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TECHNICAL MEMORANDUM

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PROPULSION AND POWER GENERATION

Presented at the Manned Planetary Mission Technology Conference, Lewis Research Center, Cleveland, Ohio, May 21-23, 1963

(NASA CR OR TMX OR AD NUMBER) (CATEGORY)

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON July 1963

[redacted]

FOREWORD

This document comprises material presented during one session of the Manned Planetary Mission Technology Conference, Lewis Research Center, May 21, 22, and 23, 1963. In order to expedite release to the conferees, the papers are being published with minimum editing and retyping of the original manuscripts. Thus the usual NASA format and style have been compromised.

The purpose of the conference was to explore the possibilities and problems of manned planetary space flight. The results and contemplations of the individual papers should in no sense be regarded as a part of NASA plans and programs. For this reason, the contents of this document are limited for the present to NASA personnel.

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By F. L. Williams

NASA George C. Marshall Space Flight Center

LAUNCH VEHICLE IMPLICATIONS FOR MARS EXPLORATION

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OBJECTIVES

The objectives of this paper are to give a brief insight into the launch vehicle implications for a manned Mars expedition and follow-on exploration; and secondly, to show the effects of such a mission requirement on the launch vehicle system. Due to the complexity and intricacy of the overall mission, vehicle system, and their interrelationships, only the very broad and salient points will be discussed.

BACKGROUND AND SOURCE OF DATA

The George C. Marshall Space Flight Center (MSFC) has, over the past three years, conducted numerous in-house as well as contracted studies in the area of launch vehicle systems and missions analysis. Of these approximately 100 studies, which have either been completed or are in progress, about 25 have dealt with the question of launch vehicles and missions including space systems design and investigations relating to manned planetary explorations. Although the prime mission of MSFC is development and operation of launch vehicle systems, it is necessary to study overall missions and space system designs in order to properly assess the launch vehicle implications and determine design requirements of such launch vehicle systems.

The major source of the data presented in this paper was in-house and contracted investigations including the following:

1. NOVA Launch Vehicle System Studies being conducted in-house at MSFC and by General Dynamics/Astronautics Division and Martin-Marietta Corporation, Baltimore Division (Ref. 1 and 2).

2. NOVA Launch Facilities Studies being conducted by the Launch Operations Center (LOC) and by Martin-Marietta Corporation, Denver Division (Ref. 3).

3. Early Manned Planetary-Interplanetary Roundtrip Expedition (EMPIRE) Studies being performed by General Dynamics/Astronautics Division, Lockheed, and Ford/Aeronutronics Division (Ref. 4, 5, and 6).

To a lesser degree, some data were used from the Advanced Lunar Transportation Studies being conducted in-house at MSFC and by contractors (Ref. 7 and 8). Also, some information from the Orbital Operations Studies under the direction of MSFC was used (Ref. 9).

MISSION REQUIREMENTS AND IMPLICATIONS

Two of the first questions which must be answered, when studying the manned Mars mission from the launch vehicle standpoint are: (1) How does one get to and return from Mars, i.e., the trajectory or flight mode? (2) What vehicle or weight is required for the space vehicle system to make such a flight? Figure 1 illustrates a typical trajectory or flight mode which might be used for a manned Mars mission. Although there are various modes that could be used, but for this paper, the one illustrated has been assumed. In general, the illustration represents roughly a 14-month total mission duration - - originating in an Earth orbit, departing and arriving at the vicinity of Mars in approximately 4 months, a 2-month staytime on the surface or in the vicinity of Mars, and a return trip of approximately 8 months duration. Figure 2 illustrates, in more detail, some of the basic assumptions as well as the base vehicle requirements and philosophy. Again, Figure 2 illustrates only a typical method by which such a mission could be accomplished in terms of the flight mode as well as the space vehicle required. As illustrated, two ships are assumed, a manned ship and a cargo ship. The manned ship would depart with one propulsion mode, illustrated by Unit 1. This propulsion mode would be required to escape the Earth's gravitational field. Unit 2, or propulsion stage 2, would be used for retro (rocket braking) maneuver into a Mars orbit. Unit 3, would be used for a propulsion maneuver from Mars orbit to Mars escape for the Earth return trajectory. Unit 4 would be the manned capsule used for hyperbolic re-entry into the Earth's atmosphere and landing on the Earth's surface. Units 2 and 3 of the manned ship could be combined into a common stage, i.e., using tank staging and reigniting the same engine. The manned ship would provide transportation for the personnel only from the Earth orbit to a Mars orbit and return.

In order to provide landing capability on the surface of Mars, a cargo ship was assumed to be required, based on the typical example illustrated in Figure 2. The cargo ship, as in the case of the manned ship, would use a rocket propulsion stage (Unit 1) to escape the Earth's gravitational field, and a propulsion stage (Unit 2) for braking maneuver into a Mars orbit. The payload for the cargo ship was assumed to be a Mars excursion module, i.e., the vehicle which would provide the capabilities for a manned landing on the surface of Mars as well as return to a Mars orbit and, of course, rendezvous with the manned spacecraft. As illustrated in Figure 2, this would be maneuvers or Units 5 and 6. Based on the assumptions regarding propulsion, i.e., nuclear propulsion for Units 1 and 2 for both ships and Unit 3 for the manned ship, and chemical propulsion for Units 5 and 6, it has been determined that the useful weight (payload) arriving in a Mars orbit for both manned and cargo ships would be approximately equal. Also, Units 3 and 4 of the manned ship would be equal in weight to Units 5 and 6 of the unmanned ship. For the purpose of this paper, it was assumed that Units 1 and 2 for both ships would be identical. The landing capability provided by the cargo ship would depend very largely on the following considerations: The type of propulsion used for Units 5 and 6; the characteristics of the Mars atmosphere; the vehicle designed; the number of people to be transported per ship; the redundancy required in terms of one ship as well as standby ships; the staytime on Mars; and the characteristics of the Mars surface. Preliminary studies indicate that, with the assumed payload to be delivered to Mars orbit by the cargo ship, the following could be accommodated:

1. Two, two-man Mars excursion vehicles, each with a two to three week surface staytime.

2. Several small probes that could be launched from orbit into the Mars atmosphere to obtain scientific data as well as for Mars surface exploration.

The weight required in an Earth orbit for one ship is illustrated in Figure 3. In view of the extended flight time, mission complexity, and the hazardous environment involved in such a trip, certain redundancies are considered necessary. As can be seen in the illustration, the weight of the ships would vary with time as well as the amount of solar activity which would be encountered over the years. For 1971, when a low solar activity is expected, the weight of the ship varies from approximately 1.0 million lb, or a little over, for a nuclear system using a Hohmann type flight mode to approximately 3.0 million lb for a chemically propelled system using Hohmann transfer. For the purpose of this paper, it was assumed that each ship would weigh approximately 1.5 million lb, which corresponds to a ship in the 1971 time period using nuclear propulsion stages and a relatively fast transfer (14 months total trip time) as illustrated in Figure 1. Also shown are the weight requirements for a ship in 1979 where not only increased velocity is required, but additional shielding is required due to the high solar activity which is anticipated during that time period. On the right of Figure 3, are given the assumptions used for this paper in terms of weight required in an Earth orbit of approximately 300 nautical miles altitude for a manned Mars landing and return. Certain redundancies will be required: Three ships are considered near minimum and four ships would constitute a desirable exploration, thus yielding 4.5 million lb for the minimum and 6.0 million lb in orbit for a nominal expedition.

LAUNCH VEHICLE REQUIREMENTS

The establishment of launch vehicle requirements at this time is considered premature. Due to uncertainties in the mode selection, trip time, type of propulsion to be used and weight of the space ship for a Mars exploration, only typical or representative requirements (or desirements) can be established. Since the launch vehicle to be used for the Mars trip will also have other missions, they too must be taken into consideration in establishing requirements. Unfortunately these other missions, such as lunar base, orbital operations, global logistics, etc., are not well defined either and, therefore, tend to complicate the "launch vehicle requirement" picture even further.

Of all the missions (or desirements) analyzed to date, the Mars mission is the most critical and places the highest requirement on very large (1.0 million lb or more) payload capabilities for the launch vehicle. Past studies in the area of economics and optimum sizes of future launch systems have indicated that "the largest vehicle is not normally the most economical, particularly if only a small number of total flights are required (approximately 100 or less)."

It was concluded that launch vehicles of the 0.7 to 1.0 million lb payload capability should be considered as the most promising next system for development and operation after SATURN V. The logic for this conclusion is the economic considerations mentioned earlier and the fact that very large vehicles (approximately 2.0 million lb payload capability) would require extensive advances in technology and considerable time and cost to achieve.

Figure 4 shows the launch attempts required versus probability for success for two sizes of NOVA vehicles: First, a vehicle which could place a 1.5 million lb spacecraft into orbit with two successful launches and, second, a vehicle which could place a 1.5 million lb ship into orbit with three successful launches. The payload capability for two packages per spacecraft would be from 900,000 to approximately 1.0 million lb per vehicle. This would provide roughly a 10 percent contingency in the launch vehicle system, as well as take into consideration the fact that each ship cannot be broken down into two identical parts. A similar contingency was assumed in the case of the three packages per Mars ship. A payload capability per launch was computed to be between 700,000 and 800,000 lb. Considering first the two successful launches per Mars ship, eight launches would be the minimum required to place four ships into orbit. As can be seen in Figure 4, the probability of success would be extremely low, roughly 10 percent. The probability of success illustrated is that of successful launch, orbital rendezvous, docking, and checkout of the spaceship itself, but does not include launch from orbit or the remainder of a manned Mars mission. Eight launches would, however, provide roughly a 60 percent probability of success that three out of the four ships would be checked out in orbit and available for launch. Due to the expense, not only of the launch system and transportation, but of the Mars ships, it is felt that a 60 percent probability would be too low. Assuming that a 90 percent probability of 3 out of the 4 ships would be a minimum, then as shown in Figure 4, 11 launches would be required. This would also indicate the probability that all of the four ships would be available for the expedition. Using NOVA vehicles with 700,000 to 800,000 lb payload capability, 17 launches would be required to provide the same probability of success as the 9 launches of the 1.0 million lb capability NOVA. To provide a bare minimum manned Mars landing and return capability, based on the two-ship scheme, it can be seen in Figure 5 that seven NOVA (1.0 million lb capability) launch attempts would be required. This would provide a 90 percent probability that two ships could be successfully assembled and checked out in orbit - - one cargo and one manned ship. Such an operation would provide no gross redundancy and is considered by the author at this time to be too marginal for consideration. Three ships are considered minimum, viz., two manned ships and one cargo ship. As shown in Figure 5, only two additional launch attempts would be required to provide the same probability of success. Such an investment is considered desirable.

Since the launch vehicles to be used for manned planetary exploration will also have other applications, a mission model has been developed in order to assess the implications of the varied mission requirements for a NOVA class vehicle. Figure 6 illustrates a typical mission model over a 10-year operational period. This mission model includes 3 manned planetary expeditions over the 10-year period. This is illustrated by the tall bars in the second to third operational year, fifth operational year, and ninth operational year. Other missions assumed were the establishment and support of a 20- and 50-man lunar base, test launchings of the manned Mars ship, as well as various orbital operations required to support the Mars ship development, and the manning of the Mars exploration. Since the manned Mars requirement places the most demanding tasks on the launch systems and facilities, it has been assumed that the manned lunar base launch requirements, the development of the Mars ship, as well as orbital operation flights, would be spread over the 10 years so as to not coincide or conflict with the manned Mars expedition. As can be seen, these other requirements, based on the assumptions made, do not result in an even launch rate over the 10-year operational period. This is primarily because of the very high launch rate required for the manned Mars mission, and the basic assumption that the overall accumulation of the Mars ship and checkout would be accomplished within a six-month period. The assumptions for the larger mission model (larger number of flights) would include 3 3-ship expeditions to Mars, the development and support of a 50-man lunar base, 3 large space stations, and the development of the Mars ships over a 10-year period. This would result in 86 launches of a 1.0 million lb payload capability NOVA and 114 launches of a 800,000 lb payload capability NOVA. The cost of such a launch vehicle program, including development, facilities, and operational flights would be on the order of $15 billion (FY 63 dollars in zero inflation rate) thus constituting a rather sizeable program. Even with the program of that magnitude, the manned Mars mission requirements are extremely critical.

The Mars mission launch facility requirements necessitate a very high launch capability which would not be fully utilized by the remainder of the mission desirements during the overall program. Based on the assumptions used, approximately one-third utilization will be made of the facilities over the complete 10-year period.

LAUNCH VEHICLE SYSTEMS

A wide range of launch vehicle systems has been studied by MSFC during the past three years. The following vehicle descriptions will include only four of those presently under consideration. The vehicles have been broken down into three classes. Class I is made up of state of the art vehicles and is illustrated in Figure 7. This vehicle would utilize 16 up-rated F-1 engines in the first stage, burning liquid oxygen and kerosene. Each of the up-rated F-1's would have a sea level thrust of 1.8 million lb. The first stage would be approximately twice the diameter of the present SATURN V first stage. The second stage would utilize two M-1 engines, each with a thrust of 1.5 million lb (vacuum), burning liquid oxygen and liquid hydrogen. The overall vehicle, including the transtage, would be approximately 240 feet high. Above this would be either a payload which would be transported to orbit or a chemical or nuclear third stage, plus payload. For orbital flights, studies have indicated that a transtage would be the most desirable solution for final velocity vector control, payload attitude control, rendezvous, and docking and, as shown in Figure 7, utilizes small aerozine $50/N_2O_4$ engines. The transtage would also be utilized to house the guidance and control systems, instrumentation, and telemetry for flight development. The vehicle would have the payload capability on the order of 750,000 lb to orbit and, if used, would require three successful launches to place the required 1.5 million lb Mars ship into orbit with the contingencies mentioned earlier. The vehicle lift-off weight (23 million lb) represents a vehicle which is optimized for orbital transportation with two stages, plus a transtage and incorporates a propulsion section recovery system. After first stage burnout, the engines and thrust structure with associated equipment would be separated from the first stage tank and follow a ballistic trajectory. A drag parachute would be used for stability during re-entry. After reaching subsonic velocity, large parachutes would be deployed and just prior to water impact retro rockets would be fired to minimize impact velocity. Studies have shown a considerable economic saving with propulsion system recovery. A cut-away drawing of the F-1/M-1 vehicle is shown in Figure 8. Numerous studies have been performed on tank configurations, such as the multi-cell tank arrangement in both first and second stages shown in the illustration.

Figure 9 illustrates an advanced NOVA vehicle concept for Class II. Such a vehicle would require the development of new engine and propulsion system concepts. This vehicle utilizes 1.0 million lb thrust liquid oxygen/liquid hydrogen engines with high combustion chamber pressure (3,000 psi) in both stages. The first stage would consist of 18 of the engines wrapped around a zero length or up to 10 percent length truncated plug. Two identical engines would be used in the second stage; however, they would be used as individual modules here. The advantage of the plug concept, if proven successful, would be altitude compensation during flight. In effect, this gives a variable expansion ratio and provides specific impulse gains during the atmospheric as well as vacuum portions of the ascent trajectory. This vehicle with its approximately 14 million lb lift-off weight, incorporating full first stage recovery by parachutes, retro rockets, and water landing is the lightest of the NOVA vehicles presently under consideration, having 1.0 million lb of payload capability.

Another Class II vehicle, which is presently being investigated in more detail, is illustrated in Figures 10 and 11. This is a two-stage to orbit vehicle with a fully recoverable first stage. This concept utilizes a shaped first stage for ballistic water recovery. It incorporates 4, 6.25 million lb thrust bell nozzle, liquid oxygen/liquid hydrogen engines with a 3,000 psi combustion chamber pressure. Such a first stage would be approximately 140 feet in diameter and 125 feet tall. This first stage concept is also under study utilizing different engine systems, i.e., forced deflection type nozzles, as well as the plug engine concept. Each of these other engine concepts results in a shorter first stage, as well as provides advantages in terms of center of gravity location for recovery dynamics. The vehicle uses two M-1 engines in the second stage and is sized for approximately 1.0 million lb payload capability into a 300 nautical mile orbit.

Figure 11 shows a cut-away view of the vehicle. The first stage has a large spherical oxygen tank in the center with a toroidal oxygen tank wrapped around the liquid oxygen tank. The first stage would burn out at a velocity of approximately Mach 5 to 6 and the re-entry environment for the first stage would be such that no heat protection would be required for the alluminum type structure. Parachutes would be deployed after reaching subsonic velocity and retro rockets are included in the nose of the first stage to reduce the landing velocity prior to water impact. Weight has been assumed for salt water protection of the overall stage, although it would not be required for the re-entry environment. The vehicle first stage would float up-right with only 15 to 20 percent of the nose being submerged in the water. Such a configuration would keep the engine and critical elements of the stage high above the ocean surface, thus protecting them from the very hostile environment. The stage would be returned to a refurbishment and checkout site prior to re-launch.

Figure 12 shows a flight profile of a Class III NOVA concept. This vehicle, an advanced unconventional system, utilizes air-augmentation during the atmospheric portion of the ascent trajectory. Air is taken in and mixed with the rocket exhaust during the early portion of the atmospheric flight as illustrated in

View 1 of Figure 12. After leaving the sensible atmosphere the mixing ring is jettisoned as shown in View 2. After achieving orbital velocity the conical payload is separated as shown in View 3. The stage effects a retro maneuver as illustrated in View 4 and re-enters ballistically as shown in View 5. By the use of large parachutes and retro rockets, the vehicle will be landed in the water as illustrated in View 6.

A view of the Class III concept is shown in Figure 13. It is basically a single-stage to orbit vehicle utilizing liquid oxygen and hydrogen as propellant with air-augmentation. Although numerous configurations are possible, the one illustrated is considered representative and employs a large liquid hydrogen tank in the rear portion of the vehicle. The toroidal liquid oxygen tank with a plug type cluster of small engines are wrapped around the periphery of the vehicle. A conical payload is shown since it provides good inlet aerodynamics. The lower half of the figure illustrates the configuration during aerodynamic ascent with the air inlet coming into a mixing chamber at the exhaust plane of the engine nozzle. In the mixing chamber aft of the engine the intake air is mixed with the rocket exhaust, thus providing thrust as well as specific impulse augmentation. Additional trade-offs are being made for pure mixing, partial mixing and burning as well as true after-burner type concepts. Although additional thrust, as well as specific impulse augmentation can be obtained by burning rather than mixing, a much greater design problem must be solved in order to take advantage of the additional gain. Design and performance data presently available indicate that pure mixing of the rocket exhaust gases with the intake air would be sufficient to make the performance of the concept attractive. The mixing ring would be jettisoned and the air inlet duct closed at approximately Mach 6 and the vehicle would have the configuration shown in the upper half of Figure 13 during the remainder of the ascent trajectory. Very high expansion ratios can be obtained by such a configuration, thus providing very high specific impulses for the liquid oxygen/liquid hydrogen rocket engine systems.

Figure 14 presents nominal trajectory data on the Class III concept. The average specific impulse over the complete ascent trajectory, as shown, would be approximately 500 seconds.

Figure 15 shows the total cost for launch facilities and operation for various types of vehicles and includes development, as well as the operational portion of the overall launch systems lifetime. As can be seen, the launch rate capability designed into the Atlantic Missile Range (AMR) facility has a dynamic influence on the overall AMR cost. As was shown in Figure 6, approximately 86 launch attempts would be required to satisfy the numerous missions for the NOVA vehicle over a 10-year period. If launch facilities were constructed to satisfy the 86launches over the 10-year period on a level launch rate basis, a capability of some 4 launches per 6 months would be required. As can be seen, the total AMR cost for such a constant launch rate would be on the order of 1.5 billion dollars. However, to satisfy the manned Mars mission, as assumed, a launch capability of some 12 launches in a 6 month period would be required for a 1.0 million lb payload capability NOVA. The requirement would raise the total AMR cost to roughly 2.5 billion dollars or impose a 1.0 billion dollar increase for AMR cost. This would represent approximately seven to eight percent of the total NOVA launch vehicle system costs to satisfy the manned Mars mission high launch rate requirements.

LAUNCH VEHICLE SYSTEM COST TRENDS

In summary, Figure 16 shows trends of launch vehicle cost parameters for various classes of vehicles. Definitions of Class I, II, and III vehicles are:

Class I - F-1, M-1 engines with recoverable first stage propulsion section.

Class II- Two-stage to orbit pure rocket system with a fully recoverable first stage.

Class III -An unconventional single-stage to orbit fully recoverable system utilizing air-augmentation.

Each of the classes are presented in terms of their operational availability and are 1974, plus or minus one year; 1977, minus one plus two years; and 1979, minus one year plus two years, respectively. The implementation cost which includes research and development, facilities, and GSE range from 5.5 billion dollars for Class I to approximately 8.0 billion dollars for Class III. The direct and total operational cost for each of the classes of vehicles are given for two program levels. The direct cost would be the cost necessary to procure, test, checkout, and launch a developed vehicle. The total cost includes the amortization of the implementation cost over the operational period, i.e., in the case of Class I the amortization of 5.5 billion dollars over 100 launch attempts during the 10-year period. The left bar for each of the classes of vehicles gives the direct and total cost for 100 NOVA launch attempts over a 10-year period. As shown for a program of this magnitude, (12 to 15 billion dollars), the Class I or earlier vehicle is the most attractive from an economic standpoint. Not only would it be the least expensive system, but it would also be available much earlier. The right bar for each class illustrates the cost associated with a program approximately four times as large, i.e., 400 flights over a 20-year period. Such a program would result in approximately a 30 billion dollar expenditure for launch vehicle systems. One factor that has not been included in Figure 16 is the cost associated with postponing the availability of the NOVA capability, i.e., what cost should be set aside for the later availability of a NOVA and the mission capabilities which it would provide the U.S.

CONCLUSIONS

After numerous studies in the majority of the areas associated with manned Mars expeditions, a large number of conclusions can be drawn. No attempt will be made here, however, to do so. On the basis of the brief data, and this paper, a few highlights in the area of conclusions can, however, be stated, as well as several critical or problem areas that relate to the overall system or mission. It should be understood that no attempt has been made to list all or even critical problem areas associated with technology or research, relative to elements of the overall manned Mars expedition system. Some conclusions and remarks relative to problem areas are listed below:

1. A launch vehicle of the NOVA class is technically feasible and could accommodate the manned Mars mission, as defined, or of the order indicated.

2. The development time for NOVA will be from 7 to 9 years after system definition and program approval. This would yield an operational system by around 1974, for a state of the art configuration, or an advanced unconventional configuration by around 1979.

3. In order to justify a NOVA vehicle, one or more of the following requirements will probably have to materialize:

   a. A large lunar base.

   b. Manned planetary landings and/or exploration.

   c. Large civilian and/or military orbital operations.

4. Of all the missions studied to date, the manned planetary is the most complex and the most demanding on the launch vehicle. From the overall mission standpoint, it is considered necessary, however, to accept additional complexities in the launch vehicle to simplify the total mission.

5. The transportation cost to deliver a Mars expedition into orbit will range from 0.75 billion dollars, assuming a very large overall NOVA program, to 1.5 billion for a large (100 launch) NOVA program. It will probably be closer to 1.5 billion dollars unless we send a lot of people to Mars. This does not include the development of the Mars ships or their procurement along with spares and assumes a lot of other people use NOVA.

6. Two critical problems from the launch vehicle system standpoint are:

a. Definition of Mars ships (wt., vol., etc.).

b. Total time allowable to accumulate ships in orbit.

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TYPICAL TRAJECTORY/FLIGHT MODE FOR MANNED MARS MISSION

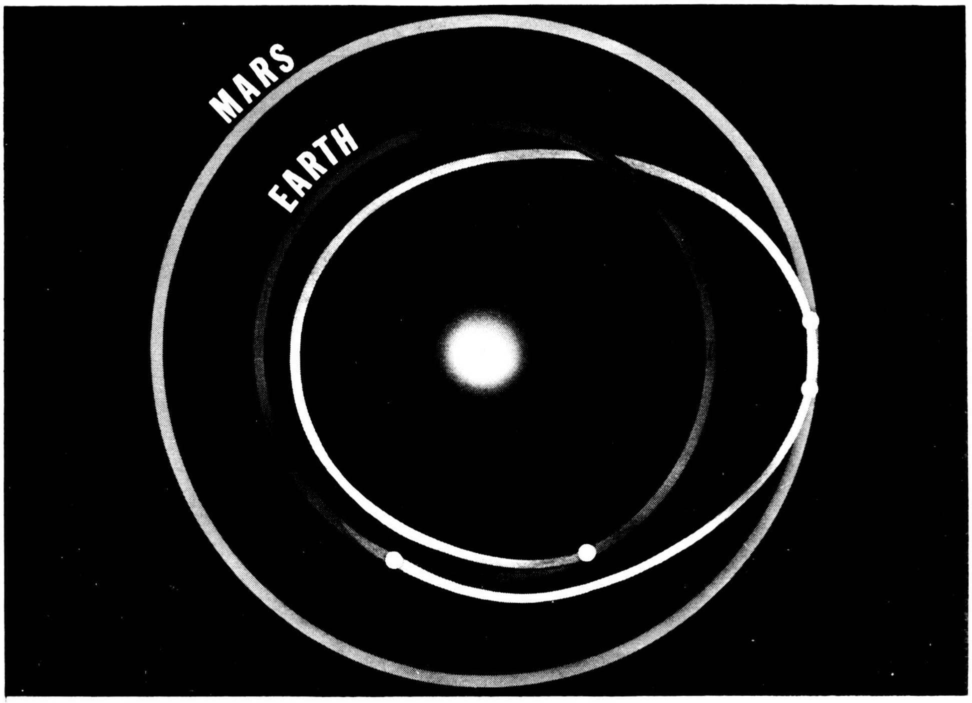


Figure 1

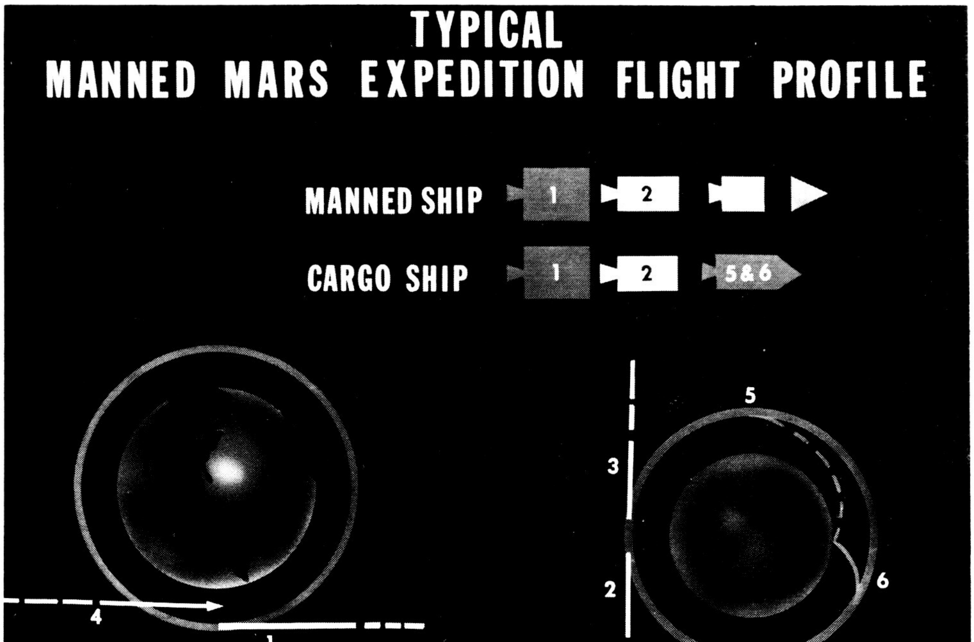
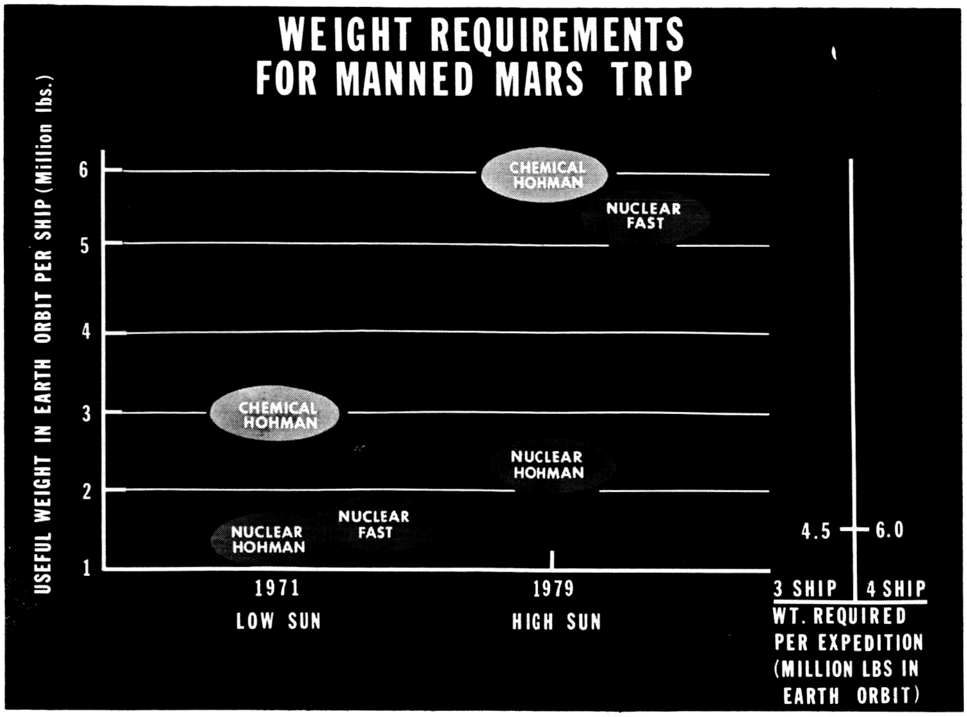


