THE POTENTIAL FOR NUCLEAR PROPULSION FOR MANNED SPACEFLIGHTS

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INTRODUCTION

Nuclear propulsion for rocket application can be separated into several categories. Probably the most obvious approach to utilizing nuclear energy is a nuclear thermal rocket consisting of a reactor made of solid material used to heat a propellant which is expelled in a hot jet to give useful thrust. Unfortunately, high specific impulse in a thermal rocket requires a high temperature in the exhaust stream. Therefore, fundamental performance limitations appear which are high temperature reactor material problems rather than limitations on the amount of nuclear energy available. It is well known that these limitations tend to give the order of magnitude of twice the specific impulse of high energy chemical systems, even when hydrogen, with its low molecular weight, is utilized as the propellant in order to yield the lowest possible temperatures. The Rover Project is developing a modest-sized rocket of this type. A doubling of the specific impulse is a very substantial performance gain, and the use of upper stages powered with these rockets, combined with conventional chemical lower stages substantially increases performance. Such stages would tend to double the earth orbital payloads of large chemical rockets such as Saturn, and decrease the size of interplanetary orbit-to-orbit shuttle vehicles by one order of magnitude[1].

Those performance improvements are worthwhile goals, and it is important that they be pursued with vigor. The situation is, however, still very frustrating on two counts. First, the cost of deploying people and equipment throughout space is still extremely high, and it would be most desirable to make a really fundamental reduction in this important basic quantity. Second, the energy potential of nuclear reactions is far higher than that which can be released in solid core fission rockets with current material limitations, so that vastly greater performance improvements are latent in the nuclear process. Attempts to get around this second frustration of "fundamental" performance limitations have, so far, split into two families. The first family utilizes a nuclear thermal rocket modified so that part or all of the energy release takes place in a gas so that no fundamental temperature restriction exists. Two sub-groups of this family are: (a) The use of nuclear bombs which detonate behind the ship and throw debris against a cunningly mounted striker plate as a method of obtaining thrust (Project Orion); and (b) The running of a contained reaction in a thrust chamber with the energy release actually occurring in the gaseous phase, the so-called gaseous fission reactor. The second family utilizes electrical methods of accelerating the propellant to avoid completely the need for high temperatures. This family has spawned a fair amount of research on various electrical propulsion schemes and, as a by-product, a large number of mission performance analyses.

A one-sentence-each summary of the recent state of such advanced propulsion follows. The present version of Orion is limited to huge payloads, and certainly was impossible to test adequately while a nuclear test ban was in effect. Electrical rockets have extremely low thrust-weight ratios due to the weight of power conversion equipment required, and their performance is consequently severely attenuated in spite of yeoman efforts in mission performance analysis. No one has yet come forth in a convincing fashion with a constructive idea for making the gaseous fission rocket work at all.

This paper will first review certain lines of reasoning which led to a possible scheme for greatly reducing the cost of space travel, even if only solid fission rockets were available. An attempt will then be made to expand the same line of reasoning to the utilization of gaseous fission rockets, an approach inspired by a number of new suggestions which have materialized recently with respect to gaseous fission reactors. Reference to Orion, nuclear electrical rockets, and fusion rockets will be made occasionally, if appropriate, but no attempt to present a completely comprehensive treatment of all types will be made. The discussion will be limited to solar system transportation requirements.

SOLID CORE FISSION ROCKETS

If one examines the reasons for the high cost of space travel, he is usually impressed with the apparent fact that everything seems hopeless. Large installations such as Cape Canaveral are evidently necessary, and expensive pieces of hardware are effectively placed almost beyond recovery on each flight, particularly after being spread around the planet and throughout local space during the operation. The very high velocities required in space flight would seem intuitively to involve such tremendous energies that it is natural to expect everything to be fantastically expensive anyway, so that the hopeless feeling tends to increase.

A really high performance rocket is capable of making fundamental improvements in this picture. If performance can be made high enough so that single-stage rockets have adequate capability, it becomes possible to re-use equipment exactly as in transport airplane practice. Although a single-stage, re-usable, high-energy chemical rocket can almost certainly be developed for earth orbital operations, nuclear energy is required for virtually all deep space missions. Except possibly for safety provisions surrounding the engine, there is no reason why a single-stage nuclear rocket should be any more complicated than a standard transport airplane, and no reason why it should be much more expensive in production, except for the price of the nuclear inventory which it must carry on board for reactor criticality requirements. True, the temperature inside the reactor (and in the exhaust jet) will be substantially higher than that to be found anywhere near a normal transport aircraft. On the other hand, the rocket accelerates out through the atmosphere slowly, and hence, is never subjected to severe external environmental factors such as supersonic transport aircraft face. In space, conditions are mild, although sometimes too radiantly beautiful. Should aerothermodynamic loads on unpowered re-entry prove too distressing, a really high performance rocket could re-enter the atmosphere on jet thrust. The high reactor core temperature, hence, is really the only major vehicle difference, and if that problem can be solved, all else settles into a normal pattern which could well be an easier environment than most modern aircraft encounter.

