assumptions for this curve yield startling results, however. The curve presented assumed a thrust-weight ratio of about 1 to be achievable with a solid-core system and also assumed a radiator temperature of 1000°K. A value of "Rover-equivalent" thrust-weight ratio of 5 was used for the remaining curves of Fig. 12-10. Although this value can be assumed as high as 20, such estimates make no allowance for pump weights or pressure shells (Ref. 11). Such items were considered in the analyses of Ref. 12, and the assumptions used here amount to adjusting the generalized expressions of Ref. 9 to match the more elaborate single point of Ref. 12.

Changing the Rover-equivalent thrust-weight ratio to 5 still indicates very low thrust-weight ratios at high specific-impulse values, as shown in the next-to-lowest curve in Fig. 12-10. A review of radiator assumptions was therefore thought to be in order. All analyses of radiator configura-

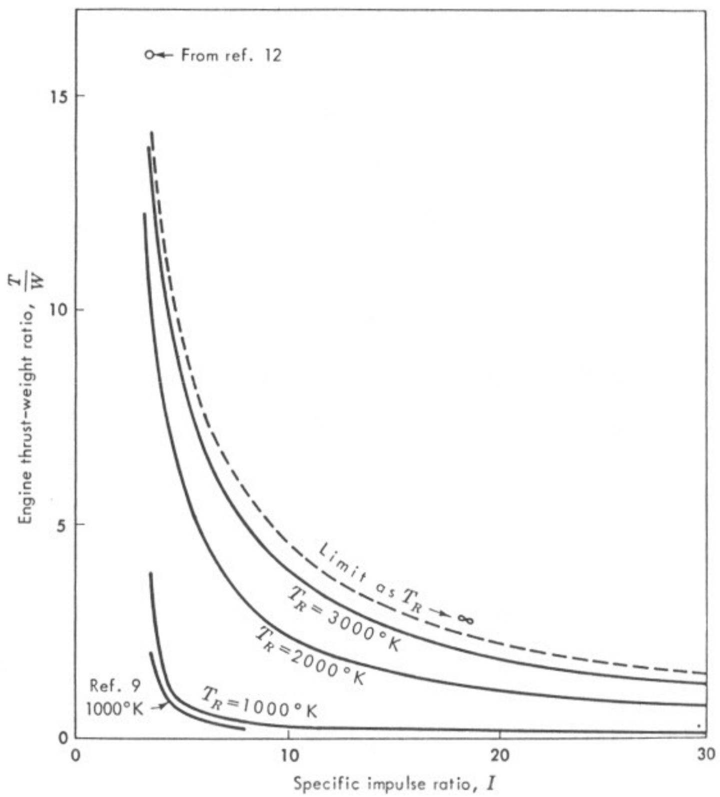

Fig. 12-10 Effect of radiator temperature on engine thrust-to-weight ratio.

tions for nuclear propulsion known to the author have centered in the requirements for nuclear electric systems for either propulsion or auxiliary power. In at least two respects these requirements are totally different from those for gaseous-fission rockets. The first point is that nuclear electric systems must be designed for long operating times (of the order of years) so that such problems as meteoroid penetration of the radiator surface must be considered in terms of long-time probabilities. This point strongly influences radiator weight. A high-thrust gaseous-fission system would operate for periods of only minutes at a time, and therefore an investigation of short-life radiators is pertinent. It is true that the radiator must survive for the total flight duration, not simply the engine-burning period, since the engine must be used for braking at the terminal. However, total flight times will be much shorter than for electrical systems, the radiators might be protected while not radiating, and the loss of a radiator segment would not be very crippling to mission performance. Second, and considerably more important, the radiator temperature of the gaseous-fission system can be as high as it is possible to build radiators. In a nuclear electric system, a balance must be struck between the efficiency of the conversion process, which requires the rejection of heat at a low temperature, and the decrease of radiator weight, which, in general, occurs at high temperature. As a result, radiators usually want to operate at a temperature on the order of three-quarters of the maximum cycle temperature, and the maximum cycle temperature is determined by the ability of either rotating machinery or thermionic systems to operate for periods of years at this maximum temperature. Therefore, radiators for gaseous-fission-rocket cooling systems should be operated at much higher temperatures than those for nuclear electric systems and might be easier to design because of the vastly shorter operating time requirements.

The upper curves of Fig. 12-10 compare the thrust-weight ratios of systems with radiators of 1,000, 2,000, and 3,000°K temperature and, for comparison, a zero area case ($T_R \rightarrow \infty$). The radiator weight per unit area may well increase stepwise as temperatures increase because of the need to utilize different materials (which are usually heavier) at higher temperatures. A constant value was assumed here, and although Ref. 11 used a 5-psf radiator-system weight, the value of 1 psf of Ref. 9 is felt more appropriate in the light of the short operating times. It can be seen that, even at specific-impulse ratios of 20, a thrust-weight ratio of over 1 is achievable with only a 2000°K (3140°F) radiator temperature. Thus, if the radiator loop were operated in the same temperature region as the primary heat-exchanger loops being considered for advanced power systems, high thrust-weight ratios result for gaseous-fission rockets. Clearly, special radiator designs appropriate for gaseous-fission reactors should be the subject of intensive investigation.

The thermal load deposited in the structure from the gas stream is

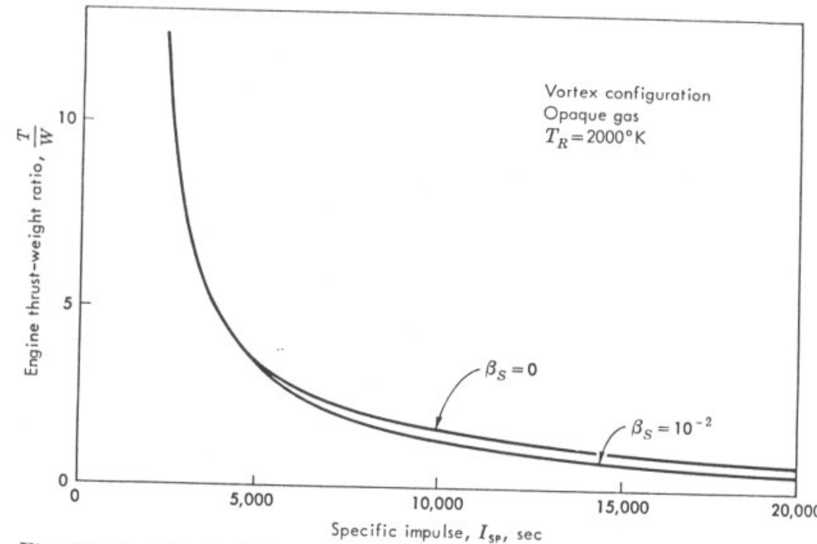

Fig. 12-11 Effect of thermal-radiation load to solid elements.

strongly dependent upon whether the gas is transparent or opaque to thermal radiation. Since only systems which transmit energy from the nuclear plasma to the propellant by radiation are being considered here, it is implied that virtually all such radiation is absorbed in the gas stream and, consequently, the opaque-gas assumption is logical. The effect of inclusion of thermal loads is shown in Fig. 12-11. The assumed value of the radiation parameter β_s of 10^{-2} is representative of a gaseous-fission system which has all fissioning plasma elements at least two optical thicknesses within the propellant mass. It can be seen that, although thermal radiation becomes increasingly important at large values of specific impulse, thrust-weight ratios of about 1 are still possible throughout the regions of interest for interplanetary transportation.

The propulsion-system thrust-weight ratios of Figs. 12-10 and 12-11 are still somewhat tentative. Even though these curves are approximate, it does appear firm that high-temperature radiators are fundamentally important. The thrust-weight ratio will decrease at the higher specific impulses, but perhaps not so much as indicated, owing to two strong compensating effects not included. These are the decrease in pump weight due to the lower fuel-flow rates required at high specific impulses, and the decrease in reactor size due to the smaller radiant-plasma area required to generate the necessary power at high temperatures. Unless these effects cause the thrust-weight ratio to become almost independent of specific impulse, however, there will be an interest in multiple-use engines which carry both the radiator required for high specific impulse and the pumps required for higher thrust at low specific impulse. Such ships can then take off on the high-thrust mode and switch to the high-specific-impulse mode after orbital speed is attained. One example of the effect of carrying such extra radiator area, compared with designing an optimum engine for each specific impulse, is shown in Fig. 12-12. It is practical to consider such two-phase engines. A sidelight of the use of thermal radiation from a fissioning plasma is that the throttling of such rockets may involve unusual design techniques, since the power per unit plasma radiant area is constant at a given temperature, and hence some area must be removed from effective radiation in order to throttle at a constant specific impulse.