Figure 2



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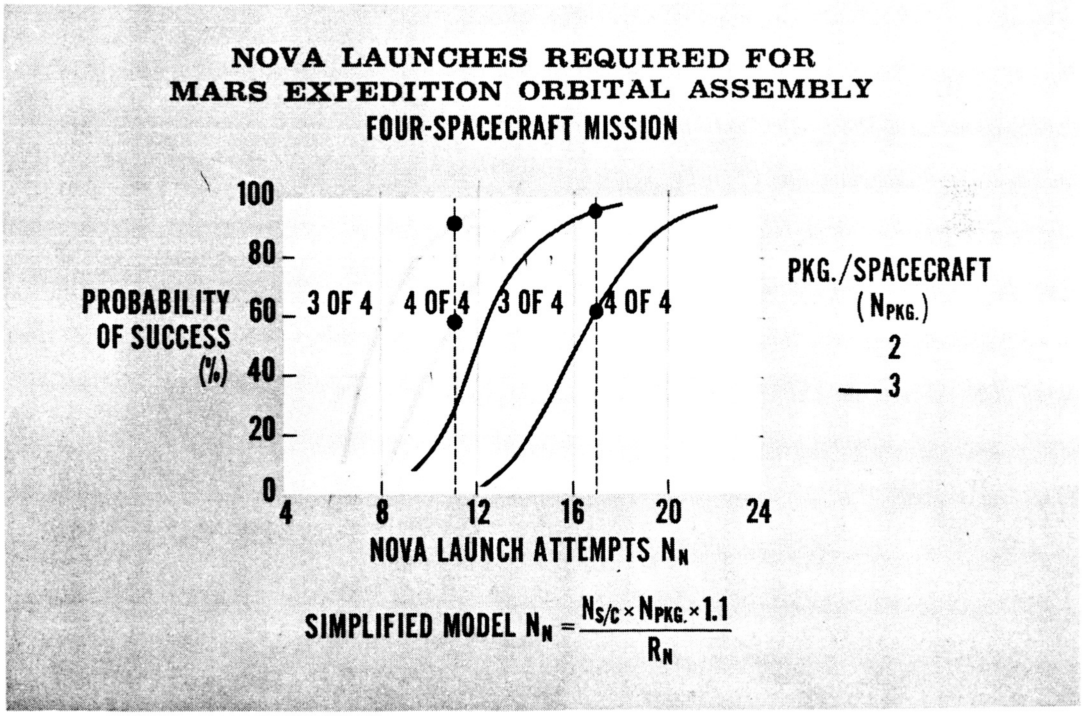
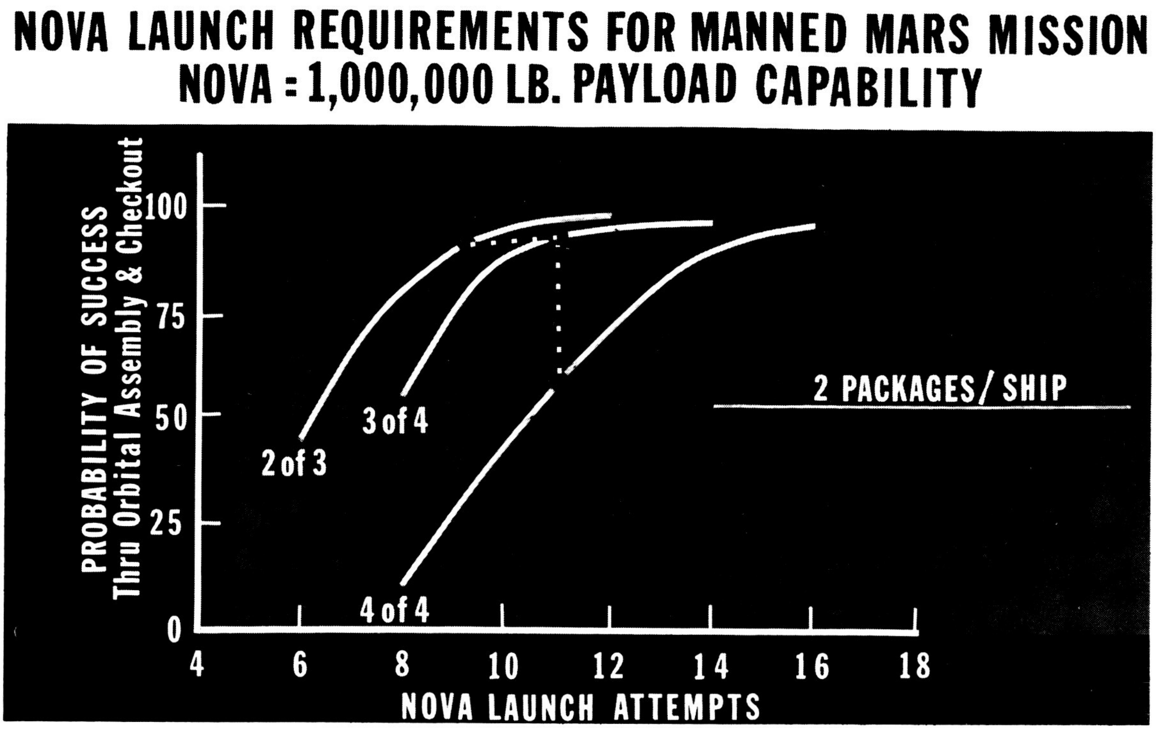


Figure 4



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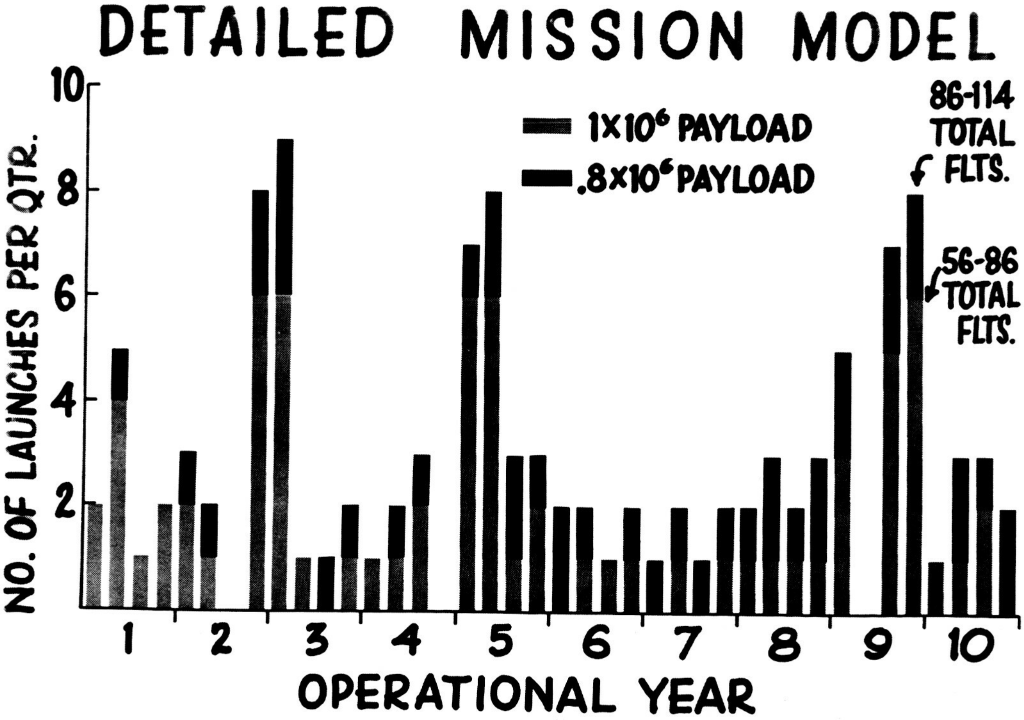


Figure 6

CATEGORY B VEHICLE DATA

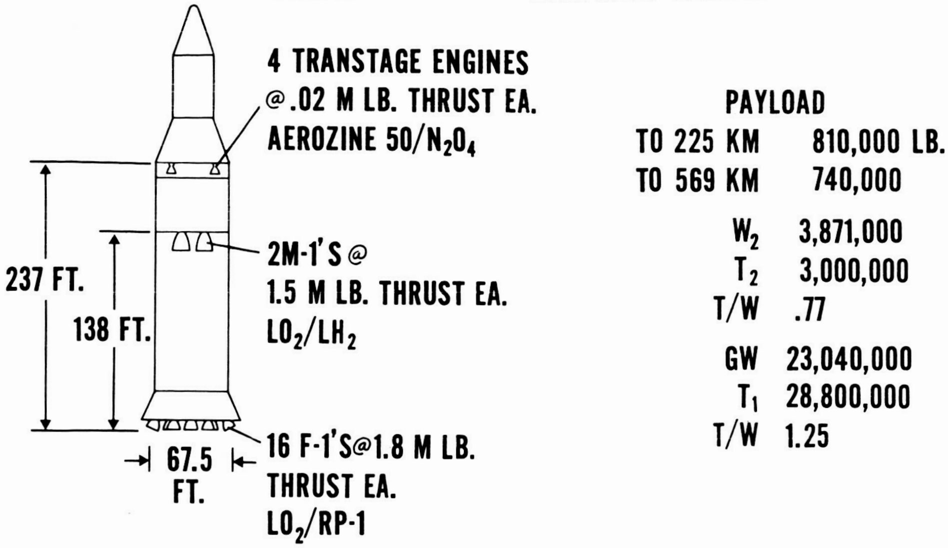


Figure 7

CATEGORY B

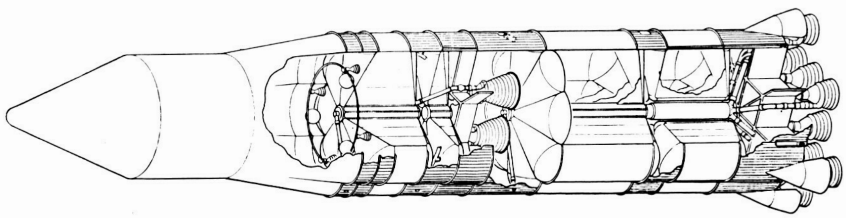


Figure 8

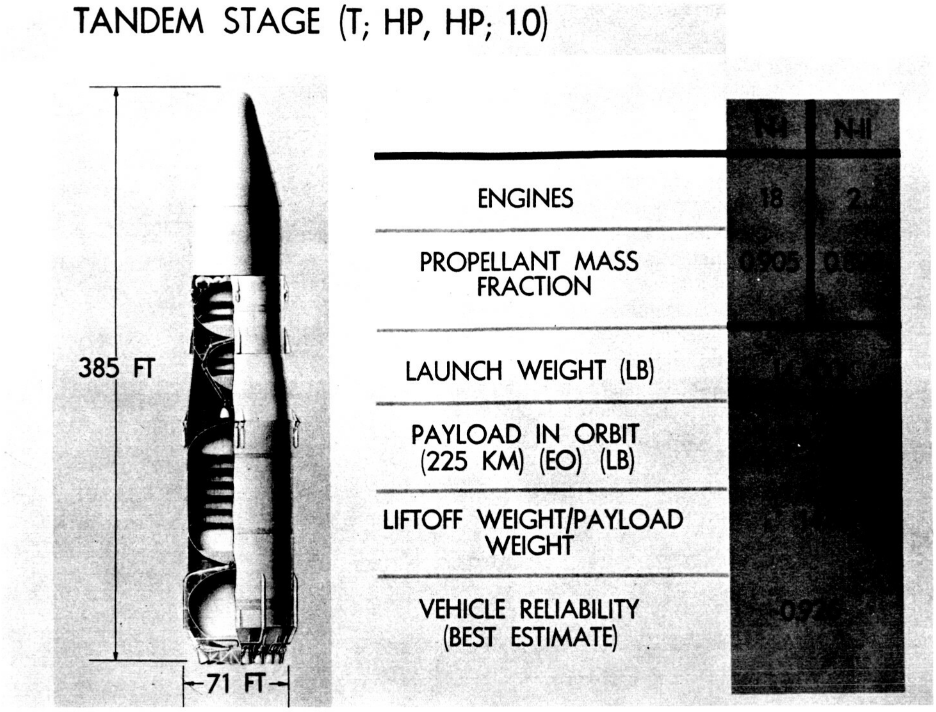
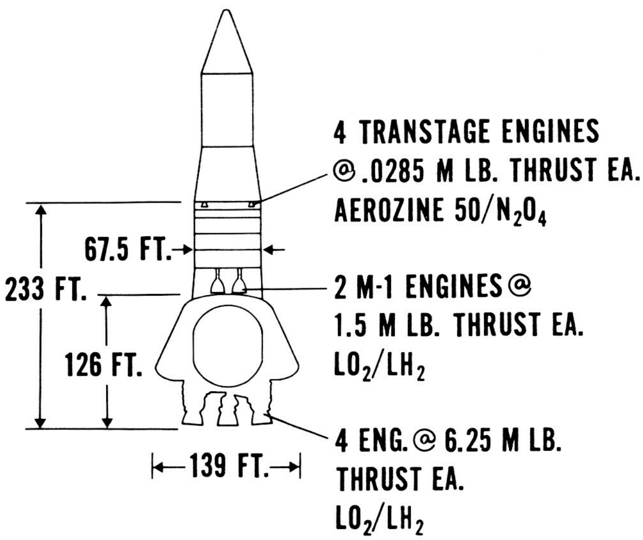


Figure 9

CATEGORY J VEHICLE DATA



PAYLOAD
TO 225 KM 1,120,000 LB.
TO 569 KM 990,000

W_2	4,445,000
T_2	3,000,000
T/W	.72
GW	20,188,000
T_1	25,000,000
T/W	1.25

Figure 10

CATEGORY J

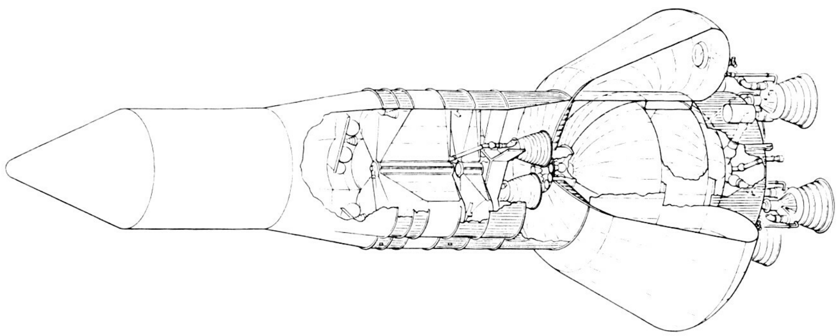


Figure 11

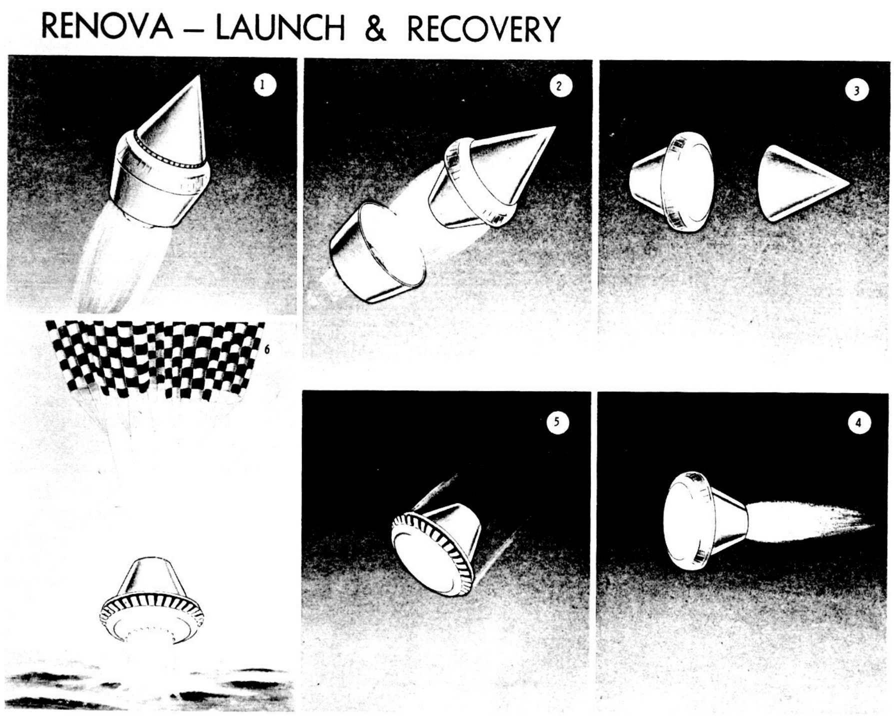


Figure 12

SINGLE STAGE TO ORBIT RENOVA

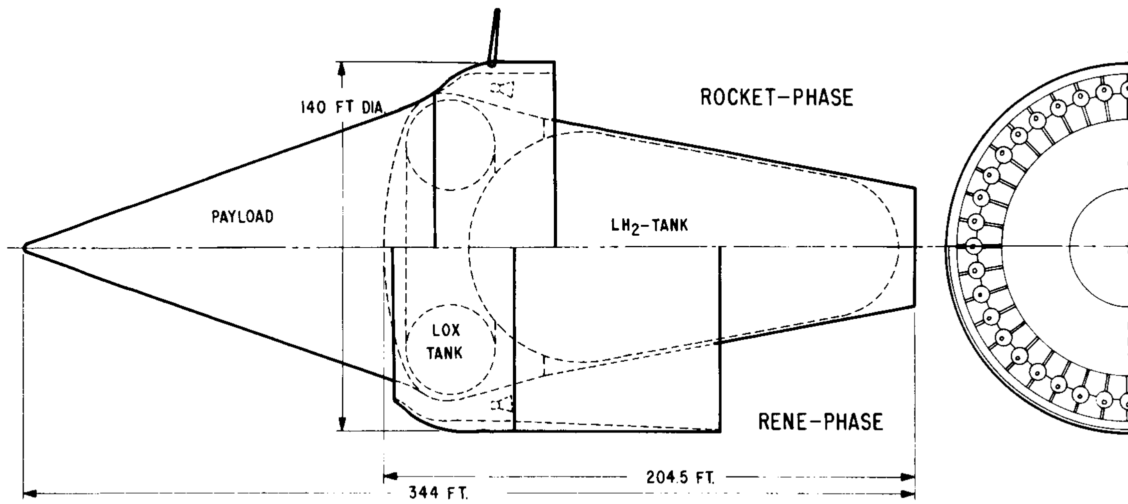


Figure 13

RENOVA NOMINAL TRAJECTORY

$W_0 = 20{,}000 \times 10^6$ LB

AIRBREATHER TO MACH=6.267

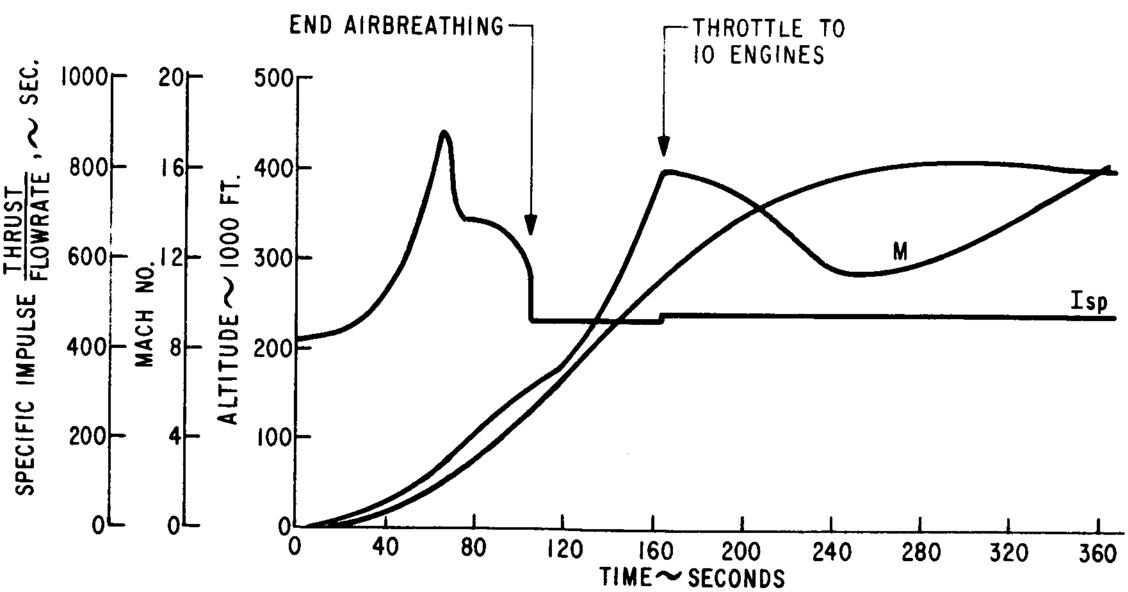
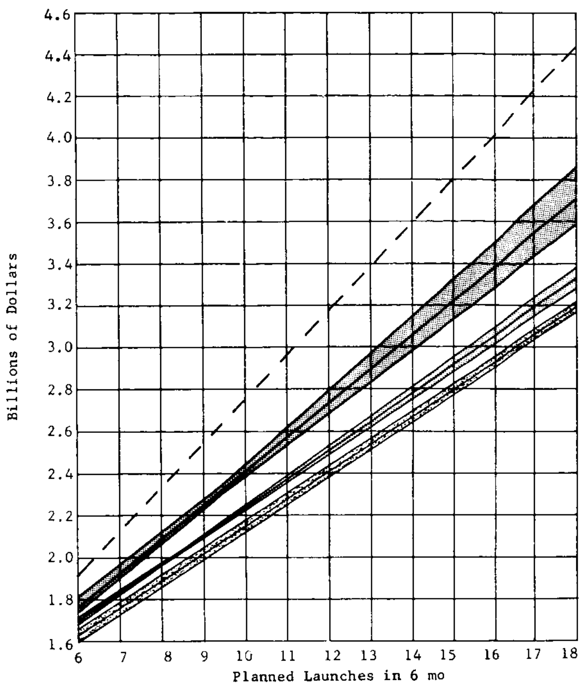


Figure 14



Code	Vehicle		
	Fuel (N-1)	Stage Qty	W_{pl} (10^6)
— —	S	2	1.0
(dark shaded)	LH_2	1 1 1/2 2	1.0
(open box)	RP-1	2	1.0
(light shaded)	RP-1	2	0.8

Note: Solid (S) does not include propellant costs.

Total AMR Costs - Hybrid Concept

Figure 15

NOVA TRENDS

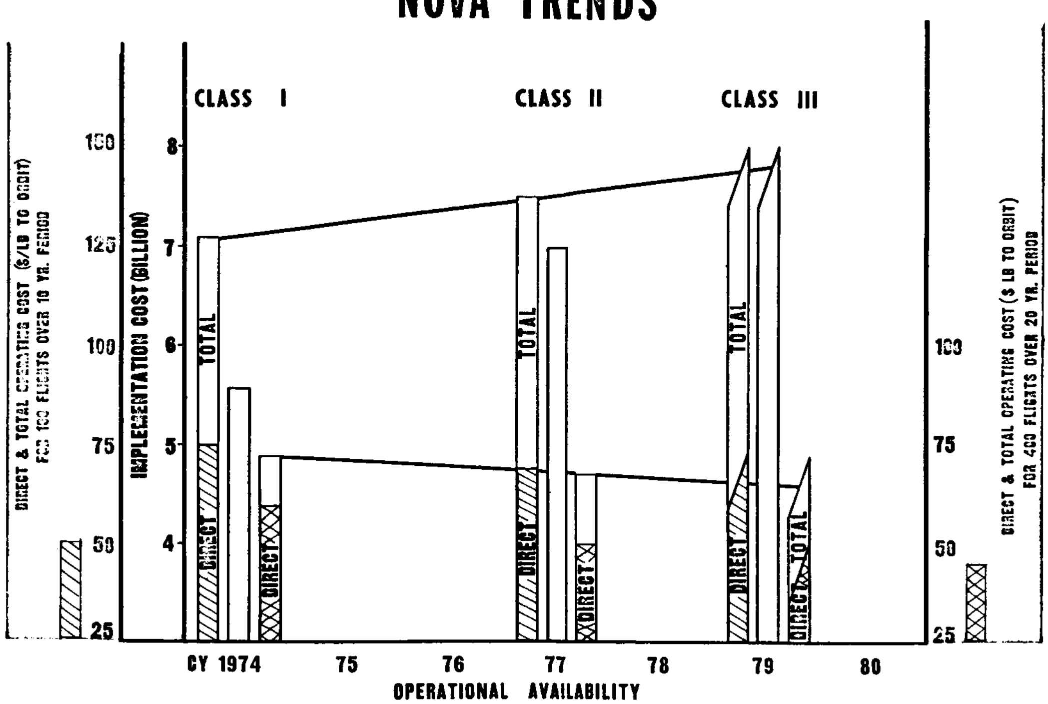


Figure 16

THE NERVA ENGINE PROGRAM

By Lester C. Corrington

Space Nuclear Propulsion Office, Cleveland Extension

[redacted]

THE NERVA ENGINE PROGRAM U

By Lester C. Corrington

Space Nuclear Propulsion Office, Cleveland Extension

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INTRODUCTION

The objective of the NERVA engine program is the development of a nuclear rocket engine for use in space missions. The letters N E R V A mean Nuclear Engine for Rocket Vehicle Application. NERVA is a long-range program that had its beginning in about 1955 when the Los Alamos Scientific Laboratory began preliminary studies on the feasibility of thermal reactors for rocket application. At about the same time, the Livermore Laboratory began parallel studies, but this dual effort was discontinued about a year later.

Los Alamos investigated metallic and graphite reactors and came to the conclusion that graphite offered more promise for early reactor development. Consequently, they narrowed their studies to graphite reactors, and this work has culminated in the building and testing of several reactors. Two principal types of reactors were built, and they are known as Kiwi A and Kiwi B reactors. The Kiwi A reactors were primarily test reactors for the investigation of fuel elements. The Kiwi B reactor was to be of a concept adaptable for flight, and the NERVA engine is based on the Kiwi B concepts.

In 1960 the NASA made the decision to initiate the development of an engine based on the Kiwi B concepts, and a joint office representing both the NASA and the Atomic Energy Commission was set up for this purpose. This joint AEC-NASA Office is known as the Space Nuclear Propulsion Office and is located at AEC Headquarters in Germantown, Maryland. It is managed by Mr. H. B. Finger.

In 1961 an industrial contractor team, the Aerojet-General Corporation and the Westinghouse Electric Corporation, was selected to carry out the development of the NERVA engine, and their activity began in mid-1961. Aerojet was made the prime contractor with responsibility for the complete engine development program, and Westinghouse was made a principal subcontractor with responsibility for the development of the reactor.

A government field office, the Cleveland Extension of the Space Nuclear Propulsion Office, was set up to manage this development program. This office is located in Cleveland because of the availability of technical backup at the Lewis Research Center.

KIWI REACTORS

Description of Kiwi Reactors

Figure 1 shows a diagrammatic cross section of the Kiwi B reactor. It con-

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[redacted]

sists of a homogeneous graphite core surrounded by an insulating graphite shell, a beryllium reflector, and 12 rotating-type control drums located in the reflector. The reactor is contained in an aluminum pressure vessel to which is mounted a propellant-cooled nozzle. In operation, the liquid-hydrogen propellant enters a manifold at the nozzle exit, passes through the coolant passages of the nozzle, through the reflector, through the core where it is heated, and then out through the nozzle to produce thrust.

The core is simply a graphite heat exchanger that is heated by nuclear fission that, in turn, heats the hydrogen that flows through the coolant passages. The diameter of the core is 36 inches and the length is 52 inches. At the design point the core delivers 1120 thermal megawatts to the hydrogen, and the average exit gas temperature from the core is 4090° R. These operating conditions represent a major extension from the current state of the art.

The core is made up of a large number of graphite fuel elements loaded with enriched uranium. Los Alamos investigated several different concepts of fuel elements and the interest is now centered around the concept shown in figure 2. In this concept, known as the Kiwi B-4 concept, the core is made up of simple hexagonal fuel elements, each with 19 coolant holes. In cross section, the hexagon is 0.75 inch across flats, and the coolant holes are approximately 0.095 inch in diameter. In initial experiments, these fuel elements were extruded with a mixture of uranium oxide, graphite flour, and a binder. After extrusion, they were baked by a process which graphitized the entire element and changed the uranium oxide to uranium carbide. Because of problems involving a hydrolysis reaction when uranium carbide is exposed to air containing moisture, a new loading concept has been developed, which is illustrated in figure 3. Small spheres of uranium carbide about 0.002 to 0.005 inch in diameter are coated with pyrolytic graphite to a coating thickness of about 0.001 inch. The pyrolytic graphite provides a very dense and impervious coating that protects the uranium carbide from contact with the atmosphere and thereby prevents the hydrolysis reaction. These small beads are mixed with graphite flour and a binder and are extruded and then graphitized to form the fuel elements.

At the design operating temperature of these fuel elements, there is a reaction between the hydrogen propellant and the graphite. The hydrogen combines with carbon to form volatile hydrocarbons such as methane. To surmount this problem, a coating has been developed for lining each coolant passage. This coating is niobium carbide. It is applied by a vapor-deposition process to a thickness of about 0.002 inch, and in laboratory tests it has performed with reasonable success. This development of a good coating has been a major accomplishment in the program.

