Z E N I

By: Maxwell W. Hunter, II
Member, Professional Staff
National Aeronautics and
Space Council
Washington, D.C.

Presented to:

The Executive Secretary
National Aeronautics and
Space Council
March 1964

ZENI

Maxwell W. Hunter, II
Member, Professional Staff
National Aeronautics and Space Council
Washington, D.C.

INTRODUCTION

This report examines the question of manned transportation throughout the solar system. Reference 1 examined the question of unmanned exploration of the entire solar system. This report is a complementary document to some extent. At first thought, such space exploits seem like rather far-out objectives, but it can be shown that they may not be as remote in time as most imagine. Furthermore, the techniques required to achieve operation throughout the solar system for reasonable expenditures are the same as those required to reduce drastically the operational costs of space transportation in the near space regions. The attainment of either objective would greatly alter the currently held concept of space exploration.

The key to drastic cost reductions is the utilization of reusable transport ships, just as in other transportation systems. This requires high performance engines even when the mission velocity requirements are low. The achievement of reusable ships with both sufficient redundancy to obtain reliability comparable to transport aircraft and the ability to abort during emergencies from all flight conditions, requires additional weight compared to current rocket design practice. This weight is not compatible with the achievement of high performance with chemical propellants. Indeed, solid core nuclear rockets are not really adequate to the task either.

Gaseous fission engines are one way around the limitations of solid core nuclear rockets, and are, in my opinion, the most attractive current solution. Orion has some possibilities. The extremely low accelerations of nuclear electric rockets, however, are an intolerable penalty for any reasonable transportation system. They will only be of interest if all high/thrust systems fail. I expect this point to become increasingly obvious as the impact of the potential existence of gaseous fission engines finally dawns on the ever-reluctant American space transportation "specialists."

This report, then, will cover primarily the current state of gaseous fission nuclear rocket development, and will attempt to show the

This shows that high ISP is required
Vortex method

calculated by standard transport airplane procedures. The results were spectacular, as shown in Fig. 1. Nuclear powered spaceships with perfect gaseous fission engines would cover the entire solar system with reasonable costs if they could be operated with transport aircraft procedures.

This report will examine, again, the entire solar system as a transportation problem. The previous work was apparently not very convincing, presumably because the engines seemed so remote in time. In addition, the implications of transportation-type ships simply were not grasped. Since then, much work has been done on both engines and ships, and more justifiable assumptions are possible. It is instructive to compare the assumptions of four years ago with current restrictions as we go. It turns out that today's possibilities come far closer to achieving ideal rockets than the authors of Reference 2 ever dreamed could occur within four years.

This shows that as time goes on totally new concepts will evolve making many topics in this report out of date

Types of Gaseous Fission Rockets

Most of the earlier ideas for 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 hopefully retain virtually all of the fuel on board while exiting all of the propellant. Hydrogen was normally assumed as propellant since low temperatures are always reassuring, 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. Weight of magnetic equipment was, as always, a problem, and the details of vortex stability and containment with any substantial diffusion rate have remained vexing.

not good because of low ISP

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 fission plasma, rather than direct intermixing. The containment problem, therefore, is not one of separation but rather one of the prevention of mixing; a fundamentally different problem. Vortex stabilization problems will certainly differ when propellant is not diffusing through the core. They will also be strongly altered if a high density propellant such as water is used rather than hydrogen.

A co-axial 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.[3] Although philosophically simple, this system cannot be expected to yield very good containment. The scheme has even been suggested, in the "glow-plug" reactor, that the fission plasma be contained in a quartz

Figure 4 shows a current estimate of the variation of the thrust/weight ratio with i_{sp} achievable for a gaseous nuclear rocket system with radiator using[7] as a basis an assumed thrust/weight ratio of 20, at an i_{sp} of 2500 seconds. The values fall off substantially at high specific impulses compared to the assumptions of four years ago, but are still greater than one to beyond 10,000 seconds specific impulse.

Mixed H_2O or H_2 give same ISP at 10,000 sec

(2) It should be pointed out that the use of water, ammonia, methane, or other non-hydrogen working fluids should be seriously considered in gaseous fission engines from the start. Not only are better ship designs permissible due to small tankage sizes and ease of propellant storability, but the use of a higher density propellant might well ease the fuel containment problem if a vortex system is used. If so, it could result in smaller, lighter engines. It might also result in earlier development programs if the ability to prove adequate containment occurred at an earlier time.

An interesting result in Fig. 4 is that the value of thrust/weight ratio as the specific impulse approaches 10,000 seconds is independent of propellant used. This is because the reduced propellant flow at such a high specific impulse results in such small thermal capacity in the incoming fuel that the engine must be almost completely cooled by the radiator system. The propellant to fuel burned ratio required for a given specific impulse is independent of propellant used, as noted previously. Hence, the engine uses the same amount of energy to generate a given specific impulse, the same fraction of energy must be rejected by the radiator, and the radiator area is unaffected by the type of propellant used.

making possible to use this radiation to heat more gas for a second jet exhaust (2)

(2) It seems clear that an engine design cooled by radiator alone should be investigated. Such an engine might be easier to develop since a major interaction between propellant and cooling system would be severed. Furthermore, such an engine could more easily use a variety of propellants. This could be very helpful in early planetary exploring.

this would give higher thrust/weight at high ISP of main exhaust jet (Fig 4)

A limitation on specific impulse of 10,000 seconds has been shown tentatively in Fig. 4. This is due to an unfortunate tendency of the propellants examined to date. Although adequately opaque to absorb the radiant energy at medium-high temperatures, they apparently become transparent at very high temperatures. This can be seen by noting the decrease in opacity at high temperatures in Fig. 6 of Reference 8. At the moment seeding the flow, which is very effective at low temperatures, does not look promising at high temperatures.

One other point of interest in connection with the thrust/weight ratios of gaseous fission engines is the power conversion weight thus achieved. Electrical propulsion enthusiasts feel extremely optimistic when power conversion weights of the order of 10 lb. per kilowatt are mentioned. A

A gaseous fission engine is basically a fluid flow development problem, and configuration changes between tests should proceed almost as easily as on chemical engines. Furthermore, once the fission product load has been ejected from the engine at the end of the run, the engine itself will be radioactive only to the extent to which it has been activated during the run. Compared to having fission products a permanent feature of the structure, this would greatly reduce the handling cycle time. It cannot be too strongly emphasized that with a cycle time between test runs of the gaseous fission rocket much shorter than solid core rockets, development of the former, once started, can proceed at a much more rapid pace.

Pulse Rockets, Electrical Rockets, and Fusion

Pulse rockets, both internal and external such as Orion, are frequently advocated as future space propulsion engines. They give promise of both high specific impulse and relatively high thrust/weight ratio. There are, however, several reasons for not considering them further in this report. With the nuclear test ban in effect, the detonation of such bombs behind a ship would almost certainly be considered a violation. Even if this were not a problem, the basic fuel cost of any bomb propellant system would seem to be much higher than that of a well-contained gaseous fission system. Each detonation, of which thousands are required, requires the assembly of a critical mass, all of which is lost. It is doubtful if pulse systems could ever compare economically with gaseous fission unless almost pure fusion bombs were used.

Nuclear electrical rockets also suffer economically, but for a different reason. The fuel consumption of such rockets is given by the perfect containment curve of Fig. 3. Because of the very low acceleration of such rockets, however, the actual total impulsive velocity required for comparable missions is approximately three to four times that required for a high/thrust system. We must on the average, therefore, utilize a specific impulse three times higher with an electrical system, and this forces the cost of fuel much higher as shown in Fig. 3.

Fusion rockets are another matter. The possible fuel cost of two different types of fusion rockets are shown in Fig. 5, in addition to the gaseous fission fuel cost curves. One suggestion for a fusion rocket, is based on operation with Helium-3 as fuel.[9] Helium-3, in general, is as expensive as fission fuel since it, too, must be manufactured. Thus, the upper fusion cost curve applies, and although this is obviously better than fission, it is not startingly so. If, as estimated in Reference 9, the resulting thrust/weight ratio of the system is only one-hundredth, then the specific impulse would have to be about twice that for a comparable fission rocket, and virtually all the gains due to utilizing fusion would be wiped out.

The procedure used in this report was to extrapolate properly calculated data out to high velocities by simply fairing the curves into the straight line assumption. Since we are interested very much in the region of a few hundred thousand feet per second, we cannot rely exclusively on the high speed assumption. In fact, we must include planetary gravity effects at low velocities to insure accuracy in that region.

Minor Planet Gravity Fields

The velocities needed to escape from the minor planets are all so small that even at only 100,000 fps ship velocity, they can be ignored. Figure 6 indicates the magnitude of velocity loss under the ideal assumption of all velocity input at the planet surface. This is, of course, not possible due both to the existence of planetary atmospheres and to the fact that as the ship accelerates, it gets farther from the planet. These effects are negligible, however, at the velocities of interest.

Major Planet Gravity Fields and Launch from Satellites

Due to their hostile environments, we would expect to operate from the natural satellites of the major planets rather than from their surfaces. A gravity gain can be made in this case. The ship would be launched from the satellite on a trajectory with a peri-apsis close to the primary planet and then make its major velocity input close to the primary. By this maneuver, it is actually possible to tap energy from the planetary gravity field. The planet accelerates the ship as it approaches peri-apsis, but the subsequent high speed escape attenuates the gravity deceleration during departure.

The maximum gain which one can make this way is given by the escape velocity at the planetary surface, minus whatever velocity is required to escape from the base satellite on an orbit with peri-apsis close to the primary planet. The maximum values for the major planets and for Earth are shown in Table I.

TABLE I

Planet/Satellite	Maximum Velocity Gain (fps)
Earth/Moon	28,400
Jupiter/Io	170,000
Saturn/Titan	103,000
Uranus/Miranda	63,000
Neptune/Triton	69,000

In the case of minimum travel times, the trajectories do not deviate much from straight lines. They do travel longer distances than the minimum distance between the planets, but they also make effective use of planetary orbital velocities. Apparently these effects compensate highly.

The greatest deviations from straight lines occur for trajectories during adverse times of the synodic period when it is necessary to reach around to the other side of the Sun from the launching planet. The actual trajectory is bent by the Sun and thus travels a longer path to the other side. It is also accelerated by the Sun so that the average velocity is higher. Thus, compensating effects are present, and they also apparently compensate far more than one would expect.

This can be done this summer

Since there is little doubt that the high speed assumption will become more accurate as the velocity is increased, Fig. 8 indicates that we may use it with reasonable confidence as a method of calculation over the entire region of interest for spaceship design. It is so used for the remainder of this report. It will be interesting to see how well properly calculated data up to 1,000,000 fps match the curve of Fig. 8.

Atmospheric Braking

In addition to the previously discussed assumptions, an allowance for the effects of atmospheric braking is desirable. Based on various aerodynamic braking studies, it was assumed that 1.5 times the planetary escape velocity could be braked aerodynamically if a planetary landing were being made, but that only 0.5 times escape velocity could be successfully braked in an atmospheric pass as part of a satellite landing operation.

6

This mostly amounts to the unwarranted assumption that the results that have been obtained at Earth may be applied to other planets in proportion to the planetary escape velocity. It is at least plausible that larger planets with larger atmospheres could brake more velocity than Earth. As in the use of planetary gravity fields, the atmospheric effects are substantial at low velocities, but are a minor correction at high speeds, when most of the braking must be with rockets.

Minimum Travel Times

The minimum travel times between Earth and the minor planets and satellites of the major planets are shown in Fig. 9. The total velocity to both launch and brake at arrival is shown. It is clear that velocities of the order of 500,000 fps are sufficient to make accessible even the remote solar system within reasonable travel times.

Except for Pluto, refueling bases at the major planets are much more needed than at the minor ones, as can be seen by Fig. 10. Refueling bases could be expected to be located on the surfaces of all the minor planets, although it may require some design effort in the case of Venus and Pluto.

The major planets are a different situation. Their surfaces are extremely forbidding as far as we know, to the extent that we are not even sure they have solid surfaces. It makes sense in that case to establish bases on one of the satellites of each of the four major planets. If we do decide to penetrate to the surface of these planets, then the velocity requirements for doing this when operating from one of the satellites is a reasonable number. Thus, bases on the larger planets' satellites not only greatly facilitate the convenience of transportation, but also present a reasonable base for surface exploration, if required.

Our own Moon may, in fact, represent a useful auxiliary base in this respect, although the velocity gain compared to Earth launch is only on the order of 28,000 fps.

So far, only travel between Earth and the other planets has been discussed. There is also the question of travel between planets other than Earth. The use of bases in other parts of the solar system to aid in the exploration of the even more remote portions should be considered. In fact, such considerations might well dictate the strategic location of bases.

At first thought it would seem to be a good idea, for instance, to use a base on one of the farther planets, say Saturn, to permit further exploration of the more remote planets like Pluto. Although this is an intriguing thought, such deep bases will have only limited utility. The reason is the extremely long synodic periods which exist among the outer planets since they move so slowly around the Sun. Figure 11 is a plot of synodic periods between all the planets in the solar system. In the worst case of all, the synodic period between Neptune and Pluto is slightly over 500 years. In addition to the long synodic period, the difference between travel at the optimum time of the year and the worst time of the year becomes more extreme the farther the planet is located from the Sun. This is quite easy to visualize when thinking of straight-line trajectories.

One way of illustrating this is shown in Fig. 12 where the effects of basing on selected planets is shown for a constant ship velocity. It is true that a deep space base will be closer than Earth to the other deep space objects when in favorable position, but equally true that it will be much farther away during the worse conditions. Surprisingly enough, the base wants to be reasonably close to the Sun, once again emphasizing that the Sun is the center of the solar system. Although Mercury might be the best

This approach was actually used in the 1960 study, and Fig. 1 represents one of the results. Figure 1, however, contains assumptions as to degree of reuse achieved by the vehicle which, although consistent with transport aircraft practice, may not apply to space transportation. At least if they do, their application must be better documented. We will, therefore, break down some of the component assumptions in Fig. 1, and try to reconstruct a more convincing story.

$\lambda' = \frac{\text{fule weight}}{\text{engine weight}}$

Weight Assumptions

The usual way of estimating weight involves estimating the parameter λ' which is defined as the ratio of fuel weight to total propulsion system weight. λ' can be estimated from a knowledge of tank weights, engine thrust/weight ratios, and a suitable allowance for miscellaneous items. This is a quite acceptable parameter where the rocket is mostly fuel, but does not adequately estimate the weights for vehicles of large payload and low fuel weight. This report makes use of an additional parameter, tentatively called λ'', which bears the same relation to payload weight that λ' does to fuel weight. It is estimated from engine thrust/weight ratios and the structural and miscellaneous weight necessary to house the payload.

Reference 2 used a similar approach, although λ'' was not specifically labeled as such. The basic parameters assumed were an engine thrust/weight ratio of 30, an initial vehicle thrust/weight ratio of 1.25, a weight of basic structure equal to 5 percent of the total propulsion system weight, and an unknown payload structural allowance to be subtracted later from the useful load. Thus, λ' was 0.908 and λ'' was 0.958. These weights are probably achievable. We have, however, become extremely conservative in our structures in recent years.