One other interesting fact arises from the gaseous-fission thrust-weight values presented. It is normally considered that such rockets would use hydrogen as a propellant. As long as radiators could not be used to improve gaseous-fission-reactor specific impulse because of their weight, it was still important to use hydrogen, both because of the relatively high specific impulse it provides for a given structural temperature and also because of the high heat capacity and, therefore, degree of regenerative cooling which it permits for a given structural temperature. It is clear that these restrictions do not apply to systems employing high-temperature radiators. It is thus interesting to look at more suitable propellants from the point of view of cost, density, and space storability.

Many substances come to mind, such as diborane, ammonia, or, the most convenient of all, water. Availability on other planets might well become the single governing item on propellant selection. The performance of a gaseous-fission reactor operating on water, compared with operating on hydrogen, is shown approximately in Fig. 12-13. This curve does not utilize an accurate estimate of enthalpy of water at high temperatures but is probably a lower limit, since complicated dissociation modes at high temperatures were ignored. In addition, allowance was not made for the fact that water pumps would be of considerably lighter weight than hydrogen pumps for these systems. It is clear that the water rocket should be actively pursued.

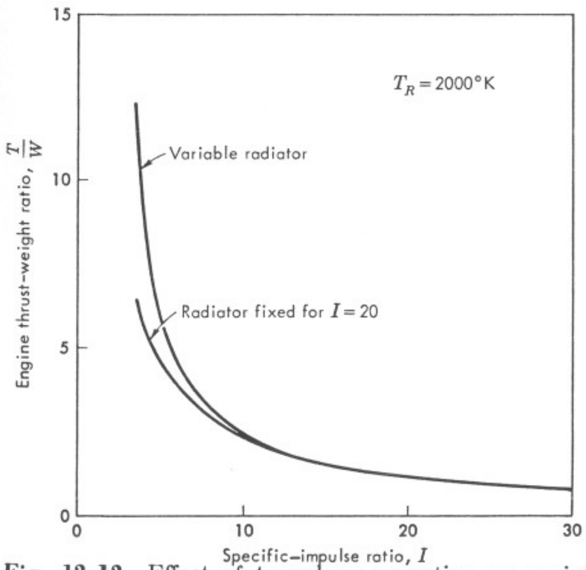

Fig. 12-12 Effect of two-phase operation on engine thrust-weight ratio.

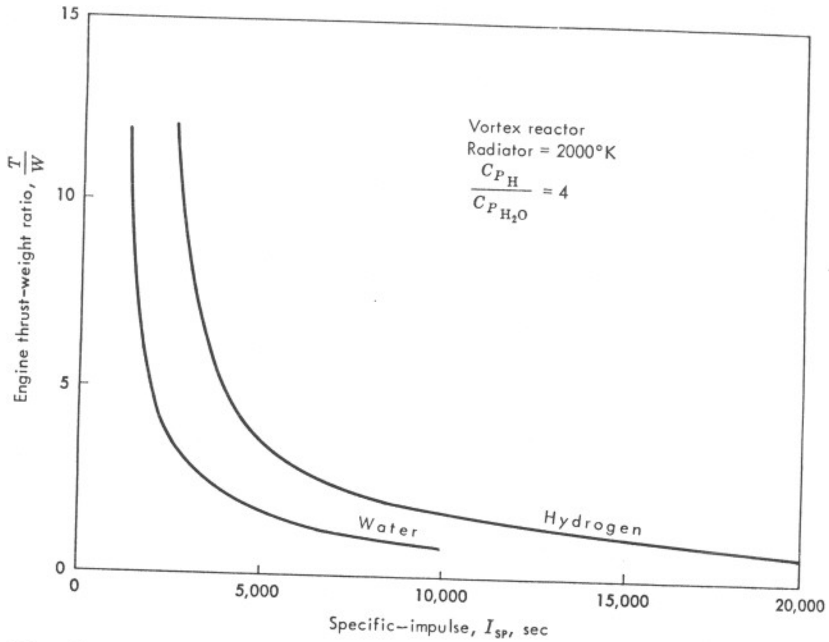

Fig. 12-13 Engine performance for hydrogen and water propellants.

12-3 Shielding

The weight of shielding required to protect any space vehicle and its occupants from both natural radiation and the radiation produced by the reactor is an important design consideration. It will not be discussed in detail here, since shielding for natural radiation is covered in other portions of this volume. However, one point concerning radiation from the reactor is of fundamental interest. It would seem intuitively that shielding weights for a nuclear space vehicle would be entirely prohibitive, in view of the fact that shielding weights have been one of the major problems of the nuclear-aircraft program. Figure 12-14 illustrates the pertinent facts involved. Although the reactor power of the aircraft is

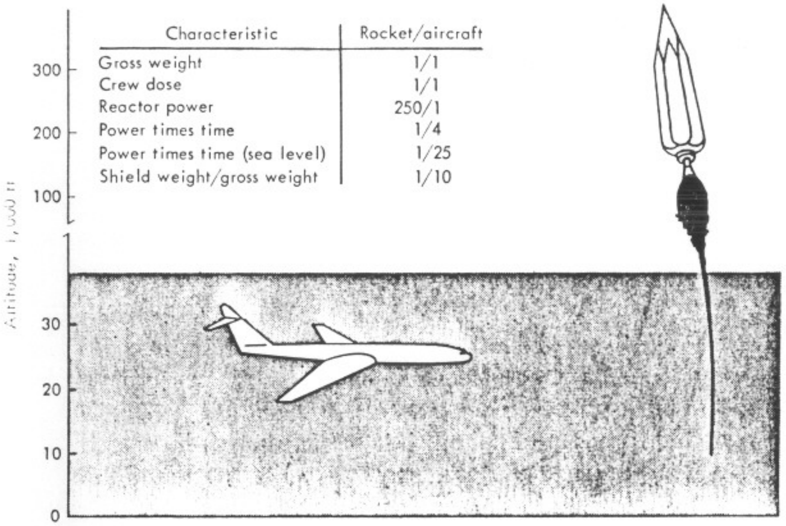

Fig. 12-14 Nuclear-vehicle-shielding comparison.

several hundred times lower than that of the spaceship, shielding weight is a function of power times operating time. The purpose in building nuclear aircraft is to obtain long-duration flights so that the operating time is days or weeks. The nuclear spaceship, however, uses its main propulsion system for perhaps only 10 min at the beginning and end of each voyage. If we compare typical values of power times time, the nuclear spaceship is characterized by a value only one-quarter of that for the aircraft. Furthermore, the airplane by definition must always operate in the Earth's atmosphere and hence is continually subject to the radiation scattered back from the Earth's atmosphere. This scattered radiation accounts for the largest contribution to the shielding weight. The spaceship, on the other hand, climbs quickly out of the atmosphere. Estimates of the equivalent power times time that each device would experience operating at sea level (a measure of the total scattered radiation) indicate a factor of 25 in favor of the nuclear spaceship. It is not surprising, therefore, that the shielding weights required for the nuclear spaceship are an order of magnitude smaller than those for nuclear aircraft. Further reductions in shielding can be made by utilizing such items as cargo, food, the life-support system, and propellant in the tanks as shielding material (Refs. 13, 14).

12-4 Earth-orbital Missions

The missions possible with nuclear rockets will be examined by starting with the lowest-energy mission, namely, merely placing objects in low

Earth orbit. The total impulsive velocity required to place objects in low Earth orbit, with appropriate allowance for drag and gravity losses, is of the order of 30,000 fps. Since this chapter will be dealing with very high-energy propulsion systems, however, a total impulsive velocity of 60,000 fps, enough to permit reentry and landing by rocket retro thrust only, will also be considered.

The propellant to launch gross-weight ratios required for single-stage vehicles for the two values of impulsive velocity mentioned are shown as a function of specific impulse in Fig. 12-15.