The fuel elements are assembled into clusters of seven for installation in the reactor, as illustrated in figure 4. Six uranium-loaded elements are placed around the central element that is made of unfueled graphite. This cluster of elements is supported in the reactor by a metallic tie rod that extends axially through a hole in the central element. This hole in the central element is lined with pyrolytic graphite for insulation purposes and then with a thin stainless-steel tube for retention of the pyrolytic graphite in case of delamination or flaking. Low-temperature hydrogen coolant passes through the annular space be-

[redacted]

[redacted]

tween the stainless-steel tube and the tie rod.

Figure 5 shows a cutaway view of the fuel-element cluster. At the hot end is an unfueled graphite block that has the same external shape as the seven-element cluster. The fuel elements rest on this graphite block, which is supported by the cooled metallic tie rod that extends the entire length of the cluster. This tie rod is in turn supported by a core support plate at the inlet end (cold end) of the reactor core. The fuel elements in the cluster are retained at the cold end by a metallic cluster plate that makes provision for installation of orifices in each of the fuel-element flow passages.

A cross section of a Kiwi B-4 reactor is shown in figure 6. The core consists of about 260 of the seven-element clusters illustrated in figure 5, and these clusters are mounted to the aluminum core support plate by the tie rods as illustrated. The core is surrounded by a graphite barrel, and this barrel provides the mounting for the core lateral support system. This system keeps the fuel elements tightly bundled together to prevent hydrogen from flowing between them and to support the entire core against excessive lateral movement. It consists of a series of graphite strips that run the full length of the core that are faced with a layer of pyrolytic graphite for insulating purposes. These graphite strips are loaded against the core periphery by metallic springs mounted in the graphite barrel.

Also illustrated in figure 6 are the control rods mounted in the beryllium reflector. These control rods are made of beryllium cylinders about 4 inches in diameter with sheets of boral covering about 120° of the circumference. The control rods are rotated by externally mounted actuators.

Kiwi Reactor Tests to Date

A total of six Kiwi reactors have been tested up to the present time by Los Alamos at the test site in Nevada. The first three were of the Kiwi A type and were run primarily for the investigation of fuel-element concepts. The fuel elements were operated at near the design point temperature but at a power level of only about 100 megawatts. The propellant (coolant) used in these initial tests was ambient-temperature gaseous hydrogen.

The next two reactor tests were of a Kiwi B-1 design that had a fuel-element concept substantially different from the one illustrated in figure 2. The first of the Kiwi B-1 reactors was operated at near the design-point temperature and at a power level of about 250 megawatts and was also cooled with gaseous hydrogen. The second of the Kiwi B-1 series represented the first reactor test in which liquid hydrogen was used as the propellant. Serious flow instabilities were encountered in the two-phase flow regime during the startup. The reactor attained temperatures and power levels near the design point for a very brief period of time, but the core was damaged extensively during the run.

The first and so far the only test of the Kiwi B-4 type reactor (the type that appears to be of most interest for the NERVA engine) was run in December of 1962. The propellant used for this test was liquid hydrogen, as in the last Kiwi

[redacted]

B-1 test. Through prior modifications to the propellant system, the hydrogen instability problem that had been experienced previously was overcome, and the hydrogen flow during the Kiwi B-4 test appeared to be very smooth and steady throughout the entire test.

During the Kiwi B-4 test, the reactor power was brought up to a level of about 250 megawatts at which point it was held for about a minute for calibration of nuclear instrumentation. The core temperature at this hold point was about 2500° R. Following this hold, an attempt was made to go up to the design-point temperature and power level, but indications of core damage resulted in the reactor being shut down considerably short of its design-point goals. Disassembly of the core revealed substantial breakage of fuel elements, although the tie rod support system was essentially intact. Analysis of the failure has indicated that the probable cause was a lack of adequate lateral support of the core by the lateral support system. Loads resulting from core vibration apparently caused considerable breakage of fuel elements. The lateral support system has since been redesigned in an attempt to overcome this vibration problem.

NERVA ENGINE

Description of NERVA Reactor

The reactor for the NERVA engine is based on the concept of the Kiwi B-4 reactor, but with a number of changes in design to make it suitable for withstanding flight loads and to incorporate a shield for the protection of engine components and the propellant in the stage tank against nuclear radiation. Figure 7 shows a cross section of the major elements of the NERVA engine reactor design as it exists at the present time. The core is composed of the seven-element clusters supported in the same manner as in the Kiwi B-4 reactor. The lateral support system has been redesigned to overcome the core-vibration problem experienced in Kiwi B-4 and to support the core against lateral accelerations of up to 4 g's that may be experienced in ground handling and in flight.

The shield in the NERVA reactor is contained entirely within the reactor pressure vessel. It is designed for an overall energy attenuation of about 10, and centerline measurements show a gamma-flux attenuation of about 20 and a neutron-flux attenuation of about 140. The shield consists essentially of a large number of stainless-steel tubular capsules mounted axially very closely together and filled with lithium hydride and borated stainless steel. The propellant flow path for this reactor is similar to that of the Kiwi B-4 reactor except that the full propellant flow passes through the shield for cooling purposes before it enters the reactor core.

The NERVA reactor program is strongly oriented toward research and development testing of components prior to full-scale reactor tests. This component development program is underway at the present time, and the first full-scale hot test of a NERVA reactor is scheduled for early next year. Fabrication of the long-lead-time parts for this reactor is now underway at Westinghouse. This test will be followed with tests of additional reactors at about 3-month intervals.

Description of NERVA Engine

A schematic representation of the flow system for the NERVA engine is shown in figure 8. This engine is based on the so-called "hot-bleed" cycle. In this cycle, liquid hydrogen flows from the propellant tank into the pump where the pressure is raised to about 1000 pounds per square inch. It then flows through the coolant passages in the nozzle, through the reflector, through the radiation shield, through the reactor core, and out through the nozzle. A small portion of the hot hydrogen is bled off at the inlet to the nozzle, is mixed with cold hydrogen from the nozzle coolant tubes to reduce its temperature to a safe level for the turbine, and is then ducted to the turbine that drives the pump. The discharge from the turbine is ducted to two swivelable nozzles that are used for vehicle roll control.

A full-scale mock-up of the NERVA engine is shown in figure 9. The turbopump is mounted within the thrust structure just above the reactor. Liquid hydrogen flows from the propellant tank down through the inlet line to the turbopump, through an external line to the nozzle, through the nozzle coolant tubes, and then through the internal parts of the reactor and out through the nozzle. A hot-gas bleed line leads from the nozzle inlet to the turbine, and the exit gas from the turbine flows out through the two roll-control nozzles.

The entire engine is gimbaled about a point in the inlet line to the pump. One of the gimbal actuators is visible in the photograph. The spheres near the top of the engine contain high-pressure hydrogen gas for actuation of various devices prior to and during the bootstrap start of the engine. Also visible are the 12 control-drum actuators at the top end of the reactor. The overall length of the engine, as illustrated, is about 23 feet.

Performance Goals

The following design-point operating conditions have been established as the initial goal:

Reactor thermal power delivered to propellant, Mw	1120
Average reactor-exit gas temperature, °R	4090
Reactor-core-exit pressure, lb/sq in.	550
Hydrogen flow rate, lb/sec	75
Thrust, lb	55,600
Overall system specific impulse, sec	750
Operating time at full power (with two restarts)[a], min	20
Engine weight, lb	14,500

[a]The operating time goal for development engines is 60 min to permit the testing of each engine several times.

Development Program

A program is now underway for the initial development for feasibility evaluation purposes of all the principal components of this engine. The reactor represents the main component and probably the most difficult development problem in the entire engine. Other components that appear to present major problems are the turbopump, the nozzle, and the control system.

The turbopump will consist of a single-stage centrifugal pump driven by a two-stage axial-flow turbine. The bearings will not be lubricated but will be cooled by liquid hydrogen. Axial loads on the bearings will be limited by a thrust-balancing piston. A significant problem in the turbopump is the development of hydrogen-cooled bearings that are resistant to nuclear radiation.

The nozzle will be of the conventional tubular construction, but because it has a very high contraction ratio from the point of attachment to the pressure vessel down to the throat, it incorporates a heavy pressure shell. Significant problems in the nozzle are the high heat-flux rates in the nozzle throat (twice as high as in chemical rocket engines) and a difficult fabrication problem because of the requirement of a heavy pressure shell.

In the engine control system, the reactor power will be controlled by neutron flux sensors with a reactor-exit temperature trim. The propellant flow will be controlled by pressure sensors that measure the reactor-exit pressure. Significant problems in the control system include the development of temperature sensors that will operate above 4000° R, the development of adequate neutron detectors, and the development of pressure sensors, wiring harnesses, and other components to operate reliably in a high flux radiation field.

Another significant problem in the design of the NERVA engine and its supporting equipment arises from the fact that once an engine has been run it is highly radioactive. The engine must be designed in such a way that maintenance, disassembly, and reassembly operations can be done remotely, and remote-handling equipment must be designed to perform these functions. Although this does not represent a problem for flight engines, because these engines will not be operated prior to flight, it represents a major problem for development engines because of the requirement for repeated tests on each engine.

The date for the first test of the engine assembly will depend on the rate at which success is achieved in the reactor program. For planning purposes, it is assumed that the first engine test will take place in late 1965 or early 1966. This test will probably be followed by subsequent tests at about 3-month intervals during the first year and at closer intervals during subsequent years.

NERVA ENGINE DEVELOPMENT FACILITIES

In the NERVA program, the reactor testing and the engine testing will all be done in Nevada in a remote desert area called Jackass Flats, about 95 miles northwest of Las Vegas. The test site has been initially developed by Los Alamos Scientific Laboratory for Kiwi reactor testing and is now known as NRDS, the Nuclear Rocket Development Station. It is located in a remote area because of the possibility of radioactive fallout during testing. Figure 10 shows a portion of the Jackass Flats area. Some of the existing reactor test facilities are shown but are difficult to see because of the great distances involved. The engine test facilities will be in an area near the left side of this photograph.

The facilities layout at the test site is shown in figure 11. Test cell A is the test cell in which all of the reactor testing to date has been done. Test cell C is also a reactor test cell and is now in its final stages of activation. These two reactor test cells are operated from a control point about 2 miles away.

The reactor MAD (maintenance, assembly, and disassembly) building consists essentially of two cold assembly bays, one large hot disassembly bay, and a number of small hot cells. The disassembly bay is a heavily shielded room equipped with the necessary remote-handling equipment to carry out complete disassembly operations on the reactor.

One engine test stand, ETS-1, is well along in construction and another engine test stand, ETS-2, is in the advanced planning stage. These two test stands will be operated from an underground control point located about 1000 feet away. An engine MAD building for maintenance, disassembly, and reassembly operations is now in the early stages of construction.

Facilities for the ground testing of complete nuclear stages are shown near the left side of figure 11. These facilities are now in the early planning stages. One of them is planned to be suitable for either engine or stage testing.

All of the test stands and the two MAD buildings are interconnected by a railroad system. Reactors and engines are moved to and from the test stands on these railroads by remotely controlled locomotives. Figure 12 shows a reactor on its test cart being moved to the test stand. In this particular case, the nozzle had not yet been installed on the reactor. The stepped plug on the forward end of the test cart fits into an opening in a heavy shield wall at the test stand, and the test connections are made on the far side of this shield.

Test cell A is shown in figure 13. The two cylindrical tanks at the upper right are liquid-hydrogen Dewars with a capacity of 28,000 gallons each. The shed mounted on tracks is for the protection of the reactor and personnel from the weather during test-stand operations. During testing, this shed is moved well away from the test stand.

Figure 14 shows a reactor at the test stand ready for test. These reactors are tested in an up-firing position for simplicity in the disposal of the exit hydrogen gas. The test cart carries a heavy shield just below the reactor, and the control-drum actuators are located below this shield. Hydraulic actuators are used for reactor test purposes but will not be suitable for use on the engine because of radiation effects. It is planned that electropneumatic actuators will be used on the engine.

An aerial view of test cell C is shown in figure 15. The two spheres are liquid-hydrogen Dewars with a capacity of 50,000 gallons each. The long cylindrical tanks are high-pressure gas bottles. This facility also contains a high-capacity liquid-hydrogen flow loop for component test purposes.

The reactor MAD building is shown in figure 16. This building contains two assembly bays and one large shielded disassembly bay. It also contains a number of small hot cells for use in post-mortem examination of reactor components.

Figure 17 is an artist's sketch of the first engine test stand, ETS-1. The run tank on this stand has a capacity of about 70,000 gallons, and this is augmented by a spherical storage Dewar with a capacity of about 250,000 gallons. The engine is installed on the test stand by a rail-mounted vehicle operated from a heavily shielded cab. During operation, two halves of a cylindrical shield are brought together to form an enclosure around the engine that can be made inert. The engine fires downward into an exhaust duct that diffuses the jet and turns it through an angle of about 120°. This diffuser is water cooled. The entire test stand with the exception of the exhaust duct is made of aluminum because of neutron activation considerations. The exhaust duct, however, will remain highly radioactive and must be shielded before personnel can approach the test stand. This shielding is accomplished by movable shield doors over the exhaust-duct vault.

The present appearance of ETS-1 is shown in figure 18. This test stand is well along in construction and will be ready for operation about the middle of 1965.

GROWTH CAPABILITY OF NERVA

The reactor power is limited primarily by thermal gradients in the core, by temperature levels, and by structural strength. Reactor temperature levels are limited by the high-temperature capability of the fuel matrix in the graphite and by coatings for the prevention of fuel-element corrosion. Operating time is limited primarily by corrosion and is therefore to some extent a trade-off with temperature level.

Design analyses of the NERVA engine indicate that the growth of power level will probably be limited by nonreactor components such as the turbopump and nozzle. These components have been designed for growth to about 1500 megawatts, whereas the reactor appears capable of ultimate growth to significantly higher power levels.

If the operating time of 20 minutes is maintained, it appears that the ultimate propellant temperature will be limited by the reactor to about 4500° R.

With these increases in power and temperature levels, the engine should have a thrust of about 74,000 pounds and an overall system specific impulse of about 830 seconds. Any increases in power beyond this level will require major redesign and development of engine components and systems, and there is currently no engine development activity in this direction. However, the advanced engine program is investigating ways and means of advancing the technology for the eventual design and development of higher power nuclear rocket engines.

DIAGRAMMATIC CROSS SECTION OF KIWI B REACTOR. POWER, 1120 Mw; GAS EXIT TEMPERATURE, 4090° R

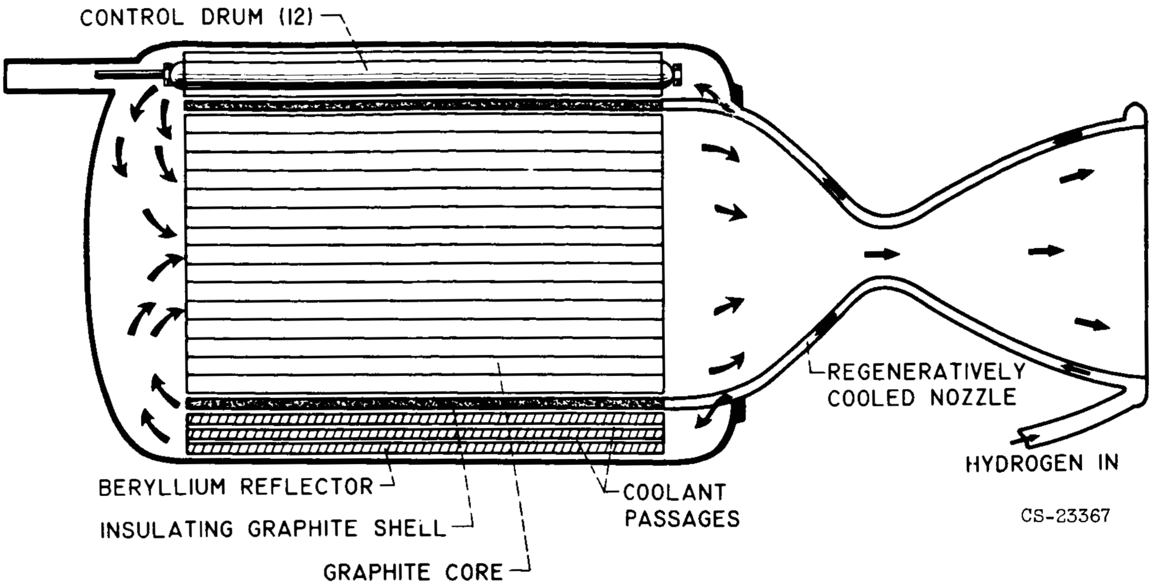

Figure 1

KIWI B-4 AND NERVA FUEL ELEMENTS SHOWING 19 COOLANT PASSAGES APPROX. 0.095 IN. IN DIAMETER COATED WITH NIOBIUM CARBIDE

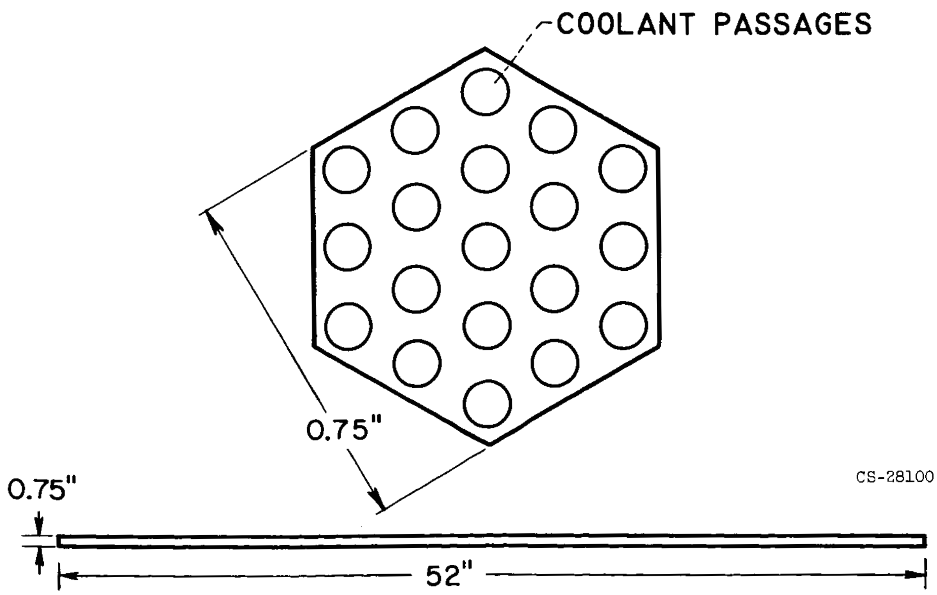

Figure 2

FUEL PARTICLES USED IN MANUFACTURE OF FUEL ELEMENTS
URANIUM CARBIDE SPHERES; 0.002- TO 0.005-INCH DIAMETER;
COATED WITH PYROLYTIC GRAPHITE 0.001 INCH THICK

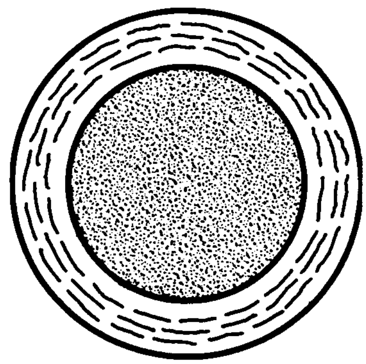

Figure 3

KIWI B-4 AND NERVA FUEL ELEMENT CLUSTERS

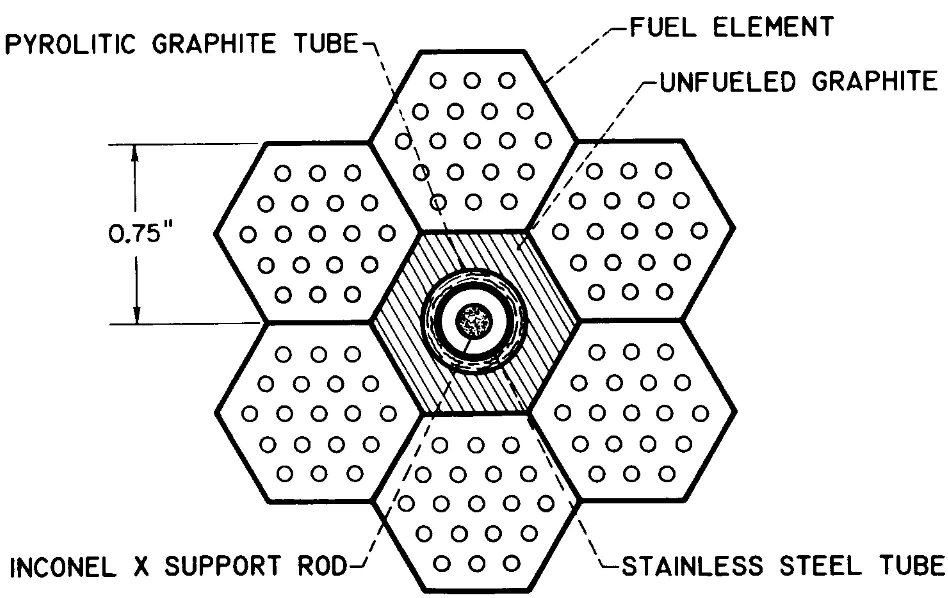

Figure 4

[redacted]

CUTAWAY VIEW OF KIWI B-4 AND NERVA FUEL ELEMENT CLUSTER

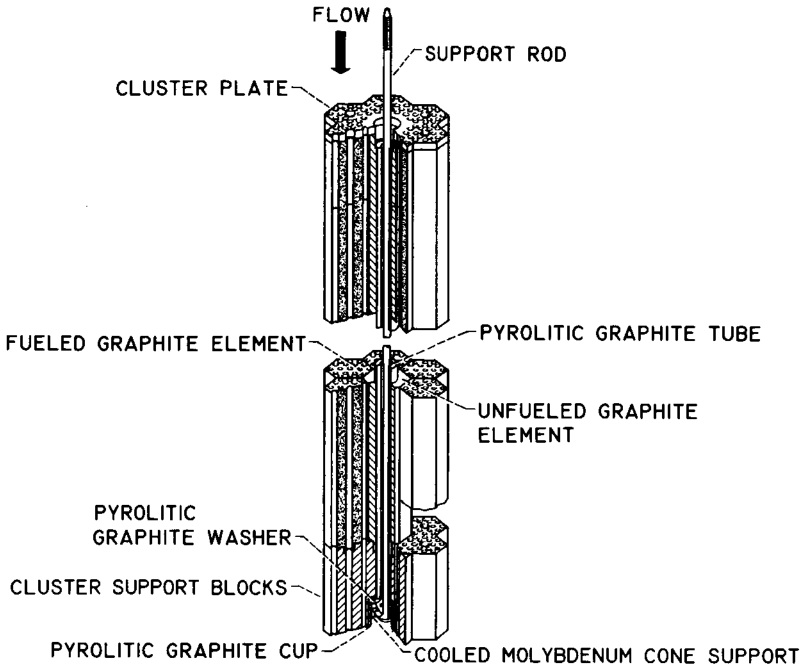

Figure 5

CROSS SECTION OF KIWI B-4 REACTOR

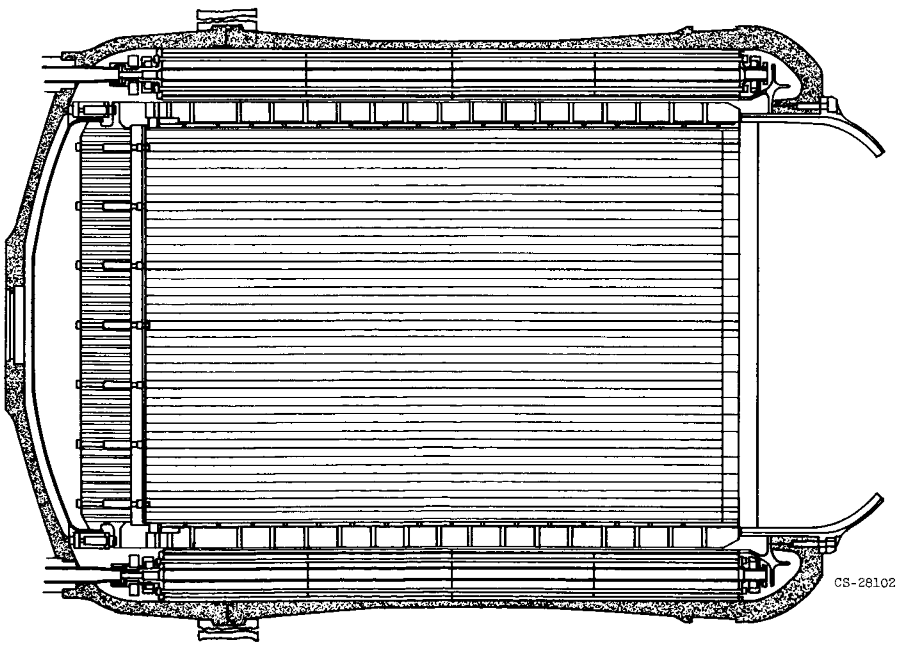

Figure 6

[redacted]

[redacted]

CROSS SECTION OF NERVA REACTOR

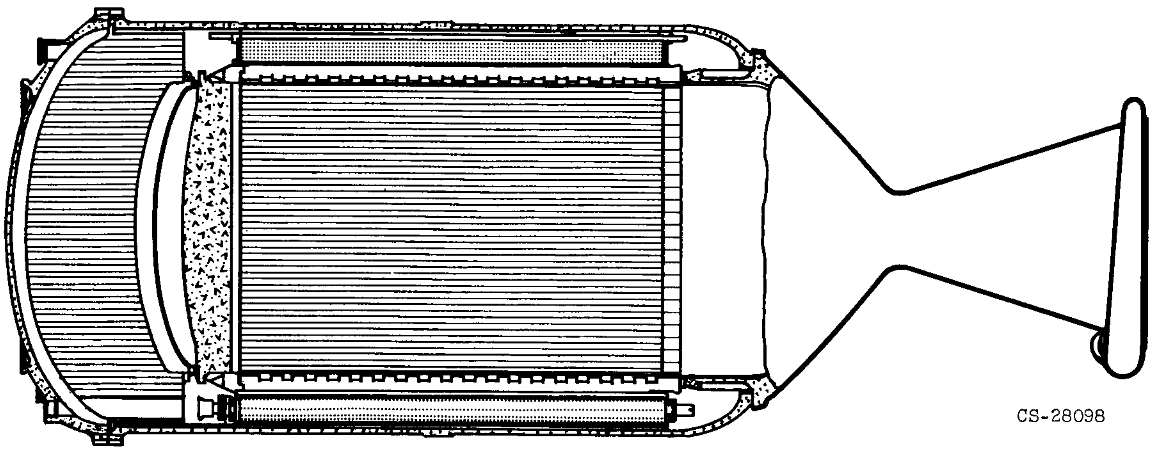

Figure 7

SCHEMATIC DIAGRAM OF NERVA ENGINE FLOW SYSTEM

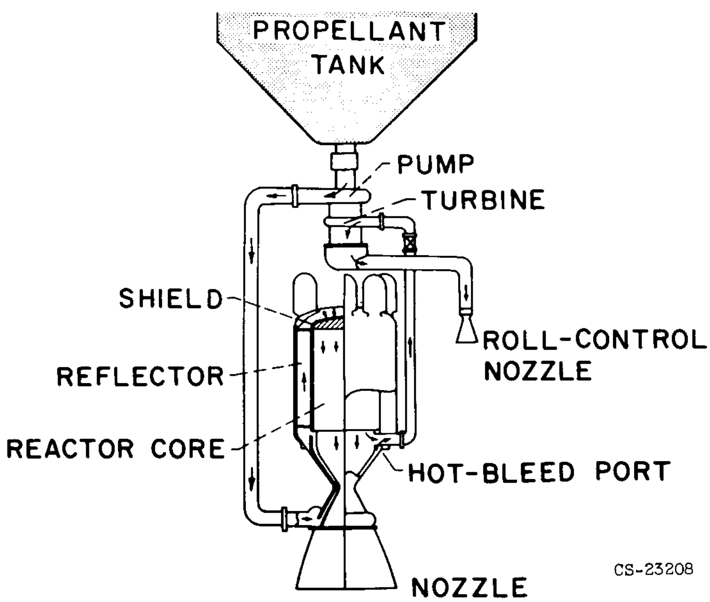

Figure 8

[redacted]