In order to provide relatively non-controversial structural estimates, the following rationale will be used. The engine thrust/weight ratio is assumed at 20, with initial vehicle thrust/weight ratio of 1.25. λ' is assumed to be 0.85 which means an 8.75 percent allowance for structure and miscellaneous. λ'' is also assumed to be 0.85, which means an allowance for payload cargo compartment and related structural items similar to that for fuel tankage.

λ' and λ'' can be converted to an effective λ' as a function of payload/initial weight ratio. This parameter is shown in Fig. 13 for the assumptions of Reference 2 and for the assumptions of this report. The Figure shows clearly the importance of including the λ'' term in the high payload ratio region appropriate to single-stage vehicles.

Also shown in Fig. 13 is an example of a design study for a completely reusable transport rocket.[12] In this case, reasonably careful design of a solid core engine. Gaseous engines with better containment would be much better. It is also evident that gaseous engines with space radiators, but with specific impulse limited to 10,000 seconds, can drive ships up to about one-half million feet per second and still maintain reasonable fuel cost. The attainment of a fuel separation ratio of 10^{-4} is almost as effective as perfect fuel containment.

The optimum fuel cost curves for gaseous fission engines with radiators were obtained by determining the optimum specific impulse for each velocity and separation ratio. This is necessary since too low a specific impulse will result in excessive propellant cost while too high a specific impulse will result in excessive fuel cost.[2] The optimum specific impulse is much higher than 10,000 seconds for all velocities beyond a few hundred thousand feet per second. Hence, these curves represent a future capability presently unattainable due to the propellant transparency problem at high temperatures previously mentioned. If it were not for this, gaseous fission ships could be driven to almost one million feet per second before fuel costs became a limitation.

use extra propellant
a to hold radiation TD
keep temp OK.
This will provide
extra thrust

Structural Cost Amortization

The 1960 study assumed a large number of reuses per vehicle, somewhat analogous to the number of times a transport airplane is reused. A transport aircraft is actually utilized about 50 percent of the time, and average flight durations are less than 4 hours. It is clear, therefore, that such vehicles are used over one thousand times per year. However, space travel durations are much longer, and it is obvious that the interaction between travel duration and number of reuses must be considered.

For the lunar mission, it is clear that large numbers of reuses are feasible. Typically, one hundred flights per year (50 each way) can be envisioned on the basis of two-day travel times, one day turn around time at each terminal, with Sundays and two weeks off for vacation. Over a ten-year ship lifetime, one thousand uses will be achieved.

One can get a feeling for the number of interplanetary uses by assuming a certain ship total life. Typically, transport aircraft are designed for forty thousand hours (4.6 years) total life. On the basis of slightly less than 50 percent utilization, such a vehicle would last for 10 years. They always last much longer, but the amortization time of the airframe is usually about forty thousand hours, since new equipment always becomes available in even shorter time.

Selecting a suitable lifetime for a spaceship presents a considerable technical dilemma. One viewpoint would simply take 10 years as above.

impulse limited to 10,000 seconds, the solar system as far as Jupiter is available with travel times not exceeding 4 months. Inner solar system travel times need not exceed 2 months. The advantage of optimum specific impulse becomes more evident at Saturn and beyond.

It must be realized that Figs. 17 and 18 represent a certain set of assumptions, and both better and worse situations may well occur. Even the case of specific impulse limited to 10,000 seconds requires gaseous fission engines with radiators, and most people today would rather agree to engines without radiators. In that case, the velocity increment achieved will be only about 25 percent of the curves shown. Furthermore, the economic penalty of, if necessary, ejecting a critical mass of fuel in the process of shutting down the engine has not been included. This will be on the order of $100,000 per shutdown.

On the other hand perfect containment might be achieved. We might design ships for each velocity increment, rather than use the single design assumed here. Furthermore, one can get a greater utilization of vehicles by the expedient of refueling the vehicles which go on deep space missions. This is preferable to multi-stage vehicles, since a fleet of ships used for refueling can also be used for other missions. No attempt will be made here to present detailed curves showing the effects of refueling. Cursory checks show that over 200,000 fps can be added for reasonable cost with only two refuelings.

The greatest conservatism of all in Figs. 17 and 18 is, of course, in the magnitude of the ordinate scale. Costs beyond $12 per pound have not been plotted so that the entire set of curves is about 100 to 1000 times lower than virtually all space cost analyses to date. This must be clearly remembered as we discuss the performance of early versions of these ships.

#18 This shows that better to use transportation on 300 day missions etc cargo & crew

Transportation Development Philosophy

The use of "transportation" type vehicles has a large number of implications for spaceship development. In the first place, most of our rocket designers today are experienced only in "ammunition" thinking where recovery and reuse are indeed a rare phenomena. "Transportation" systems not only achieve very high reuse, they contain sufficient redundancy to permit flying with partial equipment failures, and also have the ability to abort successfully from any flight condition. It is this last capability which is very important to the development program of such ships.

The savings in development cost of not losing ships continually, is obvious, yet, our ammunition thinkers are so used to the massive throw-away that they usually claim that recoverable equipment would be more state of the ship after main engine operation. In these items, gaseous fission engines are much more useful than solid core nuclear propulsion devices. For instance, Fig. 19 is a plot of the safe ignition altitude for a gaseous nuclear engine.[14] It indicates that only 5000 feet is sufficient to meet very stringent safety numbers even in a catastrophy. Furthermore, the ship never carries a load of fission products with it. If it aborts from above 5000 feet, it is not radioactive at the ground. A reusable gaseous fission ship could be flight-tested safely without most of the elaborate test procedures of chemical rockets. The fact that its basic performance is far higher has nothing to do with its basic safety.

In fact, a gaseous fission ship using water as a propellant is practically unique in transportation development history. I know of no other device which is quite such a flying fire extinguisher. In case of a real crash, this ship deluges the crash area with the order of 100,000 gallons of water. Perhaps we should investigate using fire extinguisher fluids as propellants. We have not yet begun to think!

A Possible Development Analogy

A common mistake in development thinking seems to be a tendency to relate the basic performance achieved by a device with its development difficulty. It seems so logical to assume orderly progress in development programs. Actually many major programs are not a result of orderly progress. One of the most recent interesting examples is the development of the ICBM. These ballistic missiles penetrate to their targets at a Mach number of 25. Orderly progress would have dictated that we build first fleets of supersonic bombers, then fleets of hypersonic bombers. Only after that would we consider whether or not Mach number 25 penetrators were desirable.

The fact is that Mach number 25 ballistic missiles are considerably easier to build than hypersonic bombers (and evidently Mach number 3 bombers). Their performance is attained in a different manner, with different engines (not breathing air) and in a different flight region (out of the atmosphere). They are not a Mach number 25 airplane.

The gaseous fission spaceship has many analogous elements. It is easy to achieve 500,000 fps out in space, as long as the engine is capable of it. A ship which never carries fission products aboard need never fight the safety problems of solid core nuclear rockets, or even the analogous problems of nuclear airplanes. An abortable transport rocket is a different development job than the building of larger ammunition. We must look at the gaseous fission ship in terms of its difficulty of development, not in awe of its possible accomplishments.

Radiators

A similar design interaction is evident in the use of radiators required for the propulsion system. Since it is desirable that the ship be built of high temperature materials to permit reentry, it is quite possible that the same surface area can be used to radiate energy. The energy would come from the outside during reentry, and from the inside during engine operation. Thus, the weight penalty involved in the design of a reusable ship with radiators is likely to be very much less than the sum of the separate weight penalties of the reusable structure and the power plant radiators.

use radiators as source of new heat for new propellant

If the radiator temperature is of the order of 5000°F. then about one-half megawatt of energy is radiated per square foot of surface area. In the example shown, about 4000 square feet of radiator area is available by utilizing half the normal surface area of the ship. This would permit achieving about 30 percent of the thrust/weight ratio shown in Fig. 4 at 10,000 seconds specific impulse without additional radiator area.

The large effect of this interaction is further emphasized on the curves of Fig. 4. It is evident that if the radiator temperature is 5000°F., and the unit weight of the radiator is 1 lb. per square foot, then the thrust/weight penalty of adding radiators is not very great. If, however, the radiator area is small enough that the normal structure of the ship becomes the radiating surface; then the radiator weight penalty is reduced virtually to zero. Thus, for reusable ships, the highest curve of Fig. 4 may be the appropriate engine curve with high temperature radiators.

Auxiliary Propulsion

10

do not let the ships land on planet

The auxiliary propulsion of such a ship also presents interesting design interactions. If launch safety requires not igniting the main engine until 5000 feet altitude is achieved, and this altitude is obtained by chemical rocket boost, then about 7 percent additional gross weight of the ship will be required. This represents a very reasonable solution, and should result in a very small empty weight penalty. Therefore, it would not decrease the maximum speed of the rocket very much.

A number of additional requirements exist if the auxiliary propulsion system is to aid in a really convenient operation. The first ships which explore the solar system will operate, for the most part, from totally unprepared facilities. Only on Earth will they operate from a spaceport or airport. Therefore, it is necessary that the ship be able to take off without requiring a launch complex, and without any take-off roll. It is also desirable for ease of cargo handling that the cargo doors be close to the ground when the ship is at rest.

One configuration which combines all of these possibilities is shown in Fig. 21. In this case, retractable auxiliary chemical engines are mounted compartment is made into an escape capsule. It would also seem that the pilot's compartment might as well be made into a swing nose configuration for ease of access to the cargo. I see no reason to wait until the third or fourth generation of space transports to make use of techniques which have been found to be convenient in terrestrial transports, particularly considering the lack of equipment at the non-terrestrial terminals.

The selection of the auxiliary chemical engines is also of considerable interest. It would seem, off hand, that high energy chemicals should be used because of over-all efficiency, but ease of space storability would be pertinent. For real convenience of operation, a fuel which reacts with water such as sodium, lithium, or phosphorous should be considered. Unfortunately, such propellants tend to have quite low specific impulses. An intriguing application of hybrid rocket technology would be the pumping of part of the main water supply through a lithium chamber for this application, if the performance penalty were not too severe.

Earth Launch Penalty

The performance curves so far have all stressed the use of rather high specific impulses. The weight estimates, however, assumed an engine thrust/weight ratio of 20. According to Fig. 4, an engine with thrust/weight ratio of 20 will have a thrust/weight ratio of only 5 at 10,000 seconds specific impulse, and it will be that high only if the radiator weight penalty is already included in the reusable structure and the radiator is large enough. With the 4000 square foot radiator operating at 5000°F. previously discussed, the spaceship of Fig. 20 will have an acceleration when fully loaded of only 0.10 g at 10,000 seconds specific impulse.

1/5

The assumption of engine thrust/weight ratio decreasing with increasing specific impulse is a major change from the constant value of the 1960 study. Once orbital velocity is attained, 0.10 g is reasonably adequate to achieve the various gravity attenuation and gains previously discussed.[17] Since about 0.20 g will be required for normal take-off from the large natural satellites of the solar system, only a small reduction in specific impulse at take-off will be required. Take-off from Mars and Mercury will require reducing the specific impulse to about 5000 seconds for the first 10,000 fps velocity, still a rather small penalty. Thus, the only real penalty associated with the low thrust/weight ratio is the necessity to climb out of the Earth and Venus gravity fields with a specific impulse low enough to achieve initial thrust/weight ratio of 1.25.

orbiting ships will eliminate this shortcoming

For fully loaded Earth launch, climb-out requires a specific impulse of 2500 seconds. At lower launch weights, the specific impulse can be increased. The effect of obtaining the first 20,000 fps at appropriate specific impulse is shown in Fig. 22. This, then, is the added propellant penalty that the ship would suffer when operating from the surface of Earth. The penalty would be somewhat smaller from Venus.

Solar System Safety

Gaseous fission spaceships have a major interaction with the entire solar system. Since a certain amount of fission products will be ejected in the exhaust, the disposal of these products is an important problem. During each launch of a 500 ton spaceship, the fission products generated within the engine at altitudes less than 200,000 feet are about equivalent to those of a 3 kiloton bomb.[18] Fission products equivalent to a 7.5 kiloton bomb will be generated by the time orbital velocity is reached. Considering the fact that about 90,000 kiloton equivalents had been exploded in the atmosphere up to 1960, it is clear that thousands of launches per year would be a small number in terms of bomb testing problems. The bomb tests prior to 1960 have been estimated to increase the natural background radiation at sea level by less than 5 percent over the subsequent 30 year period.[19]

Although spaceships present a very small problem in terms of general Earth atmospheric contamination to date, we are extremely sensitive on this subject. Furthermore, we do not wish to contaminate any planet's atmosphere, or radiation belts, or all of space. Hence, the previous discussion of chemical boost to orbit, and the continuing interest in the "glow plug" reactor.

At a specific impulse of 3000 seconds, the average exhaust velocity of the engine is approximately 100,000 fps. This is not only far higher than Earth escape speed of 37,000 fps, it is much higher than the 40,000 fps which must be added to the Earth's speed around the Sun to achieve solar system escape. When operating with a high specific impulse mode, the exhaust speed would be several hundred thousand feet per second. It is clear that for rockets of this sort, the fission products from the exhaust would be thrown completely from the solar system, as long as flight programs are arranged so that the exhaust jet is not directly pointed at a planet, or its radiation trapping belts. Hence, far from being a solar system hazard, such spaceships will dispose of their fission products automatically in a much more scientifically and humanely acceptable fashion than any system which is limited to disposal on this planet.

Emergency Rescue Operations

Any ship which has a heliocentric velocity greater than 138,000 fps at 1 A.U. will be operating at greater than solar system escape speed. The types of ships discussed here operate virtually throughout their whole lives at speeds beyond solar escape speed. Kinematically, they are not in the solar system.

Hence, in case of trouble, these ships will sink forever into deep space in much the same manner that an ocean vessel can be lost permanently by sinking into the ocean. If, when it is time to apply braking thrust, a ship should have an engine failure, it will be lost. Conversely, it is possible to rescue such ships or at least their crews by similar ships stationed at the terminal.

If the spaceship achieved $3.00 per pound operating cost, it would require $30 million per year for the operation. On the other hand, 300 Saturn V's per year would cost $30 billion. Sooner or later, we'll get this straight. Our current ammunition technology can only give us a minor piece of space. With true spaceships, we can open the whole solar system for massive use within current budgetary frameworks.

Table II lists the cargo delivered and cost per year to various locations in the solar system based on the previous discussion. It is clear that substantial tonnage of cargo can be delivered within current budget frameworks. An interesting checkpoint is the 50,000 tons/year which we currently ship to Antartica. The same amount per year can be delivered to the Moon by 10 gaseous fission powered spaceships.

TABLE II

CARGO DELIVERY CAPABILITY

Planet	Cargo (Tons/Year/Ship)	Cost (Million $/Year)	Specific Impulse (Sec)
Moon	5,000	15	2,500
Mercury	400	20	10,000
Venus	400	20	"
Mars	280	16	"
Io (Jupiter)	280	35	Optimum
Titan (Saturn)	150	20	"
Miranda (Uranus)	74	12	"
Triton (Neptune)	49	9	"
Pluto	35	7	"

Scientific Data

There are a number of interesting uses for such a large operation which have been well delinated in much science fiction writing, but are rarely discussed in scientific journals. One obvioususe is the pursuit of scientific knowledge itself. Considering the large terrestrial scientific installations which exist, and the amount of scientific personnel who continually make field trips to apply their knowledge to on-the-spot direction of operations, it seems rather ridiculous to assume that whole other planets can be satisfactorily explored by remote means. The Moon is almost as large as North and South American combined, and Mars is about the same size as the total that quality diamonds are worth approximately $2 million a pound, in case anyone is curious. We, however, will not mention diamonds further here, thereby ignoring the fact that some geologists expect to find them in the vicinity of lunar craters (formed by the very high temperatures and pressures of the meteoroid impacts).