It can be seen from Fig. 12-15 that the amount of fuel which must be used, even when reentry is done completely by rocket braking, is a remarkably low value at high specific impulses. The fuel load actually can be lower than that currently carried by transcontinental jet-transport aircraft. Although this seems startling, the energy requirements are not really much different. If one assumes a jet transport to be cruising for 6 hr at an L/D of 12, this means that the engines were applying one-twelfth the weight of the airplane for the 6-hr period. If this could have been applied in field-free space, the vehicle would have accelerated $\frac{1}{12}\,g$ for 6 hr and would have generated 58,000 fps, essentially the maximum total velocity considered here. Thus normal cruising aircraft utilize the same order of magnitude of energy as required for a modest space transport system, essentially because such aircraft fight gravity incessantly during their whole flight. In space operations, high-thrust rockets fight gravity quickly and efficiently in the early part of their flight and coast afterward. Space flight is expensive, therefore, not because the energy

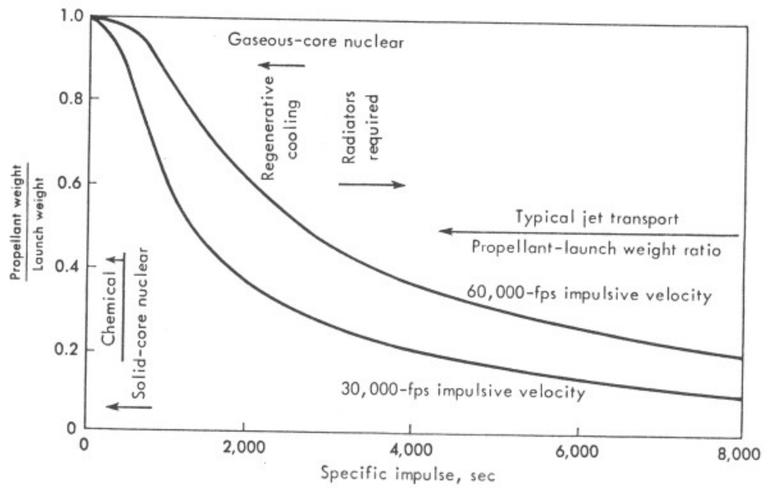

Fig. 12-15 Required fuel weights for space-launch vehicles, single-stage.

required is high, but because no way has yet been found to package and utilize these energies out in space.

Only the nuclear rocket achieves high enough exhaust velocity to create the necessary performance in a single-stage vehicle and can even reduce fuel-load requirements to the same magnitude as those of conventional transport aircraft. The key to achieving low-cost operations, then, is the design use of the large weight fractions available to achieve the best trade-off between the payload delivered to orbit and the structural weights devoted to enhancing long life, minimum cost of refurbishment, and convenience of operation. It is possible to take the basic nuclear-rocket data of Figs. 12-1 and 12-2 and combine these with both mission-performance requirements and estimates of the cost of hardware to make an overall estimate of the direct operating cost of a space transport (Refs. 4, 6). Many different assumptions are possible with respect to launch reliability of vehicles, reliability of recovery, cost of refurbishment, and vehicle total life. In order to delineate the highest potential of nuclear propulsion, the assumptions used in the remainder of this discussion are that single-stage space vehicles will be as reliable (both upon launch and return), will be as easy to maintain, and will have the essentially zero refurbishment costs of jet-transport airplanes. These are felt to be extreme assumptions by the missile-development community, but they are not necessarily extreme if a truly adequate space engine were to make its appearance and, consequently, the weight fractions of Fig. 12-15 became available.

12-5 Lunar Missions

Cargo costs for a round-trip lunar mission are shown in Fig. 12-16, drawn for two different values of impulsive velocity both with and without aerodynamic braking on return to Earth. Two values of re-use, 100 and 1,000 times, are indicated. The vehicles are designed to carry propellant for the complete lunar round trip with no lunar refueling. The cost applies to the entire flight. Operation at specific impulses as low as 800 sec are shown to be feasible, although payload costs are somewhat high, on the order of $30 per pound. For specific impulses in the 1,500-sec region payload costs may be as low as $1 per pound. The rapid decrease in the cost curves between 800- and 1,500-sec specific impulse is one illustration of a case where the development of a liquid-core reactor might be justified, because of lower operating costs on this mission, if undue trouble were experienced in developing either high-performance solid-core or gaseous-fission reactors.

The theoretical indication of cargo costs as low as $1 per pound has very strong implications for the future of space operations. Only a few years ago, it was stated that, even if one discovered pure diamonds on

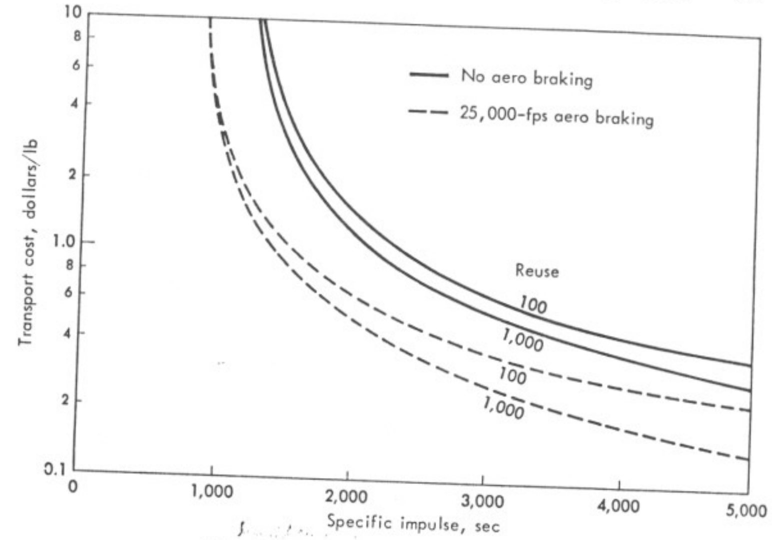

Fig. 12-16 Cargo transport cost, lunar round trip.

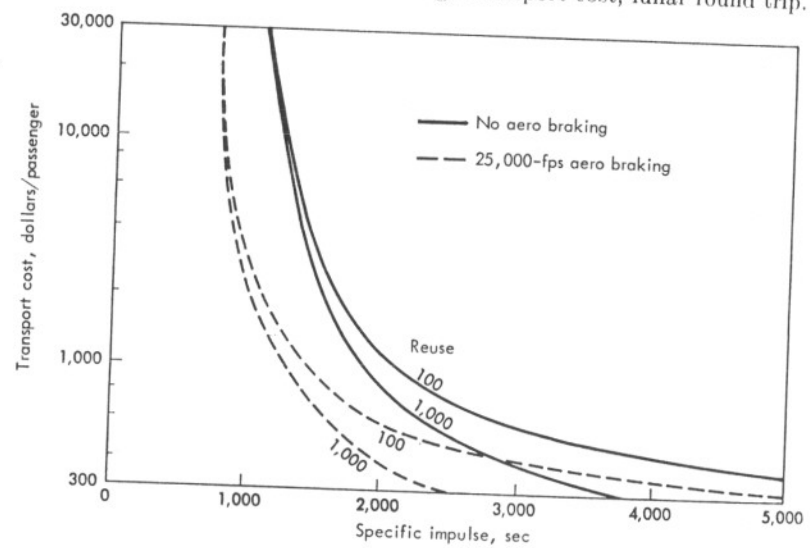

Fig. 12-17 Passenger transport cost, lunar round trip.

the Moon, it would always be too expensive to return them to Earth owing to the fundamentally high energy requirements. It is difficult to estimate the price of a pound of diamonds, but, in general, somewhat over 1 million dollars is reasonable. Therefore, not only diamonds at over 1 million dollars per pound, but uranium at 5,000 dollars per pound, gold at 500 dollars per pound, and much modern equipment at current market prices could be brought from the Moon at a profit. If we can achieve such low costs, we must drastically revise our thoughts on the nature and magnitude of future space operations.

It is possible to extend the estimation of costs into the area of passenger transport. Figure 12-17 shows the results of the application of such an analysis to a lunar round trip. The analysis has included time-dependent functions such as pilot costs, stewardess costs, and food supplies, in addition to basic propulsive costs. Curves are shown for re-use of 100 and 1,000 times. Operations can be seen to be feasible, although the cost is high when specific impulses under 1,000 and no aerodynamic braking are used. These direct operating costs drop to as low as $500 per passenger for lunar travel. The direct operating cost is not to be confused with actual ticket prices, which require the addition of indirect costs. Although the indirect costs probably would not exceed those characteristic of current airline practice, the actual amount to be added for such a mission is obscure.