FULL-SCALE MOCKUP OF NERVA ENGINE

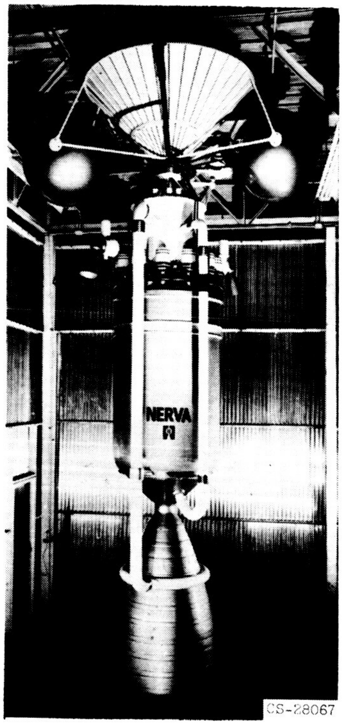

Figure 9

PORTION OF NUCLEAR ROCKET DEVELOPMENT STATION AT JACKASS FLATS, NEVADA

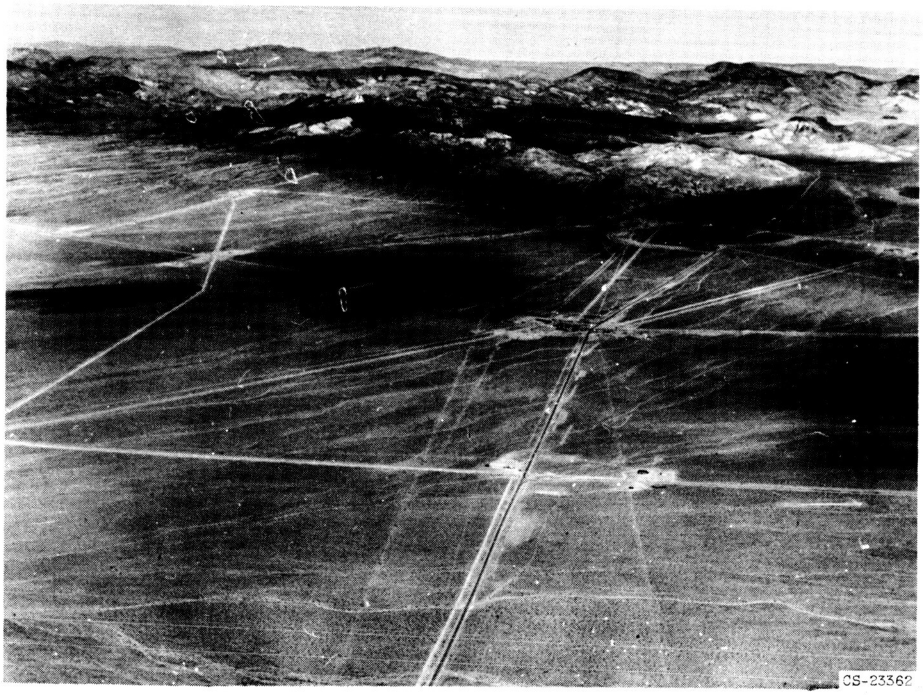

Figure 10

FACILITIES LAYOUT AT NUCLEAR ROCKET DEVELOPMENT STATION

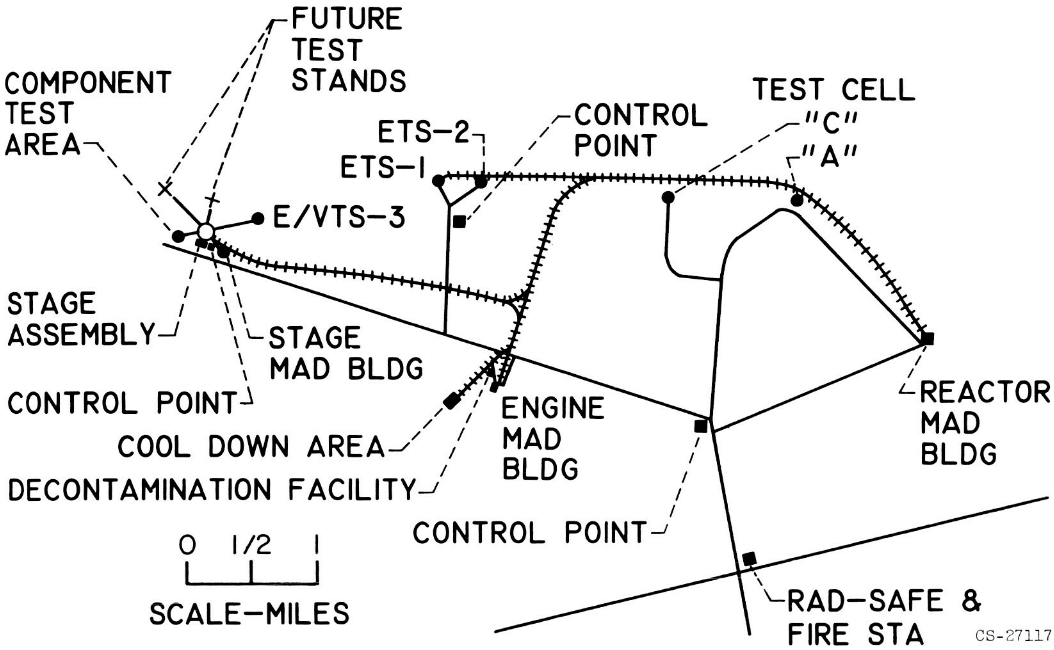

Figure 11

REACTOR IN TRANSIT TO TEST STAND (NOZZLE NOT YET MOUNTED)

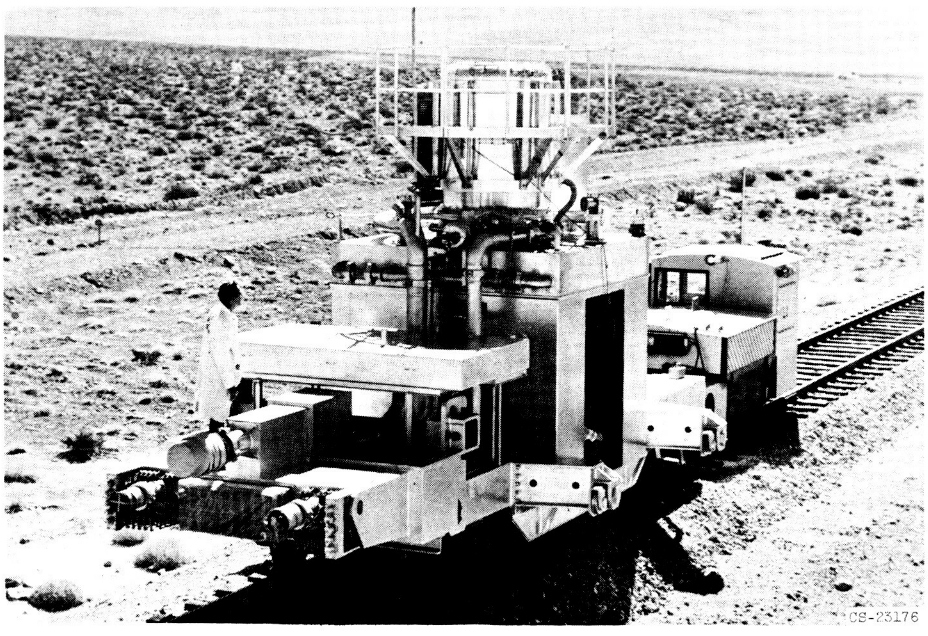

Figure 12

TEST CELL A

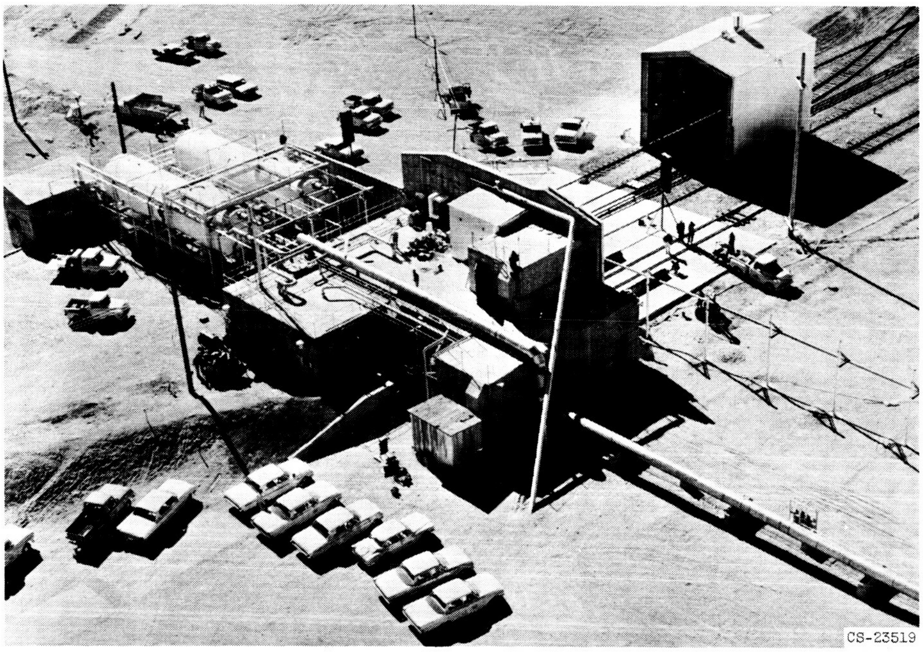

Figure 13

REACTOR AT TEST CELL A

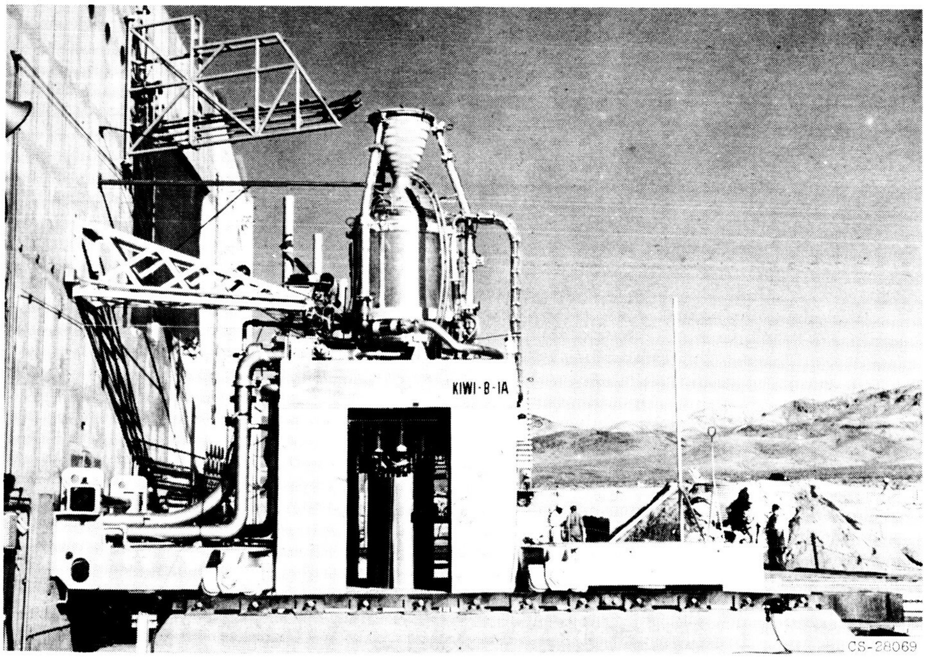

Figure 14

TEST CELL C

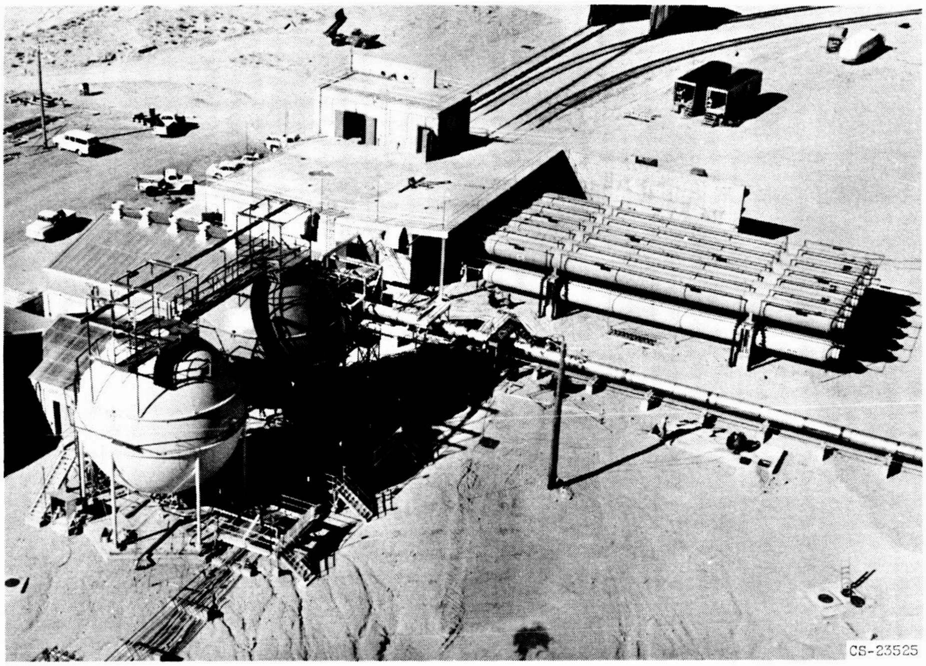

Figure 15

REACTOR MAINTENANCE, ASSEMBLY, AND DIS-ASSEMBLY (R-MAD) BUILDING

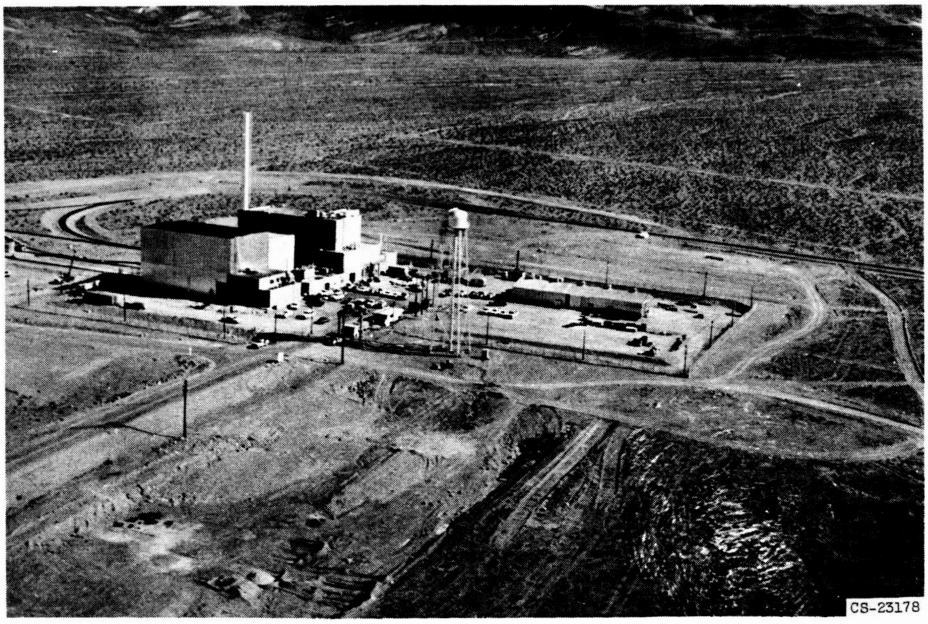

Figure 16

ARTIST'S SKETCH OF ENGINE TEST STAND (ETS-1)

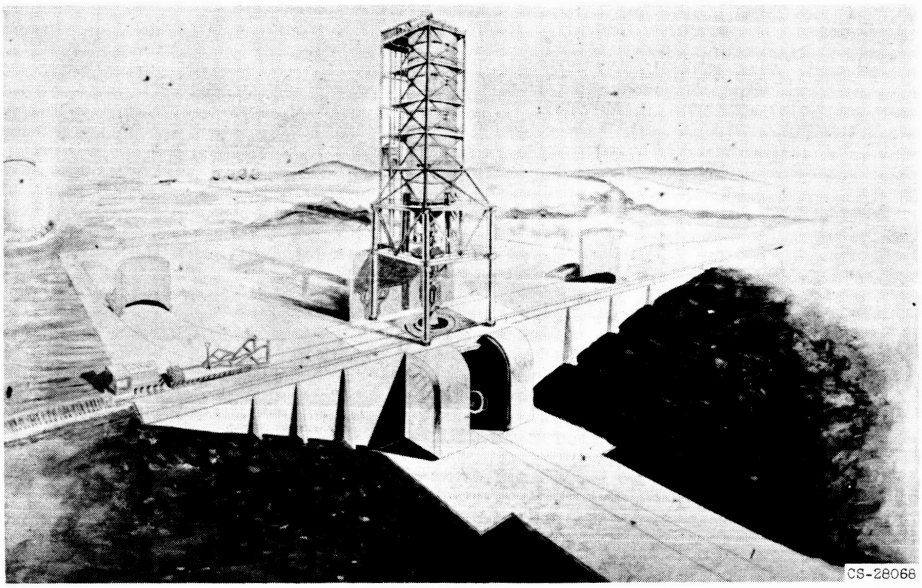

Figure 17

ENGINE TEST STAND 1 (ETS-1) AT PRESENT

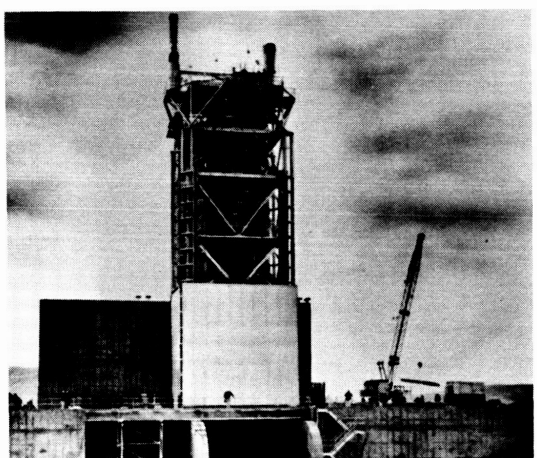

Figure 18

WATER-MODERATED REACTOR CONCEPT

By Frank E. Rom

NASA Lewis Research Center

[redacted]

WATER-MODERATED REACTOR CONCEPT U

By Frank E. Rom

NASA Lewis Research Center

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INTRODUCTION

Because of the importance of nuclear rockets for manned planetary flight, the Lewis Research Center has been studying alternate nuclear rocket concepts that have the potential for meeting the high performance requirements of these missions. Lewis has concentrated its advanced nuclear rocket research program on a tungsten-water-moderated concept. The purpose of this presentation is to describe this concept and give a brief résumé of some of the work that has been done in support of it.

NUCLEAR ROCKET REQUIREMENTS

Before getting into the details of the tungsten-water-moderated concept, some of the requirements of a nuclear rocket should be considered that lead to its choice. They are: (1) high temperature to produce the highest possible specific impulse; (2) recyclability, the ability to start up and shut down of the order of 100 times; (3) long life, that is, operating life of about 10 hours; and (4) developability, that is, the capability of being developed into a reliable man-rated powerplant with the least expenditure of cost and time.

The requirement of high temperature is, of course, obviously important to high-energy space missions such as 1-year manned trips to the nearby planets. Recyclability is of importance, not so much for missions that require restarts, but for a development program where thousands of tests are required to develop the required reliability for man-rated systems. Recyclable nuclear rocket reactors are essential to prove reliability of the entire nuclear powerplant within a reasonable amount of time.

Long operating life (compared with chemical rocket life) is required to provide the large total impulses required for lunar ferry and high-energy manned planetary missions. In addition, long lifetimes are also desirable in the nuclear rocket development program to minimize the number of reactors required and the attendant long delays between tests if the reactors must be changed after each firing.

Perhaps the most important item on the list is that the nuclear rocket powerplant be developable. If it cannot be developed into a reliable man-rated system, it makes little difference what specific impulse can be attained, how many times the reactor can be recycled, or how long an operating life it may achieve. Besides recyclability and long life, the key to a system that can be developed is finding a concept that can be exhaustively tested in ground test facilities. Important in a test program is the ability to test reactor components and subsystems by themselves to a high degree of perfection before as-

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sembly into a complete powerplant. That is not to say that complete powerplant tests are not necessary. Not at all; complete powerplant tests are necessary to find the problems due to interactions of components. All difficulties that are found by component testing save much more costly and time-consuming full engine tests. Therefore, a reactor that has a high degree of component separability is highly to be desired.

TUNGSTEN-WATER-MODERATED CONCEPT

After studies involving many reactor concepts, including fast and thermal, a thermal system using water as the moderator and tungsten enriched in the 184 isotope as the fuel element material appeared to be the most attractive.

The basic concept that evolved is very similar to the HTRE No. 1 reactor from the days of the Aircraft Nuclear Propulsion Program. A schematic of this reactor is shown in figure 1. It is composed essentially of an aluminum tank of water, which contains aluminum tubes joining aluminum headers at the top and bottom of the tank. Water fills the space surrounding the tubes. Fuel elements composed of concentric cylindrical plates of nichrome containing fully enriched uranium dioxide are held within the aluminum tubes. Air passes through these fuel elements, is heated, and is discharged from the bottom of the core. A fibre-type silica insulation about 1/10 inch thick contained within thin stainless steel cylinders running the full length of the core reduces the heat loss from the hot gases and fuel elements to a negligible fraction of the reactor power. The heat transferred to and generated in the water by neutron and gamma heating is removed by passing the water through an air-cooled radiator. This reactor can be converted to a nuclear rocket by utilizing hydrogen as the coolant, instead of air, and replacing the nichrome with tungsten fuel elements. A water-to-hydrogen heat exchanger replaces the air-cooled radiator. There is more than enough cooling capacity in the hydrogen propellant to cool the water moderator.

Figure 2 shows a photograph of a full-scale model of a tungsten-water-moderated nuclear rocket. An aluminum pressure vessel is completely filled with water except for the aluminum tubes, which contain the tungsten fuel elements and flowing hydrogen. The water-to-hydrogen heat exchanger is divided into six equally spaced segments, one of which is shown at the top. The hydrogen from the nozzle, which would be located to the right, enters the tubes of this heat exchanger, which removes the heat generated in the water. The hydrogen then enters the reactor-inlet plenum. From this region the hot hydrogen is expanded through the nozzle to produce thrust at specific impulses of 800 to 900 seconds. The water moderator is circulated through the core and heat exchanger by means of a water pump and inlet and outlet water plenums.

Figure 3 shows a sectioned view of a typical fuel element. The fuel element material in the form of five concentric cylinders of clad tungsten - uranium dioxide material is shown sectioned in this photograph. The fuel cylinders are supported and spaced by fuel support pins. The front support pins to the left pass through and are fastened to a tungsten fuel support tube. The fuel support tube runs the entire length of the reactor and provides a gap between it and the water-cooled aluminum tube, which is filled with stagnant hydrogen. The stagnant hydrogen gap about 1/8 inch thick contains a thin molybdenum radiation shield. This technique of insulation reduces the heat loss from the fuel cylinders and hot hydrogen to a fraction of 1 percent of the full reactor power. Considering the fact that 6 to 7 percent of the reactor power is generated within the water anyway by neutron and gamma heating, this small additional heat load is hardly of any consequence. The fuel cylinder stages are only about $1\frac{1}{2}$ to 2 inches long to minimize thermal expansion problems. About 20 to 30 stages are required for a full reactor length.

It is apparent that the high-temperature problems of this reactor concept are concentrated within individual isolated small fuel elements. The remainder of the reactor is made entirely of aluminum, which is water cooled at all points. The core structure being fabricated of a ductile metal and operating at essentially a constant temperature should present a minimum of problems in development. In fact, the entire water side of the reactor, which represents the major structural components, can be developed to a high degree of perfection without resorting to nuclear testing until all the major problems have been eliminated.

The individual fuel element can also be developed to a high degree of perfection relatively rapidly because of its small size. This can be done by means of out-of-pile electrically heated tests, hot hydrogen flow tests, furnace tests, vibration tests, and other similar tests. In-pile flow tests can also be carried out to more nearly simulate the complete nuclear rocket environment. Because of the relative ease and low cost of developing single fuel elements, several alternate designs could be developed simultaneously and the best one chosen for the final hot reactor development, but only after the major bugs have been worked out in the fuel element development program. Advanced fuel element concepts, perhaps for higher temperature or higher power density, can continue to be developed while the first nuclear powerplant is undergoing final development. These advanced fuel elements can eventually be used to replace the existing elements without major core modifications to uprate the performance. A reactor concept with such flexibility in development should provide the least risk in achieving a useful man-rated nuclear rocket with a minimum of time and cost.

MATERIALS RESEARCH

The most important problem area of any nuclear rocket concept is providing fueled materials capable of retaining fissionable material at temperatures in the range of 4500° to 5000° F while heating hydrogen at extremely high flow rates and heat fluxes. Accordingly, the effort on materials at Lewis has received the greatest attention. The first endeavor, when the advanced nuclear rocket program was started in late 1955, was to find high-temperature fuel element materials that had the potential of the highest temperature, recyclability, and long life. The first step involved screening more than 30 high-temperature ceramic and metallic materials for compatibility with uranium dioxide (the most refractory uranium compound). Fortunately, it was found that the most refractory metal was compatible with uranium dioxide and hydrogen, which was to be the propellant. It was thought that metals would have the most promise for recyclability and long life because of their great ductility and high strength when compared to ceramic materials. After selecting the tungsten - uranium dioxide combination for the primary materials research program, ways for manufacturing useful shapes were explored. A powder metallurgy dispersion of uranium dioxide in a tungsten matrix offered the greatest promise. Figure 4 shows how this material is made at present. Uranium dioxide powder is mixed with tungsten powder and cold pressed and sintered into a compact of about 90-percent theoretical density. This compact is then sandwiched between foils of pure tungsten about 1 or 2 mils thick. This "sandwich" is then rolled in a hydrogen atmosphere at temperatures approaching 4000° F. The rolling process increases the density to greater than 99 percent of theoretical and also bonds the clad metallurgically to the meat. The bond area cannot be distinguished in photomicrographs of the joint. It was found that high density and clad were required to prevent disastrous uranium dioxide vaporization above 4000° F.

Samples of elements such as these have been tested by electrically heating to temperatures above 5000° F in vacuum and in flowing hydrogen for varying amounts of time ranging from minutes to hours for large numbers of rapid thermal cycles with no adverse effects. In-pile tests were conducted about 3 years ago in the Westinghouse Test Reactor before it was shut down. Samples about 1 inch square, 30 mils thick, with 20 volume percent of fully enriched uranium dioxide in the meat, with only the flat faces clad with about 2 mils of pure tungsten, were tested in evacuated capsules such as shown in the exploded view of figure 5. Cooling was provided by thermal radiation through the molybdenum radiation shield to the water-cooled stainless walls of the capsule. Each of seven specimens was run for 4 hours at temperatures varying from 4720° to 5430° F. All of the specimens including the one operated at 5430° R for 4 hours indicated no loss of uranium dioxide through the clad faces. Figure 6 shows photomicrographs made of this specimen. The right-hand portion shows a region along the flat surface with the 2-mil clad. The uranium dioxide, which shows up as gray spots in this figure, shows no migration at all through the clad. On the other hand, a section through the unclad edge shows uranium oxide loss to a distance of 7 or 8 mils from the unclad edge. Besides indicating no effect of radiation on the readily observable properties of the fueled material, the in-pile tests conclusively show the need for the clad and that cladding is effective in reducing uranium dioxide loss to nothing at temperatures of at least 5430° R for 4 hours of operating time.

There is still much work to be done in utilizing this basic fueled material in suitable heat-transfer and flow configurations required for use in a nuclear reactor. The Lewis program is heavily concentrated in this area at present.

By means of hot hydrogen flow tests of tungsten fuel element designs, electrically heated fuel element simulation, thermal cycling tests, and vibration tests, Lewis is attempting to determine the operating limits and potential of fuel element geometries. Particularly important is the ability of the fuel element to withstand the large aerodynamic forces imposed by the very high hydrogen flow rates required for high power density.

There are other potential problem areas such as heat-exchanger operation without freezing of water, maintenance of the stagnant hydrogen insulation layer, insulation of the hot end reactor header plate from the hot hydrogen in the reactor exit plenum, and provision of a simple, reliable control system. Each of the anticipated problem areas has been considered to some extent. In some cases experimental work is being done or has been done to determine whether the anticipated problems exist, and, if they do, how serious they might be. From the work that has been done on these problems to date, Lewis is confident that they can be overcome.

ANTICIPATED PERFORMANCE AND CONCLUDING REMARKS

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Based on electrically heated and in-pile experimental materials and fuel element research, the use of tungsten - uranium dioxide fuel elements should permit operation with peak fuel element temperatures up to 5000° F for at least 4 hours of operation with no loss in fuel. The tungsten-water-moderated reactor utilizing these elements should produce hydrogen outlet temperatures of 4000° F and perhaps 4500° F with further development. The corresponding specific impulse would be 825 and 880 seconds, respectively, for hydrogen temperatures of 4000° and 4500° F.

Based on simulated fuel element test results the reactor should be completely recyclable and have an operating life measured in hours. Early design studies indicate thrust-to-weight ratios of greater than 10, perhaps approaching 25, at power levels of 1000 megawatts up to 10,000 megawatts and higher.

Even though the anticipated performance is based on some experimental experience, especially with regard to operating temperature, cyclability, and life, the actual thrust-to-weight ratios of a reliable man-rated system can only be determined when the actual powerplant is finally developed and all the compromises and problems of a complete powerplant are finally made and solved. Based on the studies carried along for the past 7 years, Lewis feels confident that the water-moderated-tungsten concept can be developed into a reliable man-rated nuclear rocket propulsion system useful for manned planetary flight.