It is felt by the author that most people tragically underrate this point. Five-stage rockets of the large man-and-cargo variety tend to spawn large, complicated launching systems, fantastic numbers and types of engines, vast real estate developments, and similar complicating phenomena. As a matter of fact, one of Hunter's additions to Parkinson's laws is: "The management structure (total number of companies and agencies) involved in large manned rocket programs is at least directly proportional to the gross weight times the number of stages of the vehicle." Most of us intuitively, if not always consciously, understand that Parkinson would certainly have discovered his laws in the space launch vehicle business if the British Admiralty and Foreign Service had not already provided his source of inspiration. We are duty-bound to keep a stiff upper lip, however, and should not be blind to the fact that it never takes more than one adequate propulsion system to give a simple, reliable vehicle which might be no more difficult to prepare for the next flight than the average, equally complicated, transport plane (regardless of the exhaust temperature when the throttle is opened).

It is clear from the above discussion that sufficiently high rocket performance capabilities might well yield tremendous cost reductions in both hardware and overhead costs. One of the more startling aspects of nuclear rocketry is that, in addition, the energy costs turn out to be relatively reasonable. At first thought it would seem that the energy expended in generating 60,000 fps total velocity for a lunar expedition would be far greater than any normal transportation requirement. The total velocity is almost twice earth escape speed, and even earth escape speed had not been obtained with any appreciable payload only 5 years ago. It turns out, however, that this is about the same amount of energy which a transport airplane utilizes in suspending itself in the air long enough to make a flight from Los Angeles to New York. If one assumes a jet transport to be cruising for 6 hours at an L/D of 12, this means that the engines were applying 1/12 of the weight of the airplane for the six hour period. If somehow this could have been applied in field-free space, the airframe would have accelerated at 1/12 of a g for 6 hours and would have generated 58,000 fps, essentially the total velocity required for a lunar transport. Thus normal cruising aircraft utilize the same order of magnitude of energy as required for a modest space transport system, essentially because they fight gravity incessantly during their whole flight. In space, transports fight gravity quickly and efficiently in the early part of their flight and coast through space to their destinations. Looking at it another way, a Nova-type vehicle would use about 10^7 pounds of chemical fuel to give 2×10^4 pounds of payload a lunar round trip. This is 500 pounds of propellant per pound of payload and, at an average energy of 3000 KCAL/KG, is approximately 1000 KW-HR of energy per pound of payload. Since the current rates for commercial electrical energy (in Washington, D. C.) are about 1¢ per KW-HR, $10 is the terrestrial price of the energy per pound of payload. Hence, space flight is expensive not because the energy 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. If it were possible to build a nuclear rocket with perfect fuel containment, the price of uranium burned to generate the required energy would be only a fraction of a cent per pound of payload. This is much lower than the chemical rocket example quoted, since not as much energy must be expended accelerating inefficient fuel to high kinetic energies. The difference becomes even more pronounced at higher velocities.

These various points have been explored, various ways of presenting them have been given, and analyses have been made as to velocity requirements for various future space missions[2-6]. Perhaps the most concise way to summarize this data is shown in Figure 1, where dollars per pound of cargo delivered is shown vs velocity of mission for chemical rockets, solid fission rockets, and two examples of gaseous fission rockets. Figure 1 assumes values of hardware costs consistent with current airframe practice, weight ratios of vehicles consistent with current rocket practice, and the achievement of re-use consistent with transport aircraft experience[6]. Chemical rockets can be used as orbital trucks without causing more than a few dollars per pound increase in operating cost for the missions. The only effective way to keep costs in bounds for high velocity flights, however, is with high performance nuclear rockets. The additional assumptions in the gaseous fission cases are rather difficult to meet since they presume perfect containment of the reacting fuel and high thrust-weight ratios, regardless of the specific impulse obtained. The difficulty of meeting either of these assumptions explains why gaseous core rocket development to date has been confined to theoretical and some experimental detailed investigations, rather than any large scale development. It can be seen that even if only the lower performance gaseous fission rocket is achieved, interesting additional missions are feasible compared to solid fission systems. Therefore, the state of gaseous fission rockets will be investigated.

One other point should be elaborated from Figure 1 before proceeding. One cannot operate in the high velocity regions by the simple pyramiding of chemical rockets in any reasonable fashion. This is due to the logarithmic nature of propellant weight requirements. This effect is shown by the example in Figure 2, where number of launches required for 60,000 fps, 120,000 fps, and 180,000 fps missions are shown for chemical rockets, solid fission rockets and gaseous fission rockets. Figure 2 assumes all higher performance missions to be achieved by refueling rather than staging. If staging were used, the values shown represent roughly the total weight to be launched, rather than number of launches.

GASEOUS FISSION ROCKETS

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 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 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. 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.

Another family of systems has orignated 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. This is 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 [7]. The scheme has even been suggested, in the "glow plug" reactor, that the fission plasma be contained in a quartz (or similar material) bottle[8]. It is possible to cool the bottle to reasonable temperatures while still heating the propellant to high temperature by radiation, if the bottle transmits most of the radiant energy. This system would yield perfect containment, and is hence a very exciting thought. New ideas are afield in the gaseous fission reactor arena, and an attempt will be made to catalogue the limitations and capabilities of some of these schemes.