12-6 Near-planet Missions

The velocity requirements for minimum flight times to the various planets are shown in Fig. 12-18. Travel between Earth and Mars is concentrated upon in the following examples, as it explains all the features of travel to other places. It is assumed that a really good transport system must involve relatively short flight times; otherwise, the equipment is tied up too long in transit, the cost of sustaining passengers mounts, and the

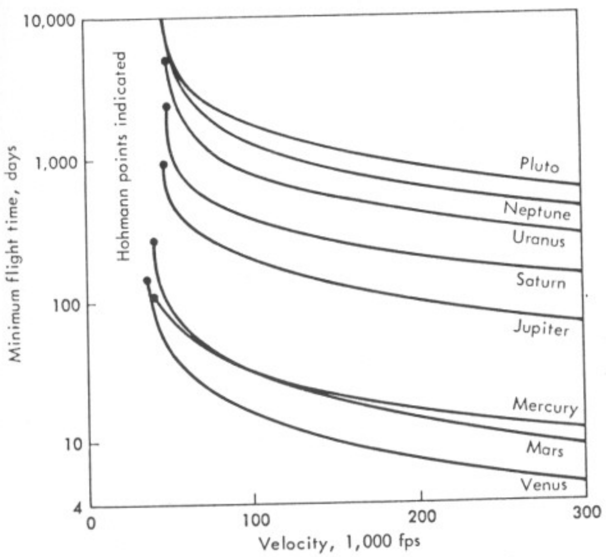

Fig. 12-18 Earth-to-other-planets ballistic transfer, minimum flight times for interorbital transfer.

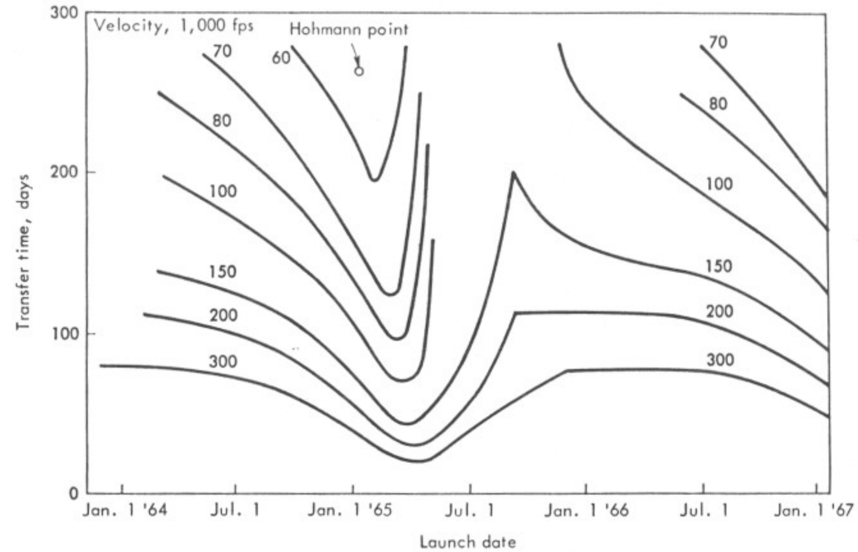

Fig. 12-19 Impulsive-velocity requirement for Earth-to-Mars transfer and Martian landing.

general overall inconvenience of the system becomes great. The continual development of terrestrial transportation systems to higher and higher speed regions exemplifies this point. If the Mars operation were performed at minimum energy, the flight time would be 9 months each way; furthermore, the operation can be performed only once every 780 days. This is no way to run a railroad!

If velocity capabilities of the order of 100,000 fps were available, the minimum flight time to Mars would be of the order of a month, which is a much more acceptable value. Actually a total of 200,000 fps is required for 1 month because of braking requirements of Mars. At the latter total speed an exploration to Pluto requires about 5 years each way, so that if this velocity capability is ever achieved, deep-space exploration could be conducted with this class of vehicle.

Consideration of minimum flight times to the planets, however, reveals only part of the story. Since it would be desirable to conduct operations throughout the year, year-round flight capabilities are of interest. Figure 12-19 shows the transfer time to Mars as a function of launch date for different velocity requirements. The cycle repeats itself every 780 days, as previously mentioned. The velocity shown on this curve is the total velocity requirement for a one-way operation, including both take-off from Earth and landing at Mars. With speeds of the order of 70,000 or 80,000 fps fairly reasonable exploration can be conducted, since one-way travel times are only about 4 to 6 months. Speeds between 150,000 and 300,000 fps are required to achieve fairly respectable year-

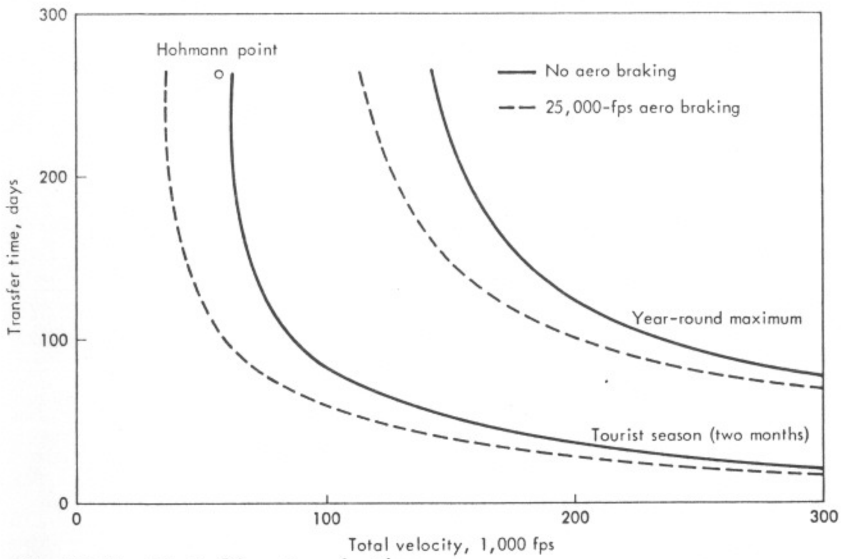

Fig. 12-20 Earth-Mars transfer time.

round operation. At 150,000 fps, for instance, the average flight time during the unfavorable portion of the year would be of the order of 4 months. No matter how fast the rocket travels, there is always a time in the synodic period when the planets are in favorable positions for the transits, and thus the flight times are vastly reduced. In interplanetary-travel schedules, therefore, there will be a tourist season when either faster flights may be made or larger cargos may be carried, with consequent savings in cost.

Data from the previous figure are replotted in Fig. 12-20 to give the transfer time vs. the total velocity for both a 2-month tourist season and the year-round maximum. The basic curves are shown for both zero aerodynamic braking and 25,000 fps of aerodynamic braking. As previously mentioned, if one had a really high-performance space engine, one could solve the reentry problem by simply standing the rocket on its tail and backing it slowly down through the atmosphere. To attempt to do this with a marginal propulsion system involves a severe penalty, as shown by the large effect on transfer time of aerodynamic braking in the low-velocity region of the curves. However, when impulsive velocities reach the point where transfer times are of the order of a few months, it makes relatively little difference whether aerodynamic braking is employed or not. It is, in fact, probably desirable to avoid it in order to avoid structural-weight penalties or possible use of coatings which might have to be replaced after each flight. A solution may be to use lightweight ablating coatings which would not be required for normal oper-

ation but which would be available for emergencies when it became necessary to make severe atmospheric reentries. In this case replacing the coating would be an acceptable reconditioning cost. One other point should be made: a source of working fluid on foreign planets is highly desirable. If year-round operation with a maximum trip time of 4 months is desired, 200,000 fps total velocity one way and approximately 400,000 fps round trip are required. If a source of hydrogen exists on Mars, the ship is designed for only 200,000 fps. Thus, although early expeditions can function nicely by operating only during the tourist season, and possibly by shipping hydrogen in tankers for the return flight, one of the first objectives for early exploration should be the establishment of working-fluid production facilities on the other planets. This may not be logistically difficult if the propulsion reactors could be modified to serve as power sources for the production process.