Conf. R.D. Author

HTRE NO. I CORE

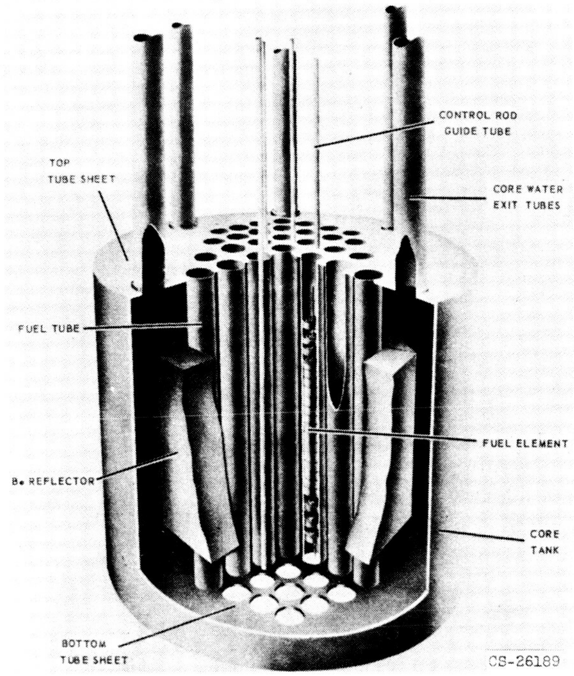

Figure 1

TUNGSTEN-WATER-MODERATED REACTOR CONCEPT

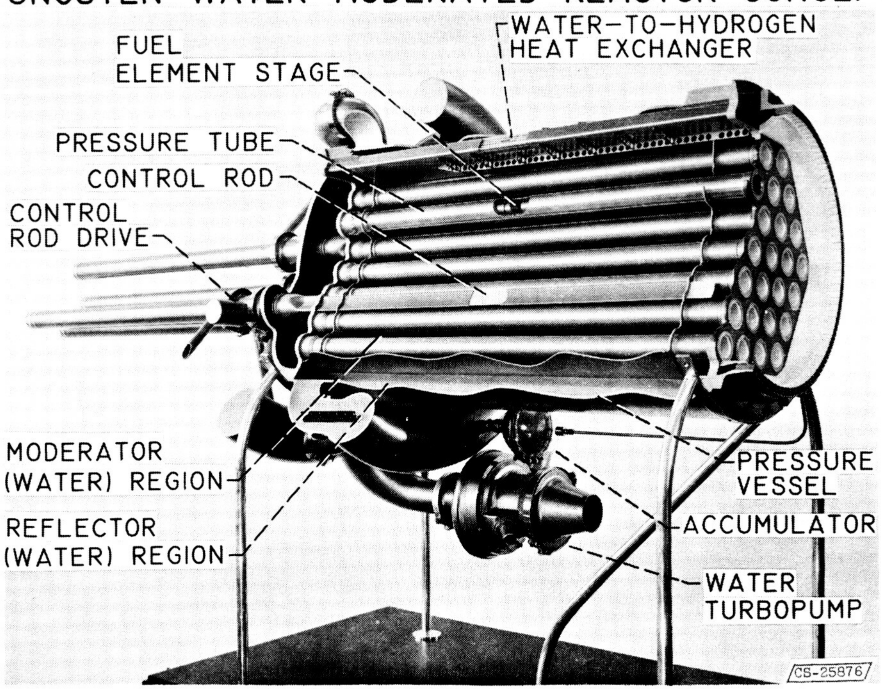

Figure 2

TYPICAL FUEL ELEMENT ASSEMBLY FOR TUNGSTEN-WATER-MODERATED REACTOR CONCEPT

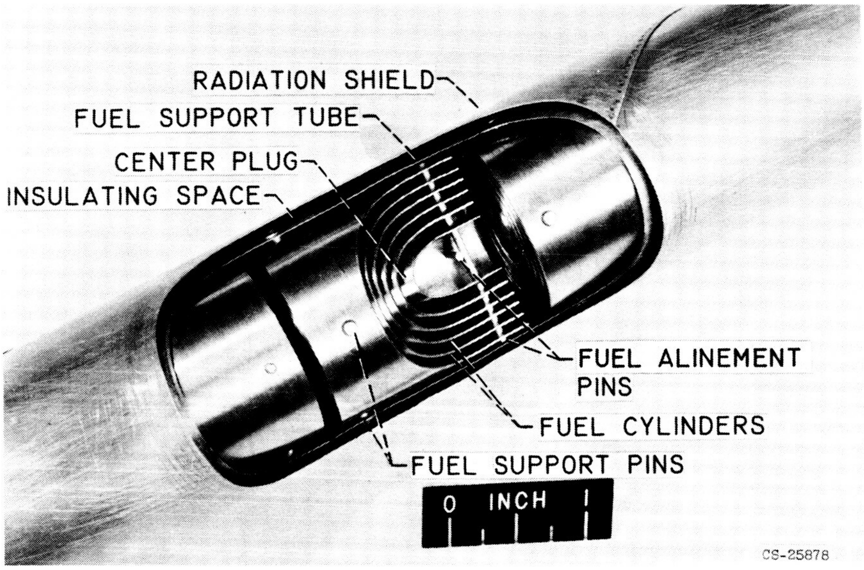

Figure 3

TUNGSTEN - URANIUM DIOXIDE FUEL ELEMENT

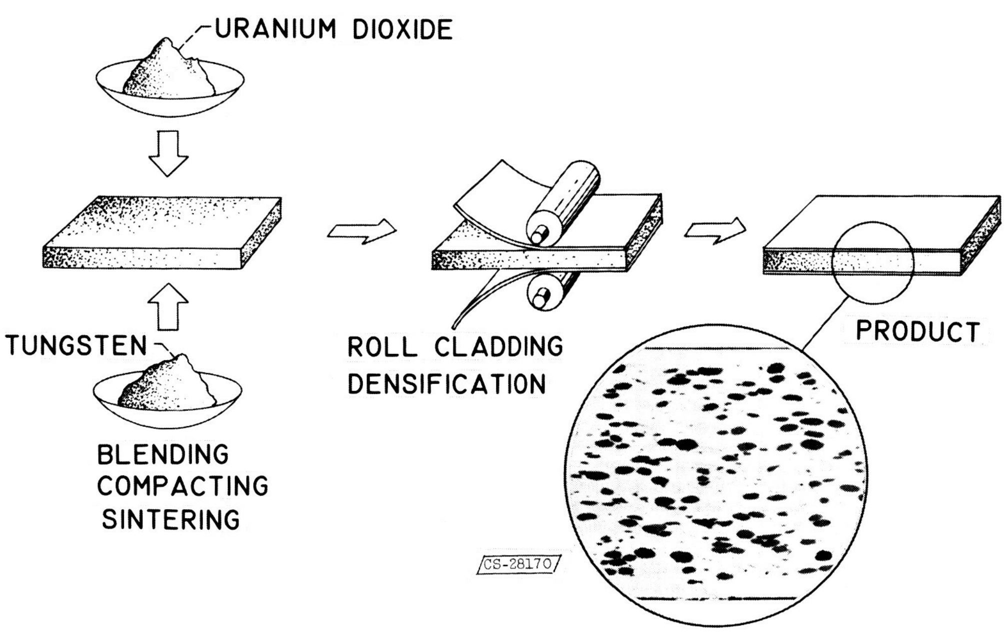

Figure 4

IN-PILE CAPSULE TESTS

SPECIMEN	APPROX. CENTER REGION TEMP, °R	APPROX. SURFACE HEAT FLUX, BTU/SEC IN.^{2}
1	4900	0.74
2	4720	.63
3	5050	.84
4	5200	.94
5	5430	1.12
6	5330	1.04
7	5250	.98

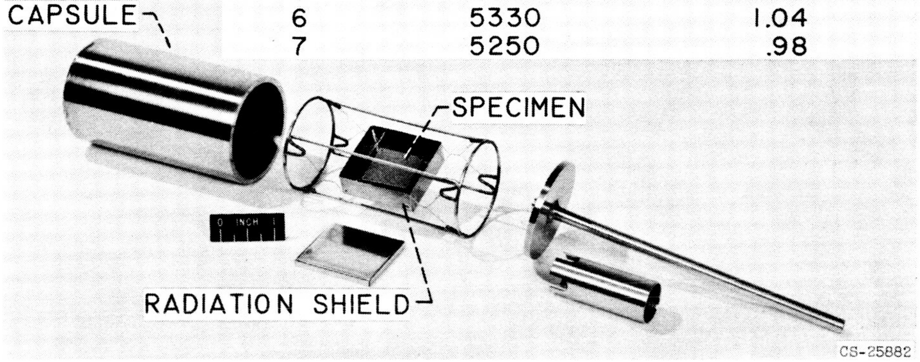

Figure 5

TUNGSTEN - URANIUM DIOXIDE FUEL ELEMENT SPECIMEN

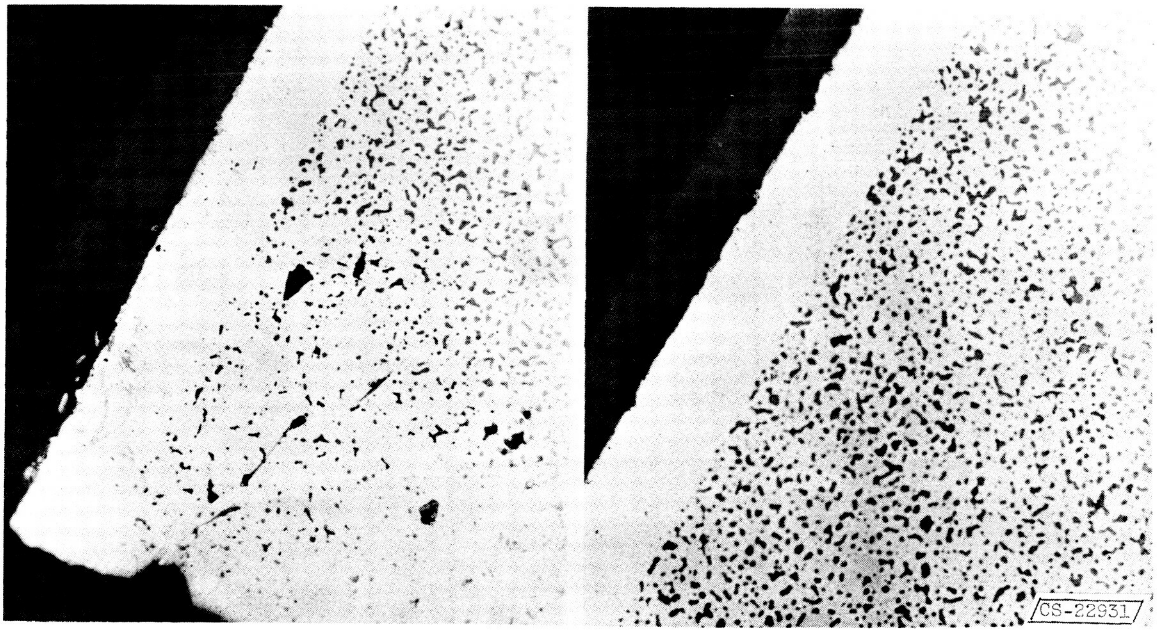

Figure 6

PRELIMINARY STUDY OF A NUCLEAR ROCKET SYSTEM FOR MANNED PLANETARY MISSIONS

By John H. Povolny

NASA Lewis Research Center

PRELIMINARY STUDY OF A NUCLEAR

ROCKET SYSTEM FOR MANNED u

PLANETARY MISSIONS

By John H. Povolny

NASA Lewis Research Center X65 50155

The primary purpose of the NERVA nuclear rocket currently under development is to demonstrate the feasibility of nuclear rocket propulsion. Once developed, it may have application in missions such as a manned lunar landing or space station supply. Unfortunately, it is too small for manned interplanetary missions. This being the case, it was decided to examine a second-generation nuclear rocket suitable for manned interplanetary flight in order to get an idea of component characteristics and problem areas. This examination is, in general, based on current technology and includes the graphite reactor concept, a nozzle fabricated from Inconel X tubing, and conventional turbopump blading and bearings. Some of the preliminary results of this examination are reported herein.

Conf. R.D, Author

The manned planetary mission on which the study of a second-generation nuclear rocket system is based is illustrated in figure 1. Some of the ground rules for this mission were as follows:

(1) Seven-man mission with four men on Mars for a 40-day stay

(2) Separate nuclear engines for all three propulsion phases starting with departure from Earth's orbit

(3) Empty tanks and used engines jettisoned

(4) Noncoplanar, elliptic planetary orbits considered

(5) 1979 Departure date

The component weights assumed were as follows:

Earth return vehicle, lb	14,000
Space vehicle quarters and radiation shield (solar and nuclear), lb . . .	57,000
Mars landing vehicle, lb	96,000
Consumables, lb/day	45

The mission study involved an optimization of energy requirement, weight, trip time, perihelion, and so forth. Much more detail will be given in papers by Mr. R. W. Luidens. A summary of the results obtained is presented in figure 2, which shows first-phase power requirement as a function of trip time. The region of interest includes power requirements ranging from about 4000 to 24,000 megawatts. The low end of this range will presumably be taken care of by the growth potential of the NERVA engine, but the high end will require a new engine development program. Inasmuch as 15,000 megawatts represents an order of magnitude increase in size over the NERVA, it was decided that this would be a good place to start the examination.

The effect of some cycle parameters on the performance of nuclear rockets is presented in figure 3 as a plot of engine specific impulse against rocket chamber pressure for both the "topping" and "hot-bleed" cycles. These cycles differ principally in the source of the turbine gas supply. For the hot-bleed cycle, a small amount of hot gas is bled off at the reactor exit, blended with some cold gas to a temperature of about 2000° R, and then supplied to the turbine. The turbine is preferably a multistage unit operating at a pressure ratio in the range of 10 to 20. For the topping cycle the propellant is partway heated in one or more of the engine components (such as the nozzle, core support, shield, etc.); most of it is then passed through the turbine, after which it is returned to the cycle to be heated to the rated propellant temperature. The turbine for this cycle is one stage and operates at a relatively low temperature and pressure ratio (1.8 to 2.0).

The performance obtained with the topping cycle is represented by the dashed line (fig. 3); the numbers on the line indicate the turbine-inlet temperature requirement. Although chamber pressure has no effect on specific impulse for the topping cycle, the turbine-inlet temperature requirement is a direct function. For pressures less than or equal to 500 pounds per square inch absolute the turbine energy requirement can possibly be met by the heat extracted from the nozzle, core support, and shield. For pressures greater than 500 pounds per square inch absolute an additional heat source is needed. Inasmuch as the only remaining source is the reactor, a two-pass reactor will thus be required. This gets considerably more complicated than the single-pass reactor.

The performance obtained with the hot-bleed cycle is represented by the two solid lines. The upper curve is for an overall turbopump efficiency of 50 percent, and the lower one is for an efficiency of 30 percent. (The lower curve is representative of current performances.) The numbers on the curves represent the percent turbine bleed required if the turbine is operated at a pressure ratio of 20. For the hot-bleed cycle, specific impulse is an inverse function of turbine bleed, which, in turn, is a function of chamber pressure and turbopump efficiency. For every 1 percent of turbine bleed, there is a loss of about $4\frac{1}{2}$ points of specific impulse.

The information in figure 3 may be summed up as follows:

(1) For the topping cycle, it appears desirable to keep chamber pressure low so as not to complicate the reactor.

(2) For the hot-bleed cycle, it appears desirable to keep chamber pressure low and turbopump efficiency high so as to minimize specific impulse losses.

The effect of cycle pressure on the estimated weight and size of the turbopump required for a 15,000-megawatt engine is presented in figure 4. Inasmuch as turbopump weight is primarily a function of pump discharge pressure, this is the variable against which weight is plotted. The curve shown is based on existing technology and is applicable to both axial and centrifugal units.

Examination of the curve reveals that turbopump weight and size are quite sensitive to pump discharge pressure. For example, for a pump pressure of 1000 pounds per square inch absolute the turbopump would weigh about 12,000 pounds and would be 7 to 10 feet long. For a pump pressure of 3500 pounds per square inch the weight would increase to about 42,000 pounds and the length from 10 to 15 feet. It is estimated that turbopump weight could be reduced by half with the development of advanced blading and bearings. If this is so, then it is apparent that advanced technology becomes more and more desirable as pump discharge pressure is increased. For example, at a discharge pressure of 3500 pounds per square inch absolute, advanced technology would save about 20,000 pounds, whereas at 1500 pounds per square inch absolute and less, the savings would be less than 10,000 pounds.

The effects of changes in chamber pressure on turbopump weight are considerably influenced by the cycle selected. This can be illustrated with the aid of figure 4, which presents plots of chamber pressure against pump discharge pressure for both the hot-bleed and topping cycles. (It will be recalled that the topping cycle requires higher pump pressures because the turbine pressure drop is in series with that of the other components.) For an increase in hot-bleed-cycle chamber pressure from 500 to 1500 pounds per square inch absolute, the turbopump weight would increase about 6000 pounds (from 12,000 to 18,000 lb), whereas a similar change in topping-cycle chamber pressure would increase the weight about 22,000 pounds (from 16,000 to 38,000 lb). Thus, from the standpoint of turbopump weight, chamber pressures of the order of 500 pounds per square inch absolute appear preferable.

The variation of nozzle-throat heat flux and tube-wall temperature with chamber pressure for the 15,000-megawatt engine with Inconel X tubes and operating at a chamber temperature of 4500° R is presented in figure 5. It should be remembered that these curves are for a series of optimum designs and are not the performance of one design. The significant item to note in this figure is that as chamber pressure is increased, wall temperature can be held to a desired value until the coolant passage chokes. After this, the heat-flux requirements cannot be met on the coolant side, and a considerable increase in maximum wall temperature is encountered. For example, a wall temperature of 3500° R is indicated for a chamber pressure of 1500 pounds per square inch absolute. This is far in excess of current material capabilities. If the maximum wall temperature is limited to the current practice of 2000° R, then the chamber pressure is limited to about 700 pounds per square inch absolute. If it is assumed that future material improvements will permit a maximum wall temperature of 2600° R, the limitation can be extended to about 1000 pounds per square inch absolute. Ceramic coatings, ablative coatings, film cooling, and so forth, may permit higher pressures if they are required, but considerable development will be required in order to apply these techniques.

The effect of the chamber pressure in the 15,000-megawatt engine on nozzle-throat tube-wall strain is presented in figure 6 for the same conditions as the lower tube-wall temperature curve in figure 5. Two curves are shown, one for the tangential and one for the longitudinal strain. The tangential strain is a result of the pressure difference and temperature gradient across the hot wall of the tube (see sketch in fig. 6). The longitudinal strain is a result of thermal stresses in the hot wall caused by the difference in temperature between the inner and outer nozzle walls. It can be seen that the longitudinal strain far exceeds the tangential strain; however, both exceed the elastic limit (for the range considered) and thus are in the region of plastic deformation. Up to about 700 pounds per square inch, the longitudinal strain is constant with chamber pressure, because the hot-wall temperature is constant. Beyond this, there is a rapid increase in longitudinal strain because of the rapidly increasing hot-wall temperature. The value of longitudinal strain (0.0165) indicated for pressures under 700 pounds per square inch absolute is typical of that encountered in the Inconel X nozzles currently under development for NERVA. Thus far, none of these nozzles has proved satisfactory, and it is quite possible that nozzle technological improvement is required even if low chamber pressures are to be utilized in the advanced nuclear rocket.

The effect of chamber pressure on exhaust-nozzle length and weight for the 15,000-megawatt engine is shown in figure 7 for nozzle expansion ratios of 25, 50, and 100. Included for reference are points that indicate the lengths of the NERVA (6 ft) and the M-1 (21 ft) exhaust nozzles.

Changes in nozzle weight and length with reductions in chamber pressure are relatively minor down to a pressure of about 1000 pounds per square inch absolute; below this value, they increase at a rapidly accelerating rate. The rapid weight increase is a result of the constant tube-wall thickness (0.0090 in.), which, in turn, is necessitated by longitudinal strain considerations. If fabrication techniques can be devised to alleviate the longitudinal strain, appreciable weight reductions might be realized at the lower pressures. Even with current technology, however, the nozzle weights at a chamber pressure of 500 pounds per square inch absolute do not appear excessive when it is considered that the higher area ratios provide greater impulse.

Of perhaps greater significance is the size of the nozzle, because it not only affects engine weight but influences interstage structure weight as well. At a chamber pressure of 500 pounds per square inch absolute, a 25:1 conventional nozzle would be about 20 feet long, whereas a 100:1 nozzle would be about twice that or 40 feet long. Thus, if low chamber pressures and high area ratios are to be employed, serious consideration should be given to unconventional, folded, or multinozzle arrangements.

A preliminary look at a graphite reactor for a 15,000-megawatt engine indicated that it would weigh about 12,000 pounds (excluding the shield) and that the weight would be essentially unaffected by chamber pressure. The shape or volume of the reactor, however, probably would be affected by pressure level, with the lower pressures requiring larger diameters. Inasmuch as larger diameters will mean a larger and heavier shadow shield, it was decided to take a quick look at this component. This look indicated that the weight of a shadow shield for a 15,000-megawatt engine will vary from about 5500 pounds at a chamber pressure of 1500 pounds per square inch absolute to about 8000 pounds at a chamber pressure of 500 pounds per square inch absolute. Inasmuch as the change in shield weight is only 2500 pounds, this effect is considered to be insignificant.

. Another factor to consider is the effect of diameter on reactor controllability. As the diameter becomes larger, it becomes increasingly difficult to control the reactor by means of a simple peripheral rotating-drum control system. One method of handling this situation would be to use sliding control rodes dispersed throughout the core or more complex "island type" control drums. Another method would be to operate at higher pressures thereby permitting smaller diameters. It is to be emphasized that these considerations plus many others must be taken into account in the optimization of the reactor and propulsion system.

The variation of the sum of the weights of the reactor, the nozzle, and the turbopump with chamber pressure for the 15,000-megawatt engine is presented in figure 8 for the three nozzle expansion ratios previously considered. (The weight of the shield is not included.) It should be recognized that although these three components are the principal constituents of a nuclear rocket, items such as the interstage structure and impulse trade-offs associated with cycle and nozzle selection could also influence the selection of the optimum chamber pressure. For both cycles and the nozzle area ratios considered, the minimum combined reactor, nozzle, and turbopump weights occur at about 500 pounds per square inch absolute chamber pressure. Variations in chamber pressure from about 300 to 1000 pounds per square inch absolute, however, do not have an appreciable effect on the combined weights because of the compensating effects of the turbopump and the nozzle. The higher weights required for the topping cycle are a result of the higher pump pressures that this cycle requires. For the range of chamber pressures considered (500 to 1500 psia) the difference in weight is not too great and is more than compensated for by the higher specific impulse obtained.

This preliminary study of some of the variables and components of a 15,000-megawatt nuclear rocket engine suitable for a possible manned Mars mission has indicated that the selection of the cycle and the engine design chamber pressure will probably not be determined by engine weight considerations (if the chamber pressure falls in the range of 300 to 1000 psia), but more probably by reactor control and nozzle heat transfer, stress-strain, and size considerations. The selection of the optimum cycle and design pressure is quite a complex problem with many interrelated variables that undoubtedly will be influenced by technological developments of the future. This being the case, it is desirable that a continuing and more detailed study be pursued.