The specific impulses theoretically achievable with fission rockets are basically very high, but extremely difficult to achieve in practice since very high temperatures are required. Economy is a problem, since separation (ratio of mass flow rates of unburned fuel to propellant in the exhaust) must be of the order of 10^{-4} to reduce fuel cost to a value comparable to propellant cost, Figure 3. Separation ratio as lax as 1.0, however, does not strongly deteriorate specific impulse. The propellant costs in Figure 3 are for a hydrogen/uranium mixture, assuming a hydrogen price of 25¢ per pound, and a uranium price of $5,000 per pound. It can be seen that for very high performance, fuel costs become very high even with perfect containment. Hence, a fusion rocket would be very desirable due to low fuel costs as well as lack of fission product production.

An analysis was previously performed by making use of the data of Figure 3, and assuming that it would be possible to create fission rocket propulsion systems with thrust-weight ratios comparable to solid fission systems over the entire performance spectrum[6]. This was a very great assumption indeed, for high temperatures not only force gaseous phase heating which complicates containment, they also generate such intense thermal energies that any spurious fluxes may be of large magnitude, and their consequent dissipation may create basic performance limitations. The reason for such a theoretical exploit was to see if a limit on desirable performance existed for solar system transportation, regardless of engineering difficulties. The results were that even for relatively fast transits as far as the planet Pluto (less than two years), specific impulses of much beyond 20,000 seconds were not desirable. This was the reason for showing 20,000 seconds as the highest specific impulse in Figure 1. It can be seen from Figure 3 that this result could have been anticipated, considering the rapid increase in fuel cost which occurs at higher specific impulses. Figure 4, reproduced from Reference 6, shows that optimum operating costs (with high re-use assumptions) were achieved with single-stage vehicles with fuel to gross weight ratio varying from 20% at low velocities to 60% at high velocities.

The problem of achieving even 20,000 seconds with high thrust-weight ratio is still difficult. 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 tends to limit engine performance seriously, as previously indicated. The portions of the rocket system which must remain solid 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 [9, 10]. The thermal loads are intense, and are determined by whether or not any fissioning takes place in the solid material, the fraction of energy that appears in nuclear radiations which will heat the solid surfaces, and by the question of whether the hydrogen propellant is transparent or opaquė 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 Figure 5 related to the fraction of energy release (ζ) which appears as thermally effective nuclear radiation. This fraction is assumed to be 10% in Meghreblian's work, 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 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 Figure 6.

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=o), both for the case of no thermal radiation from the gas (ρ_s=o), and for the case of representative values of thermal radiation from an opaque gas, is shown in Figure 7, reproduced from Reference 10. The opaque gas assumption is used as being appropriate to systems where the propellant is heated by radiation rather than diffusion. The radiator power fraction (γ) is the energy which must be rejected from the radiator system compared to 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 Figure 8 as the lowest curve[9]. Specific impulse ratio is defined as the as the ratio of specific impulse achieved to that of the propellant operating 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 to this curve yield, however, startling results. The curve presented assumed a thrust-weight ratio of about one 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 Figure 8. Although this value can be assumed as high as 20, such estimates make no allowance for pump weights or pressure shells[11]. Such items were considered in the analyses of Reference 12, and the assumptions used here amount to adjusting the generalized expressions of Reference 9 to match the more elaborate single point of Reference 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 Figure 8. A review of radiator assumptions was therefore thought to be in order. All analyses of radiator configurations for nuclear propulsion known to the author have centered around 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 only operate for periods of 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. Secondly, 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. 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 Figure 8 compare the thrust-weight ratios of systems with radiators of 1000, 2000, and 3000° K temperature and, for comparison, a zero area case ($T_R \to \infty$). The radiator weight per unit area may well increase step-wise as temperatures increase due to the need to utilize different materials which are usually heavier at higher temperatures. A constant value was assumed here, and although Reference 11 used a 5 LB/FT^2 radiator system weight, the value of 1 LB/FT^2 of Reference 9 is felt more appropriate in light of the short operating times. It can be seen that even at specific impulse ratios of 20, a thrust-weight ratio of over one 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 ability to reject heat through a radiator and still keep a high thrust-weight ratio, leads to other interesting viewpoints on desirable propulsion systems. If a vortex containment system is used, although the containment may be very good, it is doubtful that it can be made perfect. As indicated, imperfect containment will adversely affect fuel costs. If a "glow plug" system can be made to work, it will have essentially perfect containment. It will have a lower peak performance, however, since the material in the plug itself will be heated by nuclear radiation, and, hence, the fraction of energy dissipated in solid heating will be higher than if no glow plug were used. If in a vortex system, 10% of the energy were dissipated in heating the solid elements, perhaps an additional 3% would be dissipated in the glow plug. The two cases are shown in Figure 9, which indicates only a modest degradation of the glow plug with respect to the vortex system. Utilization of the high temperature radiator obviously strongly influences this conclusion. Note that Figure 9 is plotted vs specific impulse with 2000° K material temperatures rather than specific impulse ratio as in Figure 8.