An interesting sidelight with respect to the early views of space flight expounded decades ago is evident. It was formerly assumed that it would be so difficult to achieve the velocities required for space flight that only minimum possible velocities would be used. Under this assumption, most of the velocity generated is needed to break free of the gravity fields surrounding each planet, and only a small amount is required to transfer from planet orbit to planet orbit. This type of minimum-energy operation, as indicated previously, yields extremely long flight times and very awkward operations. When high velocities are utilized, as suggested here, to reduce travel time, they become so great because of the large distances involved that they are the dominant

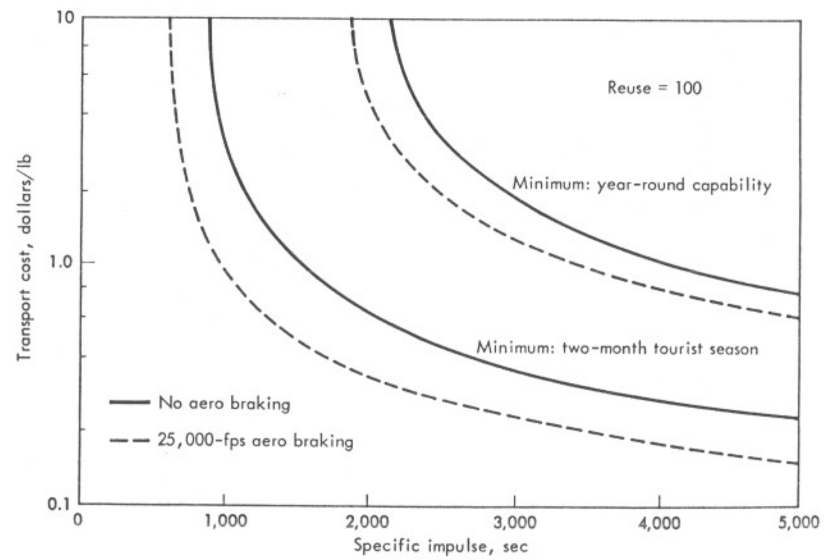

Fig. 12-21 Cargo transport cost, one-way Martian trip.

velocity requirement. In fact, these ships would move through the planetary gravity fields so quickly that the gravity losses would be reduced to a small fraction of the losses experienced with the minimum-energy case. Thus, in very low-energy space flights the gravity fields of the planets create the major energy requirements, but in high-velocity space flight the planetary gravity fields are only a minor inconvenience.

Cargo transport costs to Mars are shown in Fig. 12-21. These were calculated under the same assumptions as the lunar cost curves. Unmanned cargo ships were assumed. The order of magnitude of cost of delivery of cargo to the near planets thus may be not much greater than lunar cargo delivery, and the previous comments concerning the effect of low costs upon lunar operations must also be applied to at least the near-planetary region.

12-7 Nuclear Electric Rockets

It is frequently considered that interplanetary transportation is the logical operating region for nuclear electrical-propulsion systems. Nuclear electric systems are basically attractive, since they accelerate the working fluid electrically and hence possess a straightforward way of circumventing the restrictions placed upon thermal rockets by the high temperatures required for high performance. Although this is theoretically a straightforward solution to the dilemma, its effectiveness is greatly limited by excessive weight. The critical factor in this weight, to date, is the high unit weight of electrical-power conversion equipment. Even extremely advanced systems provide estimates of power-conversion-equipment weights of the order of 10 lb per exhaust kilowatt, and the performance of optimized systems has accelerations of the order of magnitude of only 10^{-3} g.

Slow escape from a gravitational field with a low-acceleration (microthrust) device invokes a kinematic-efficiency penalty compared with a high-acceleration (macrothrust) device. The magnitude of this effect is indicated in Fig. 12-22, where the impulsive velocity required to escape from orbit is shown as a function of vehicle acceleration (Ref. 5). Also shown are curves for given amounts of hyperbolic excess speed, i.e., the speed remaining at a distance from Earth which is sufficiently large so that the effect of Earth gravity is negligible. The impulsive-velocity penalty of the low-thrust system in attaining escape speed is further increased when large hyperbolic excess velocities are required. It is helpful to use Fig. 12-22 in defining high thrust. For the case of escape from circular orbit, it can be seen that appreciable kinematic inefficiency is involved only if the thrust-to-weight ratio falls below the order of ½. Hence, in general, high thrust is considered to be a thrust-to-weight ratio in excess of ½.

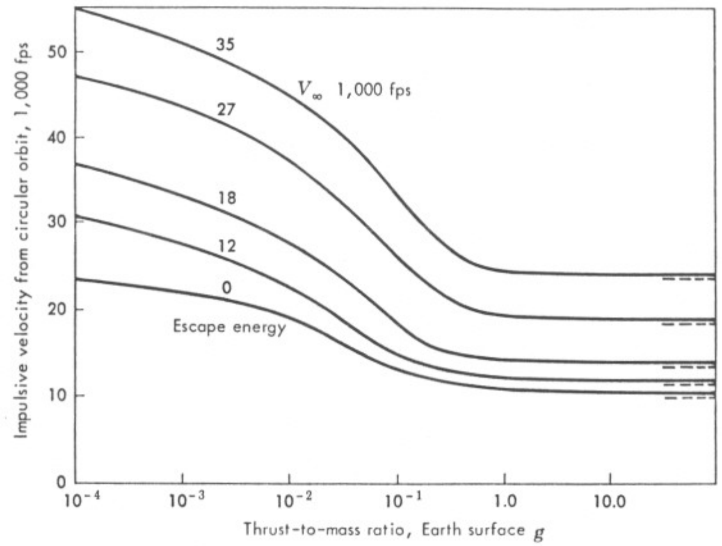

Fig. 12-22 Total impulsive velocity, escape from circular orbit.

It would be possible to make a payload comparison between vehicles at constant total flight energy (hyperbolic excess velocity) directly from Fig. 12-22, but in this case the trip times would be grossly different. The acceleration time of the microthrust vehicle may even exceed the total travel time required by its high-thrust equivalent. For example, it requires nearly 200 days to accelerate to a hyperbolic excess of 35,000 fps at a thrust-to-weight ratio of 10^{-4}. However, for the same hyperbolic excess generated by a high-thrust vehicle, the total travel time to Mars at optimum times would be only on the order of 80 days. It is felt necessary for a proper comparison between systems to base them on equivalent travel times.

Total impulsive velocities required of microthrust and macrothrust vehicles to perform a round trip to Mars at favorable constellation times are plotted in Fig. 12-23. This is a result of a two-dimensional analysis which assumes departure from a low-altitude circular orbit about Earth, transferring to a low-altitude circular orbit above Mars, and return to a low-altitude circular orbit about Earth. In the case of microthrust, it is assumed that the vehicle accelerates continuously at a constant rate, initially accelerating up to some peak energy and then decelerating. In the case of the high-thrust vehicle, the required impulsive velocities are optimized. Departure from an Earth orbit along any particular trajectory implies a particular increment of velocity at departure and a certain Martian approach velocity, with a certain amount to be dissipated in establishing a circular orbit around Mars. The sum of these two has been minimized by selection of a particular trajectory. Total travel time

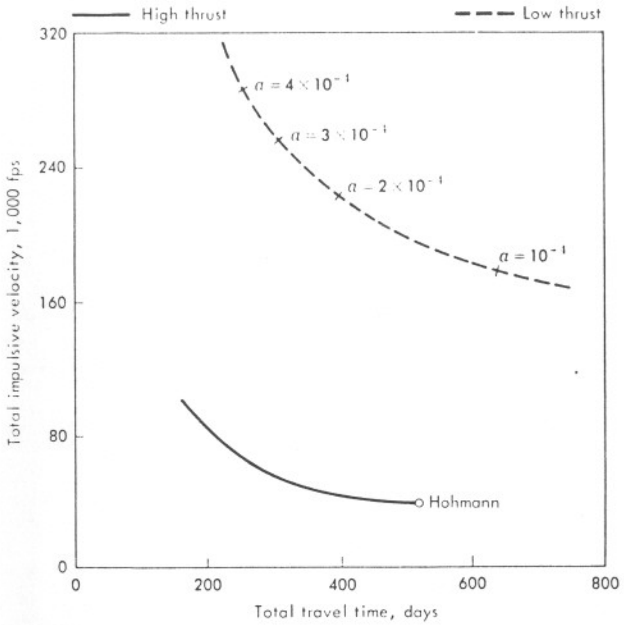

Fig. 12-23 Mars round trip.

is precisely that; it does not include any required period of waiting in a Martian orbit to allow establishment of the return trajectory, and the assumption is made that travel times to Mars and from Mars are identical. These restrictions make the curves somewhat unrealistic but apply equally to both microthrust and macrothrust, so that the comparison is valid. As can be seen in Fig. 12-23, the total impulsive velocity required for microthrust is approximately four to five times larger than the high-thrust case for equal travel times. This difference, then, is caused both by the previously cited inefficiency of gravitational escape under microthrust and by the fact that a microthrust vehicle must accelerate to much higher peak energies to compete on an arrival-time basis.