TYPICAL EARTH—MARS TRAJECTORY

TRIP TIME, APPROX. 500 DAYS
STAY AT MARS, 40 DAYS

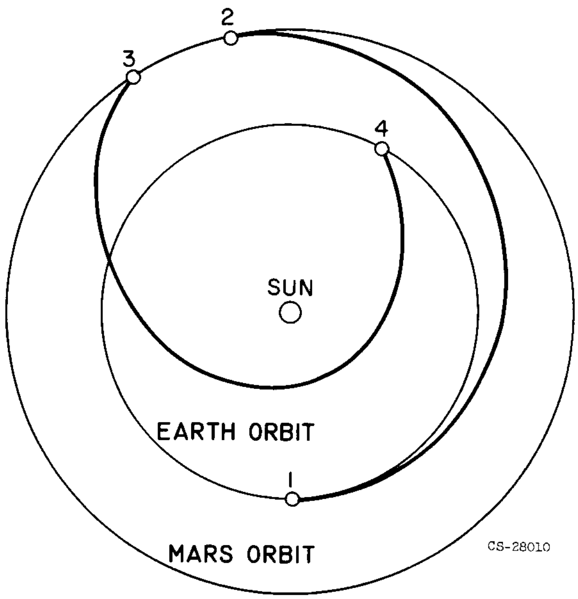

Figure 1

RANGE OF POSSIBLE POWER REQUIREMENTS FOR A MANNED MARS MISSION

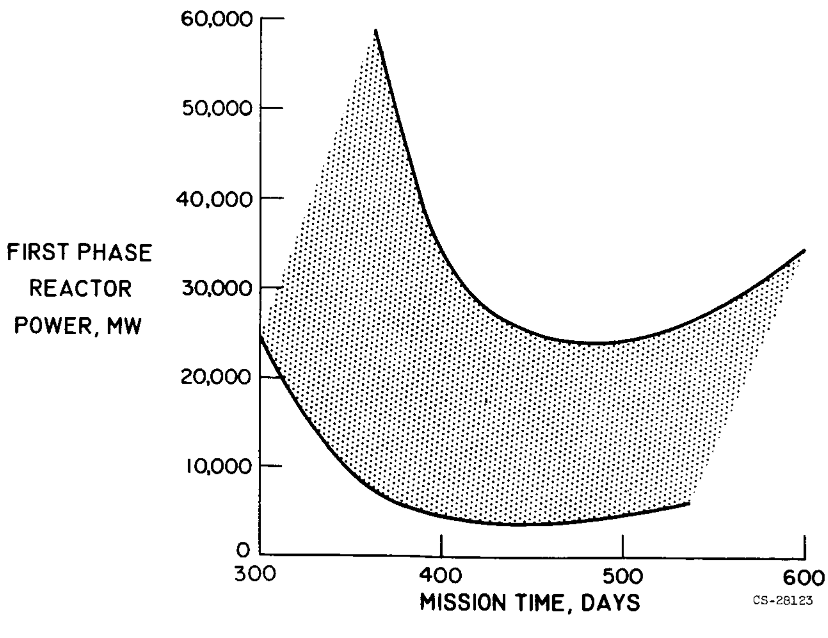

Figure 2

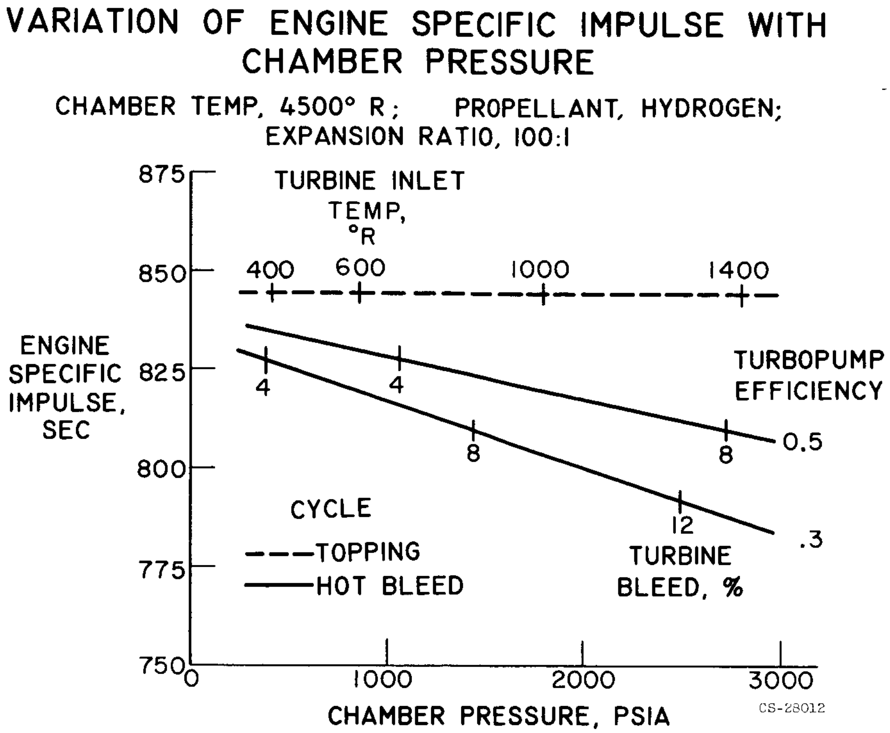

Figure 3

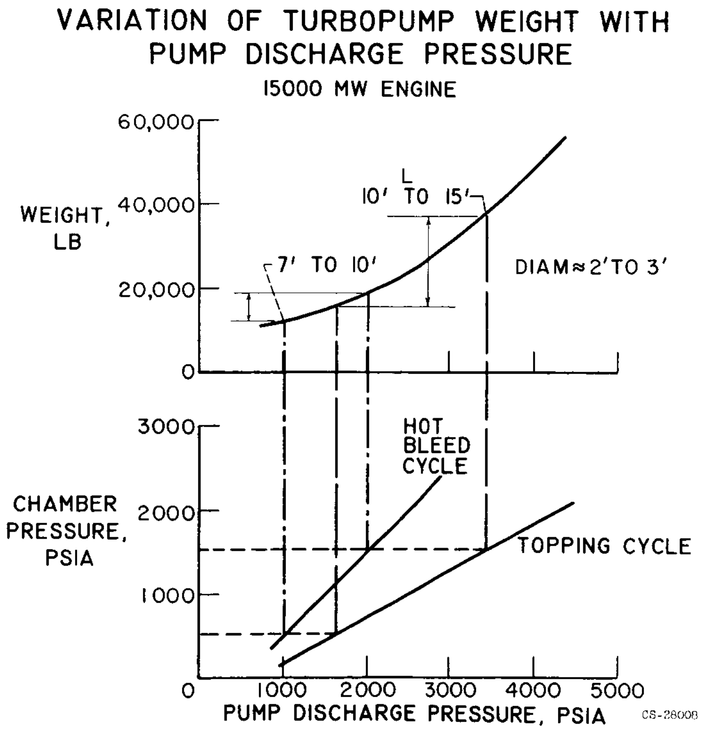

Figure 4

VARIATION OF NOZZLE-THROAT HEAT FLUX AND TUBE-WALL TEMPERATURE WITH CHAMBER PRESSURE

15000 MW ENGINE
CHAMBER TEMP, 4500° R; MATERIAL, INCONEL X

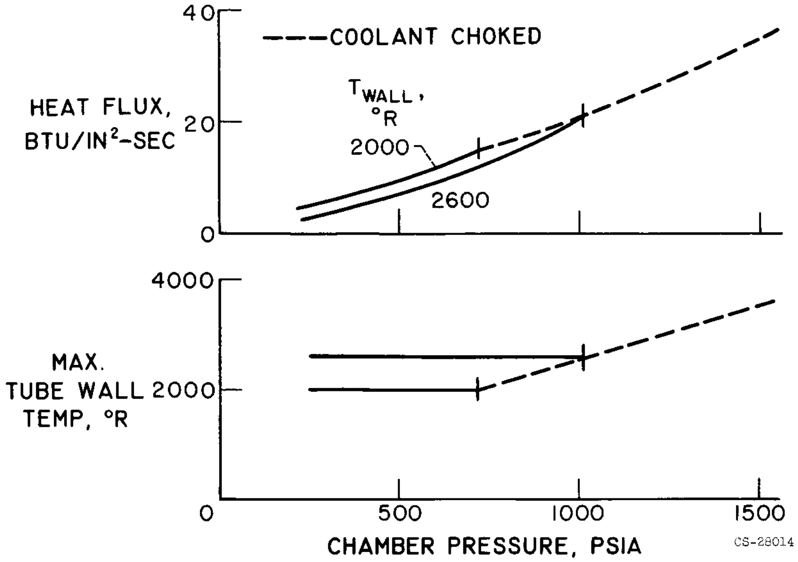

Figure 5

VARIATION OF NOZZLE-THROAT TUBE-WALL STRAIN WITH CHAMBER PRESSURE

15000 MW ENGINE
CHAMBER TEMP, 4500° R; MATERIAL, INCONEL X

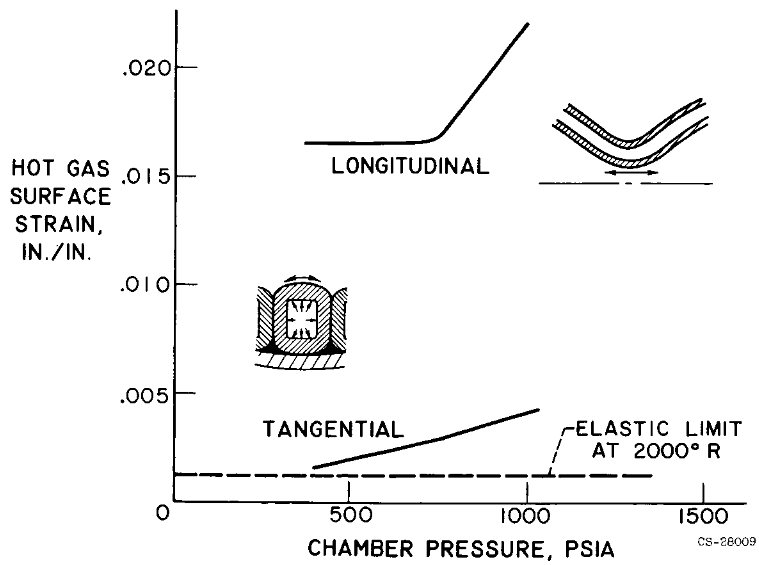

Figure 6

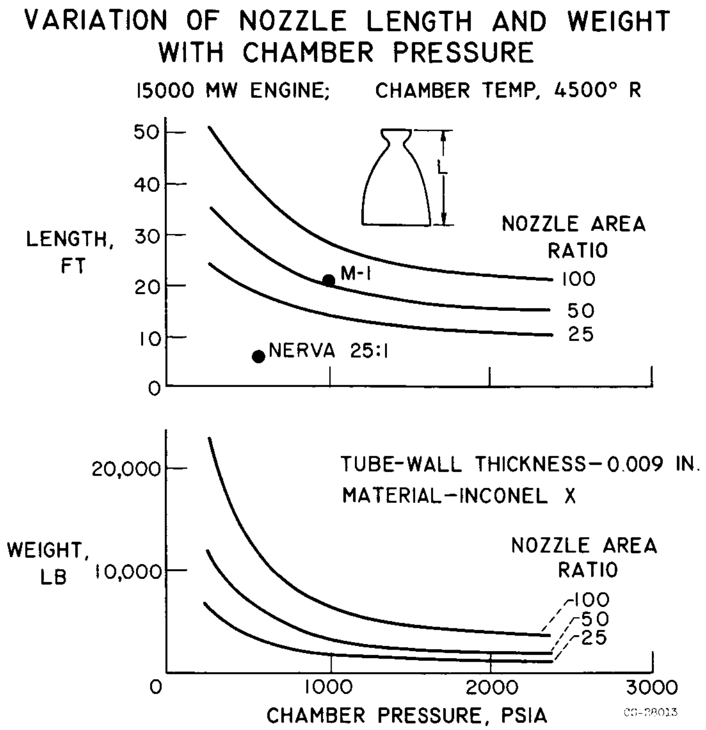

Figure 7

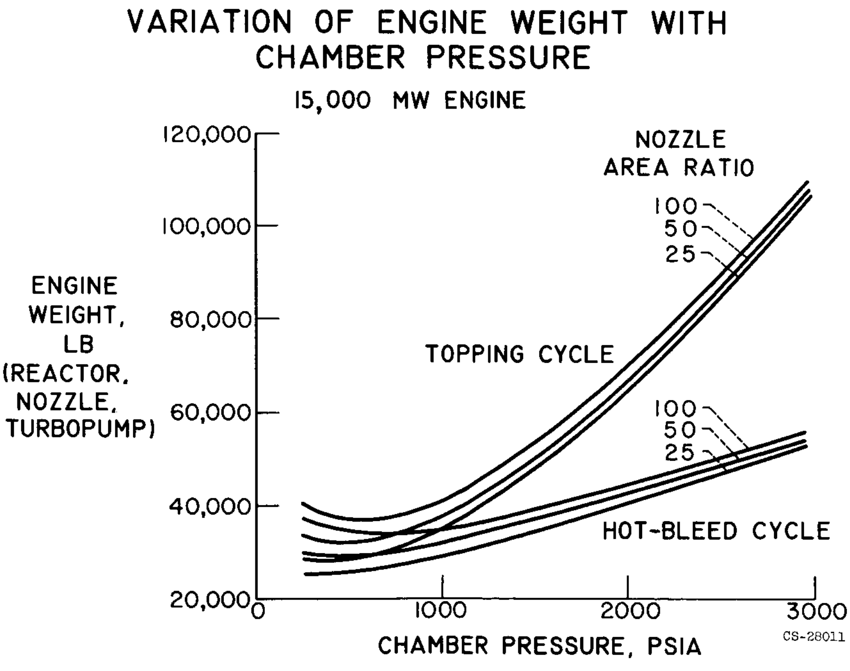

Figure 8

ELECTRIC PROPULSION FOR MANNED MISSIONS

By Robert J. Denington, Russell D. Shattuck,
and William J. LeGray

NASA Lewis Research Center

ELECTRIC PROPULSION FOR MANNED MISSIONS U

By Robert J. Denington, Russell D. Shattuck,
and William J. LeGray

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INTRODUCTION

Missions of Interest [redacted]

Electric propulsion is of interest for manned interplanetary missions, primarily because it offers the potential of delivering and returning relatively large payload fractions from the planets. This can be achieved because of the high specific impulse obtainable when the propellant is accelerated by electrical means rather than thermally, as in chemical and nuclear rockets.

The benefits of reduced propellant consumption are not, however, obtained without cost. For a given thrust level, increased electrical power must be put into the exhaust jet to obtain a higher specific impulse, and the powerplant weight increases correspondingly. When thrust level and specific impulse are optimized in order to balance propellant weight and powerplant weight while achieving a maximum net payload, the vehicle acceleration is typically found to be on the order of 10^{-3} to 10^{-4} g for most interplanetary missions. As a result, the electric spacecraft cannot take off from a planetary surface but must be boosted into orbit by a high-thrust engine. In addition, as a result of the low thrust, the electric vehicles require relatively long propulsion times to perform the missions of interest. In contrast to high-thrust operation, which consists of short power bursts followed by long coast periods, the electric system operates for long periods of time to provide equivalent velocity increments.

Electric Engine Defined

A typical electric engine is shown schematically in figure 1. It consists of an electric thrustor having a propellant feed system, the electrical power-conditioning equipment, and the electrical power supply. A powerplant consists of the latter two.

The electric thrustor accelerates the propellant to a high velocity, which produces specific impulses in the 1000- to 40,000-second range. The propellant feed system stores the propellant and controls its flow to the thrustor. Typically, the system consists of the tankage, a pressurizer, the flow-control provisions, and, in most cases, a propellant vaporizer. The electrical power-conditioning equipment converts the power supply output to the voltage and frequency required by the thrustor. It consists of inverters, transformers, rectifiers, and circuit breakers in the more complex installations. The electrical power supply consists of a heat source, an energy-conversion system, and a means of rejecting waste heat into space.

General Engine Specifications

The interest in an electric engine depends on its competitiveness relative to other propulsion schemes. The nuclear rocket appears to be the next major advance in propulsion, so a comparison of nuclear rockets and electric propulsion can serve as an illustration of the performance required of the electric engine. In figure 2 relative initial payload is plotted against mission time for a Mars round trip for nuclear rockets and electric propulsion. Three nuclear rocket curves [redacted]orable launch dates (1971, 1986, 2001, etc.), a less f[redacted]0), and specific impulses of 800 and 900 seconds. The [redacted] electric engine is shown parametrically as a function of powerplant specific weight for only one launch date as, in general, changes in launch dates have little effect on missions with high-specific-impulse engines.

From the comparison, it is apparent that low-specific-weight (10 to 15 lb/kw) electric engines capable of operating reliably for 400 to 500 days are required to compete successfully with nuclear rockets for the Mars mission. For longer higher energy missions, planetary missions (such as Jupiter or Pluto), or where large quantities of electric power are required at the destination, the electric engine is competitive at even higher specific weights.

The electrical power levels of interest for manned interplanetary flight vary, depending on the mission profile, payload requirements, and specific weight of the engine. A typical 450-day Mars mission for a net weight in Mars orbit of 170,000 pounds and a return weight to Earth orbit of 120,000 pounds can utilize a 20-megawatt, 11-pound-per-kilowatt power system. If heavier shielding weights are required for crew protection, the power requirements could increase. Or, if specific weights of the powerplant are lower, the power may increase. In general, the power levels for electric propulsion missions can be crudely estimated by assuming that one-third of the spacecraft weight in orbit will consist of powerplant, one-third of propellant, and the remainder of structure and payload. It appears that the power range of interest for the manned Mars mission is 5 to 40 megawatts, where the 5 megawatts correspond to missions with small spacecraft using electric propulsion from Earth escape and the 40 megawatts are for larger spacecraft starting from Earth orbit.

While it is likely that large electric engines, if developed, will be designed for manned missions, there are many useful unmanned missions that could be performed during the powerplant "man rating" phase and after the engine is developed. The power levels of interest for the unmanned missions are related to booster capability. Typical examples are:

(1) Starting with a Saturn C-5 payload of about 200,000 pounds in Earth orbit, 5 megawatts of electric power can be used to provide a large Jupiter satellite.

(2) With a Saturn C-5 payload of 60,000 pounds placed beyond Earth escape, 1 megawatt of electric power can be used for a smaller Jupiter orbiter.

(3) With a Saturn C-1B, 28,000 pounds can be placed in a 300-mile orbit. A spacecraft of this size with about 400 kilowatts of electric power can be used to perform missions such as a solar orbiter or Mercury orbiter missions.

In general, the unmanned missions are longer, requiring 700 to 800 days for Jupiter orbiters and up to 1200 days for Pluto probes. For these higher energy missions, the electric engine can compete successfully with a nuclear rocket with powerplant specific weights up to 30 pounds per kilowatt.

Preliminary analysis indicates that the engine should, also, be of modular construction and that a useful module size is about 5 megawatts electric. One module would consist of many thrustors, a single reactor, and one or more power-conversion systems. Two to four 5-megawatt-electric modules could then be used for a manned mission, and one 5-megawatt-electric module at rated power or derated module for the unmanned missions.

Other general requirements for the electric engine can be inferred from these mission considerations. To operate in the space environment, the engine must tolerate vacuum conditions, meteoroid impacts, and space radiations. It can only reject waste heat by thermal radiation in the vacuum. As most space power systems must reject 7 to 10 kilowatts of thermal energy for each kilowatt of electric power generated, the radiator becomes prohibitively large at high powers unless operated at high temperatures. As an example, if heat is rejected at 600° F, approximately 1 acre or 40,000 square feet of radiating surface are required per megawatt of electric power generated. While at 1500° F, only 2000 square feet are required. Since radiators typically weigh 1 to 5 pounds per square foot, the achievement of lightweight power systems requires that heat be rejected in a 1200° to 1800° F temperature range. The high heat-rejection temperatures in turn imply a requirement for high heat source temperatures of 2000° to 3500° F, depending upon the power-conversion system.

An electric engine must utilize a nuclear heat source. Chemical heat sources are too heavy for the long missions. Solar heat sources require large collectors (about 70 sq ft/kw), which are much heavier than nuclear sources at higher powers. Isotopes are not available in sufficient quantities for manned missions, and a fusion reactor is not yet conceivable. Thus, the nuclear fission reactor heat source is the only one of interest.

A closed-cycle power system that incorporates its own heat-transfer media must be used. While small quantities of propellant are continually exhausted from the engine, a simple thermodynamic calculation will disclose that the propellant weight flow is insufficient for transferring heat in any adequate power-generating system because of the inefficiencies in the electrical and thrustor systems.

The stringent reliability requirement for manned missions, coupled with the need for long life at high temperatures, imposes a severe burden on the engine. It is doubtful whether this reliability and life can be achieved without extensive redundancy and in-flight repairs.

In summary, the general requirements for the electric engine, based on both manned Mars and unmanned missions, are tabulated in table I. The following discussion describes current thrustor and electric powerplant concepts, summarizes their development status, and outlines some of the problems to be resolved.

ELECTRIC THRUSTORS

Thrustors of Interest

To attain high specific impulses without the need for the extremely high exhaust-gas temperatures of nuclear rockets, the thermal energy of the nuclear heat source is first converted to electrical power at more manageable temperatures. This power is then used to produce vehicle accelerations by energizing electric thrustor devices. As a result, the impulse-producing momentum of the expellant is determined by electrical rather than thermal considerations, and much higher exit velocities and lower propellant mass-flow rates result. On the other hand, thrust levels are low for the same energy consumption, which results in long thrusting times.

Electric thrustors are not limited in their capability to generate high specific impulses. However, mission optimization has indicated a need for thrustor devices that operate over a wide range of specific impulse (1000 to 20,000 sec). It is doubtful whether such a range can be covered efficiently with a single type of thrustor. Consequently, a variety of electric thrustor devices is currently being considered.

For thrust generation, three classes of devices are currently being investigated - electrothermal, electrostatic, and electromagnetic. Potentially, electromagnetic accelerators should be competitive, at least, in the low-specific-impulse range of 1000 to 5000 seconds. This is because the integration of ionization and acceleration processes promises the efficient utilization of the ions produced. However, the operating efficiencies of these devices to date have been relatively low (10 to 30 percent), and developmental problems exist in an area where there is little experience. Electrothermal thrustors (resisto jet and arc jet) and electrostatic thrustors (contact ionization and electron bombardment ionization) have demonstrated much higher operating efficiencies, and, for this reason, only the electrothermal and the electrostatic thrustors will be discussed in detail.

Thrustor Descriptions

The electrothermal thrustor devices heat the propellant electrically and expand it through a nozzle to produce thrust. On the other hand, the electrostatic devices ionize the propellant and accelerate the ions electrostatically. Variations of these thrustor concepts are shown in figure 3. Electric heating of the propellant in the electrothermal devices may be accomplished by using a resistance heat exchanger or by striking an arc to heat the propellant to high thermal energy levels.

The electrostatic thrustors develop thrust by accelerating the ionized propellant by using electric fields. The production of ions in the electrostatic device can be achieved either by contact ionization, which occurs on hot surfaces, or by bombardment of the propellant with high-energy electrons to detach electrons from the propellant atoms.

To achieve high-specific-impulse performance, the electrothermal devices utilize hydrogen or ammonia as the propellant. The heating surfaces in the resisto jet are refractory metal coils and operate at temperatures of about 4200° F. The contact ionization thrustor utilizes cesium propellant vapor and produces ions by diffusing cesium through a 2000° F porous tungsten plug and forming ions on the surface of the tungsten. The electron bombardment ionization thrustor has been demonstrated with mercury, cesium, argon, and xenon as the propellant. Both cesium and mercury have been used in operational prototype devices. Cesium appears to be more attractive because cathode sputtering is reduced and propellant utilization is higher, and, as a result, longer life and higher overall rocket efficiency are promised. The noble gases are being considered primarily to reduce the materials corrosion problems.

Although the electrothermal devices that have been operated have produced only low-level specific impulse (850 to 1050 sec) and have not demonstrated high efficiencies, they remain attractive because they produce a specific impulse in this range, and their electrical power input requirements are characterized by low voltages and high currents. On the other hand, the electrostatic devices have produced specific impulses ranging from 5000 to 7600 seconds with power efficiencies ranging from 62 to 82 percent but require direct-current voltages on the order of a few thousand volts.

Development Status and Problems

Prototype versions of the four thrustor devices that have been described have been operated in the laboratory. Actual performance figures, or realistic performance estimates believed possible within the next year or less, are given in table II. It is seen that a wide spectrum of specific impulses is being successfully pursued ranging from 1000 to about 10,000 seconds and at power efficiencies up to 82 percent. The specific areas are about 0.001 square foot per kilowatt for the electrothermal thrustors and about 0.1 square foot per kilowatt for the electrostatic devices. Specific weights as low as $\frac{1}{4}$ and $1\frac{1}{2}$ pounds per kilowatt for the electrothermal and the electrostatic thrustors, respectively, are indicated. Lifetimes of a few hundred hours have been demonstrated by complete thrustors, and some of the more critical components have shown lifetimes of a few thousand hours.

The thrustor specific areas that appear feasible indicate that some deployment of ion thrustors will be necessary to propel manned spacecraft after launch to Earth orbit or escape. Probably, a minimum specific area will occur for 30- to 50-kilowatt thrustor modules and will be about 0.1 square foot per kilowatt.

The specific weight of the electrostatic thrustors is higher than desired, the lightest being 1.3 pounds per kilowatt. This thrustor is the only one in table II that was designed with weight in mind, and even lighter designs are thought to be possible. (This unit, although designed, has yet to be built and tested.) The mission analyses performed to date at the Lewis Research Center have assumed that thrustor weights will be about 5 percent of the powerplant weight. If the thrustor should weigh more, the power supply will have to be correspondingly lighter.

Each thrustor device has its own requirements for new technology, which result because of life, power efficiency, or propellant utilization deficiencies, or various combinations of the three. Problems related to the four devices of interest will be discussed separately.

Electrothermal thrustors. - As an electric propulsion device, the resisto jet is promising for several reasons: (1) It is electrically simple, (2) it can be adapted to almost any power supply voltage (a.c. or d.c.), and (3) it does not produce electrical noise. Unfortunately, the temperatures to which the propellant must be heated approach the melting point of tungsten. The high temperatures and high flow velocities make erosion a serious problem for long lifetime.

Development of the arc jet has been emphasized more than that of the resisto jet. Because the devices are similar, many of the materials and component problems, which are common to both, are being investigated.

Electrically, the arc jet is more complex than the resisto jet, but less complex than the electrostatic thrustors. The arc jet is more difficult to start and operate stably, largely because of the negative resistance characteristic of the arc. It produces more electrical noise and is not as flexible in adapting to power supplies as the resisto jet. Because of the low specific impulses produced by the electrothermal devices, arc jets are not presently being considered as serious candidates for manned interplanetary missions, but rather for Earth satellite attitude control or lunar ferries. Because they may eventually even be attractive for manned missions, arc jets appear in this discussion.

Electrostatic thrustors. - The contact ionization thrustor is probably the oldest concept for electric propulsion. This concept is simpler, electrically, than the electron bombardment thrustor, and its propellant utilization can potentially approach 95 percent. The power efficiency for this device is severely limited because heat is radiated from the ionizer surface directly to space. Since the heat loss is essentially constant for a given ionizer temperature, it is necessary to operate at a high specific impulse to raise the ratio of beam power to total power input. This implies that for the contact ionization thrustor a lower limit of specific impulse exists below which its power efficiency is not acceptable. In order to improve the power efficiency, it is desirable to operate the ionizer at a high current density to minimize its heat radiating area. Unfortunately, this leads to an increased loss of un-ionized propellant. This balance between the propellant loss and the heat loss occurs near 95 percent propellant utilization.

Operation of the ionizer at high temperatures brings about serious, irreversible property changes in the ionizer. The porous material continues to sinter and, as the pore sizes increase, propellant losses increase.

The propellant loss is of consequence for two reasons: The first is the payload weight penalty, and the second, and most serious, is due to the existence of neutral particles in the exhaust beam. When un-ionized atoms exist in the accelerator space, a predictable portion of these neutrals interacts with the high-velocity ions, and charge exchange results. As a result, an ion having essentially zero velocity suddenly appears at a place in the accelerator space where it should have a very high axial velocity along the line of thrust. Thus, the new ion can only accelerate toward the accelerator structure where it impacts at high velocities and causes erosion and subsequent loss of the critical electrode shape. This damage can limit the life of the accelerators to a few hundred hours. Much research is being conducted in each of these areas, and progress is being made.

The electron bombardment thrustor is being developed because it avoids the high-temperature ionizer problems and is not as sensitive to current density as the contact ionization thrustor. It does, however, introduce other almost as serious problems.

Propellant utilization is not as good as with the contact ionization thrustor, typically ranging from 80 to a little over 90 percent. Poor propellant utilization introduces similar problems with the electron bombardment thrustor as with the contact ionization thrustor.

Thrustor Performance Potential

The thrustor devices that are currently being developed promise to meet a whole spectrum of specific impulse requirements for various missions. A comparison of power efficiency plotted against specific impulse for the devices that have been discussed is given in figure 4. Although a specific impulse of only 1000 seconds has been experienced to date, the electrothermal devices have the potential capability of 1500 seconds. For the contact ionization devices, porous tungsten ionizers capable of producing higher current densities along with possibilities for improved propellant utilization promise to provide efficiencies approaching 90 percent. Improved electron bombardment emitters and possibilities for improved propellant utilization promise efficiencies up to 90 percent for the electron bombardment devices. Experience with prototype electron bombardment thrustors indicates that specific weights may be 1 pound per kilowatt or less. Specific areas for ion thrustors are currently about 0.1 square foot per kilowatt, which appears applicable to manned missions if provisions are made to deploy some of the thrustor modules.

In summary, three thrustor concepts are nearing engineering phases of development. The two electrostatic thrustors are scheduled for flight testing in 1963 and 1965; the first test is scheduled for summer 1963. Ion accelerating efficiencies are near theoretical, but there is much room for improvement in the process of propellant ionization. Efforts to improve ionization and propellant utilization efficiencies to reduce power requirements and increase payload and life capability, respectively, are being performed under several NASA contracts. In addition, heater coil, nozzle, ionizer, and electron bombardment cathode material capabilities are being investigated by NASA.

The most troublesome problem has been the electron emitter lifetime. Long-life emitters have been built, but they represent the results of a very rugged brute-force approach, and poor power efficiency has resulted. A NASA contract currently exists to develop a cesium electron bombardment ionization source, which employs a cesiated cathode. Since the propellant is also cesium, the cathode can be self-heated by bombardment with energetic cesium to maintain its temperature. Such a device promises long-life performance, but has yet to be demonstrated for more than a few hours.

Flight Test Programs

Within the next 2 years and beginning this summer, flight tests of electrostatic thrustor devices are planned. The first, SERT-I, is to determine thrust capability and the possibility of neutralization of the departing particles in space. Next, the SERT-III test will check out the attitude control capability of ion thrustors by incorporating them in a 24-hour synchronous satellite. Third, the SERT-II test is not as well defined, but current plans are to incorporate it into a 180-day Earth orbit satellite to study long-term effects of the space environment. The tests will incorporate 0.5-, 1-, and 3-kilowatt ion thrustors, respectively. SERT-I will have a contact ionization and an electron bombardment thrustor on board, SERT-III will include several contact ionization thrustors, and SERT-II is presently scheduled to be a contact ionization thrustor. Because electron bombardment thrustors are progressing rapidly, they may be used for the SERT-II test instead.

ELECTRIC POWERPLANTS

Power Supplies of Interest

The electrical power supply is the heaviest subsystem of the electric engine, potentially the least reliable, and the most difficult to develop. At present, there is no power system flight hardware under development that approaches the performance required for electric propulsion applications. There are, however, technology programs, such as the Air Force's SPUR, the AEC's SNAP-50, and NASA's advanced system, that may some day provide sufficient data to design such hardware.

Three nuclear-reactor-powered space electric-generating systems are under development: the SNAP-10A, the SNAP-2, and the SNAP-8. The SNAP-10A is a 500-watt, 900° F nuclear reactor system incorporating thermoelectric power conversion and weighing about 800 pounds per kilowatt. SNAP-2 is a 3-kilowatt, 1200° F nuclear reactor system with a mercury Rankine cycle power-conversion system and weighs approximately 250 pounds per kilowatt. SNAP-8, a 35-kilowatt, 1300° F mercury Rankine cycle system, is similar to SNAP-2 and weighs about 150 pounds per kilowatt. All these specific weights include nominal shielding for electronics. The specific weights and power levels are summarized in figure 5, where they are compared with the requirements of the advanced powerplants. It is apparent that considerable advance in the state of the art is needed to achieve the performance required, which for electric propulsion is less than 30 pounds per kilowatt at temperatures exceeding 2000° F.

Because the power supplies of interest all require a nuclear reactor heat source, the primary difference between power supplies is in power conversion. Power-conversion schemes of potential interest are the Rankine cycle, the Brayton cycle, thermionic conversion, thermoelectric, and magnetohydrodynamics. Of these five schemes, only the Rankine cycle, thermionic, and magnetohydrodynamics presently show promise for manned missions. The Brayton cycle requires relatively low radiator temperatures, and, consequently, the powerplant is too heavy. The thermoelectric-conversion scheme operates at low temperatures, is relatively low in efficiency, and is consequently too heavy. Magnetohydrodynamics shows promise, but the technology is presently not far enough advanced to be discussed realistically.

Rankine Cycle Powerplant

The Rankine cycle appears to be the best dynamic conversion scheme for space electric propulsion applications because: (1) It has relatively high efficiencies closely approaching the Carnot efficiency, and (2) it rejects and adds heat at constant temperature, which reduces the weight of the heat-rejection system Rankine cycle. Disadvantages are: (1) the requirement of relatively active cycle fluids for operation at high temperatures, and (2) the introduction of problems of two-phase flow in zero gravity and moisture handling in the turbomachinery. Figure 6 is a schematic of a typical Rankine space nuclear powerplant incorporating multiple turboalternators and radiators. Based on limited present-day knowledge, this version of the Rankine cycle appears to be the best choice for an advanced nuclear space electric system and should be considered only as a reference cycle. Other combinations of the components are possible, and, as technology advances, this reference concept may change.

Powerplant description. - The basic Rankine cycle consists of a boiler, a turbine for converting high-pressure vapor into mechanical energy, a condenser, and a pump to recirculate the condensate to the boiler. Potassium, sodium, cesium, and rubidium are potential cycle fluids. To date, although a particular cycle fluid has not been selected, the emphasis in the technology programs has been on potassium.

The heat source in the reference cycle is a liquid-cooled fast reactor with the coolant circulated by a motor-driven rotating pump. Lithium appears to be the most promising coolant for the heat source. A liquid-cooled fast reactor is incorporated in the reference cycle because it provides the most compact, lightweight reactor and shield combination.

The reference cycle incorporates a liquid coolant loop for the primary heat-rejection system. It cools the condenser and transfers the heat to the radiator, where it is rejected into space. This coolant loop is desirable because it facilitates segmenting of the radiator, which thus reduces the probability of failure due to meteoroid puncture. The coolant loop also simplifies powerplant startup and spacecraft integration by reducing the requirements for preheating the radiator to provide high condenser pressures needed for pumping and should concentrate the handling of two-phase flow in a few relatively small heat exchangers. Sodium potassium (NaK) and lithium are the more likely candidates for the fluids for all coolant loops. Lithium has more desirable heat-transfer characteristics, and NaK has advantages because its lower melting temperature should reduce freezing problems previous to startup.

The powerplant will also require a number of secondary heat-rejection loops for cooling the alternator, controls, power conditioning, reactor shield, reactor reflector, pump, reactor control-drive motors, and bearings. These loops may incorporate segmenting for redundancy to improve reliability.

The powerplant requires a control system for startup to compensate for reactor burnup, control reactor-outlet temperature, prevent overspeed of the turboalternator, and control alternator frequency and voltage. Although an extensive study of the control problem has not been made, it appears that a control concept similar to that used on SNAP-8 may be suitable. This involves reflector control that varies neutron leakage to control the reactor power level and compensate for burnup. Frequency and turbine rotational speed are controlled by a parasitic electrical load control that varies the alternator electrical load. Voltage is controlled by varying the field current in the alternator windings.

An additional difficulty that will be encountered on long missions at high powers is excessive burnup in the reactor. The powerplant may have to be operated at low power (idled) during coast phases of the trajectory rather than operated at full power with excess electrical energy dissipation in the parasitic load to minimize burnup. While controls for idling have not been investigated, there is a possibility of incorporating a control valve at the boiler inlet, which along with suitable reactor control would reduce boiler flow and the reactor power required to maintain a given outlet temperature during the idle phase.