For many other materials, it seems to me that the asteroid belt is of great interest. There is a pretty good chance that the asteroid belt is, in fact, a shattered planet. The iron meteorites presumably come from the core, and the stony meteorites from various parts of the mantle. Thus, looking at it as a natural resource, the asteroid belt is a pre-mined planet, although admittedly a small one. (The total material in the asteroid belt has been estimated to be only 3 percent of the Moon's mass.)

It is a great deal cheaper to bring materials back from the asteroid belt at the transportation prices in this report than it is to penetrate down to the Earth's core. In fact, it is quite easy to see technically how to bring cargo from the asteroids. At the moment, we do not see how to get to the center of the Earth. It is easily conceivable that some day specialized refining industries using processes based on the almost unlimited vacuum of space will spring up in the asteroid belt, so that partially or completely processed materials are shipped back, rather than simply raw materials.

One can argue, however, that we don't want to go to the Earth's core, or maybe we don't really need to go very deeply below its surface. We do, however, continually worry about the future depletion of the natural resources of this planet. Substantial segments of the Earth's surface have been damaged in mining operations, and the search for materials does go on even deeper into the Earth's surface, causing ever more damage. Presumably, at each step, some risk of damaging the Earth's ecological balance occurs.

I do not understand why those Terrestrials who are concerned with the preservation of our natural resources, whether it be for parks, or simply for preserving a habitable Earth, do not consider the space program as one of their big hopes for the future. Space is the one place where we can obtain resources with no damage whatsoever to the Earth's ecological balance or its natural beauty. The concept of developing space resources in the future is no more dreamy, nor less scientifically noble, than attempts to create power from fusion reactions, or to move into the sea for food supply purposes.

The world production of nickel in 1962 was 400,000 tons. An iron asteroid one mile in diameter containing 9 percent nickel would contain almost 2 billion tons of nickel, or enough to supply current needs for about impulse. Table II, however, assumed different engines for different applications with the advanced engines utilized for deeper space missions. The ship of Fig. 20, with a full payload, will have about 340,000 fps velocity capability with 10,000 seconds specific impulse. Without the use of radiators, but with methane propellant, such a ship would have about 60,000 fps capability, enough for the lunar transport described herein.

We should carefully examine the use of radiators right from the beginning. It must be realized, though, that the early ship, if it must fly without radiators, is still an excellent lunar transport and also makes a good early inner planet explorer. And it is a major step in the direction of developing a good spaceship. It is, in fact, a true spaceship -- it merely is not as good a one as can be built with later versions of the engines.

Another, even lower performance version of the ship of Fig. 20 could be of extreme interest. If the ship were powered with an engine without radiators, but operating on water, its specific impulse would only be about 1200 seconds. This means an impulsive velocity of only 38,600 fps. 30,000 fps, however, is enough to obtain orbital velocity, and this is a ship which can reenter at will. Hence, it can operate between any points on Earth at orbital speeds carrying 20 percent of launch weight as payload, and needing only water as a propellant.

Its payload fraction is higher than a subsonic jet transport. Its flight time to any point on Earth is at most 45 minutes. And, as shown on Fig. 3, its fuel cost with perfect containment is only 0.3¢ per pound of propellant. This is almost a factor of 10 lower than the price of kerosene. In fact, the fuel cost to deliver cargo to 2000 miles would be less than one cent per ton-mile and to deliver cargo half way around the world, it would be one-sixth cent per ton-mile. These are very impressive numbers.

agree with this

Perhaps this is the real use for the glow-plug gaseous fission engine, for this domesticated spaceship would have lower operating cost than any current subsonic jet or projected supersonic or hypersonic transport. As a commercial transport, or a cargo craft, it should be superb. And as a deployer of troops in emergency, it seems hard to beat.

A REUSABLE TEST VEHICLE

Omit

12

The problem of developing spaceships is looked upon by most people as mostly an engine development program. Basically, in my opinion, this is true. It is only true, however, if proper vehicle thinking has been underway in the meantime. It is a major thesis of this document that the latter is not true.

If we have an effective λ' of 0.94, then it is quite clear that high pressure, high energy technology would enable an orbital launch weight to payload ratio of about 14. This means that a 110,000 lb. launch vehicle (the weight of the Thor) could put 8000 lb. (the weight of a Gemini capsule) on orbit. The RL-20 is a 250,000 lb. thrust engine, so that it is quite clear that it, combined with a high λ' vehicle, could put the order of 14,000 lb. on orbit. This would be a very adequate two-manned installation for test vehicle purposes.

The question, then, is how can one get the effective λ' of 0.94 required? The immediate reaction is that this requires a two-stage vehicle. However, another technique exists which could be described as a 1 1/2 stage system which is the inverse of Atlas; namely, throwing away tanks rather than engines. In airplane terminology, this is the use of auxiliary fuel tanks. This concept has been examined (called Rhombus) for very large vehicles.[21] In the Rhombus case, a large vehicle which already had a high λ' was made better by the use of disposable tanks. This is not what is suggested here.

The disposable tank technique can be used to give a high effective λ' when one of the components is not particularly high to start with. We can thus examine the question of making a totally reusable vehicle such as the Astro concept of Reference 12 into an effectively high λ' vehicle.

Figure 26 shows effective λ' as a function of the velocity at which the auxiliary tanks are jettisoned for several values of basic vehicle velocity increment assuming that the fuel tank λ' is 0.95. This fuel tank value should be readily achievable even for hydrogen tanks. As indicated in Fig. 13, the vehicle of Reference 12, which was worked out in some design detail, showed a λ' of almost 0.85 in the 180,000 lb. size class. Since 0.85 is not enough to achieve orbital capability, particularly with normal chemical engines, the solution suggested in Reference 12 was the usual one of going to two-stage vehicles.

The curve of Fig. 26, however, makes it quite clear that the use of 0.95 λ' disposable tanks would give an effective λ' of about 0.94 with basic vehicle λ' of 0.85 if the tanks are jettisoned after attaining 42 percent of the impulsive velocity. The installation of a single RL-20 would be actually a lighter weight propulsion installation than the single J-2 and dual RL-10 installation used.[12]

Jettisoning the tank after 42 percent velocity still means that 67 percent of the fuel is carried in the auxiliary tanks. A more easily designed vehicle might be one with smaller auxiliary tanks. If 52 percent of the fuel is in auxiliary tanks, then the effective λ' is 0.92, and a 200,000 lb. vehicle could put a 10,000 lb. payload on orbit.

CONCLUSIONS

Gaseous fission engines give promise of specific impulses of 2500 seconds at high thrust-to-weight ratios without the use of space radiators. With space radiators, this value can be increased to 10,000 seconds.

The use of water as a propellant is feasible if radiators are used. It may be highly desirable, both from advantages in spaceship design, and from fuel containment advantages if a vortex system is adopted. Other propellants should also be investigated, as well as an engine cooled completely by space radiators and capable of using several different propellants.

Single-stage spaceships with these engines could generate a half-million feet per second velocity capability, with fuel costs less than $10.00 per pound of payload.

Such vehicles could be operated in a transportation network covering the whole solar system as far as Saturn with average travel times of eight months. The average travel time could be reduced to five months either by refueling techniques or even more advanced engines. Average travel times around the inner planets should not exceed two months.

Almost no celestial mechanics calculations have been performed at
13 the operational velocities of these ships. Such calculations are needed.

A single ship of one-half million pounds take-off weight could deliver 5000 tons of cargo per year to the Moon, the equivalent of 300 Saturn V launches. Ten such ships on a regular shuttle could deliver the same amount of supplies to the Moon as are delivered to the Antarctic each year in support of the U.S. explorations there.

Tremendous cost savings would be possible with such ships provided that the opportunity to develop transportation-type vehicles is pursued. With a high performance engine, the vehicle is no longer marginal and the weight necessary for component redundancy and abort capability, long common in Earth transport practice, can be tolerated. This, in turn, should greatly reduce development costs if pursued from the beginning rather than as an afterthought of ammunition development philosophy.

REFERENCES

1. Hunter, Maxwell W., II. Future Unmanned Exploration of the Solar System. December 1963. To be published in Astronautics & Aeronautics May 1964.

2. Hunter, M. W., Matheson, W. E., Trapp, R. F. Direct Operating Cost Analysis of a Class of Nuclear Spaceships. Douglas Aircraft Engineering Paper No. 1016. Presented at the 11th Congress, International Astronautical Federation, Stockholm, Sweden, August 15 - 20, 1960.

3. Weinstein, H. and Ragsdale, R. G. A Coaxial Flow Reactor - A Gaseous Nuclear-Rocket Concept. ARS Preprint 1518-60. Presented at 15th Annual American Rocket Society Meeting, Washington, D.C., December 5 - 8, 1960.

4. McKee, J. W. The Glow-Plug Gas-Core Reactor. Presented at Symposium on Gaseous Fission Reactors, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California, April 26 - 27, 1962.

5. Hunter, Maxwell W., II. Possible Nuclear Space Vehicles During this Century. (Applications and Materials Requirements from 1975 - 2000 A.D.). Presented at the National Topical Meeting on Nuclear Materials for Space Applications, American Nuclear Society, Cincinnati, Ohio. April 17 - 19, 1963.

6. Hunter, Maxwell W., II. The Potential for Nuclear Propulsion for Manned Spaceflight. Presented at National Meeting on Manned Space Flight, Institute of the Aerospace Sciences, St. Louis, Missouri, May 1962.

7. McLafferty, G. H. Investigation of Gaseous Nuclear Rocket Technology -- Unclassified Programs, B-910093-1. Quarterly Progress Report No. 1-U, September 16 through December 15, 1963, Contract No. NASw-847. United Aircraft Corporation Research Laboratories, East Hartford, Connecticut. December 27, 1963.

18. Hallet, R. W., Jr., McKee, J. W., Faller, R. J. Operational Use of Gas Core Nuclear Reactor Propulsion Systems in Large, Manned, High-thrust Spacecraft. Douglas Aircraft Engineering Paper No. 1200. Presented at 10th Annual Meeting of the German Rocket Society, Bremen, West Germany, September 23 - 25, 1961.

19. The Effects of Nuclear Weapons. April 1962. Prepared by the U.S. Department of Defense. Published by the U.S. Atomic Energy Commission.