The utilization of either system at the higher specific impulses involves more than simply the provision of high temperature radiators. In general, hydrogen must be seeded with some other material in order to increase its absorptivity sufficiently to make radiant heat transfer practical. The amount and type of such material would be expected to vary with radiating temperature. In the glow plug, in addition, higher temperatures may shift the radiation spectrum so far toward the ultra-violet that it is moved essentially out of the region of transparency of the plug material. Internal seeding to shift the spectrum may solve this problem. The plug material may have to be at a lower temperature than other solid elements to maintain transparency, and, consequently, the assumed additional 3% loss may be optimistic. It is evident that all detailed problems of these systems have not yet been solved. The essential point is that as they are solved, high thrust-weight systems will be available with high specific impulse.

The curves of thrust-weight ratio presented so far have shown performance limitations for the idealized assumption of no energy deposited in the structure by thermal radiation from the gas stream. The radiators included were required to reject the energy deposited in the structure by the nuclear radiation, and the decreased thrust-weight ratio at high specific impulse was caused both by the inclusion of these radiator weights, and by the fact that more energy is required to generate higher specific impulses. For a constant value of thrust, the energy required is directly proportional to specific impulse, and, consequently, the weight of the basic reactor increases with specific impulse due to the assumption of constant energy density in the solid portions of the reactor. With this simplified assumption, the thrust-weight ratio for the case of no radiator decreases exactly inversely with specific impulse as shown in Figure 8. The thermal load deposited in the structure from the gas stream is strongly dependent upon whether the gas is transparent or opaque to nuclear 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 Figure 10. The assumed value of the radiation parameter ρ_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 one are still possible throughout the regions of interest for interplanetary transportation.

GASEOUS FISSION PROPULSION APPLICATIONS

The previous section has indicated that gaseous fission reactors are capable of yielding thrust-weight ratios of about one up to a specific impulse as high as 20,000 seconds, if appropriately high temperature radiators are utilized. Radiator temperatures as well as the temperatures of all solid structure portions of the propulsion system can be held as low as 2000°K, and still achieve this performance. The containment problem must, of course, be adequately solved.

Many uses can be envisioned for such propulsion systems. The remaining discussion will be limited, however, to the examination of the possibility of the provision of a single highly effective versatile, space transportation system to handle all reasonable 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. For manned exploration, interest centers around 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 Figure 11, 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 only be of interest to civil engineers. Many missions of extreme interest in planetary exploration occur in the velocity regions 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 tough competitor, indeed. It is interesting to note that single-stage gaseous fission rockets of 5000 seconds specific impulse are capable of effectively penetrating these speed regions. (Even at only 2500 seconds, very interesting performance is possible.) Note in Figure 11 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 last century, and with the convenience of year-round operation to all points. A possible analogy between the frigates and galleons of old comes to mind. If slow, clumsy ships were the way to conquer the sea, this paper 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.

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 scale. 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 non-democratic 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. The problem can be seen at a glance in Figure 11. 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. If the lesson inherent in Figure 11 is not learned, our fate is to be driven violently about by our competition. The question is not only what nuclear spaceships can do <u>for</u> you, but what nuclear spaceships might do <u>to</u> you.

THE DESIGN OF SPACESHIPS

In the opinion of the author, too many words have been written about the design of aircraft and the design of guided missiles, and too few about the design of spaceships. Manned space flight is the subject of this paper, and therefore, speculation on manned spaceship design is appropriate. Guided missiles have grown up in the the throw-away-everything atmosphere of bullets. Recovery and re-use has been looked upon as something mysterious, and achievable only at great expense and low reliability, partly because very little effort has been put into it. A difficult component development problem occurs since the equipment is rarely available for examination afterwards. It is hard to get enthusiastic about recovery, when it is not part of the operational scheme. Redundancy of design has usually been held to a minimum. High acceleration loads are tolerated both on launch and re-entry, sometimes because the equipment can take them, sometimes because they cannot be avoided, and sometimes because they are very desirable for trajectory efficiency. Aircraft, on the other hand, are re-used constantly, contain many redundant items so that missions are almost always completed, and, fortunately, constantly utilize almost perfect recovery reliability. This creates a tendency to feel that anything which breathes air and has wings is always economical to operate. This thought, if extended blindly into regions of poor payload carrying ability, will lead to uneconomical operation, regardless of past history.

The true spaceship will be neither missile nor airplane. It must have the extremely high performance of the missile, in fact, much higher than any built today. It will probably be single stage. Wings and/or air breathing engines, which can only aid in the first 5% of the mission, will not be tolerated if they should unduly deteriorate the performance of the other 95% of the mission. The spaceship will be as redundant and reliable as a transport aircraft, and there is no reason why it cannot land as reliably and safely as an aircraft. Since it will carry soft cargo and normal people, it will throttle its engines after take-off, and use lifting entries at landing in such a way as never to place more than perhaps 2g acceleration on the passengers except in emergencies. It will be able to operate at variable specific impulse levels for optimum economy with different missions. Typical design fuel weight and specific impulse as a function of velocity of the mission are shown in Figure 12.