The startup system is not shown in figure 6, primarily because this area has not yet been investigated. Startup in the Rankine cycle powerplant is particularly complicated as it is necessary to start the powerplant in orbit to satisfy nuclear safety requirements, and many of the fluids used in the powerplant freeze at temperatures above the space equilibrium temperature. An additional start problem results from a pump requirement for a reasonable inlet pressure before starting. For the cycle fluids of interest, this requires a condenser temperature in excess of 1000° F. It appears that a startup concept similar to that being investigated for SNAP-8 will be suitable for the advanced Rankine system, but, as the problems are considerably more severe with alkali-metal cycle fluids, this area still requires considerable investigation.

Development status and problems. - Reactor: The AEC has two programs at Pratt & Whitney, the Lithium Cooled Reactor Experiment (LCRE) and the SNAP-50, that should provide technology for a high-temperature, high-powered, fast, liquid-metal-cooled reactor capable of operation in space. The LCRE is a ground experiment designed to demonstrate the operation of a high-temperature, lithium-cooled, UO_2-BeO fueled, fast reactor with a power output of about 10 megawatts thermal. SNAP-50 is similar to the LCRE; but, instead of being designed for a ground experiment, SNAP-50 is being developed for space by reducing weight and auxiliary requirements.

Boiler and condenser: There is presently sufficient background in boiling and condensing to design heavy boilers and condensers that can operate successfully. They are heavy because there are limited data on both boiling and condensing heat-transfer coefficients, pressure drops, and the stability of alkali metals. This lack of data requires that the component be overdesigned, which increases weight. However, data that should remove this limitation are being obtained in these areas from at least five major heat-transfer programs.

Turbine: Alkali-metal turbines can now be designed by using the same techniques as presently used for steam, mercury, and gas. The higher temperature must be considered in the structural design, and the increased problems of moisture, handling, and erosion effects must also be considered. Before optimum efficiency and life are obtained, considerable turbine testing will be required. To date, there are essentially no test data on alkali-metal turbines, but two turbine test rigs are scheduled to start operation this year that should provide some of the required test data.

Pumps: There is considerable experience under the Aircraft Nuclear Propulsion (ANP) program on the pumping of high-temperature alkali metals that can be applied to an advanced Rankine cycle system. An extension of these data is required, however, for a better understanding of the cavitation problems that can occur in a Rankine system where low pump inlet pressures at high rotational speeds are encountered. This problem might be circumvented by the use of booster pumps, subcooling, higher condensing pressures, and lower rotational speeds, if the additional complexity and weight could be accepted.

Radiators: In large space powerplants, the radiator, because of its large surface area and the meteoroid protection required, becomes the heaviest single component in the electric engine. High-temperature (1600° F) radiators can be fabricated today out of common materials (steel and/or copper with high-emissivity coatings). Lower temperature radiators (700° F) can be constructed of aluminum. With these radiators, the meteoroid protection would be provided by armor. With conventional materials, even at high temperatures, armored radiators are quite heavy; thus, development is proceeding on improved radiator materials, such as beryllium, niobium, and molybdenum, and better protection techniques that could reduce weight.

Little is presently known about meteoroid flux, meteoroid composition, or the effect of hypervelocity impact on materials and structures. An extensive program is under way in this area that includes space experiments, greater use of ground observations, and an extensive impact program. It is expected that within 3 years sufficient engineering data will be available to engineer a design for meteoroid protection. At present, meteoroid data vary by several orders of magnitude.

Power-generating and -conditioning equipment: The major electrical components are the alternator, the transformer, and the rectifiers. The magnetic and insulating materials currently available limit the alternator and transformer to operating temperatures of 500° to 800° F. State-of-the-art semiconductor materials limit solid-state rectifiers to operating temperatures of 180° to 230° F. These components must reject approximately 6.0, 0.8, and 1.5 percent of the energy they handle as waste heat, respectively. Any increase in operating temperature capability will ease the heat-rejection problem, especially for the alternator, because of its higher inefficiency.

To achieve increases in allowable operating temperatures, magnetic materials with improved high-temperature permeability and physical strength, insulating materials with improved high-temperature dielectric strength, and semiconductor or gas-filled rectifiers capable of operation at high temperatures must be developed. Programs to investigate and develop such material and devices are being conducted under NASA and Air Force contracts.

The integration of the alternator into a turboalternator unit poses several development problems. State-of-the-art alternator technology is only adequate for the development of moderate-speed, low-temperature components. To simplify the integration, it is desirable to develop high-rotational-speed machines along with the higher temperature capability discussed previously. Alternator and power-conditioning design studies are being performed under a NASA contract.

Thermionic Powerplant

Thermionic conversion currently appears to be the best static conversion scheme for space electric propulsion applications. Thermionic converters consist of two electrically insulated electrodes, one hot and one cold, as shown by figure 7. Electrons are emitted from the hot surface, transported through an ionized interelectrode vapor, and collected at the cold surface. By proper selection of materials and geometry and control of temperatures, a potential of about 1 volt is produced, which enables the electrons to do work in an external circuit.

Coupling of thermionic converters with a fission reactor heat source may be accomplished by placing the thermionic converters within, or external to, the reactor. Conceptual "out-of-pile" thermionic systems incorporate converters in the heat exchanger, or radiator. In these systems the emitter temperatures correspond to liquid-metal reactor coolant temperatures. Conceptual "in-pile" systems incorporate converters in the reactor fuel element where emitter temperatures correspond to fuel temperatures. Thermionic systems for electric propulsion application must have emitter temperatures of about 3000° F or higher. Because the in-pile concept requires a 3000° F or higher fuel material without the need for a 3000° F or higher liquid-metal containment material, only the in-pile system is currently of interest for electric propulsion application.

Powerplant description. - A schematic of an in-pile thermionic powerplant that has the potential of producing multi-megawatts of power for the missions of interest is presented in figure 8. The thermionic converters, which are an integral part of the reactor fuel elements, are heated directly by the fuel and cooled by the liquid metal circulated through the reactor.

The electric energy output of the converters is conducted from the thermionic reactor through bus bars to the power-conditioning equipment that tailors the reactor output to thrustor needs. At the multi-megawatt level, hundreds of these thermionic fuel elements must be contained within the reactor, and each fuel element may contain from 10 to 20 converters. The converters are connected in series and series-parallel networks to provide a reasonable voltage output and a measure of redundancy.

The liquid-metal coolant from the reactor is circulated by a pump to heat exchangers that are connected in parallel. Liquid-metal coolant transfers the heat from each heat exchanger to the main radiator, which is segmented. For the powerplant schematic shown, there is one radiator segment for each heat exchanger. Auxiliary cooling systems are used to maintain the temperatures of the cesium reservoir and the power-conditioning equipment at tolerable levels. In addition, a cooling system may be required for the reactor shield.

To achieve attractive efficiencies and system weights, the operating range for emitter surface temperatures must be 2800° to 3500° F, with collector temperatures of 1250° to 1900° F.

The thermionic reactor constructed by assembling arrays of thermionic fuel elements has a fast neutron energy spectrum for compactness and reduction of the shielding weight. Because the output of thermionic converters is very sensitive to temperature, nearly flat axial and radial power distributions are desired throughout the life of the reactor. Studies indicate that peak to minimum power distribution values of about 1.4 are needed for reasonable efficiencies. Small reactors can be controlled by varying neutron leakage with reflectors. However, for the large cores of interest, the control perturbations at the center of the core caused by the reflectors may be ineffective, and other schemes for control may have to be provided.

The power-conditioning equipment required for a thermionic powerplant must include inverters to convert the direct-current output of the thermionic converters to alternating current, in addition to transformers and rectifiers similar to those required for a Rankine cycle powerplant.

Development status and problems. - The technology required to design thermionic powerplant hardware is in the early phases of being developed. There is little experience in the design and construction of thermionic fuel elements, and development programs comparable to the SNAP-2, -8, and -50 are nonexistent for thermionic powerplants. The thermionic powerplant technology program is concerned with providing new materials and device configurations capable of performing reliably in highly stressed environments (temperatures up to 4100° F in the event of a converter open-circuit failure, high radiation fluxes, and corrosive active metal atmospheres) for long times.

Results from the current development program for advanced systems are applicable to the pump, heat exchanger, and radiator components of both the Rankine cycle and thermionic powerplant. The program includes a study of electromagnetic pump concepts and investigations of lightweight, high-temperature radiator materials and configurations. Components unique to the thermionic powerplant pose problems that create "make or break" areas of concern. These problems and the status of activities to solve them are discussed in the following paragraphs for the thermionic fuel element, thermionic reactor, and d-c to a-c inverter components, respectively.

Thermionic fuel element: The fuel-emitter combination or "fuel form," the insulators, and the metal-ceramic seal present the most challenging materials problems. Representative converter configurations, both single- and multi-cell modules, will have to be operated successfully in the laboratory and reactor environments. Activities to demonstrate the feasibility of a full-scale thermionic fuel element can then begin.

Fuel form: For the fuel form, handling of the gaseous fission products and compatibility of the fuel with good thermionic emitter materials are major requirements. Ideally, the fuel form should contain the fission product gases without swelling and also provide a suitable electron-emitting surface. To approach this condition, there is strong incentive to develop refractory metal-clad fuel forms having refractory metal matrices to strengthen the fuel-form structure.

In-pile screening tests of clad and bare fuel forms are currently being performed under AEC and NASA sponsorship to determine fission product containment and fuel-emitter compatibility characteristics in the radiation environment. If containment is not feasible, the products will have to be vented directly to space, vented via the interelectrode gap by flushing intermittently or continuously to space, or both. The effect of fission gases within the interelectrode space on thermionic performance is being investigated by NASA, AEC, Air Force, and Navy contractors.

The compatibility of fuel and emitter materials is also currently being investigated by several Government contractors. Tungsten, the highest melting temperature refractory metal, has been the most compatible emitter material with the candidate fuels, UO_2 and UC near a temperature of interest (3272° F) during 1000-hour furnace tests. Other advanced material combinations are promising and are being developed. Another aspect of compatibility is the diffusion of fuel constituents to the emitting surface. This phenomenon and its influence on thermionic performance are being studied under existing NASA and Air Force contracts.

The containment, compatibility, and emission-diffusion behavior of the best candidate materials in simulated operating environments for much longer times must be investigated to designate a fuel form for the thermionic fuel element feasibility demonstration. Fuel-form activity is pacing the progress of the thermionic powerplant materials technology program.

Insulators: Two different insulators are required. The thin, low-temperature insulator located between the collector and the outer clad of the thermionic fuel element must have high dielectric strength properties and be impervious to the interelectrode vapor. This insulator electrically isolates the thermionic converters from the reactor liquid-metal coolant and allows the converters to be connected in series and achieve higher voltages and lower current outputs to the power-conditioning equipment. Tests are being conducted under AEC contract to fabricate sandwiched structures and to evaluate the effects of alkali-metal attack and radiation on insulator properties.

The second insulator, a high-temperature insulator, is required to support one end of the emitter and must withstand voltages corresponding to the output of a single converter (less than 1 v). The properties of bulk insulator materials in cesium and radiation environments at temperatures of interest (up to 2910° F) are being investigated by contractors for the Navy, the AEC, and NASA. The failure of a high-temperature insulator could mean the loss of output from a single converter, whereas breakdown of the low-temperature insulator could lead to a major degradation in output power.

Metal-ceramic seals: Metal-to-ceramic joining capability is required at many different locations throughout the thermionic fuel element structure. Good bonding for low thermal resistance is required for the low-temperature insulator sandwich, leaktight metal-to-ceramic joints may be required between adjacent emitters if the fission gases are vented directly to space, and a metal-to-ceramic seal is required to contain the interelectrode vapor. The Navy's Bureau of Ships is currently sponsoring a program to develop high-temperature metal-ceramic seals. Activities for improved seals are necessarily a part of all thermionic converter programs and, also, other programs within industry to develop products (i.e., lamps, electronic tubes, etc.) for the space, military, and commercial markets.

Converter modules: The materials technology is incorporated into operating devices by constructing and testing single- and multi-cell converter modules. In the multi-cell modules, the cells have a common envelope and interelectrode vapor reservoir. With each additional cell, a new set of material and geometry problems is encountered.

Fuel forms and interelectrode vapors and geometries are being investigated in single cells. Individual cells are being connected in series-parallel arrays to investigate the character of electrical interplay between cells. End supports, intercell connections, and intercell venting structures (where thought necessary) are being investigated in dual-cell modules. After successful tri-cell module in-pile tests, thermionic fuel element mockups and demonstration units can be constructed.

For successful operation, the single- and multi-cell module structures must demonstrate that they possess the power output and life capabilities required for application. When the structures have been developed for steady-state and transient operation in simulated environments for short durations, long-time degradation effects and failure mechanisms must be eliminated or controlled.

One of the most serious failure modes is expected to be the occurrence of an open-circuit failure. Because the cells within the thermionic fuel element are connected in series, a mechanical failure, or zero load condition, which interrupts the current flow, would eliminate the electron cooling paths of all the emitters in the particular series circuit. It is estimated that 50 to 60 percent of the heat energy generated by fission is removed from the fuel forms through these paths. As a result, fuel-form temperatures would rise 800° to 900° F above the normal operating temperatures, as high as 4100° F. Mechanisms for correcting or tolerating the overtemperature conditions must be thoroughly investigated.

To date, a single-cell module has produced power for more than 8500 hours without degradation. The results demonstrate that the same thermionic electrodes and cesium plasma are capable of operating for long times in a nonnuclear environment. This module was constructed and operated by the General Electric Research Laboratory. Thirteen single-cell modules that typify thermionic fuel element structure will be constructed and life-tested under an existing NASA contract.

The performance of dual-cell modules out-of-pile and in-pile is currently being investigated under AEC contract. Several dual-cell modules simulating fueled emitters are being studied in the laboratory under Air Force contracts.

The Air Force is also sponsoring an out-of-pile investigation of nine converters in series with a common interelectrode vapor reservoir. Multi-cell modules have been operated in-pile by the AEC at Los Alamos Scientific Laboratory, but the tests were short lived, and performance degradation effects were inseparable..

An investigation of thermionic converters for nuclear marine application, being sponsored by the Navy's Bureau of Ships, is one of the largest thermionic programs in existence. NASA's solar thermionic programs to develop converter modules for space application are providing a useful background of experience.

Thermionic reactor: In addition to containing the thermionic converters, fast neutrons sustain the chain reaction within the thermionic reactor. The uniqueness of the thermionic reactor requires that methods be developed for flattening power distributions and achieving control capability. Low-power critical experiments can be performed to determine flux distributions and criticality requirements for such a reactor. However, until reactor systems similar to the one of interest have been operated, it will be difficult to accurately predict the temperature coefficients that correspond to certain changes within the reactor.

Because the economy of fast neutrons is poor, large amounts of fissionable material are required in the thermionic reactor. For the multi-megawatt power levels, core sizes are limited by thermionic emitter area requirements. Therefore, the fuel materials are heavily diluted, and additional uranium 235 is required for criticality. The increased fuel loading reduces the percentage of total uranium atom burnup required, and the diluent may improve burnup capability. Fuel volume fractions of about 0.2 to 0.4 result after the coolant and thermionic requirements have been considered. The resulting structure must provide for a suitable combination of nuclear, thermionic, and heat-transfer characteristics to achieve acceptable performance.

Problems related to the energy-conversion cycle are the design of electrical connections between fuel elements and the load, and the design of interelectrode vapor supply and regulating systems. The possible effects of electric oscillations, stray currents, or magnetic fields within the reactor must be investigated, and the ability to tolerate open-circuit failures within the thermionic fuel element must be developed.

Los Alamos is currently constructing a critical facility to mock up a thermionic powerplant. Parametric and preliminary design studies are being performed by NASA contractors to investigate thermionic powerplant designs with emphasis on the thermionic reactor. Effects that may degrade thermionic performance, dissipate available electrical energy uselessly, or create failure mechanisms are being investigated to reduce uncertainty and define development requirements.

Direct-current to alternating-current: For the thermionic powerplant, the low-voltage, direct-current output must be converted to a nearly square alternating-current waveform, transformed to the high voltage, and then rectified to the specified direct-current voltage. At present, all the electrical components are low-temperature devices that will necessitate large cooling loops unless major advancements are made in electric power-conditioning component technology.

A rough estimate made under a current NASA contract of the power-conditioning specific weight for a 1-megawatt-electric powerplant using Apollo technology (which is about 2 yr away) was 16 pounds per kilowatt. This weight does not include the weight of the auxiliary radiator or electrical bus bar required, and a 125° F coolant temperature and silicon-controlled rectifiers were assumed. About 85 percent of this weight is in the inverter stages and is due to the switching capacitors required to control the silicon-controlled rectifier devices and the support structure for these capacitors. To achieve overall powerplant weights less than 20 pounds per kilowatt, power-conditioning weights of 3 to 5 pounds per kilowatt will be required.

Available current-carrying devices for inverters are of the solid-state variety, either germanium transistors or silicon-controlled rectifiers. Although the transistors have lower power capacity, they do not require the continued application of gating signals to be maintained in an off condition, as do the silicon-controlled rectifiers. The capacitor penalty necessary to keep the silicon-controlled rectifiers in an off condition may offset their greater energy throughput capability. A study to evaluate the trade-offs involved is being made under an existing NASA contract.

Because the greatest weight penalty is imposed by the switching capacitors and the solid-state devices, NASA contracts to develop high-temperature - high power switchgear (for use as high as 1000° F) and high-temperature power tubes (for use at temperatures ranging from 930° to 1470° F) are being initiated. Solid-state devices for operation at 930° F are being studied under a current Air Force contract. The gas-filled power tubes (thyratrons) can be used in the rectifier as well as in the inverter stages. The thyratrons are of metal-ceramic construction similar to thermionic converters and should adapt to the highly stressed environments as well.

To design power-conditioning components for such an advanced power system, it is necessary to know the properties of materials for high-frequency and -temperature conditions and in vacuum and alkali metal environments. A NASA contract is being initiated to investigate magnetic materials, electrical conductors, and electrical insulators at such conditions. Thus, activity for the material and device improvements required is being sponsored under several NASA and Air Force contracts.

Powerplant Performance Potential

The performance or competitiveness of an electric engine depends primarily upon the specific weight and the attainable life of the power supply. From the present state of knowledge it is impossible to quantitatively estimate the life potential of an advanced power system, but weight estimates can be provided.

Pratt & Whitney under contract NASw-360 has prepared conceptual designs and parametric data for 1-megawatt Rankine cycle and thermionic powerplants. The program began in July 1962 and has reported primarily parametric design data (refs. 1 to 3). From personal contacts with Pratt & Whitney, additional data that include preliminary weight estimates for the early conceptual designs were obtained. These estimates were used as the basis for preparing the weights in this section. As the design work at Pratt & Whitney progresses, some changes in the weights will undoubtedly occur.

Rankine cycle powerplant assumptions. - The Pratt & Whitney 1-megawatt Rankine cycle system consists of a single reactor connected to four 250-kilowatt power-conversion loops operating in parallel. Each power-conversion loop contains a single boiler; a turbine directly coupled to an alternator; power conditioning; controls; a condensate pump for recirculating the flow from the condenser to the boiler; and 10 lithium-cooled condensers, each with its own pump and independent radiator. Schematically the powerplant is identical to that shown in figure 6. Potassium is used as the turbine fluid, and lithium is the coolant throughout. Cycle conditions, operating temperatures, component efficiencies, and component ratings are summarized in table III.

It was necessary for Pratt & Whitney to make a number of assumptions in preparing the designs. These cannot, of course, be justified because of the lack of experimental data. However, the assumptions are fairly conservative.

Pratt & Whitney used the meteorite-protection criteria recommended in reference 4 with 1956 meteoroid flux estimates (ref. 5) but neglected the spalling and thin-plate correction factors recommended by these investigations. Beryllium was assumed to be a suitable radiator material. The radiator was designed for a 90-percent probability that 75 percent of the radiator surface would be effective at the end of 16,000 hours. Recent data have indicated that the penetration criteria (ref. 4) and the flux estimates (ref. 5) are very conservative. Neglecting the spalling and thin-plate corrections and using a 90-percent probability, Pratt & Whitney is optimistic. Thus, there is a tendency for these two effects to cancel each other, although calculations confirming this have not been conducted by the investigators. The radiator design is admittedly crude. However, at this time it is not felt that more detailed or more refined calculations of meteorite protection requirements are indicated, as the uncertainties still remaining in the meteorite area are too large to warrant further refinement.

Shield design for the megawatt systems assumes that a 20-foot-diameter payload is located 50 feet from the reactor and receives neutron and gamma doses of 10^{13} nvt and 10^7 rads, respectively. This integrated dose of 10^7 rem is obviously too high for manned spacecraft, but a new shield has not been calculated as its design depends to a very large extent upon the spacecraft configuration and the shield provided for the space environment. Very preliminary estimates indicate that the shield required to limit the dose to 10 rem for a 250-foot separation distance from the powerplant weighs approximately 1 to 2 pounds per kilowatt (paper by Karp). If this is correct, the error in using Pratt & Whitney's shield design is $\frac{1}{2}$ to $1\frac{1}{2}$ pounds per kilowatt.

Rankine cycle powerplant weight estimates. - As stated before, the Pratt & Whitney design assumptions have been fairly conservative. However, there are a few additional areas the authors have taken the liberty of modifying to provide better estimates. These include the following:

(1) Boiling and condensing average heat fluxes were based on the present state of the art - heat fluxes in the 20,000 to 30,000 Btu/(hr)(sq ft) range in

contrast to Pratt & Whitney's assumption of fluxes in the 100,000 to 250,000 range.

(2) Turbine and alternator efficiencies were reduced from 83 and 95 percent to 77 and 90 percent, respectively.

(3) The Pratt & Whitney radiator incorporates segmenting in the design, but does not provide redundancy, as they assume that the mission can be accomplished if 75 percent of the radiator area required for full power remains at the end of 16,000 hours. The authors have added 33 percent additional radiator area to compensate for this loss of area so that full power is available at the end of 16,000 hours.

(4) The authors feel that the reactor design is too optimistic. Consequently, the weight of the reactor has been increased significantly to allow for the use of more conservative fuels at lower fuel burnup rates.

Weight breakdowns for the 1-megawatt Rankine cycle system, together with the numbers of each component and their duty, are presented in table IV.

The Pratt & Whitney 1-megawatt powerplant weights have been extrapolated to the 5-megawatt level by assuming that all weights scale linearly except the reactor, shield, piping, controls, and startup components. The weights of these components were estimated. The weight breakdown for a 5-megawatt system together with the component ratings and the number of components is shown in table V. Note that the weight breakdown includes all anticipated items, including power conditioning and controls. It should also be noted that, although a single 30-megawatt-thermal reactor and shield is included, all other components include a 25-percent allowance for redundancy. Thus, although four boilers, turboalternators and power-conditioning units are sufficient to provide 5 megawatts electric, five of these are provided; likewise, where 80 condensers and radiator segments are sufficient, 100 are provided. Despite this conservatism, the weight of the powerplant without shielding for manned missions is 20 pounds per kilowatt - a weight that appears to be just barely competitive with nuclear rockets for some of the more ambitious manned missions.

There is hope, however, that these weights can be improved. If just a little less conservatism is allowed in estimating the temperature and efficiency potentials of an advanced Rankine system, a considerable reduction can be obtained in weight. This is shown in table VI, where the weight savings is indicated that would result with certain components improvements. As an example, a turbine-inlet temperature of 1850° F was assumed. If this can be increased to 2100° F and the condensing temperature increased to 1400° F (14 lb/sq in. abs), the weight of the powerplant is reduced 2.4 pounds per kilowatt. Also, turbine and alternator efficiencies of 77 and 90 percent were assumed. There is some hope that these efficiencies could be raised to 85 and 95 percent, respectively. If the larger values are obtained, the weight of the powerplant is reduced another 1.4 pounds per kilowatt. Similarly with the temperature of the electrical components, it was assumed that rectifiers using semiconductors operate at temperatures of 140° F and that the alternator and transformer operate at 500° F. If these temperatures can be raised to 500° and 800° F, respectively, a weight savings of 1.9 pounds per kilowatt results. If the reactor burnup can be improved or if the boiling and condensing heat-transfer coefficients can be improved, there is another possibility of saving 1.6 pounds per kilowatt. The accumulative savings approaches 8 pounds per kilowatt. While it is unlikely that all of these improvements can be achieved, there is potential for accomplishing some and partially fulfilling others. Consequently, it is likely that a Rankine cycle powerplant weighing considerably less than 20 pounds per kilowatt at 5 megawatts is achievable.

Thermionic powerplant. - The Pratt & Whitney thermionic powerplant design is an in-pile thermionic system similar to that shown in figure 8. The collector is cooled by lithium, which is the coolant used in the radiator segments and other cooling loops. Thermionic converter performance is based on an empirical correlation of available thermionic test data (ref. 2). Power conditioning presents a rather large unknown in the design, as the only available inverter data use semiconductors. The inverter alone could weigh as much as 14 pounds per kilowatt while operating at 140° F. So in the design it was necessary to assume that higher capacity tubes or semiconductors would be developed to achieve lower power-conditioning weights. Pratt & Whitney's assumptions and the authors' corrections in the areas of meteorite protection and shielding are consistent with those used in the Rankine analysis, although there does appear to be some advantage to using niobium in the radiators and operating the main radiators at higher temperatures to provide lower radiator areas.

The design conditions for the thermionic system are summarized in table VII, and a weight breakdown for the 1-megawatt system with corrections similar to those used for the Rankine system is provided in table VIII. An extrapolation technique similar to that used for the Rankine system provides the 5-megawatt thermionic system weight breakdown, also shown in table VIII.

Comparison of Rankine and thermionic powerplants. - The specific weights of Rankine and thermionic powerplants are compared in figure 9, where the specific weights are plotted as a function of radiator inlet temperature. The thermionic systems are also shown parametrically as a function of emitter temperature. The specific weights presented for both systems do not include the redundancy that was included in the detailed weight breakdowns presented in the previous tables. The specific radiator areas for four powerplants are also shown; however, these areas do not include provisions for component cooling.

The Rankine powerplants are shown for two cases: one with a beryllium radiator, the other with niobium. Presently, only beryllium, niobium, and molybdenum appear to have the properties desired for high-temperature radiator materials (ref. 6). In estimating weights it was assumed that beryllium is resistant to meteoroid impact, even though there are some indications that it may be too brittle. The use of higher density niobium will impose a severe weight penalty on the Rankine powerplant because the powerplant cannot readily utilize the high-temperature potential of niobium unless very high liquid-metal temperatures (2500° F) can be tolerated in the reactor.

The specific weights for the thermionic powerplants are shown for both niobium and beryllium radiators. Beryllium appears to be limited to temperatures less than 1400° F by strength and sublimation. The thermionic powerplant weight is a minimum at the highest temperature at which beryllium can be used (about 1350° F), and another minimum occurs at about 1900° F with niobium radiators.

The Rankine cycle powerplant using beryllium appears to offer a slight weight advantage over a 3200° F thermionic powerplant. While the niobium Rankine powerplant is heavier than the thermionic, the differences in specific weight are not large. In any case, in considering the assumption and unknowns inherent in the analysis, there is no reason to select one power system over the other at this time.

If radiator surface is the most important selection criterion, the higher temperature thermionic system offers advantages. The primary radiator area is about 800 square foot per megawatt as compared with 3700 square foot per megawatt for the Rankine cycle. As these areas do not include secondary cooling, this 4-to-1 area advantage may not be maintained when cooling requirements for power conditioning are included, unless higher temperature (500° F) d-c - d-c converters are developed.

In summary, neither thermionics nor Rankine power systems possess any inherent advantages that would allow the selection of one over the other for intensive development at this time. Instead, each has certain advantages and disadvantages, which include the following:

(1) Thermionic powerplants may not require rotating equipment.

(2) The nuclear and power-conversion problems are separated in a Rankine powerplant.

(3) The Rankine powerplant appears to be lighter.

(4) The thermionic powerplant requires less radiator area.

(5) The thermionic powerplant can operate with lower liquid-metal temperatures.

(6) A Rankine powerplant has less complex power-conditioning problems.

CONCLUSIONS

Electric propulsion appears to be competitive with chemical and nuclear propulsion providing lightweight, long-lived engines are developed. High-specific-impulse thrustors that promise to satisfy advanced electric engine requirements are nearing engineering phases of development. Presently, small thrustors are being ground-tested, and it is expected that the electrostatic devices will soon be flight-tested. The development of 30-kilowatt thrustors has been initiated, and, if they have sufficient life, they could serve as thrustor modules along with the megawatt powerplants.

Technology programs for high-temperature, lightweight power-conditioning equipment are now getting under way. High-temperature materials and devices are being studied, but it will be many years before useful hardware is available.

At present, the equipment that can be built is much too heavy, presents cooling problems, and introduces radiation-shielding complexities.

The nuclear-electric power supply poses the most difficult problems. The research and development program is still in the early technology phase with NASA and other Government agencies and is concerned, primarily, with investigating materials, securing engineering design data, establishing component performance capabilities, determining meteoroid effects, and so forth. It is felt that this technology phase will have to continue for a number of years before meaningful hardware designs can be initiated and feasibility demonstrated.

For the two powerplants currently of interest, the advanced Rankine cycle technology is more advanced and has a broader technology base. In contrast, thermionic technology is in an earlier technology phase, and predictions of its potential are less reliable. At present there appears to be no incentive to select either a Rankine or thermionic powerplant for intensive development, as either powerplant has good potential.

It appears that lightweight, high-efficiency electric engines competitive with other propulsion schemes can be developed. It is not apparent, however, that long life can be obtained unless extensive redundancy and/or in-flight repair capability are provided. The advanced electric engines will require a long and expensive development program, especially for the electric powerplant. The program will probably last 10 to 20 years and cost hundreds of millions of dollars, if the reliability required for manned flight is to be realized.