20. Pratt & Whitney Aircraft. Space Transport Engines. PWA FR-794. West Palm Beach, Florida. September 1963.

21. Bono, Philip. The Rhombus Concept. Page 28, Astronautics & Aeronautics. January 1964.

GSA DC 64-14133

FIGURE 1

CARGO TRANSPORT COST

SINGLE STAGE VEHICLES

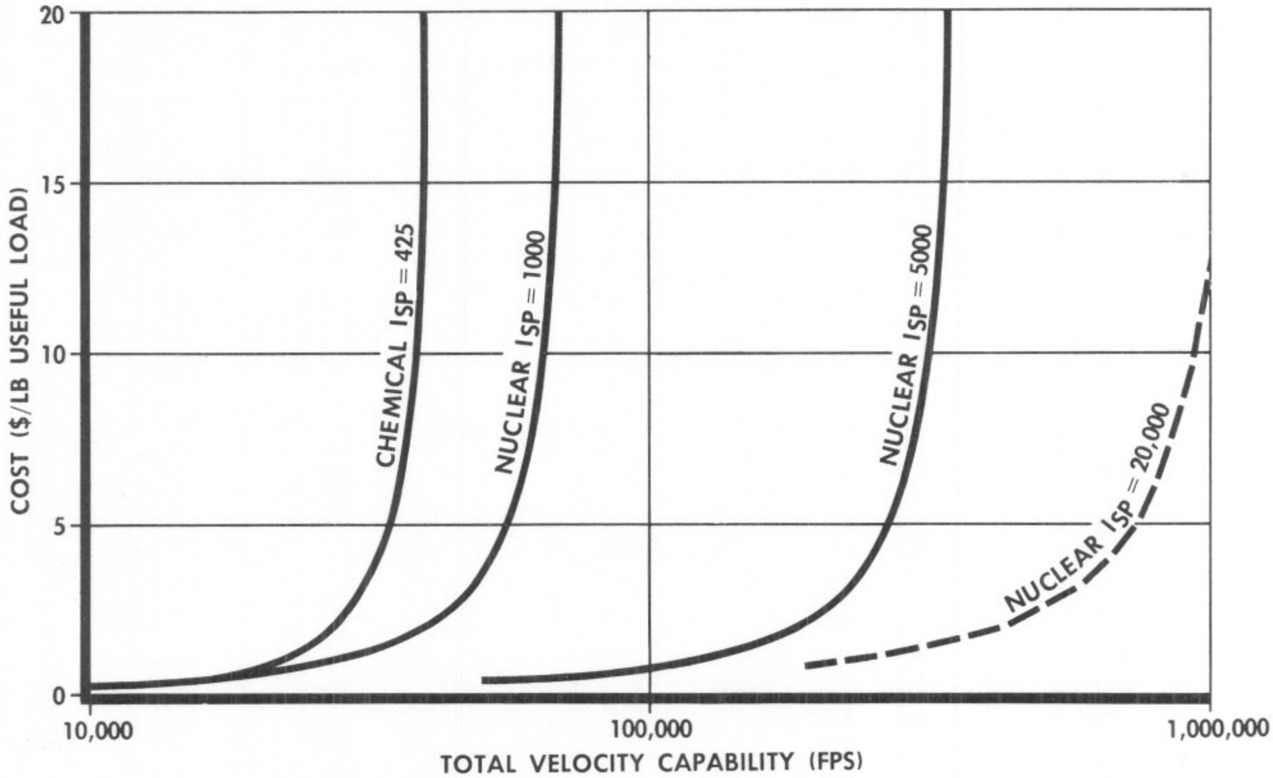


FIGURE 2

SPECIFIC IMPULSE FROM NUCLEAR REACTIONS

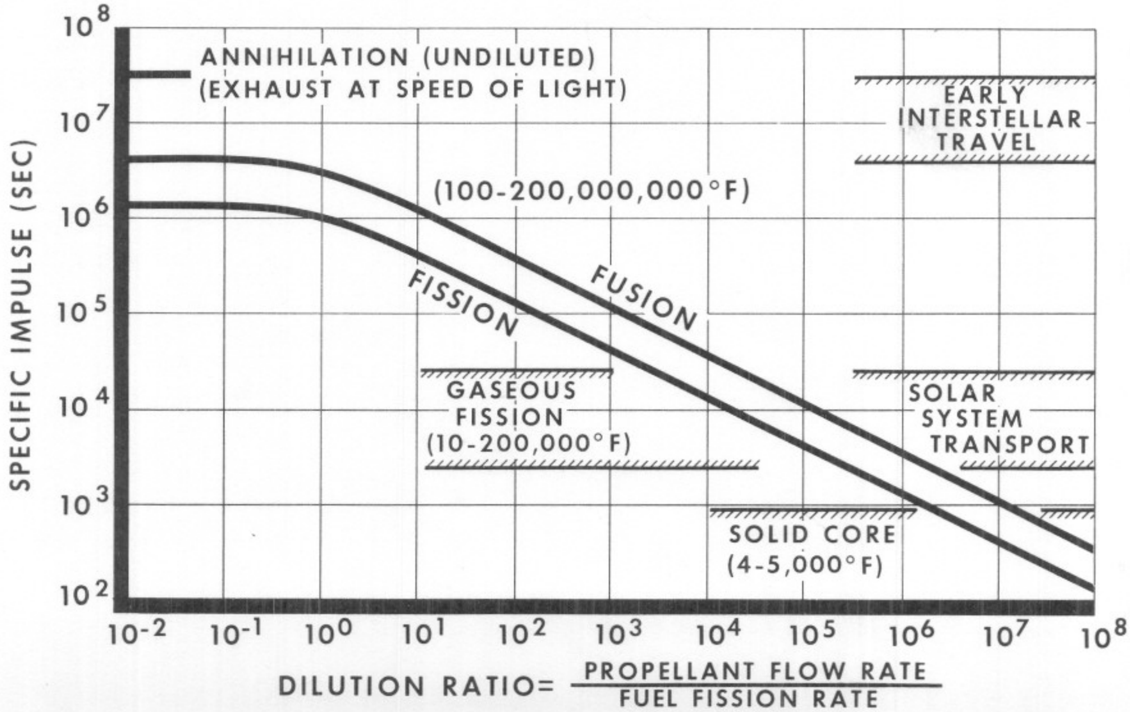


FIGURE 3

NUCLEAR FISSION ROCKET

COST OF FUEL AND PROPELLANT

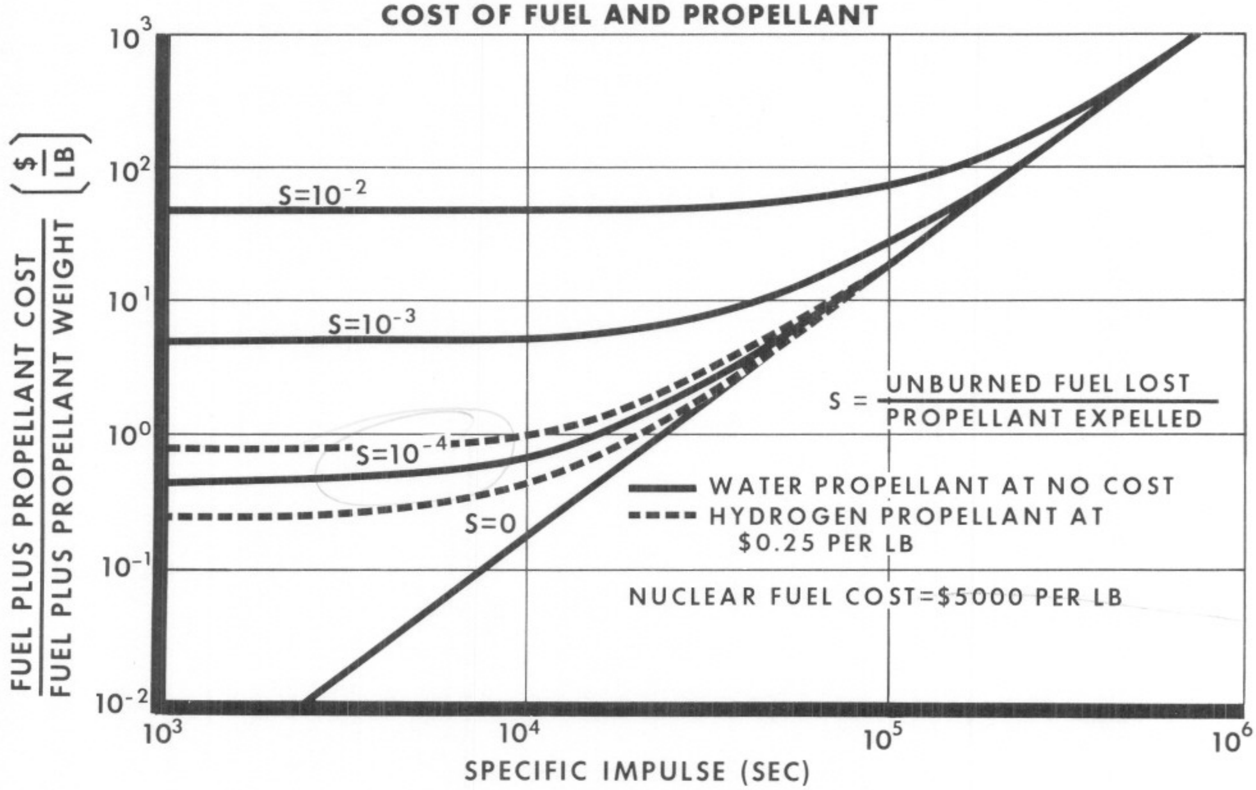


FIGURE 4

GASEOUS FISSION ENGINE

THRUST/WEIGHT RATIO

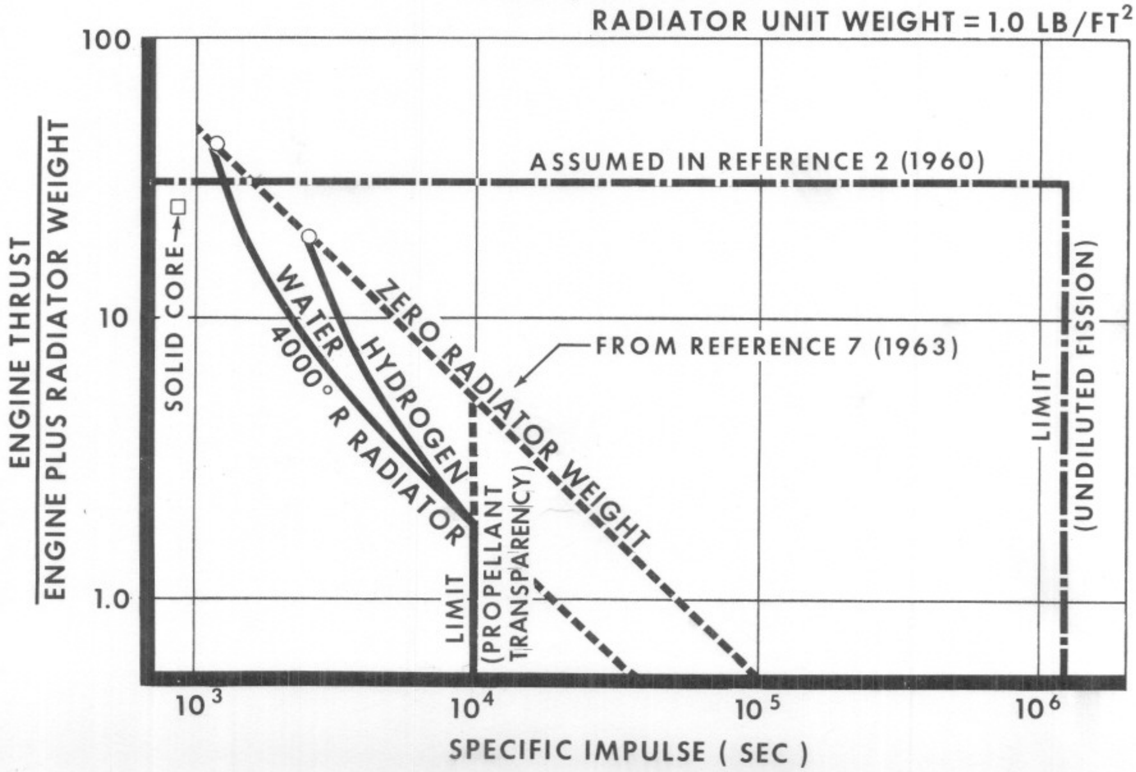


FIGURE 5

NUCLEAR ROCKETS