A computation of the total energy expended in an Earth-Mars operation for both high- and low-thrust vehicles is shown in Fig. 12-24. It can be seen that the difference between the total work of the microthrust and macrothrust vehicles differs by a factor of over 100. The unit of megawatt-days was intentionally chosen, since 1 Mw-day represents approximately 1 g of fissionable material consumed. The microthrust vehicle shown here would consume between 10 and 16 kg of fissionable material during its trip, whereas the assumed high-thrust vehicle would consume some 30 to 150 g. Total power required by the microthrust reactor is in the neighborhood of 20 Mw. Then an interesting point is apparent: for such a reactor the normal critical mass used in reactor design would be

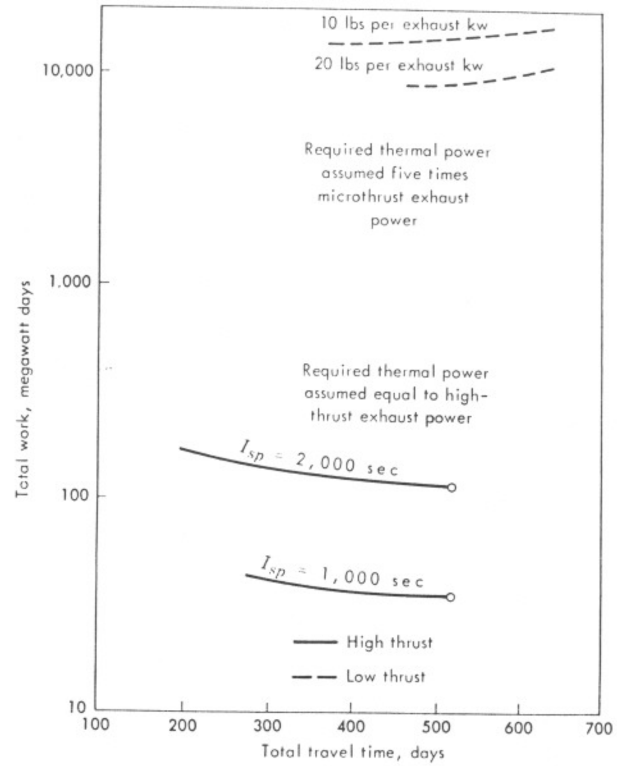

Fig. 12-24 Mars round-trip total reactor work.

on the same order as that required of expendable fissionable material. A very high percentage of useful fuel in the reactor may make it difficult to achieve the very low specific power-plant weights cited in these figures and elsewhere and would make the provision of excess fuel to permit repeated mission use very difficult. It may well be necessary to rebuild the reactor after each flight. The high-thrust vehicle, however, has a very nominal consumption of fissionable material, of the order of 100 g, for a 1,000-Mw reactor. Even a conventional reactor should therefore possess a high re-use capability.

Estimates have been made of the comparative shielding weight required by the two types of vehicle. In spite of the two orders of magnitude difference in energy utilized, the actual shield weight involved is only about twice as great for the low-thrust system. This is basically due to an exponentially beneficial effect of shield thickness.

The useful load which can be carried by the Mars round-trip vehicle is shown as a function of travel time in Fig. 12-25. Two curves are shown for microthrust, one at a power-plant specific weight of 10 lb/kw and one at 20 lb/kw. It is well to remember that the power-plant specific weight

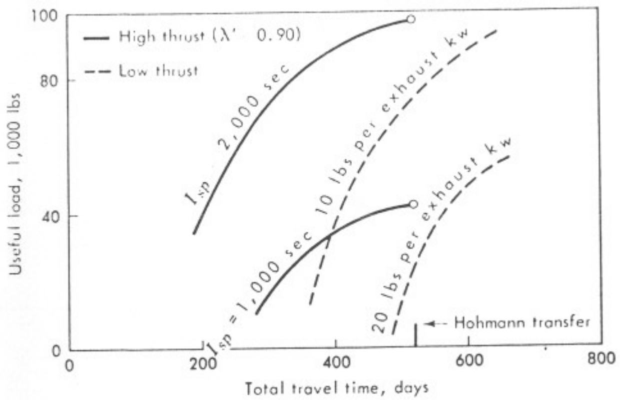

Fig. 12-25 Mars round trip. Useful load of 200,000-lb vehicle.

is the weight of the power plant per exhaust kilowatt. In considering the efficiency of the turbine, the generator, and the accelerator itself, the area between 10 lb per exhaust kilowatt and 20 lb per exhaust kilowatt may be assumed to be reasonably optimistic, since it implies only 2 to 4 lb per electrical kilowatt. For the high-thrust case two curves of different specific impulses are shown: one at 1,000 sec and one at 2,000 sec. As can be seen by Fig. 12-25, the useful loads are comparable with the low-thrust systems, having the edge at long flight times, while at short flight times the high-thrust vehicles would seem better. Further conclusions based on this curve alone usually degenerate into an argument over the achievability of microthrust machines with 10 lb/kw as opposed to high-thrust nuclear rockets with specific impulses in excess of 1,000 sec.

For the purposes of this discussion, no further resolution of the question of the relative utility of nuclear electrical rockets as opposed to high-thrust systems will be attempted. It is simply pointed out that, for any fair transportation comparison, equal mission convenience must be considered. When this is done, curves such as those of Fig. 12-25 result and usually indicate considerable performance advantage for high-thrust systems. To make a substantial change in this conclusion requires about two orders of magnitude reduction in weight of electrical power-conversion equipment, at which point the electrical rocket would also become a high thrust-weight system. The rest of this discussion will be devoted to high-thrust nuclear rockets exclusively.

12-8 Solar-system Missions

Space exploration (and competition) will spread as far away from the Earth as technical and economic factors permit. If one is to keep from

drastically revising his large vehicle program every year, he must anticipate, with vigor, the performance regions of interest. A nuclear-spaceship performance analysis has been made on the assumption that it would be possible to create fission rocket-propulsion systems with perfect containment and with thrust-weight ratios comparable with solid-fission systems over the entire fission performance spectrum (Ref. 6). As previously indicated, this is a very large assumption, both because of the extremely difficult containment problems of high-temperature fission plasmas and because of the difficulty of handling the large, spurious energy fluxes. The reason for such a theoretical exploit was to see whether or not a limit on desirable performance existed for solar-system transportation, regardless of engineering difficulties. Figure 12-26 presents some of the basic data from this study. It can be seen that an optimum ratio of useful load to gross weight can be defined on the basis of vehicle operating costs. If, at a given velocity, too low a specific impulse is used, the price of hydrogen carried becomes excessive. On the other hand, if too high a specific impulse is utilized, the improvement in load-carrying ability does not make up for the high price of uranium consumed.

It is interesting to note that these optimum regions exist at very high useful load fractions, ranging from 80 per cent at low velocities to 40 per cent at high velocities. This is the region where single-stage rockets are structurally as efficient as multistage vehicles. Staging is beneficial only at very low useful load fractions, where the structural weight is so heavy

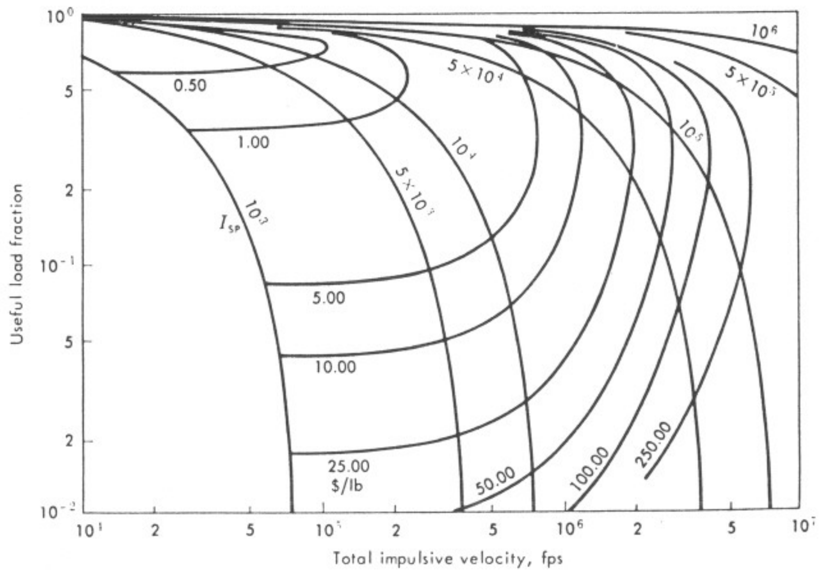

Fig. 12-26 Useful-load fraction.

with respect to the payload carried that it must be discarded in order to achieve the required performance. At high load fractions, the structural advantage of staging is greatly reduced and, in addition, is frequently overcome by the requirement for carrying extra engines in the multistage vehicle. Hence, for optimum economy of operation, as well as ease of recovery, the single-stage vehicle is best. This conclusion obviously depends upon the assumed relative price of hydrogen and uranium. If extremely low-cost hydrogen were available, multistage vehicles might be desirable. The advent of a low-cost nuclear fuel, on the other hand, would indicate single-stage vehicles with even higher load fractions.