It should be possible to make spaceship operation as safe as any other form of transportation. Since the fissionable material is always in the gaseous phase, if it should have to be jettisoned in case of emergency, it already is prevaporized for safe deposit in space or in an atmosphere. Assuming that calculations on gaseous fission reactors for booster propulsion give a reasonable estimate of reactor characteristics for spaceships, the nuclear inventory on board would be less than 10 kilograms for a BeO moderated system[13]. Thus, atmospheric contamination either in normal operation or during emergencies is truly negligible. A solid core fission rocket accumulates fission products during use, and hence is quite radioactive on return and difficult to leave, approach, unload, or maintain. A fundamental advantage to the gaseous core is that the fission products can be safely disposed in space, or the atmosphere, and the vehicle landed by aerodynamic means (auxiliary chemical propulsion would be needed on atmosphereless bodies). It is, consequently, safe to approach and work around essentially immediately after landing. A glow plug reactor would have perfect containment at launch, but retain fission products at landing, unless the plug were jettisoned. A vortex containment reactor would not be as perfect at launch, although it has the same inherent nuclear safety once aloft. Thus, the gaseous fission rocket has the potential to meet the most stringent safety requirements devised.

Landing Gear Design

In spite of a fundamentally safe operation, it is conceivable that for emotional reasons, such ships would not be permitted to take off directly from the earth on nuclear thrust. Auxiliary propulsion might, therefore, be required. If so, it could be one of three categories; namely, liquid chemical rocket, solid chemical rocket, or air breathing engines and wings. In view of the tremendous performance potential of nuclear rockets, it is perfectly clear that only minimum auxiliary chemical propulsion should be tolerated. The proper perspective is maintained if it is realized that these auxiliary chemical systems correspond, in a good spaceship, to landing gear design in a transport airplane. It should also be pointed out that the spaceship is primarily a vacuum-dwelling rocket-thrust vehicle which is, however, interested in periodic return to planetary bodies. If the proper shape and construction of propellant tanks and other structure can give better re-entry conditions, without severe weight penalties, it would be desirable. This is analogous to wing flap design in aircraft, another auxiliary function. In actual operation, this may be only a desirable emergency provision, since the extra price of fuel involved in a power letdown is not great. True spaceship designers will start to come forth whenever the potential for handling these auxiliary requirements without appreciably crippling basic spaceship performance becomes evident.

Shielding

Shielding of biological payloads from both the propulsion system and the natural environment is of extreme importance. Fairly extensive shielding analyses have been made which indicate that if multiple materials are used as appropriate, and clever use is made of propellant, food supplies, and equipment to aid in shielding, shield weight penalties are not overburdening in terms of nuclear rocket payloads[2, 14, 15]. The shield weights for nuclear space transports are reasonable even though shield weights have been one of the major problems of the nuclear aircraft program. Although the reactor power of an aircraft is several hundred times lower than that of a spaceship, shield 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 minutes 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 aircraft 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 shield 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 shield weights required for the nuclear spaceship are an order of magnitude smaller than those for nuclear aircraft. As indicated previously, the use of gaseous fission reactors greatly reduces shielding requirements after landing. The higher performance ship thus would almost certainly be easier to handle and require less extensive ground facilities than either a solid core nuclear rocket or a nuclear aircraft.

An extensive shielding analysis will not be presented here, but a few remarks based on the References (which did not consider gaseous fission reactors in detail) are appropriate. A cursory analysis of the effects of the greater energies inherent in higher specific impulses and high velocity increments indicates no substantial effect on shield weight. This is primarily due to the logarithmic effects of shield thickness; a little more shield thickness gives a lot more protection if the shield was already thick enough to have reduced the flux by many orders of magnitude. If this were not so, the shield weights on electrical propulsion systems would be completely prohibitive since such systems use much more energy to achieve equivalent performance[16]. The effects of major solar bursts are confused at the present time. If they do remain a problem (there is some evidence that even major bursts may not be), then the very short flight times or out-of-ecliptic capabilities possible with high performance gaseous fission ships becomes even more desirable.

Other Propulsion System Considerations

The propulsion system thrust-weight ratios of Figures 8 through 10 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 as much as indicated due 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. An interesting sidelight of the latter point is that the throttling of gaseous fission 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. 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 highest 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 to designing an optimum engine for each specific impulse, is shown in Figure 13. It is practical to consider such two-phase engines. Figures 12 and 13 are not consistent. Figure 12 was derived assuming substantially higher thrust-weight ratios than even the peak values of Figure 13, and no attempt was made to analyze two-phase engine operation. A more accurate derivation of Figure 13, followed by a re-derivation of Figure 12, would yield more accurate spaceship design parameters.

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 due to 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 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 for 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 to operating on hydrogen, is shown approximately in Figure 14. 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.

CONCLUSIONS

The capabilities of nuclear rockets are such that they possess great potential for future manned space flights.

Utilization of solid core nuclear fission rockets of the Rover type as upper stages on chemical vehicles gives substantial performance improvements. This development is justified.

Total velocities of several hundred thousand feet per second, with payloads sufficiently large for manned operations, will be required for future solar system exploration. Gaseous fission rockets possess the theoretical capability of achieving this performance economically.

The single-stage vehicles of high performance potentially achievable with gaseous fission rockets should be eminently suitable for manned exploration, due to overall vehicle simplicity, economy of operation, and versatility. Emphasis on very high speed capability is a better way to assure long useful life than emphasis on very large payload capability.