COST OF FUEL AND PROPELLANT

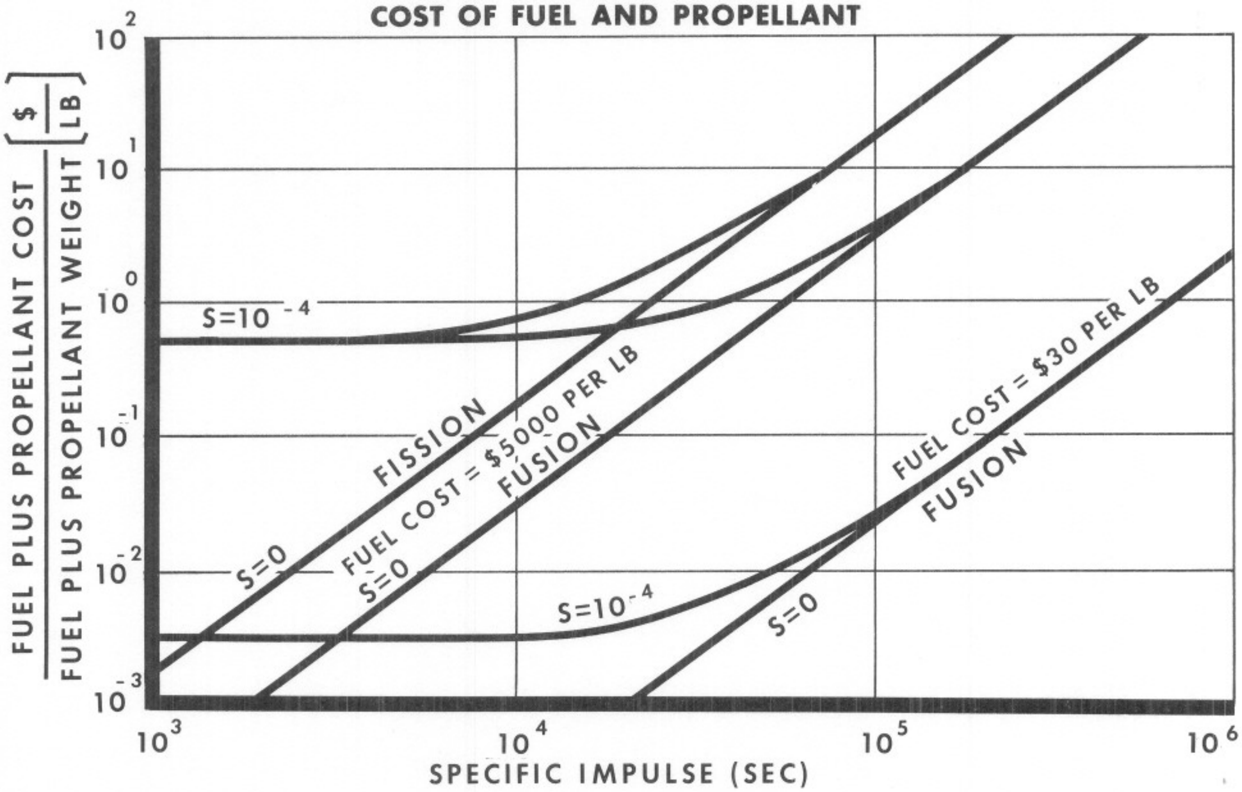


FIGURE 6

GRAVITY PENALTY

OPERATION DIRECTLY FROM MINOR PLANETS

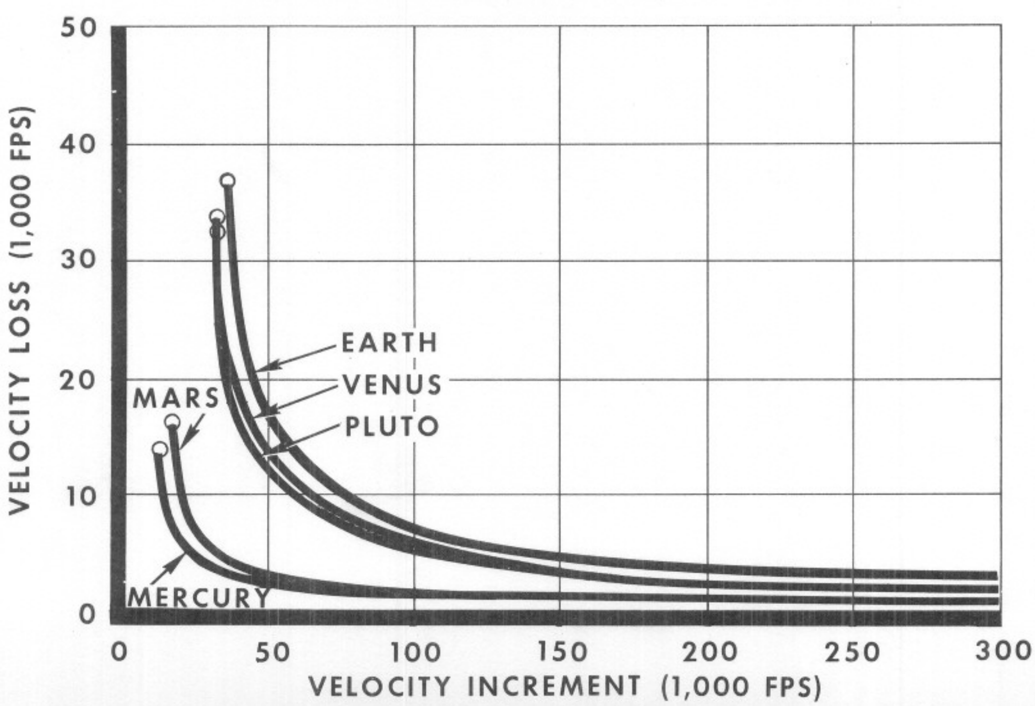


FIGURE 7

GRAVITY GAIN

OPERATION FROM NATURAL SATELLITES

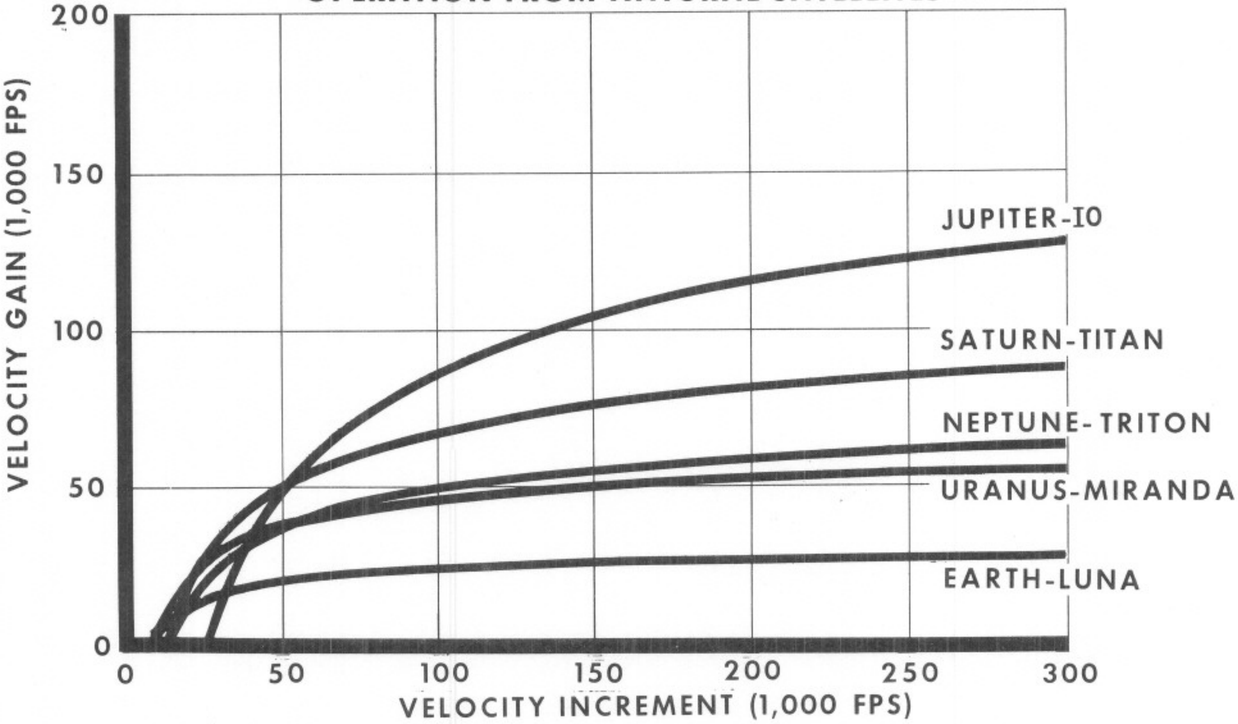


FIGURE 8

PERFORMANCE CALCULATION

COMPARISON OF SUNLESS ASSUMPTION WITH MORE EXACT CALCULATIONS

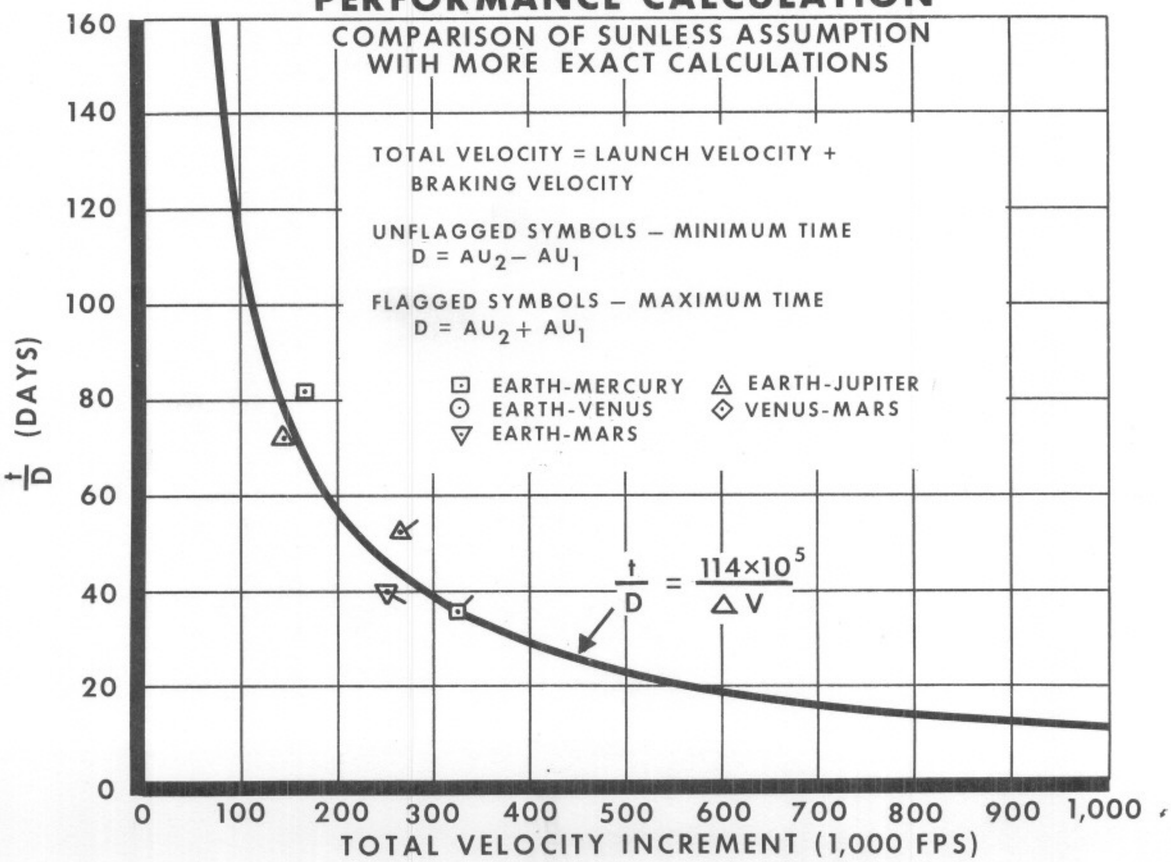


FIGURE 9

MINIMUM TRAVEL TIMES FROM EARTH
INCLUDING BRAKING REQUIREMENTS

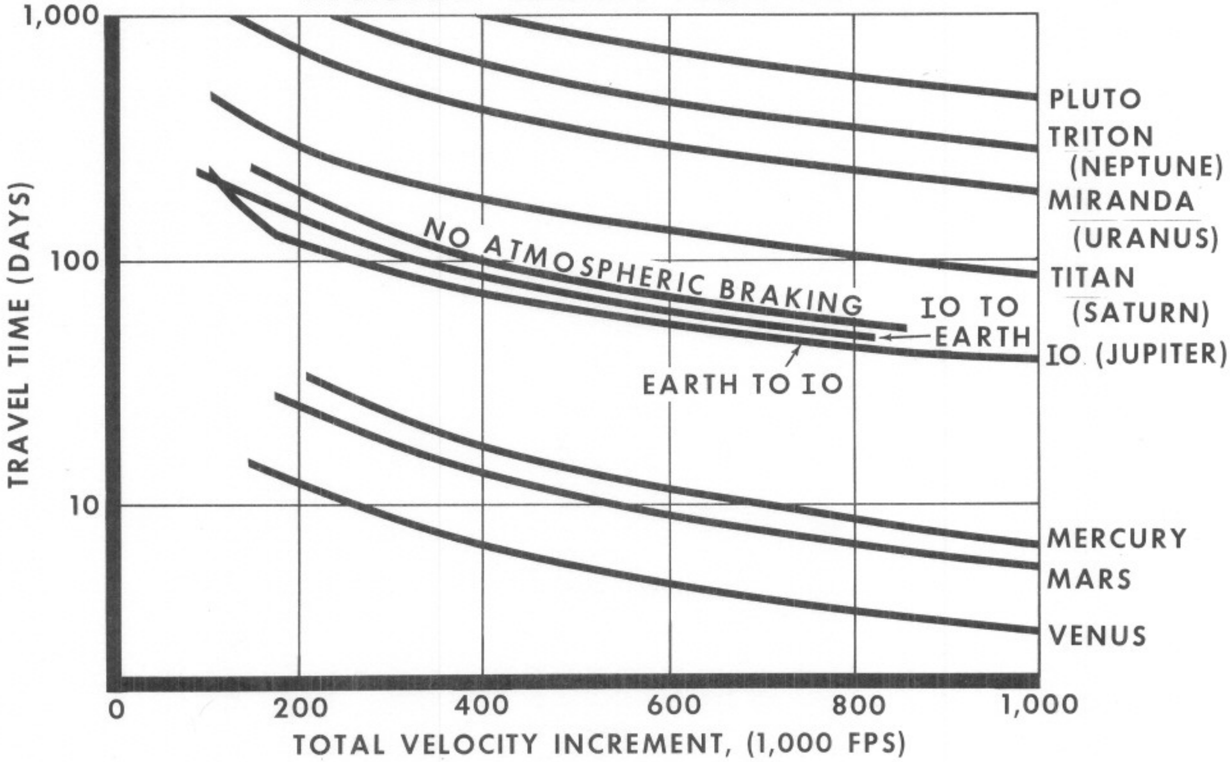


FIGURE 10

AVERAGE TRAVEL TIMES FROM EARTH
INCLUDING BRAKING REQUIREMENTS

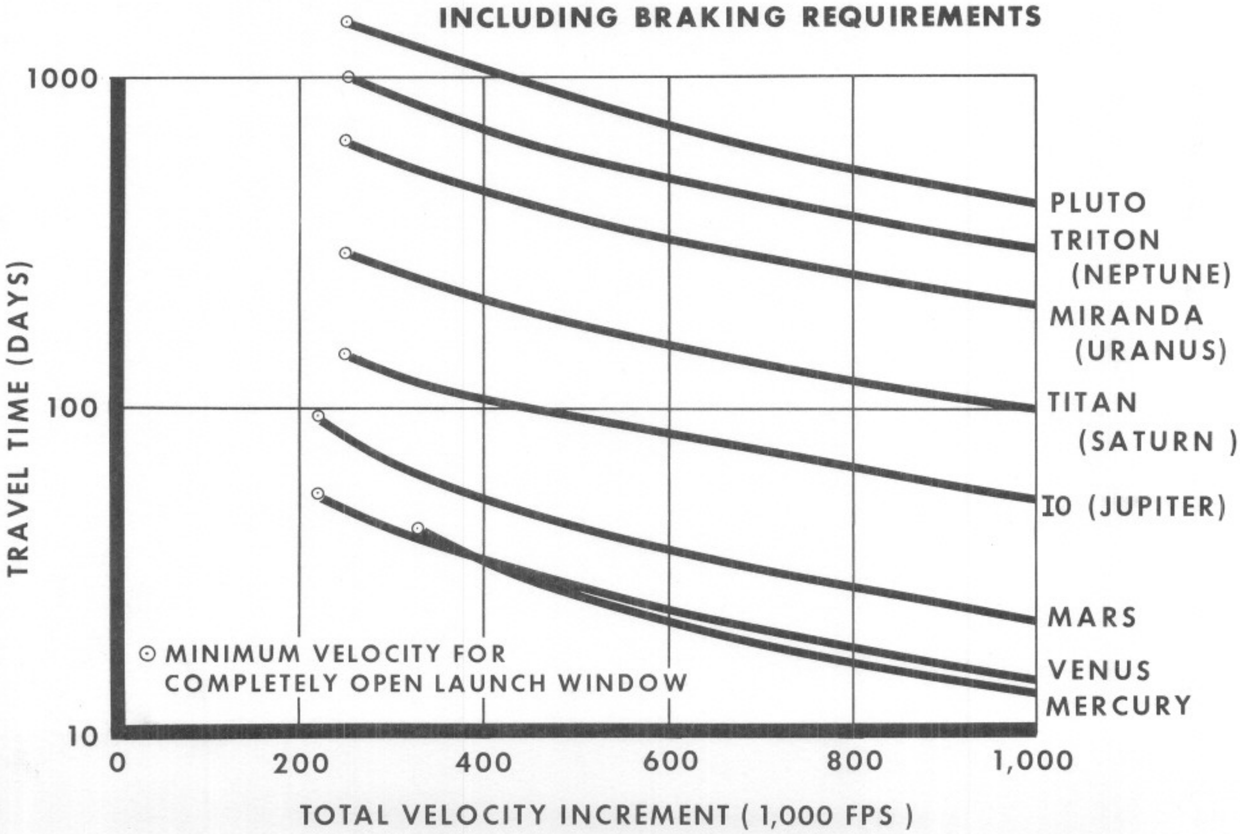