Although giving an indication of the general class of vehicle desired and a feeling for the rather high speeds that can be achieved economically, Fig. 12-26 does not in itself answer the question of the maximum performance desirable in the solar system. In order to evaluate this requirement, calculations were made of passenger cost vs. trip time for different vehicle specific impulses for trips to various planets. The results for Mars are shown in Fig. 12-27 and for Pluto, the farthest known point in the solar

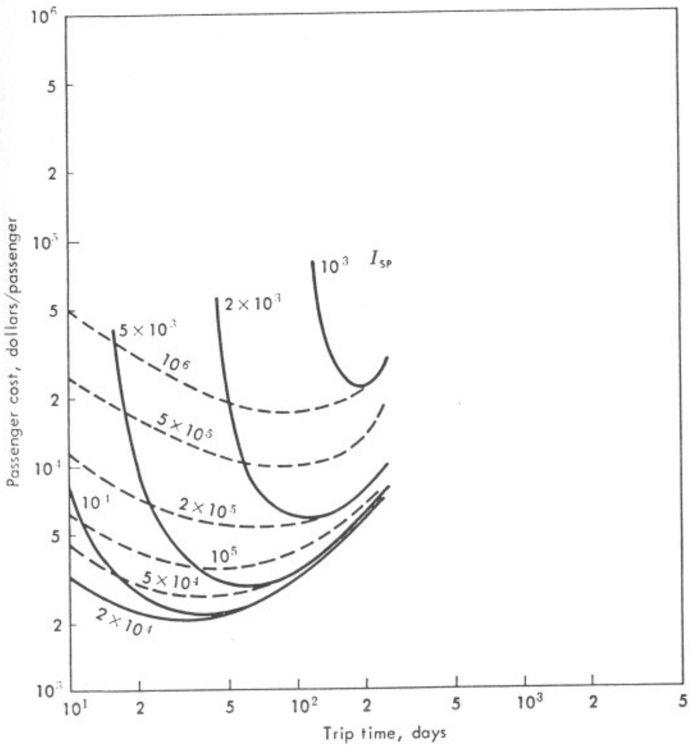

Fig. 12-27 Passenger transport cost, one way to Mars.

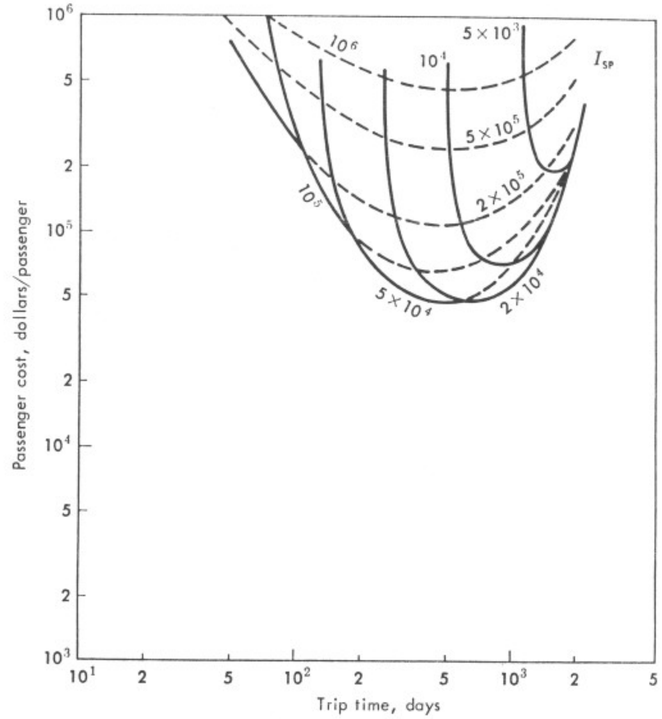

Fig. 12-28 Passenger transport cost, one way to Pluto.

system, in Fig. 12-28. Passenger costs were used, rather than the cost of cargo delivery by unmanned ships, in order to assess the value of decreased travel time. It should be noted that, even for relatively fast transit as far as the planet Pluto (less than 2 years), a specific impulse of much beyond 20,000 sec is not desirable.

For manned exploration, interest centers in relatively large payloads, perhaps on the order of 50 tons. Although this is a small payload as far as sea transports are concerned, it is nonetheless representative of a fully loaded C-133 airplane. In terms of the feats of air-transport logistics to date, it is perfectly clear that the ability to place 50 tons at predetermined locations in the solar system on a reliable and convenient schedule would constitute a very commendable early space-logistics capability. The velocity requirements for various missions of interest are shown in Fig. 12-29, along with typical payload vs. velocity curves for chemical rockets and the various classes of nuclear rockets discussed. It is clear that versatility demands the payload to be delivered over a very wide velocity region; tremendously large payload weights which are limited to low velocities should be of interest only to civil engineers. Many missions of extreme interest in planetary exploration occur in the velocity regions

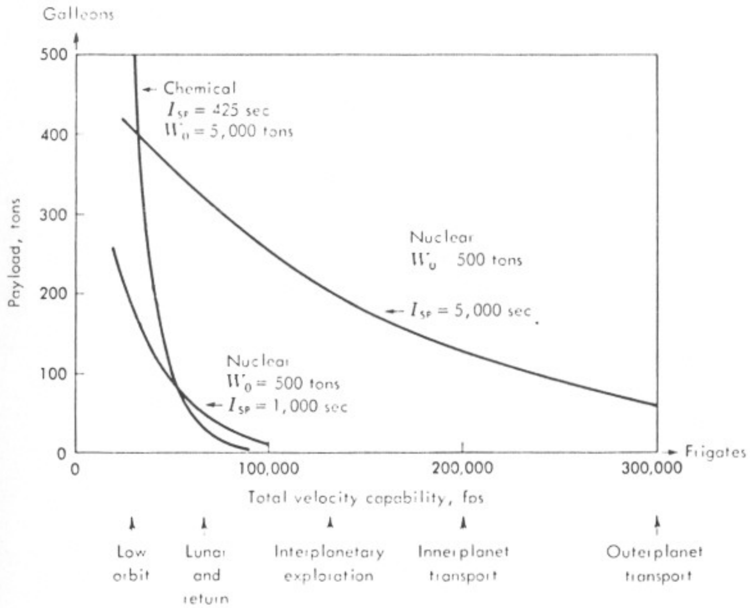

Fig. 12-29 Payload velocity capabilities. Chemical and nuclear vehicles.

of several hundred thousand feet per second. If a spaceship were available which could deliver a 50-ton payload to these speeds, it would have the desired tremendous utility and would be a very serious competitor indeed. It is interesting to note that single-stage gaseous-fission rockets of 5,000 sec specific impulse are capable of effectively penetrating these speed regions. (Even at only 2,500 sec, very interesting performance is possible.) Note in Fig. 12-29 that different gross weights were assumed for the different classes of propulsion. The velocity shown for inner planet transportation is that which gives one-way travel times of about 3 months to Mars or Venus even at the most adverse time of the synodic period. A propellant supply (but not fuel) is required on each terminal planet for such operations. It is clear that the missions which gaseous-fission rockets can perform span the whole solar arena at a quite reasonable operating cost, with travel times, at least to the inner planets, no worse than typical "windjammer" times of the last century, and with the convenience of year-round operation to all points. A possible analogy with the frigates and galleons of old comes to mind. If slow, clumsy ships were the way to conquer the sea, this chapter would be written in Spanish. Of all nations, the one most famous for clipper ships should be the first to grasp the many intrinsic advantages of short travel times in space. Historically, these advantages have invariably been overpowering.