Relatively new concepts in gaseous fission reactor design involving the heating of the propellant by radiation from the fission plasma show promise as reactor design concepts. High temperature radiators appear to be the key to the achievement of high thrust-weight ratios at high specific impulses with these systems. Appropriate research is desirable in these areas.

The improvement in thrust-weight ratio with high temperature radiators re-opens the question of propellants other than hydrogen. Water, in particular, should be investigated.

Research and development efforts should, in general, be concentrated where the potential pay-off is greatest. Single stage, very high velocity spaceships are almost the toughest competitors imaginable in the solar arena. To be unprepared for their advent could be somewhat more than disadvantageous.

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FIGURE 1

CARGO TRANSPORT COST
(SINGLE STAGE VEHICLES)

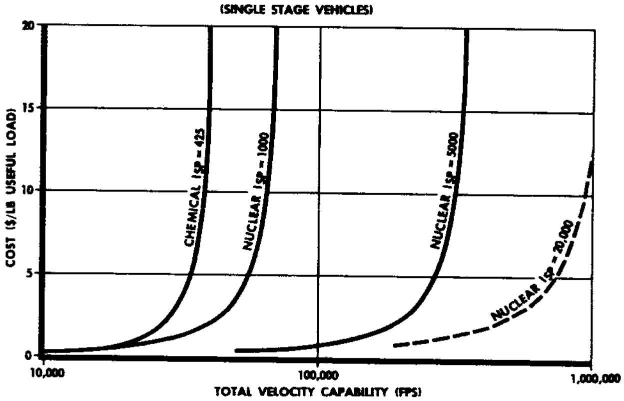


FIGURE 2

FUTURE VEHICLE REQUIREMENTS

PROPULSION TYPE	SPECIFIC IMPULSE	VELOCITY PER STAGE	NO. LAUNCHES TO ACHIEVE 60K FPS	120K FPS	180K FPS
CHEMICAL	425	30,000	18	324	5832
SOLID CORE NUCLEAR	1000	60,000	1	15	225
GASEOUS CORE NUCLEAR	5000	300,000	1	1	1