FIGURE 11

SOLAR SYSTEM SYNODIC PERIODS

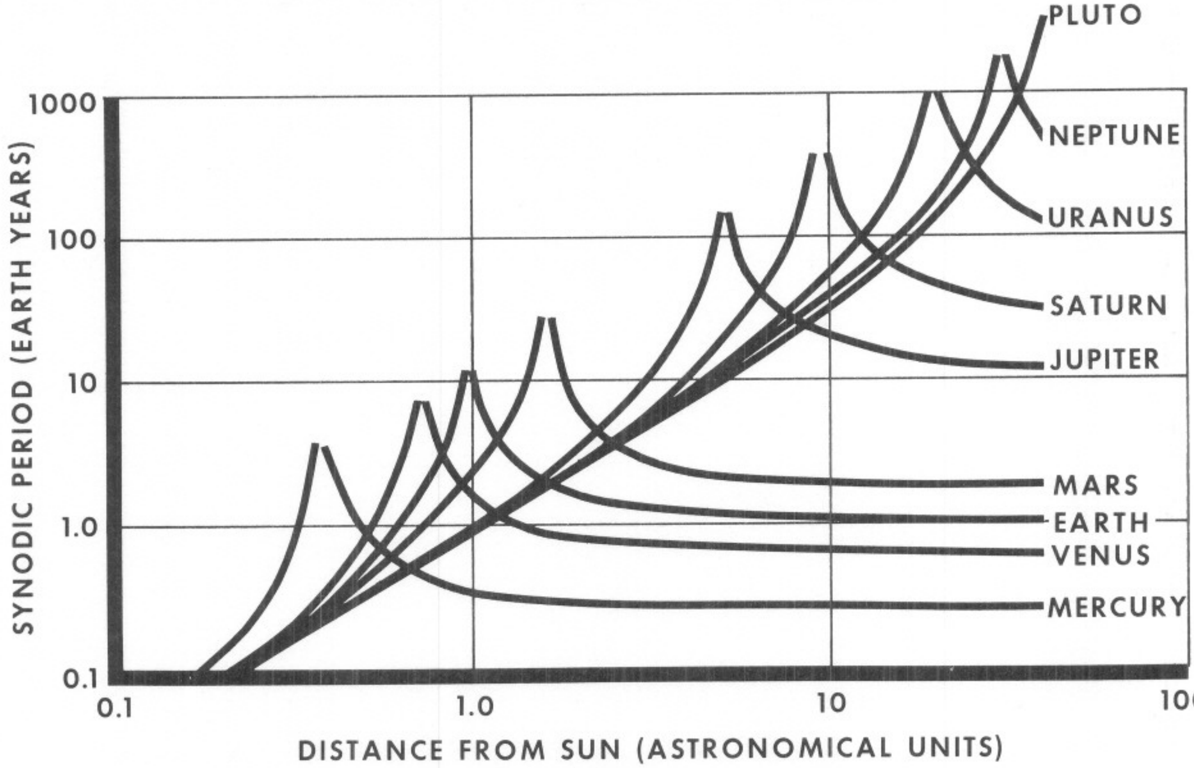


FIGURE 12

TRAVEL TIMES BETWEEN PLANETS

TOTAL VELOCITY INCREMENT = 500,000 FPS

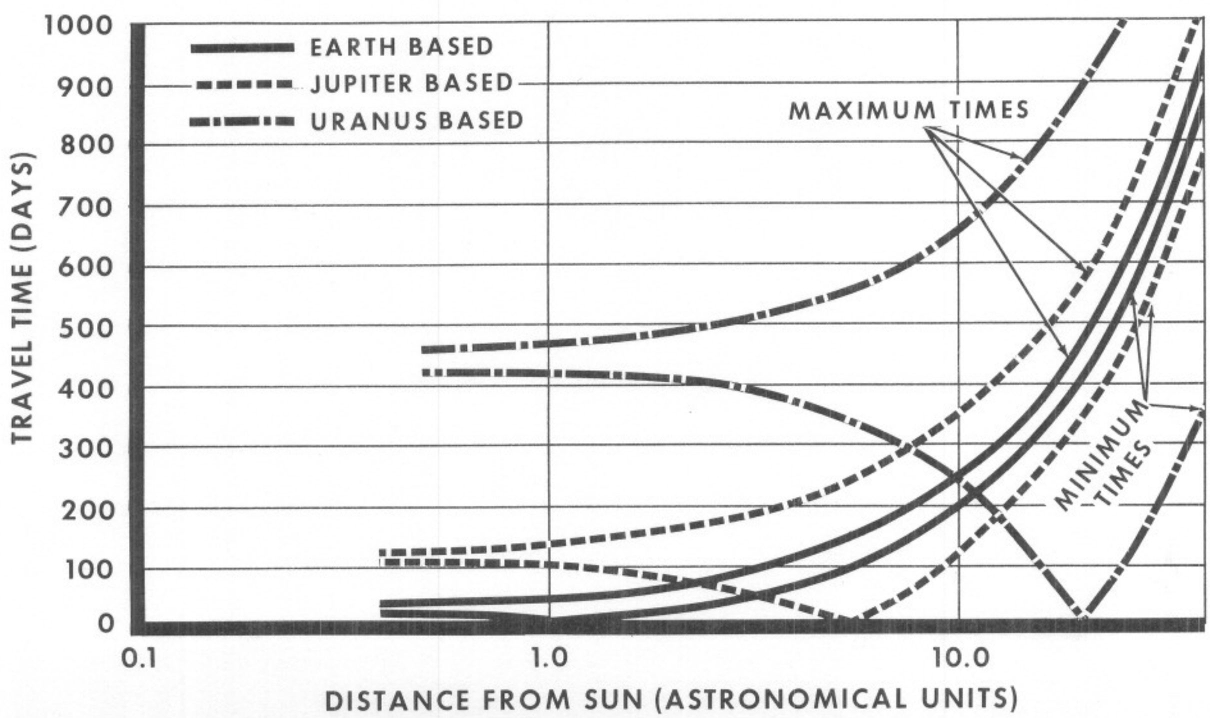


FIGURE 13

SPACESHIP WEIGHT ASSUMPTIONS

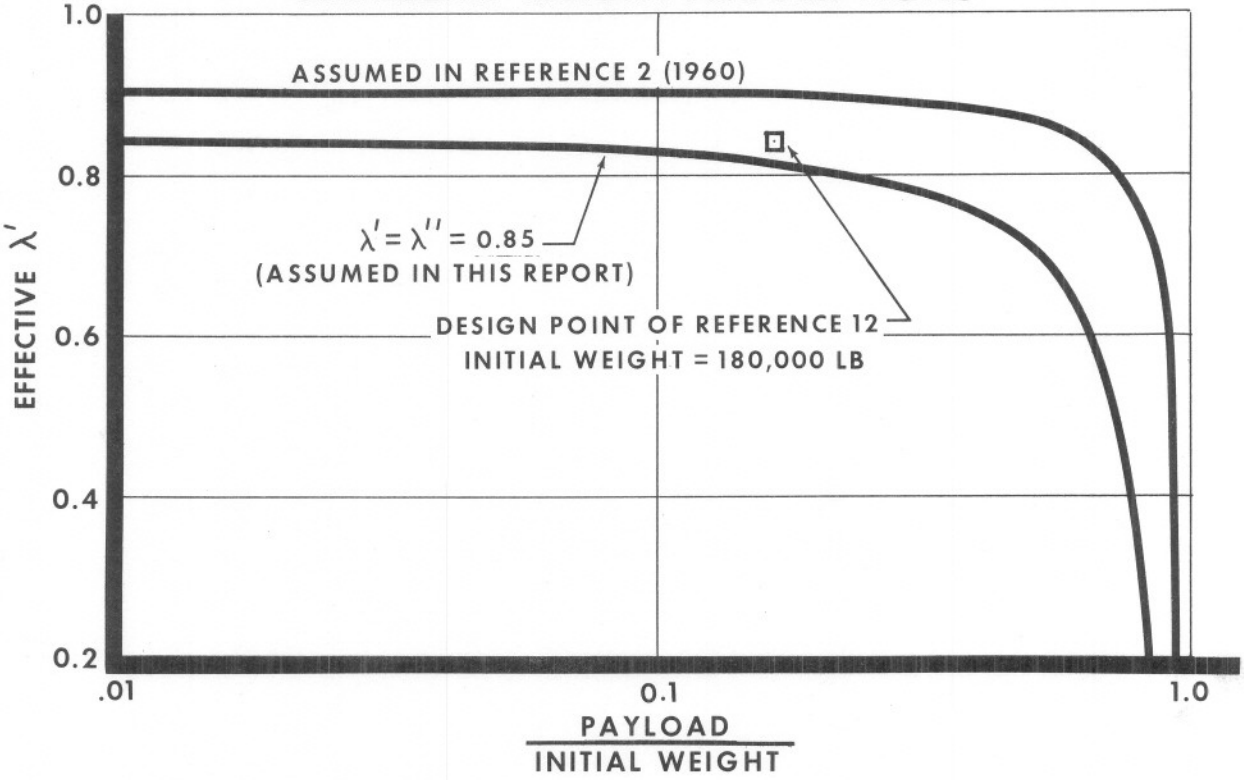


FIGURE 14

ROCKET WEIGHT PARAMETERS

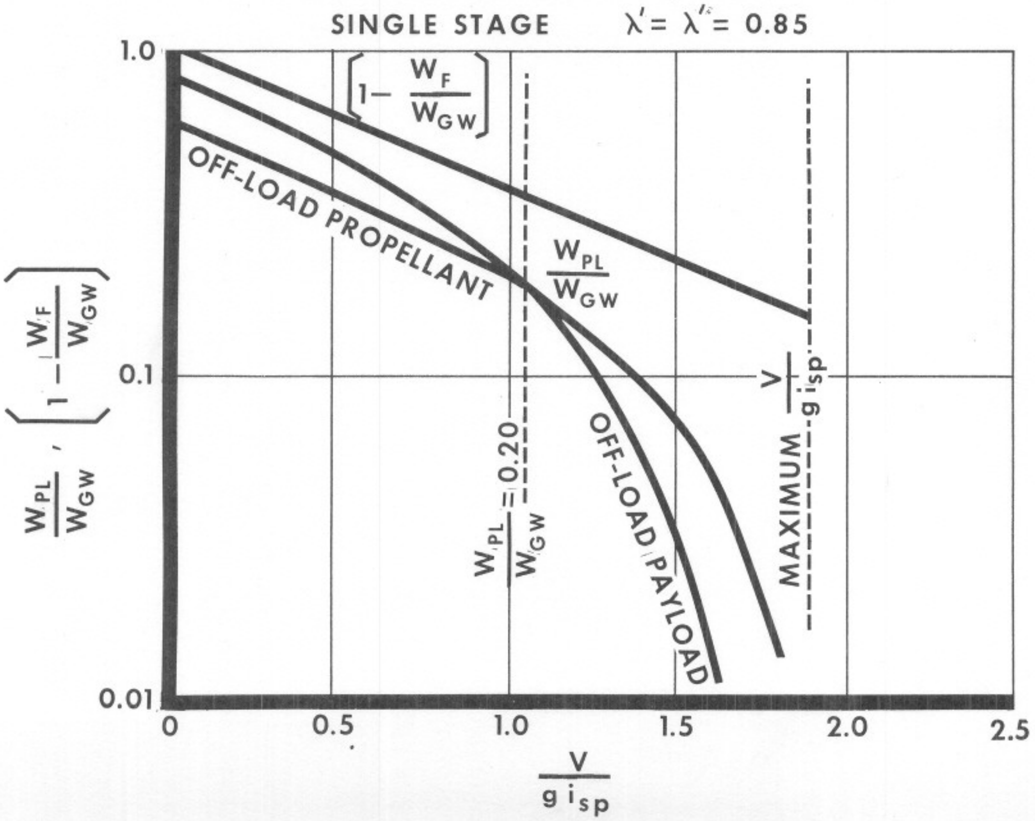


FIGURE 15

SINGLE STAGE SPACESHIPS
FUEL AND PROPELLANT COSTS

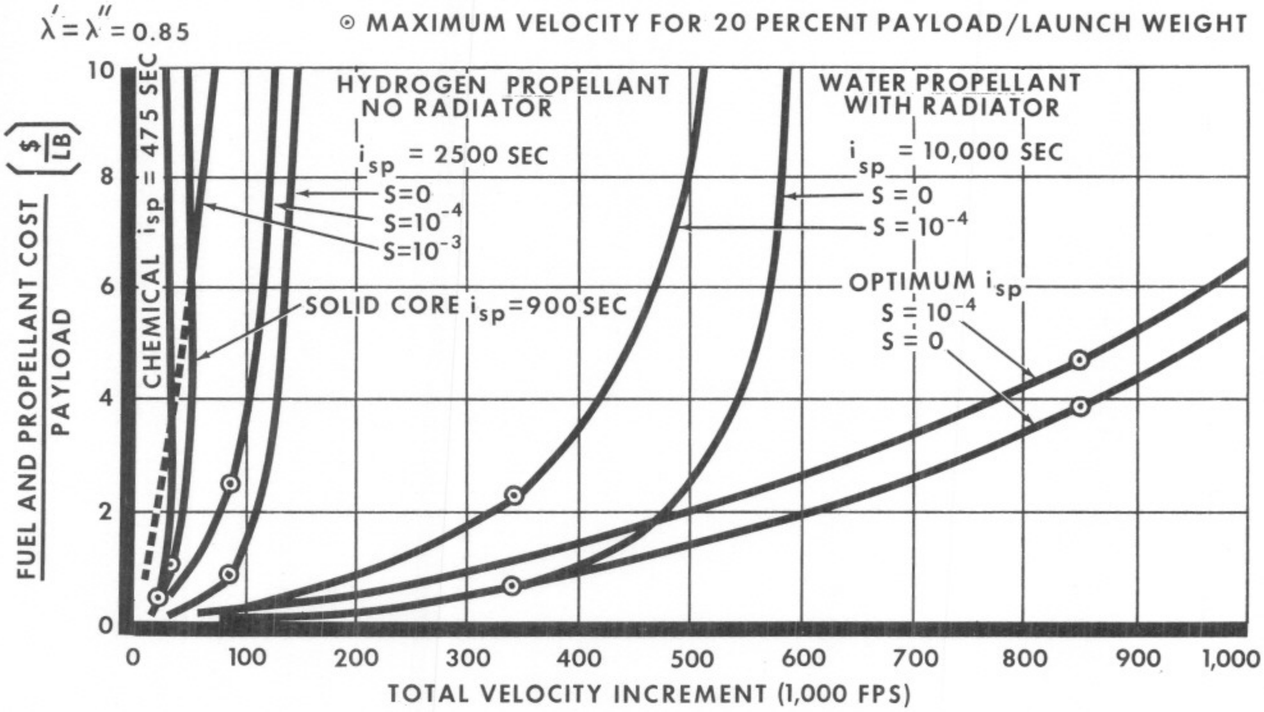


FIGURE 16

SPACESHIP PAYLOAD CAPABILITY

DESIGN POINT - 20 PERCENT PAYLOAD AT MAXIMUM WEIGHT

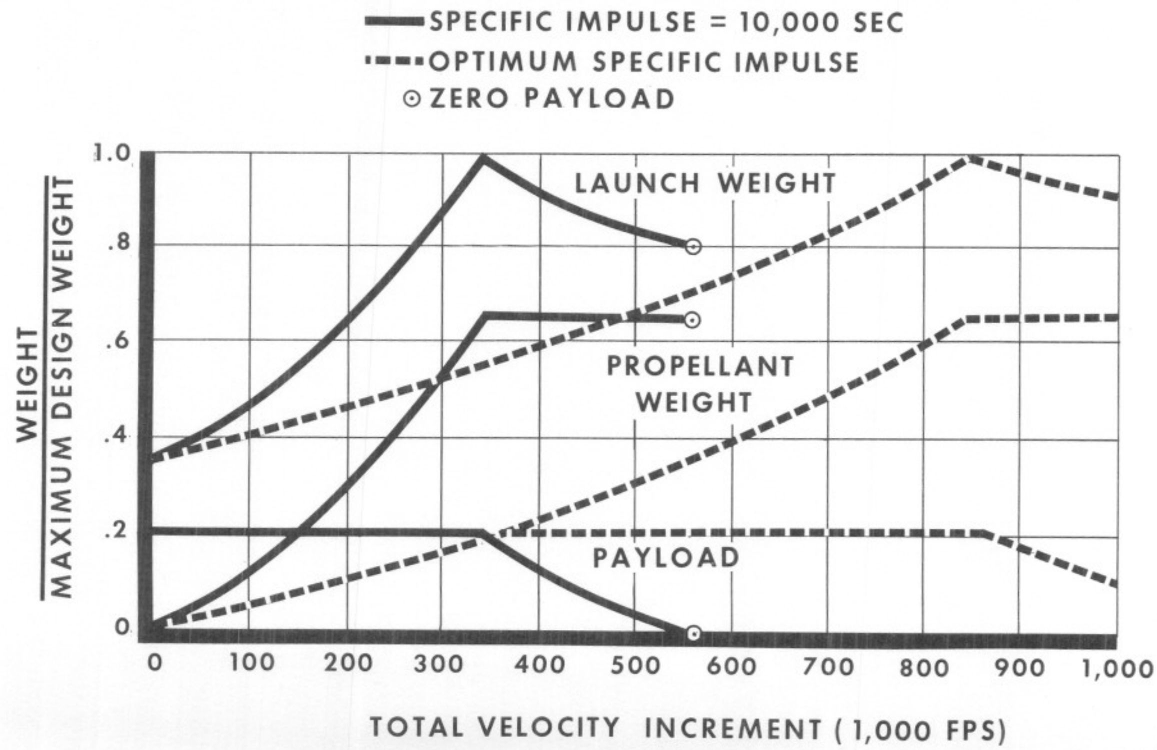


FIGURE 17