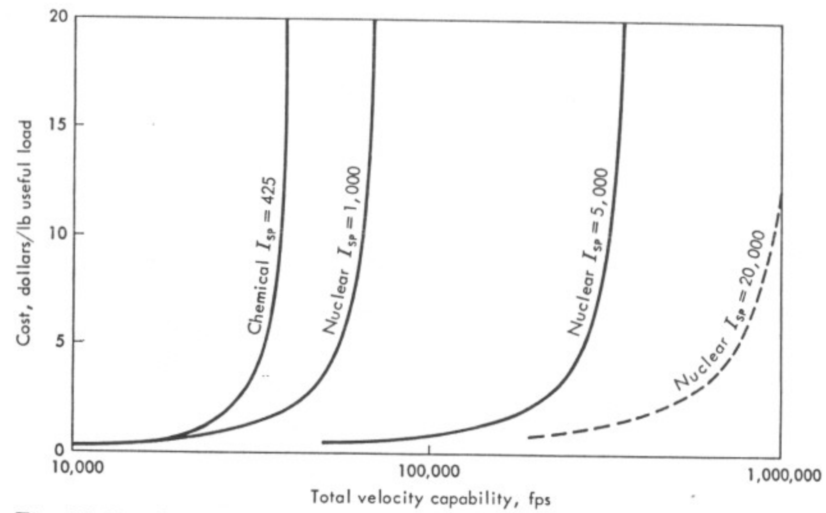

Fig. 12-30 Cargo transport cost. (Single-stage vehicles.)

Another way of illustrating the capability of such versatile vehicles is shown in Fig. 12-30. It can be seen clearly that the nature of the increase of rocket gross weight with velocity is such that, if one tries to attain excessive performance with a given propulsion system, the gross weight mounts extremely rapidly. It simply is not practical to build extremely high-performance vehicles with low-performance propulsion by continual pyramiding of rocket upon rocket. This, of course, is not a new conclusion. What is most interesting in Fig. 12-30 is the indication of the theoretical capability of nuclear rockets to achieve several hundred thousand feet per second velocity at a low operating cost. It is hard to see how to penetrate this region economically by any other means.

12-9 Safety

The operating-cost calculations in this discussion have always assumed highly re-usable single-stage vehicles. Consequently, the feasibility of launching such vehicles from the ground inevitably must be examined. Nuclear rockets do present a substantial safety problem in use. Actually, almost any high-energy device requires numerous safety procedures. If one had to choose between being hit on the head with a jet airplane loaded with several hundred thousand pounds of kerosene or a nuclear rocket loaded with several hundred thousand pounds of hydrogen, it is doubtful that one would feel a very great preference either way. The special concern with nuclear rockets, of course, is the special technological and psychological problems surrounding any use of nuclear energy.

While no attempt will be made to examine fully the safety aspects of nuclear rockets, several points will be mentioned.

It is by now apparent that nuclear-rocket engines are not bombs, that the possibilities of critical excursions are very low, and that the resulting release of fission products would be equivalent to that of only a fractional-yield bomb. Thus, even in the event of an accident on the launching pad, the resulting debris can be characterized more as an awkward cleanup problem than as a major catastrophe. The radiation flux surrounding even a large nuclear rocket on direct take-off from the ground is such that a person unprotected only 1 mile away would receive less than 1 rem dosage. Compare this 1-mile radiation-exclusion radius with the 15-mile noise-exclusion radius of a chemical NOVA-class vehicle. Thus, nuclear vehicles could take off from relatively normal launch bases and airfields and proceed throughout most of their burning phase under normal launch safety-control procedures. There is a time when the vehicle's velocity exceeds ICBM speed, but is still suborbital, so that a vehicle flight abort might cause impact anywhere on the planet. This particular critical failure time can be handled by auxiliary chemical-propulsion capabilities used either to decrease the velocity, so that the vehicle will fall within the range, or increase it to an orbital speed. If the trajectory is held unusually low—a desirable thing anyway from flight-abort considerations—the critical time is only a few seconds and only a few hundred feet per second retro-velocity must be supplied by the emergency chemical system.

Once the vehicle is well up into the atmosphere, negligible danger would exist if it were possible to deposit the fission products in the atmosphere. Hence, provision for burn-up of the fuel elements on atmospheric reentry would give us a basic solution. It is quite conceivable that this would work well, and auxiliary power units are now being designed for this method. Considerable effort might be required to ensure positive burn-up of a large solid-core nuclear rocket, however. In the case of gaseous-fission rockets, all active nuclear fuel is in the gaseous phase and is easily and safely injected into the atmosphere in vapor form in case of emergencies. In fact, it is probably impossible to prevent its injection in case of trouble.

Intuitive reasoning would say that the gaseous-fission rocket is more dangerous than the solid-core-fission rocket, since its containment would not be perfect. The opposite is probably true. As indicated above, it may be difficult to ensure emergency burn-up with solid-fuel elements. Furthermore, the fission-product build-up continues in solid cores as number of re-uses increases, and this inventory must be returned to low altitudes after each flight. Thus atmospheric disposal is complicated, and the vehicles require special handling upon return. Gaseous systems would be expected to dispose of their fission products in the high atmos-

phere after each flight so that the returning vehicles are as safe and as easy to approach and service as chemical vehicles. It therefore follows that each succeeding take-off would start with zero fission products aboard. The take-off could be made perfectly safe if auxiliary chemical propulsion were utilized.

Estimating the amount of such auxiliary propulsion requires a reasonably sophisticated analysis. The following discussion is based on Ref. 15. If we strive for a very high degree of safety, then an accident which causes the release of all fission products aboard must be analyzed by the usual atmospheric-dispersion techniques and the dose to an exposed person calculated at the worst location on the Earth's surface. In general, the worst flight time for such an accident will not be immediately at reactor ignition, since no fission product build-up has occurred. Nor will it be at the end of burning, since although fission-product inventory is presumably highest at that time, atmospheric dispersal is very effective. Hence, one must assume different start altitudes and then make calculations for accidents at various burning times until the worst case is located. The results of such a set of calculations (from Ref. 16) are shown in Fig. 12-31. It is noteworthy that, even for 3,000,000-lb-thrust rockets, only about 4,000 ft start altitude is required for complete safety.

The low safe altitudes indicated are startling. They show simply that atmospheric dispersion is very effective even at low altitudes and that only small amounts of fission products are ever aboard to be dispersed. Hence, one must remember, not only that propulsion reactors are not bombs, but also that gaseous-fission reactors are not as dangerous as normal power or propulsion reactors, since, in space operations, they will not build up fission products aboard ship. The philosophical and technical objection to flying reactors, which is basically because of the fission-product load aboard, is thus almost completely discounted. The few

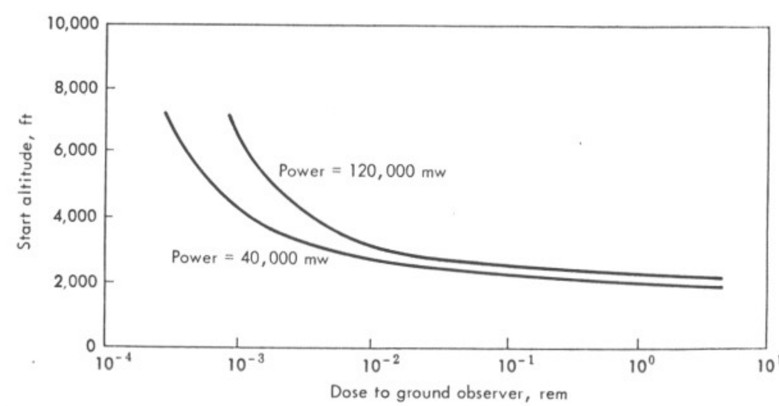

Fig. 12-31 Dose to ground observer vs. start altitude for gaseous core.

thousand feet safe altitude can be obtained with a chemical boost of the order of a few hundred feet per second, with a resulting total vehicle launch-weight increase of less than 10 per cent.

12-10 Concluding Remarks

A common complaint unites all people involved in space programs, be they technical, military, or political. The complaint is that programs are never sufficiently stable to permit the development of useful, reliable equipment in any reasonable time. It is usually felt, furthermore, that our competition is always doing a better job of such programming, and this is inevitably excused by reference to his nondemocratic decision processes. It is submitted that this excuse is badly overworked and that at best it is indeed a poor replacement for inspired advance planning. There are many missions to be performed in space. It is possible to foresee them, and it is possible to build one class of ship to do them. Unstable programs result from not facing the reality of the exotic missions as soon as they become feasible.

This chapter has tried to indicate that the potential of nuclear propulsion for space flight is so high that, properly utilized, it should enable transportation throughout the solar system at reasonable costs and on reasonable schedules. This possibility is obviously a key factor in the planning of future space flight. Even though many other difficult technical achievements will be important in the provision of space operational capabilities, it is inadequate propulsion which has limited the human race to one planet so far and it is propulsion which is at the heart of the question of the economic feasibility of extended space travel.

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