FIGURE 3

COST OF PROPELLANT FOR FISSION REACTOR ROCKET

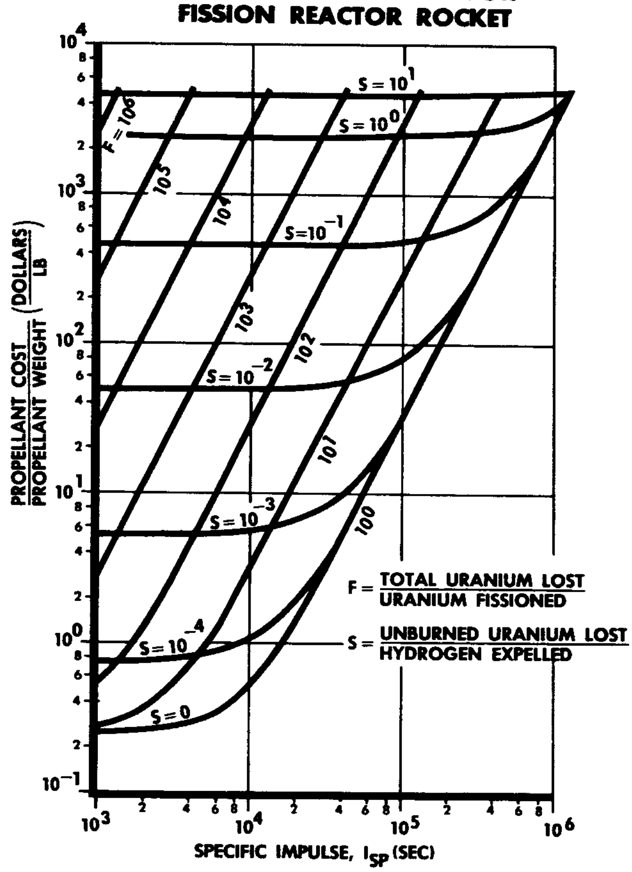


FIGURE 4

USEFUL LOAD FRACTION

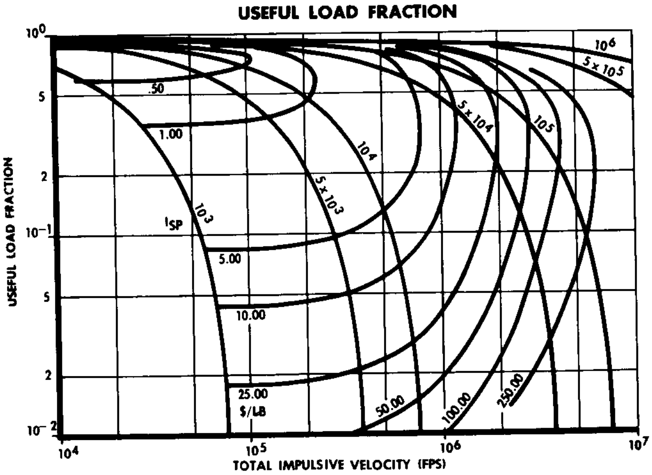


FIGURE 7

INFLUENCE OF RADIATOR POWER ON SPECIFIC IMPULSE

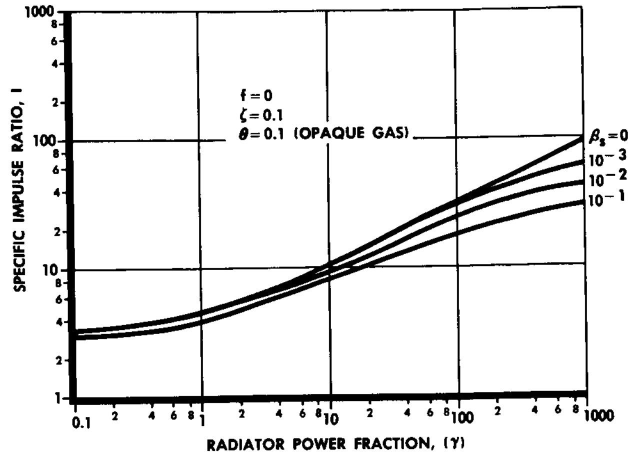
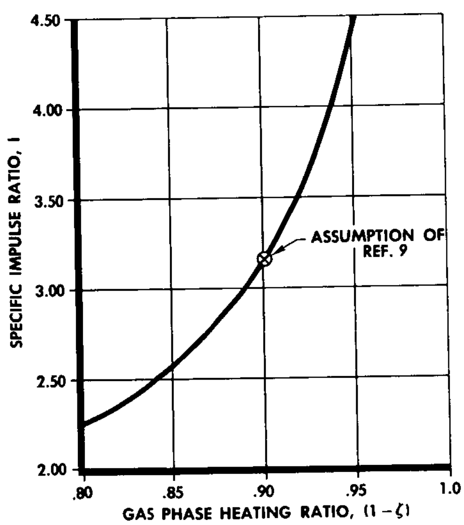


FIGURE 5

SPECIFIC IMPULSE VARIATION WITH REGENERATIVE COOLING ONLY



EFFECT OF RADIATOR TEMPERATURE ON ENGINE THRUST TO WEIGHT RATIO

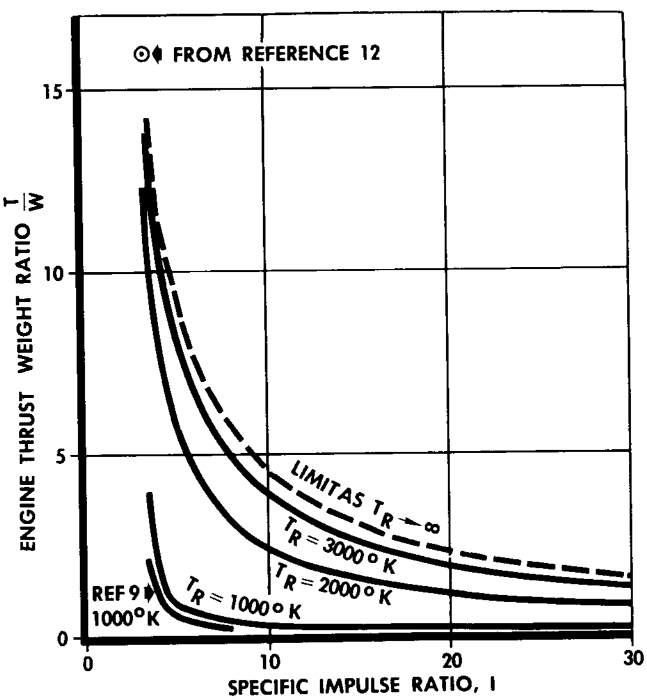


FIGURE 6

EFFECT OF REFLECTOR THICKNESS ON REACTOR PERFORMANCE

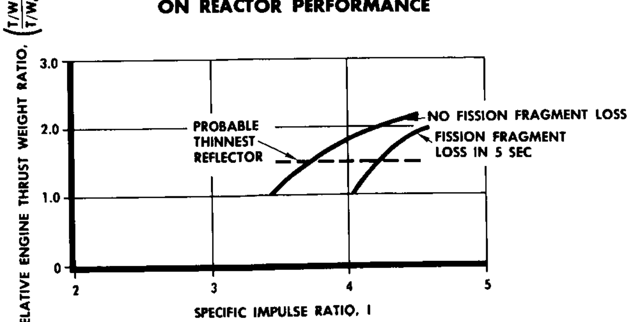


FIGURE 9

COMPARISON OF TWO REACTOR CONCEPTS

(RADIATOR AND REFLECTOR TEMPERATURE = 2000 °K)