SINGLE STAGE SPACESHIPS

FUEL, PROPELLANT AND STRUCTURE COSTS

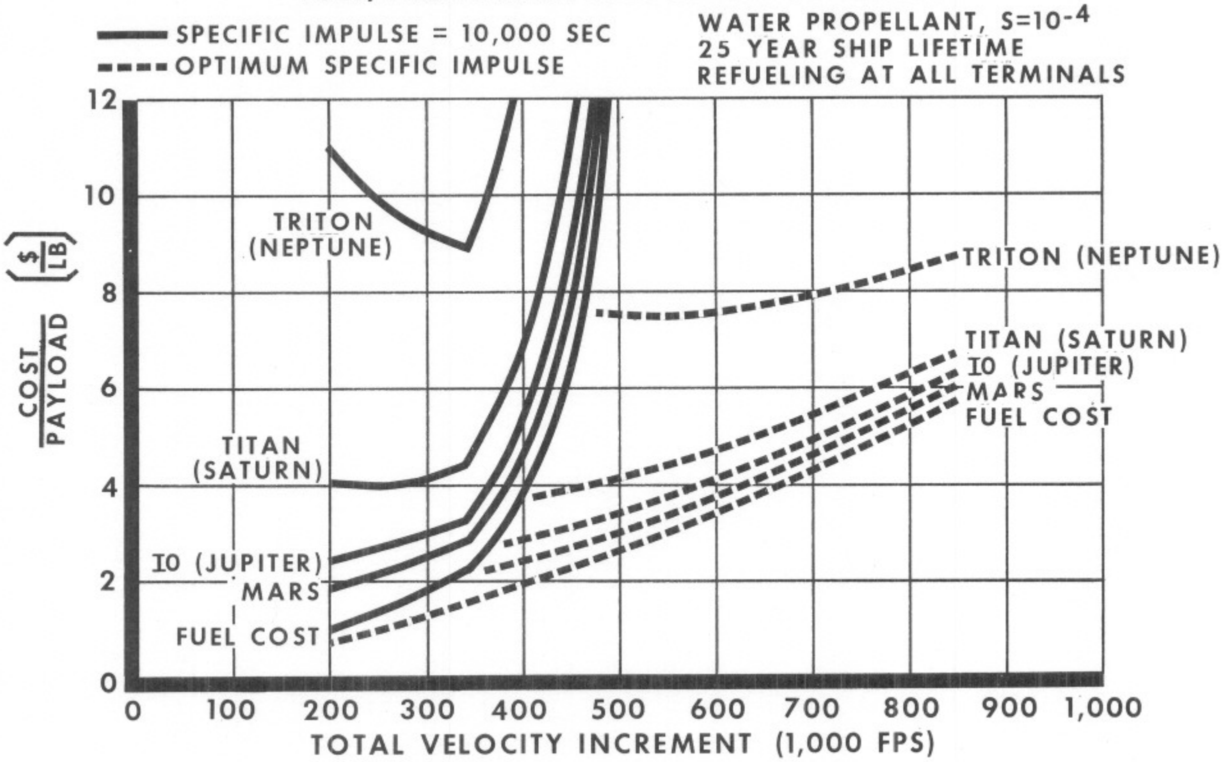
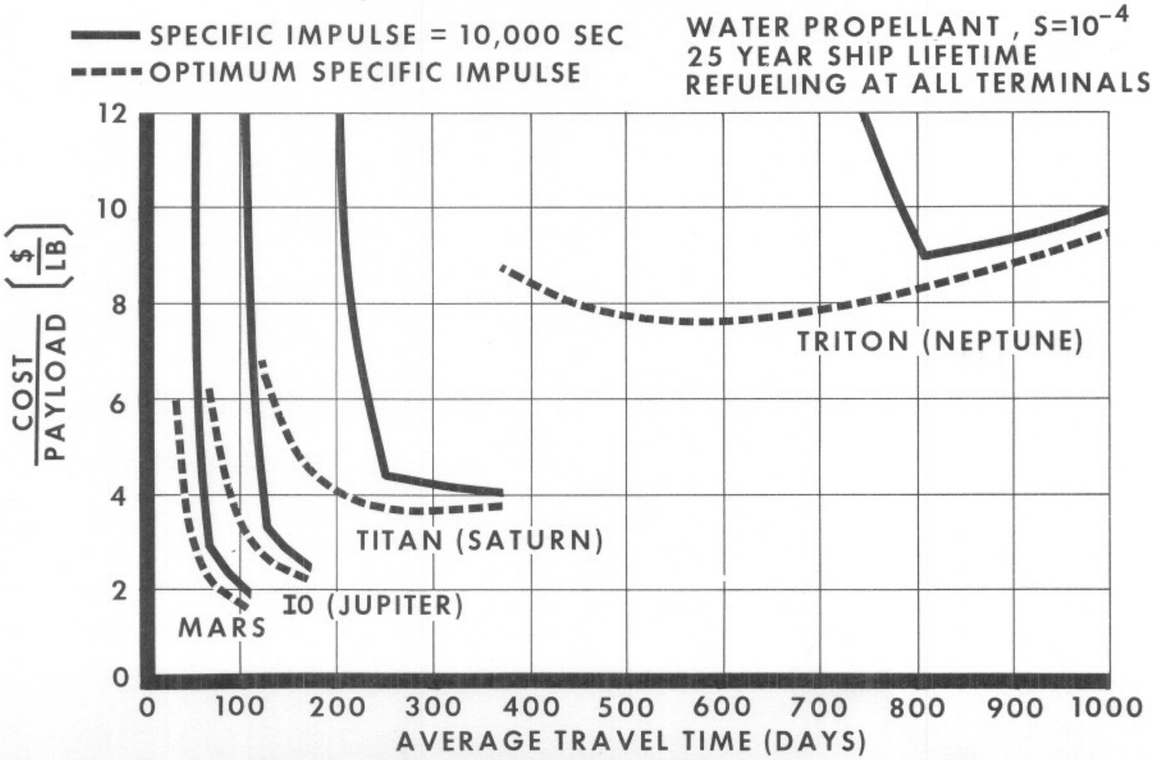


FIGURE 18

SINGLE STAGE SPACESHIPS

FUEL, PROPELLANT AND STRUCTURE COSTS



This shows desirability of optimum

FIGURE 19

DOSE TO GROUND OBSERVER VS. START ALTITUDE FOR GASEOUS CORE

DOSAGE SELECTED FOR MOST CRITICAL TIME OF GASEOUS CORE OPERATION

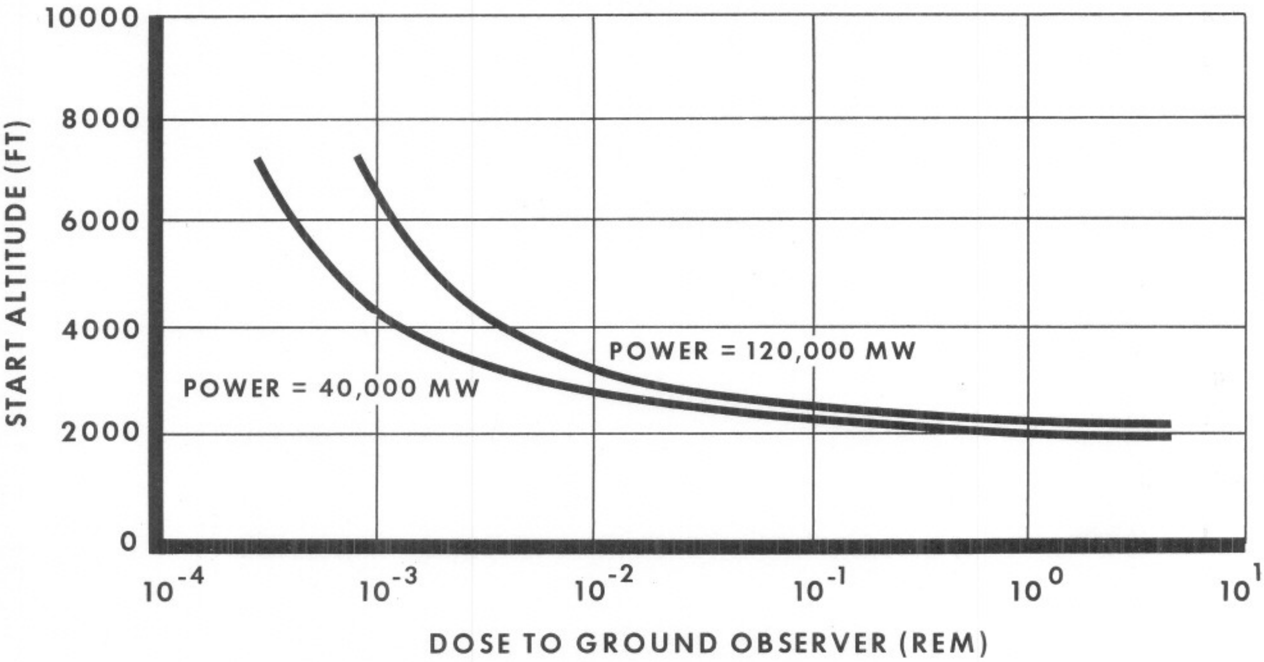
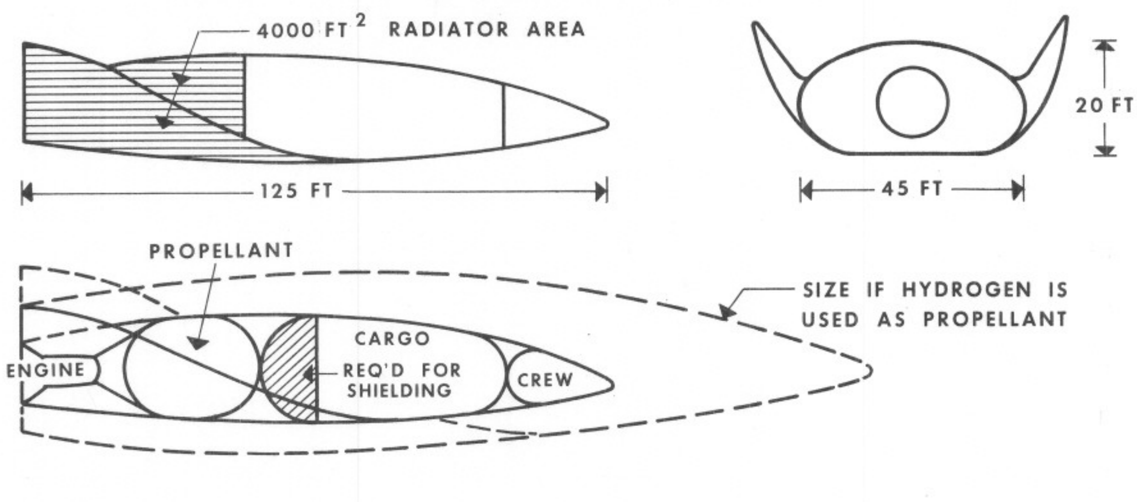


FIGURE 20

GASEOUS FISSION POWERED SPACESHIP



LAUNCHING WEIGHT = 1,000,000 LB (500 TONS)
CARGO WEIGHT = 200,000 LB (100 TONS)
PROPELLANT WEIGHT = 650,000 LB (325 TONS)

FIGURE 21

GASEOUS FISSION POWERED SPACESHIP

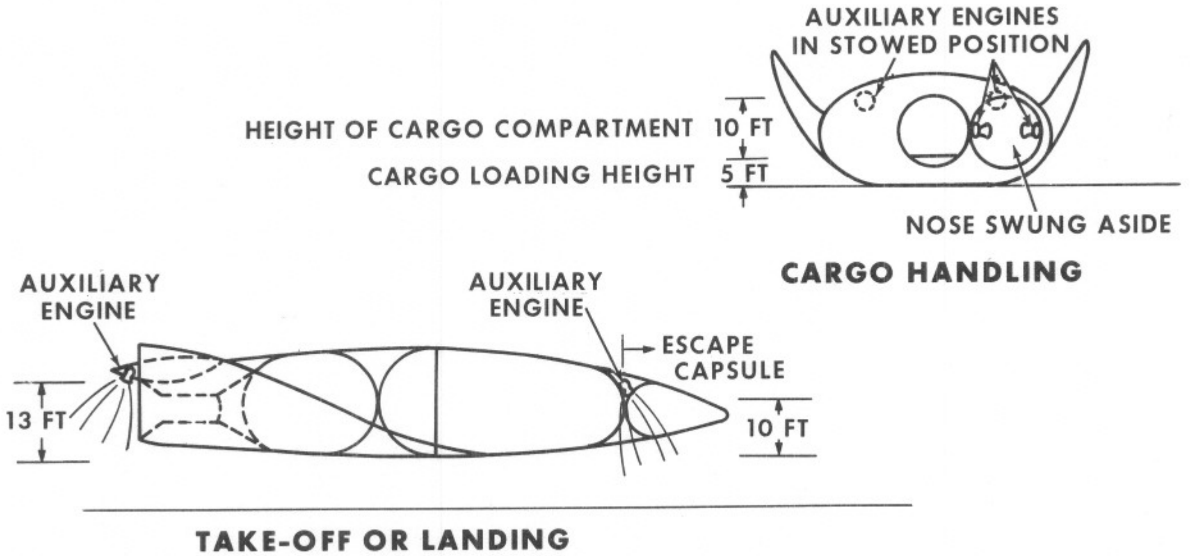


FIGURE 22

SPACESHIP PAYLOAD CAPABILITY

EFFECT OF EARTH LAUNCH

DESIGN POINT - 20 PERCENT PAYLOAD AT MAXIMUM WEIGHT
SPECIFIC IMPULSE LIMITED TO 10,000 SEC

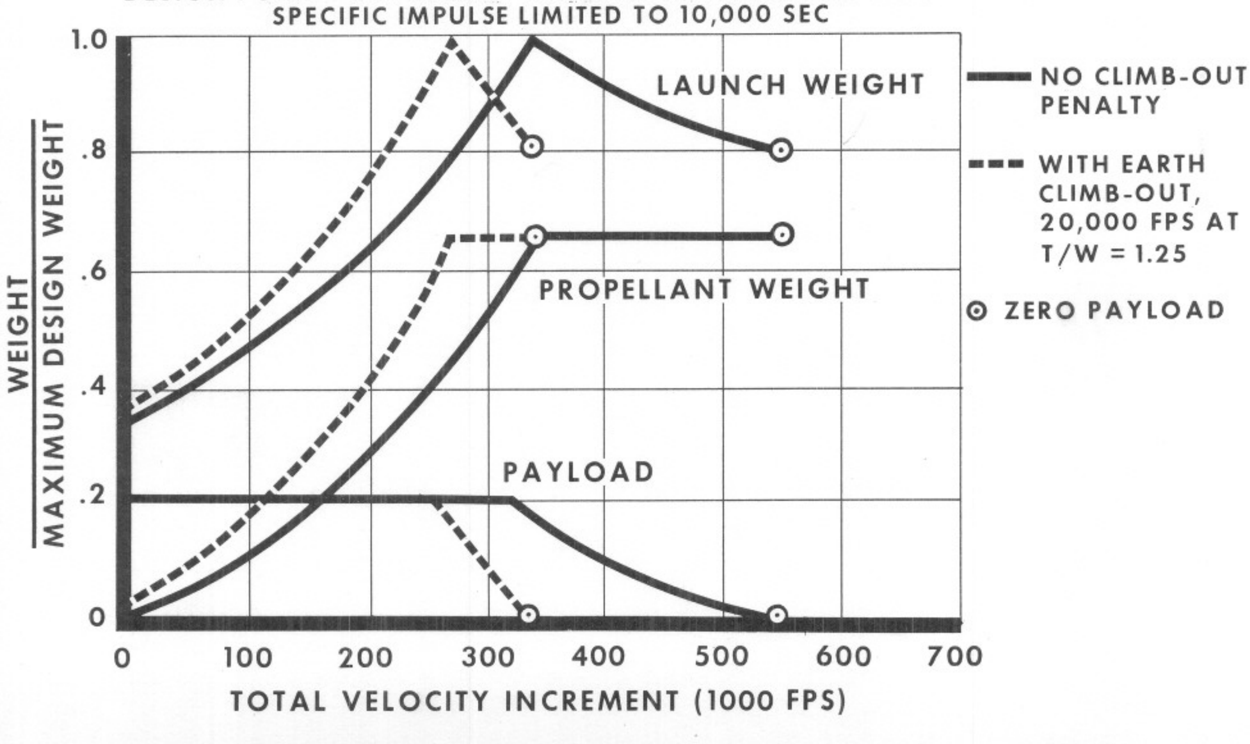


FIGURE 23

TRANSPORTATION VS AMMUNITION

COMPARISON OF RE-USE ASSUMPTIONS

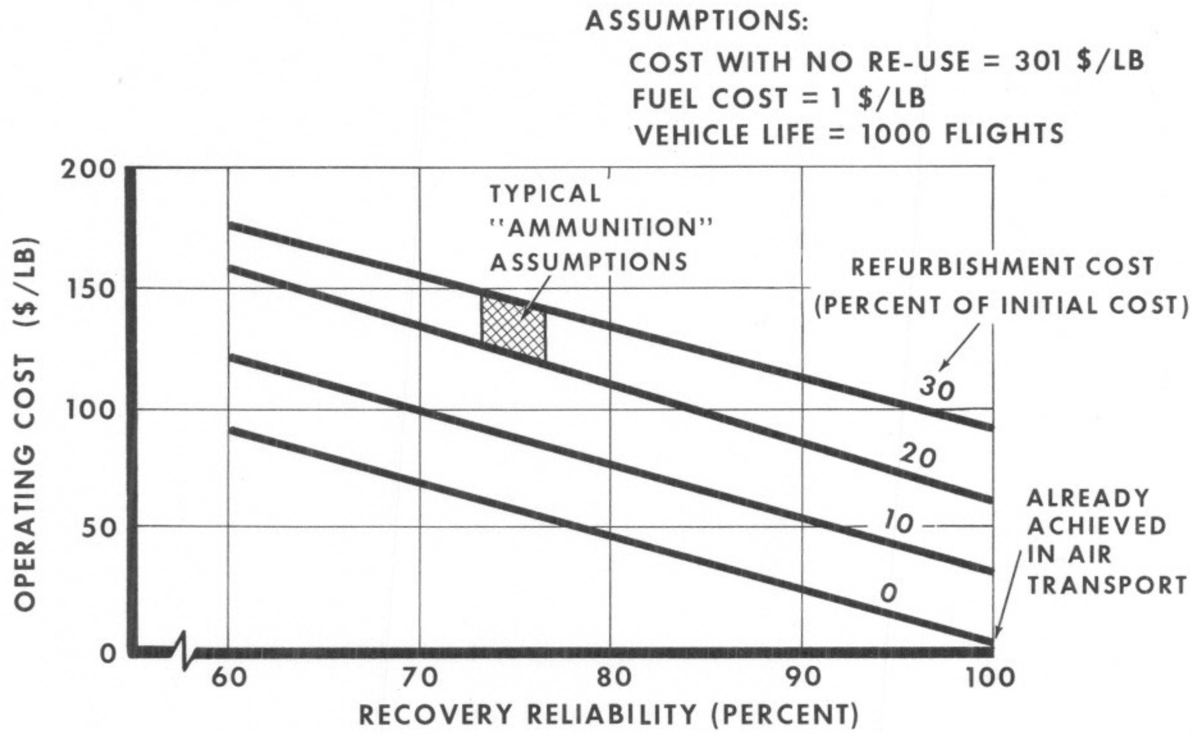


FIGURE 24

TRANSPORTATION VS AMMUNITION

EFFECT OF FUEL CONTAINMENT ON ORBITAL COST

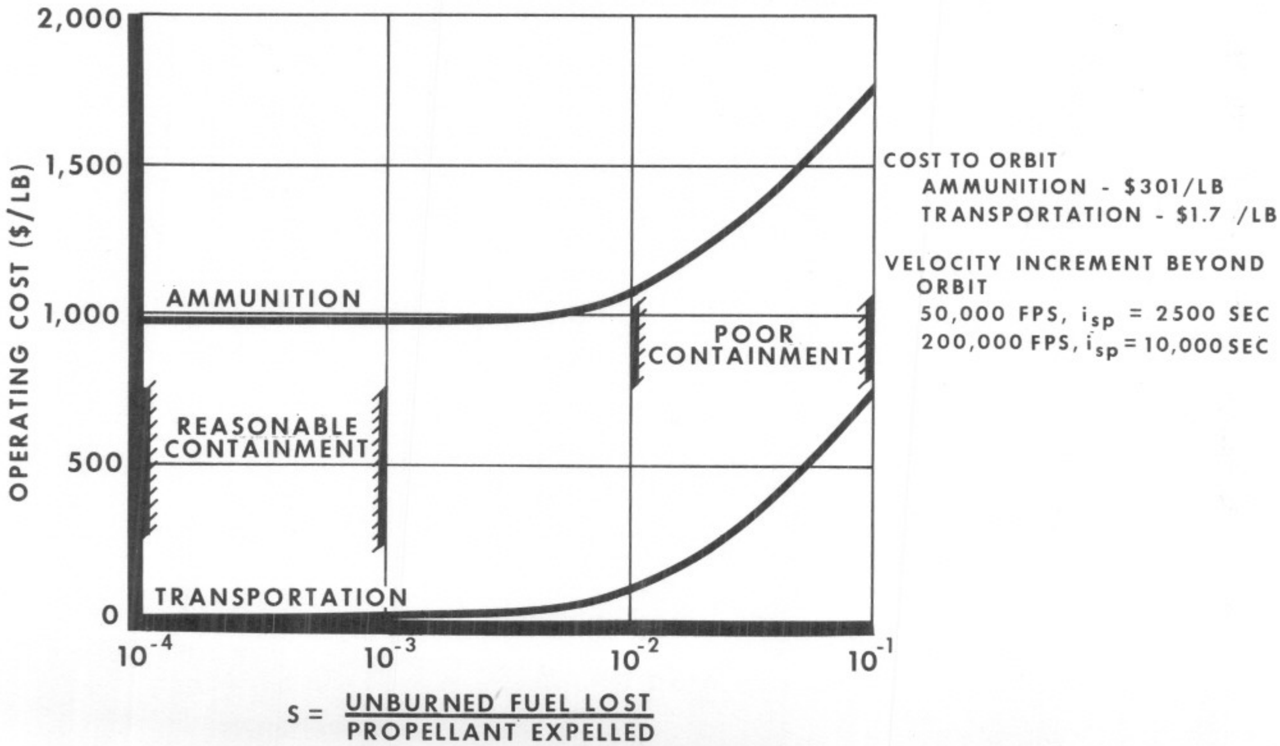


FIGURE 25

ORBITAL WEIGHT RATIO

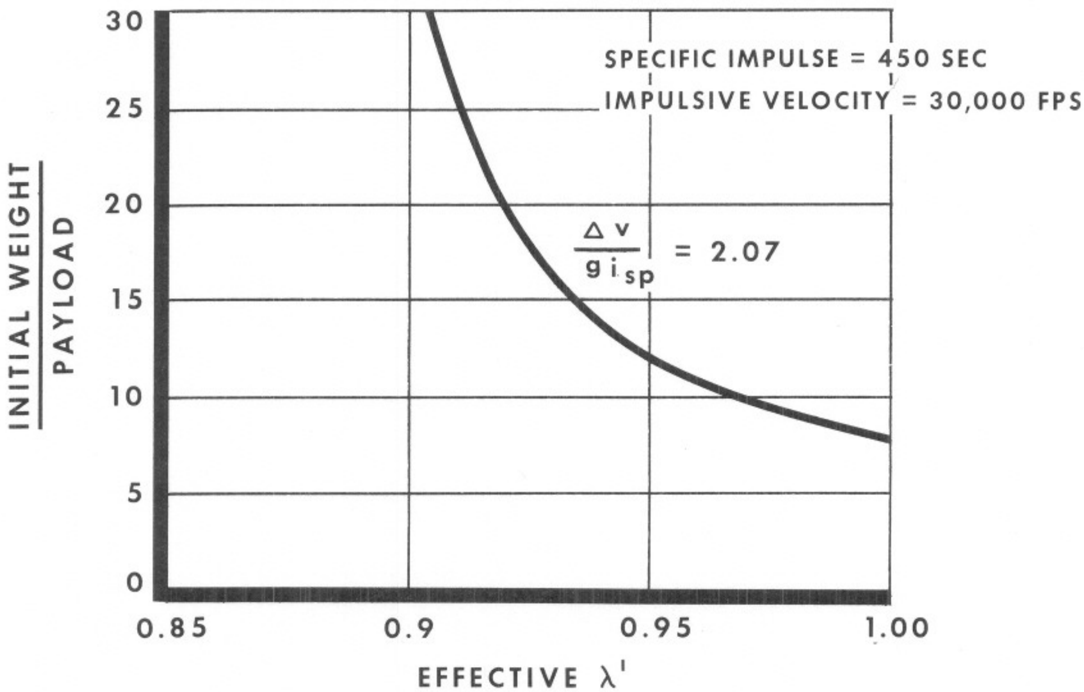


FIGURE 26

EFFECTIVE λ'

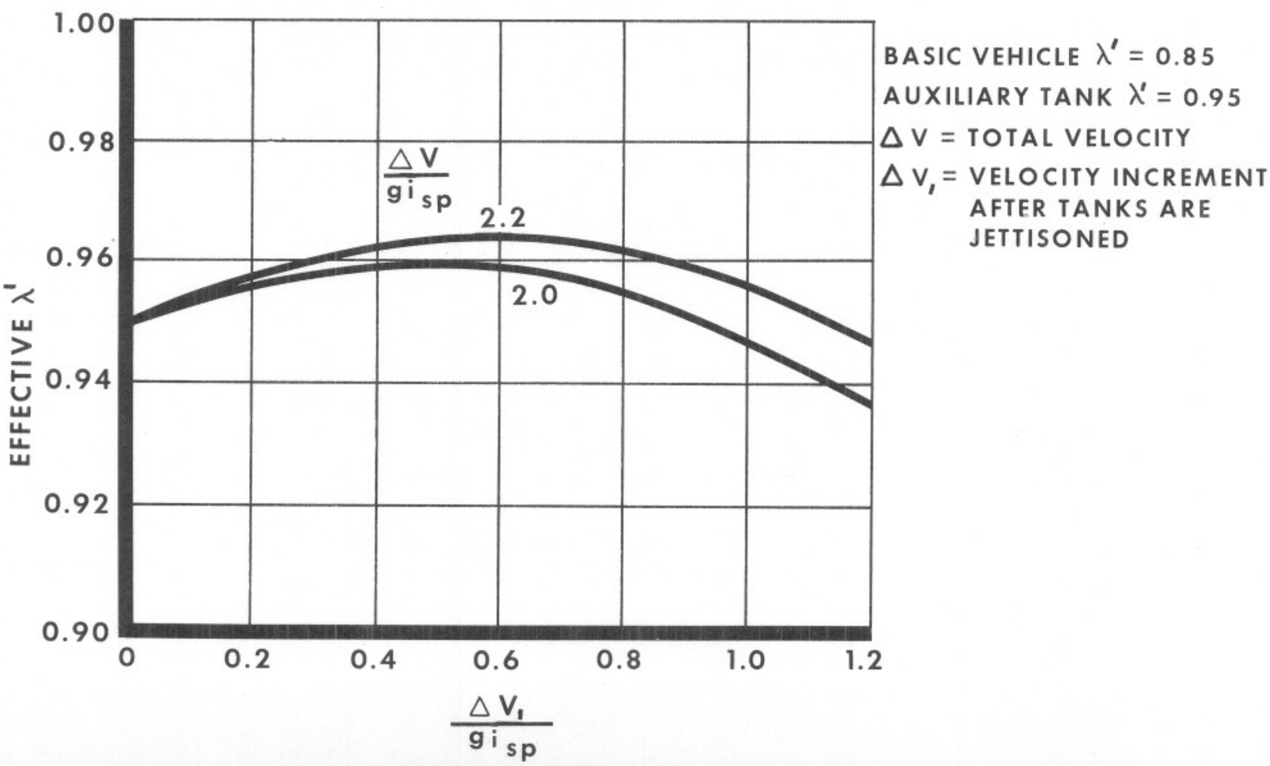
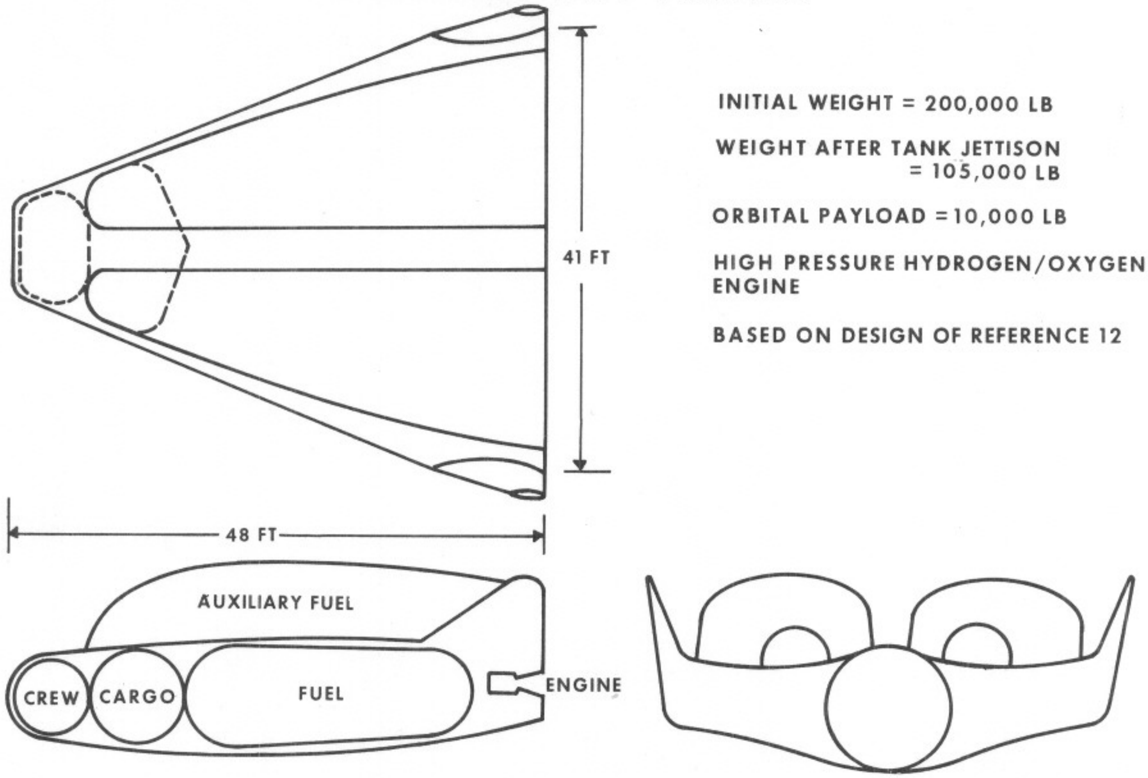


FIGURE 27

POSSIBLE TEST VEHICLE



INITIAL WEIGHT = 200,000 LB

WEIGHT AFTER TANK JETTISON = 105,000 LB

ORBITAL PAYLOAD = 10,000 LB

HIGH PRESSURE HYDROGEN/OXYGEN ENGINE

BASED ON DESIGN OF REFERENCE 12