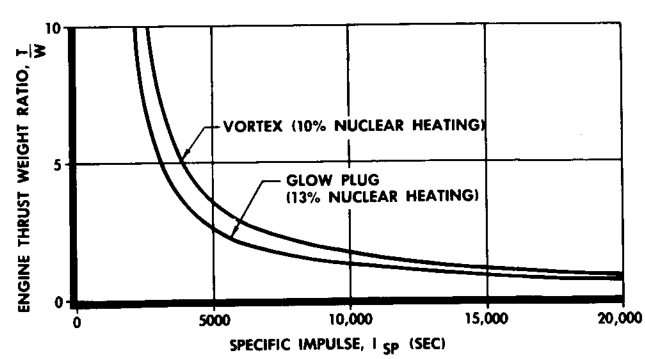


FIGURE 10
EFFECT OF THERMAL RADIATION LOAD TO SOLID ELEMENTS

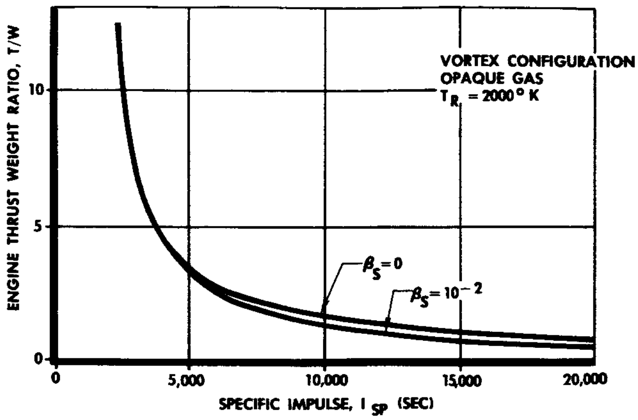


FIGURE 11
PAYLOAD VELOCITY CAPABILITIES CHEMICAL & NUCLEAR VEHICLES
(PRACTICAL STAGING)

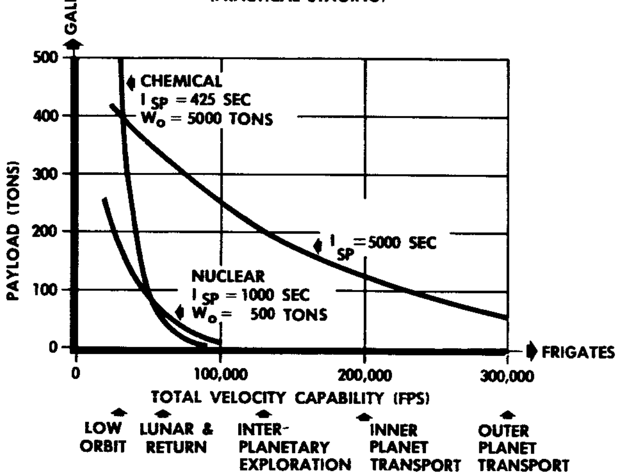


FIGURE 12
SPACESHIP DESIGN PARAMETERS

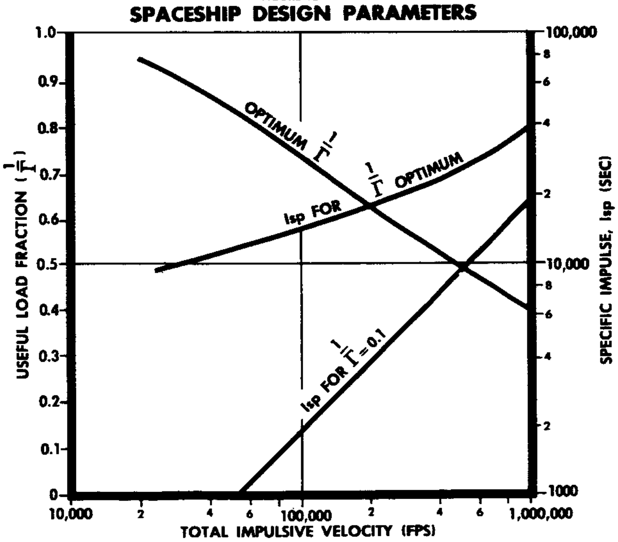


FIGURE 13
EFFECT OF TWO PHASE OPERATION ON ENGINE THRUST WEIGHT RATIO

$T_R = 2000°$ K

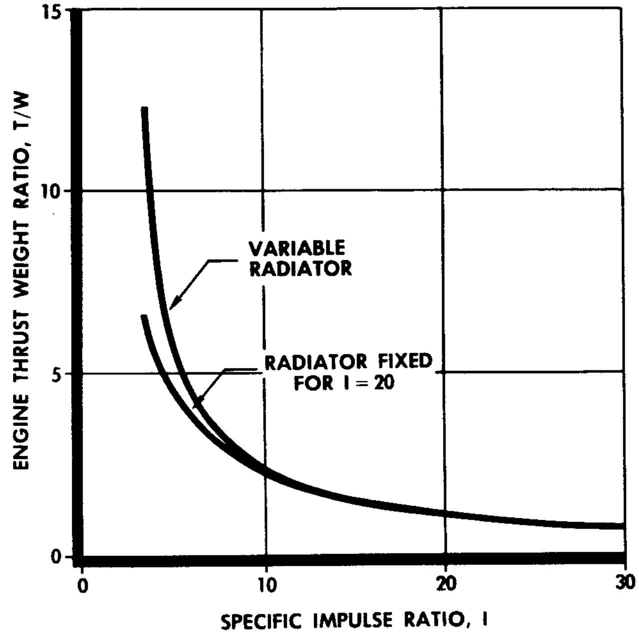


FIGURE 14
ENGINE PERFORMANCE FOR HYDROGEN AND WATER PROPELLANTS