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TABLE I. - ELECTRIC ENGINE DESIGN OBJECTIVES

	Manned	Unmanned
Power level, Mw	5 - 40	0.4 - 5
Life, hr	10 - 15,000	10 - 30,000
Specific weight, lb/kw	10 - 20	10 - 30
Reliability, percent	99+	90
Radiator temperature, $^{\circ}$F	1200 - 1800	1200 - 1800

TABLE II. - PRESENT STATUS OF THRUSTORS

Thrustor	Approximate power, kw	Specific impulse, sec	Power efficiency, percent	Specific area, sq ft/kw	Specific weight, lb/kw
Lewis Resisto jet	15.0	850	75	0.003	1.0
AVCO Arc jet	30.0	1050	45	.0004	.22
TAPCO Electron bombardment[1]	3.0	5000	75	.05	3.3
EOS Electron bombardment	1.0	5500	82	.34	4.0
EOS Electron bombardment[1]	2.3	7000	75	.10	1.3
Lewis Electron bombardment	.5	5000	70	.88	6.0
Lewis Electron bombardment	2.0	5900	75	.17	5.0
Hughes Contact ionization[1]	.5	4500	40	.54	4.1
Hughes Contact ionization[1]	2.5	7600	62	.08	4.8

[1]These performance figures have not been attained (4-63), but are believed to be realistic estimates of performance within 12 months.

TABLE III. - RANKINE CYCLE POWERPLANT

ASSUMED CYCLE CONDITIONS

Net power output high-voltage d. c., Mw	5
Alternator outputs low-voltage a. c., Mw	5.9
Reactor thermal output, Mw	30
Reactor-outlet temperature, $^{\circ}F$	2000
Turbine-inlet temperature, $^{\circ}F$	1850
Turbine-inlet pressure, lb/sq in. abs	89
Turbine-exit pressure, lb/sq in. abs	4
Radiator-inlet temperature, $^{\circ}F$	1150
Radiator-exit temperature, $^{\circ}F$	1000
Turbine efficiency, percent	77
Alternator efficiency, percent	90
Power-conditioning efficiency, percent	97
Radiator material	Beryllium

TABLE IV. - WEIGHT BREAKDOWN OF 1-MEGAWATT RANKINE CYCLE POWERPLANT

Item	Number of units	Unit rating,* Mw	Weight, lb
Reactor	1	$^{t}6$	1500
Shield	1	-----	800
Boilers	5	$^{t}1.50$	1700
Primary loops and pumps	5	-----	1200
Turboalternators	5	$^{e}.30$	1750
Condensers	20	$^{t}.30$	1550
Primary radiators	20	$^{t}.30$	3000
Structure	--	-----	4600
Secondary radiators	5	$^{t}.25$	2800
Power conditioning	5	$^{e}.25$	2700
Secondary piping	5	-----	400
Startup loops	5	-----	600
Miscellaneous and contingencies	--	-----	1400
	Total weight, lb		24,000
	Specific weight, lb/kw		24

*t, megawatts thermal; e, megawatts electric.

TABLE V. - TYPICAL WEIGHT BREAKDOWN OF 5-MEGAWATT

RANKINE CYCLE POWER SUPPLY

Item	Number of units	Unit rating,* Mw	Weight, lb
Reactor	1	$^{t}30$	3,000
Shield	1	------	1,500
Boilers	5	$^{t}7.5$	6,500
Primary loops and pumps	5	------	5,400
Turboalternators	5	$^{e}1.48$	7,500
Condensers	100	$^{t}.30$	7,700
Primary radiators	100	$^{t}.30$	15,000
Structure	---	------	17,000
Secondary radiators	25	$^{t}.05$	13,500
Power conditioning	5	$^{e}1.25$	12,700
Secondary piping	---	------	1,800
Startup loops	5	------	1,400
Miscellaneous and contingencies	---	------	7,000
	Total weight, lb		100,000
	Specific weight, lb/kw		20

*t, megawatts thermal; e, megawatts electric.

TABLE VI. - IMPROVEMENT POTENTIAL FOR 5-MEGAWATT

RANKINE CYCLE POWER SUPPLY

	Design value	Future value	Weight reduction, lb/kw
Turbine-inlet temperature, °F	1850	2100	2.4
Turbine efficiency, percent	77	85	.9
Alternator efficiency, percent	90	95	.5
Electrical equipment temperature, °F	140/500	500/800	1.9
Reactor burnup, percent	1	5	.4
Boiler average heat flux, Btu/(hr)(sq ft)	30,000	150,000	.6
Condenser average heat flux, Btu/(hr)(sq ft)	50,000	150,000	1.0
Accumulative improvement, lb/kw			+7.7

TABLE VII. - THERMIONIC POWERPLANT ASSUMED

OPERATING CONDITIONS

Emitter temperature (maximum), °F	3200
Collector temperature (average), °F	1800
Radiator-inlet temperature, °F	1850
Radiator-outlet temperature, °F	1580
Converter power output, low-voltage d.c., Mw	6.2
Net power output, high-voltage d.c., Mw	5
Theoretical converter efficiency, percent	16.7
Theoretical converter power density, w/sq cm	7.2
Average converter efficiency, percent	12.7
Average converter power density, w/sq cm	4.1
Power-conditioning efficiency, percent	93
Overall system efficiency, percent	11
Radiator material	Niobium

TABLE VIII. - WEIGHT ESTIMATES OF 1- AND

5-MEGAWATT THERMIONIC POWERPLANTS

Item	Weight, lb/kw	
	1-Mw system	5-Mw system
Reactor	4.9	3.3
Shield	1.5	.6
Primary heat exchangers	.2	.2
Pumps	.6	.6
Bus bar	.5	.5
Power conditioning	5.0	5.0
Structure	3.0	1.0
Main radiator	4.4	4.4
Secondary radiators	6.0	6.0
Total	26.1	21.6

SCHEMATIC OF ELECTRIC ENGINE

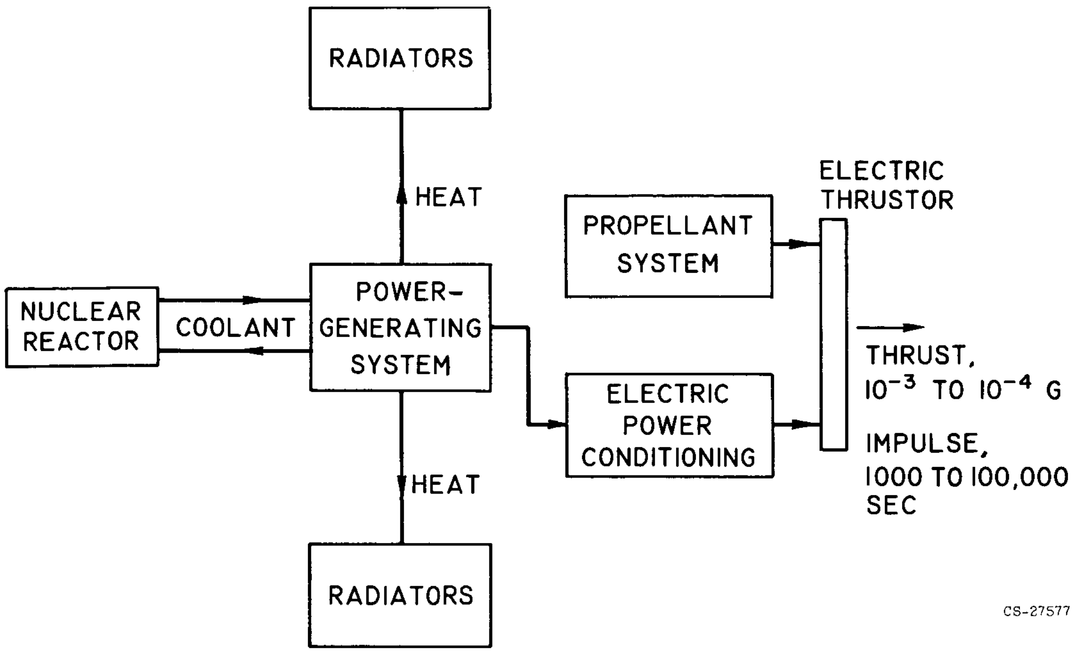

Figure 1

COMPARISON OF NUCLEAR ROCKET AND ELECTRIC ENGINE PERFORMANCE FOR MANNED MARS ROUND TRIP

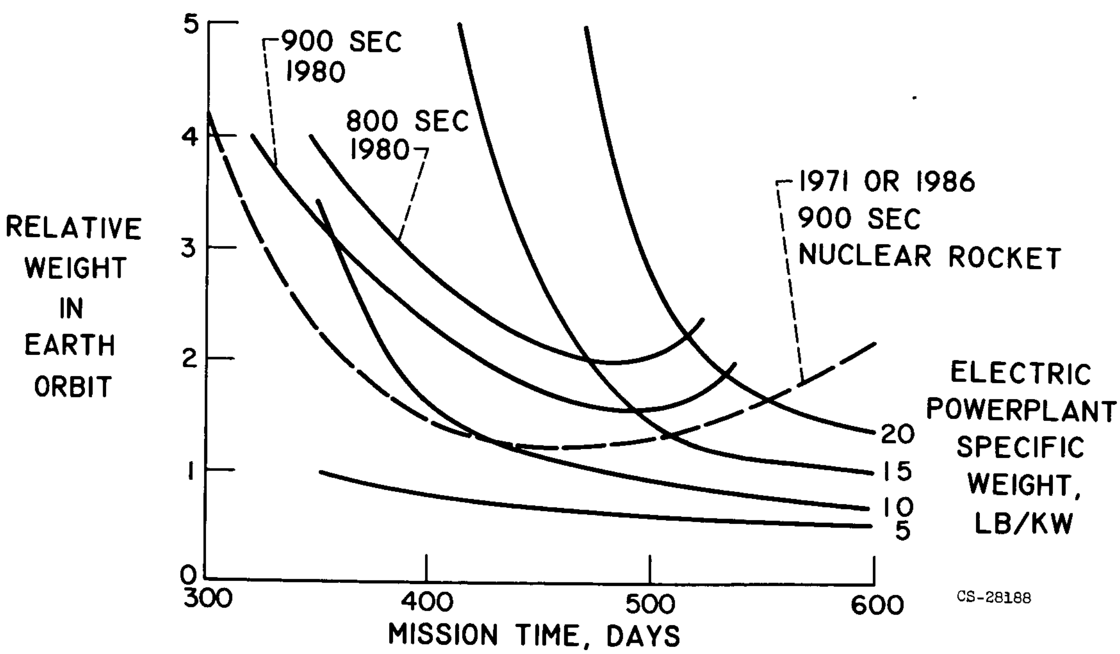

Figure 2

ELECTRIC THRUSTORS

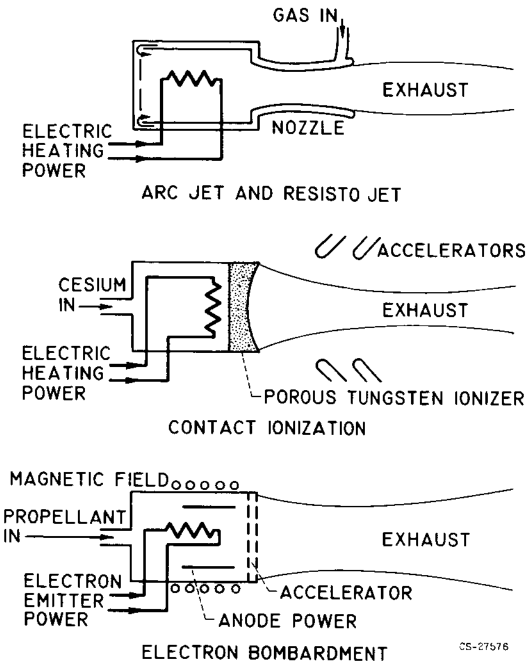

Figure 3

THRUSTOR POWER EFFICIENCY

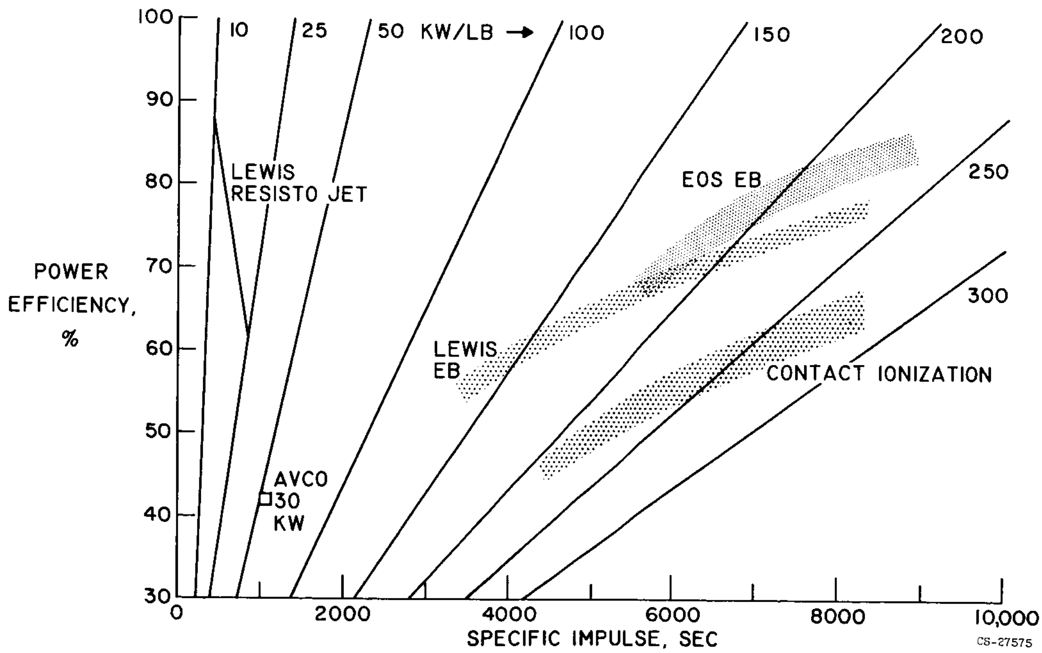

Figure 4

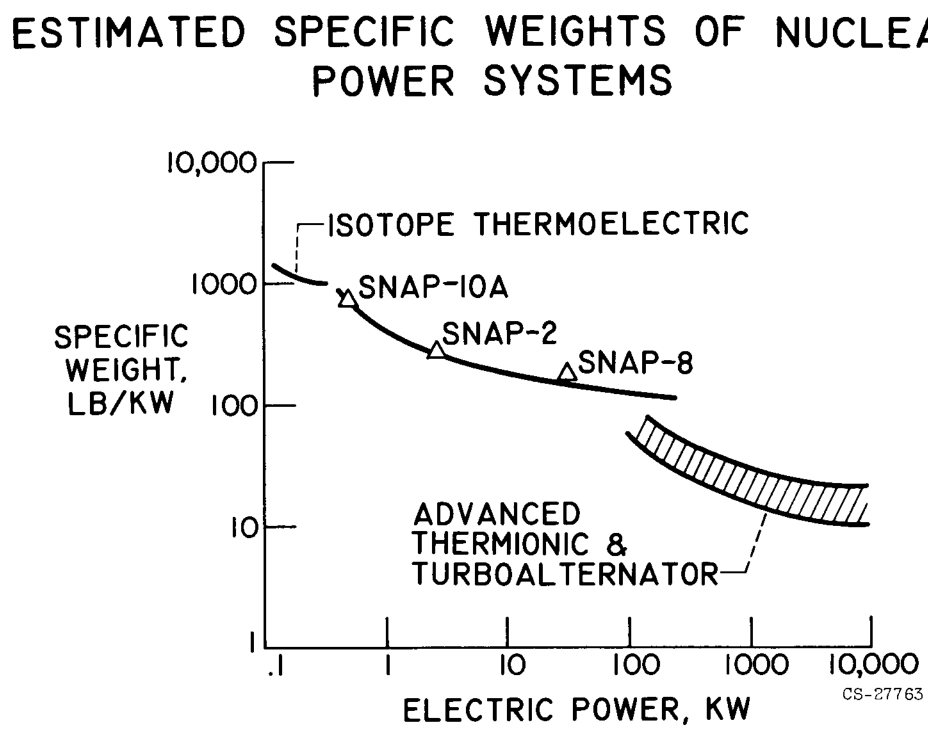

Figure 5

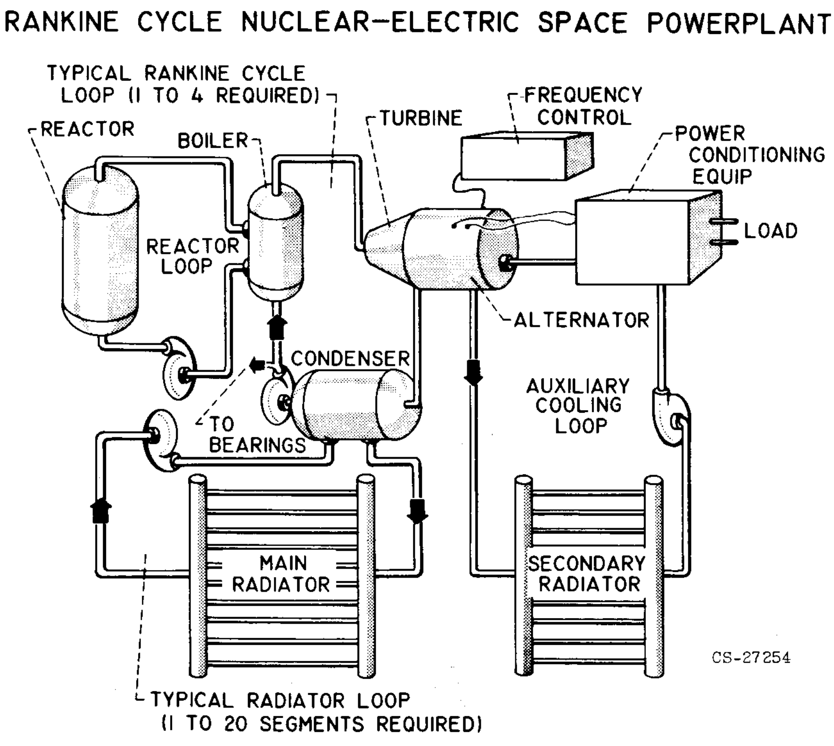

Figure 6

THERMIONIC CONVERTER

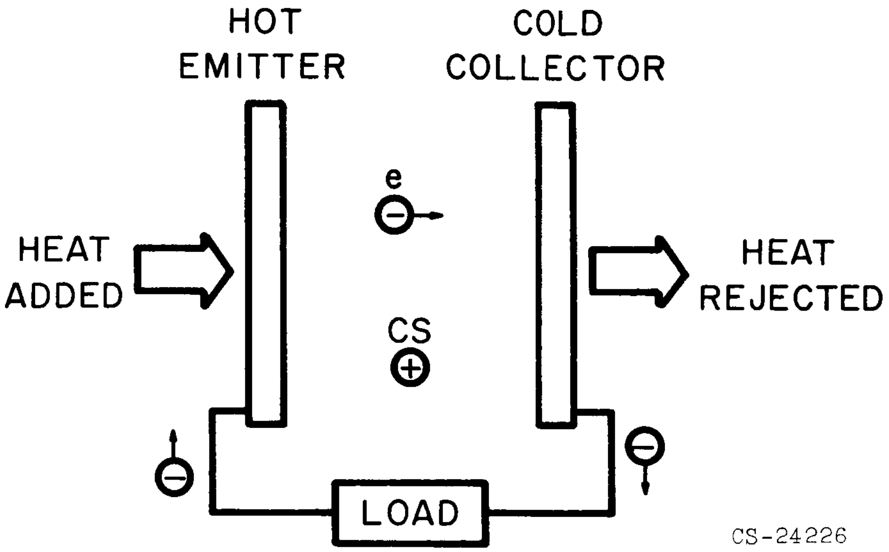

Figure 7

NUCLEAR THERMIONIC POWER SYSTEM

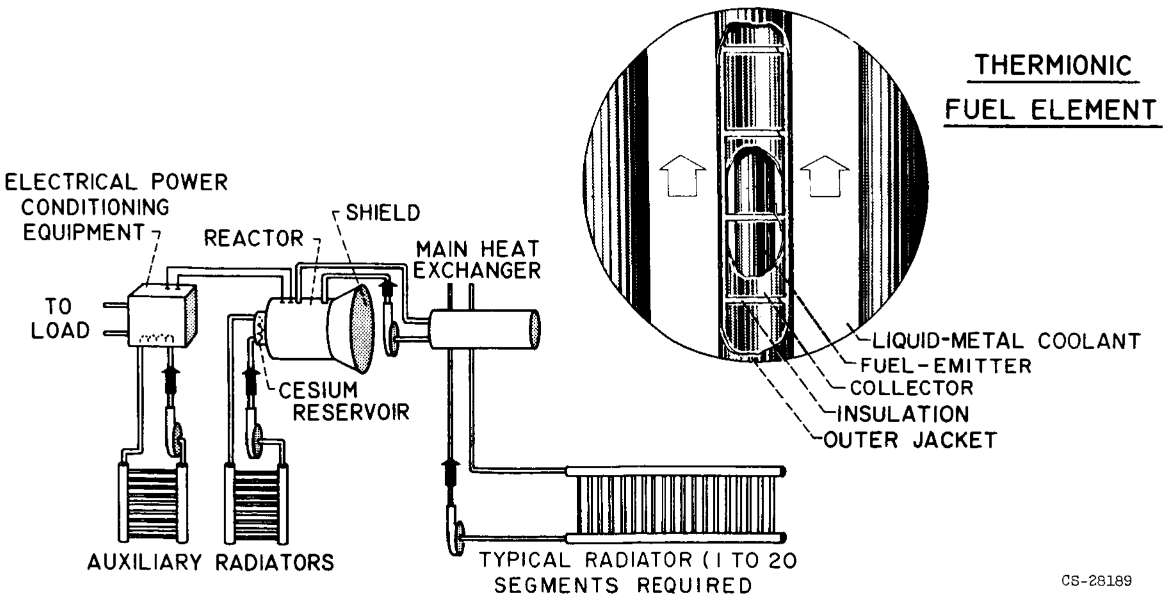

Figure 8

SYSTEM SPECIFIC WEIGHT PLOTTED AGAINST RADIATOR-INLET TEMPERATURE FOR RANKINE AND THERMIONIC POWERPLANTS

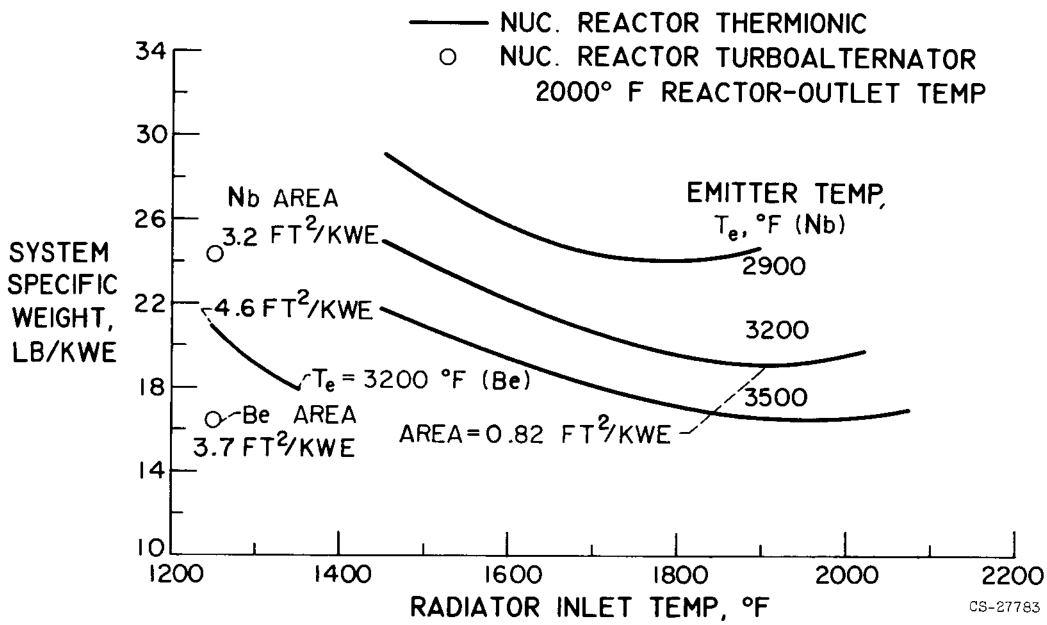

Figure 9

WILD BLUE YONDER PROPULSION SCHEMES

By John C. Evvard

NASA Lewis Research Center

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WILD BLUE YONDER PROPULSION SCHEMES U

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NASA Lewis Research Center

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This paper will include a discussion of the ORION concept, several gaseous core nuclear rockets, thermonuclear propulsion utilizing superconducting magnets, and finally a lightweight radioisotope power generation system for electric propulsion. With the exception of the latter concept, all of these schemes have much in common. The initial vehicle weights would be very large - on the order of several million pounds. The payload fractions are high - on the order of 25 to 50 percent of the takeoff weight - for near Earth missions. The development problems would be severe, and, correspondingly, the development costs would be extreme - on the order of many billions of dollars. In addition, the launching problems from Earth would be fantastic - with nuclear radiation hazards and political overtones added for good measure. However, the reward for success would be great. One can contemplate large payload fractions propelled on space missions - with thrust-to-weight ratios, at least in some cases, greater than unity and with specific impulses of several thousand seconds. The mission transportation costs would run in terms of dollars per pound of payload with clear opportunities for reasonable manned expeditions across the solar system. This is the carrot that leads the endorsement of such gigantic projects. Conf. R.D. Author

The ORION vehicle resembles a city water tower, as shown in figure 1. Rather sophisticated small nuclear bombs would be ejected at frequent intervals to a trailing position along the vehicle axis, and exploded. The explosion pressure of these bombs reacts on the pusher plate shown at the base of the vehicle. The shock load on the vehicle proper is minimized by a tuned damper system connecting the pusher plate to the cabin. Additional shock isolation beds may be required to minimize the oscillatory accelerations on the crew.

The operational sequence on the vehicle is shown in figure 2. Following ejection, the bomb is exploded when the pusher plate reaches its maximum rearward velocity. Thus, the piston inertia helps to shield the cabin from the shock load. The full explosion pressure reverses the motion of the pusher plate relative to the cabin to reset the cycle for the next explosion.

A minimum-size ORION vehicle, weighing perhaps 600 tons, is shown mounted on a Saturn booster in figure 3. The chemical first stage is required to boost ORION to a sufficient altitude in order to permit the first nuclear explosion. This altitude is perhaps above 35,000 feet. The ORION vehicle then adds sufficient velocity increment through a series of nuclear explosions to accomplish the desired mission.

The payload capability of ORION improves as the size increases, with vehicles as large as 5000 tons contemplated. Except for the nuclear bombs, the entire ORION vehicle and payload would be returned to Earth's vicinity for reuse after accomplishing the mission.

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The gaseous core reactors shown in figures 4 to 8 offer about the same kind of performance promises. To be superior to the more conventional heat-transfer nuclear rocket, the hydrogen propellant must be heated to temperatures above the melting point of most materials. This heating is accomplished in a gaseous core reactor through which the hydrogen is passed. The fluid mechanic arrangements are generally tailored in special ways so as to conserve as much of the reacting uranium gas as possible. A typical cavity might be 10 feet in diameter and 10 feet long. This cavity would be surrounded by high-temperature moderating materials to thermalize the neutrons. The propellant must cool these materials to counteract the neutron and gamma heating amounting to about 10 percent of the total energy generated. Thus, the thermal properties of the moderator set an upper limit for the specific impulse of a gaseous core reactor at about 3000 seconds.

If only uranium gas fills the core, the minimum pressure for hot criticality is about 25 pounds per square inch. With the addition of the hydrogen propellant, the wall pressures approach 1000 to 10,000 pounds per square inch.

The pressure shell to contain these pressures is sufficiently thick so that no reactor shield is required. Nevertheless, the total weight of the reactor including moderator and pressure shell is from 250,000 to 500,000 pounds. Hence, in a mission comparison with more conventional nuclear rockets, the gaseous core reactor system would likely require a fuel load of more than 500,000 pounds to capitalize on its higher specific impulse.

An early suggestion for a gaseous core reactor is shown in figure 4. Tangentially entering hydrogen passes radially inward through a gaseous uranium vortex. Hopefully, the centrifugal forces associated with the heavier uranium molecules would be balanced by the diffusion drag of the inwardly moving hydrogen. The hydrogen would ultimately move along the axis to the exhaust nozzle.

Unfortunately, the drag produced by the flowing hydrogen is so great that excessive loss of uranium will occur unless the hydrogen flow rates are limited to very low values. Hence, in a single-tube vortex reactor, only low thrusts could be obtained without excessive loss of uranium.

One way to avoid this difficulty is to use multiple vortex arrangements as are shown in figure 5. Criticality is achieved by the combination of many gaseous uranium cores. These may either be materially separated, as in the upper left diagram, or established by a matrix injection pattern, as shown in the square box drawing. These schemes were proposed by Jet Propulsion Laboratory and Space Technology Laboratory. Both have a major problem of cooling the enclosed hardware.

The United Aircraft Corporation uses an alternative approach to boost the hydrogen flow (fig. 6). Ninety percent of the incoming swirling hydrogen moves axially toward the annular exhaust nozzle on one end. This hydrogen would be seeded with additives to absorb the radiant heat from the core. Note, however, that the slower moving boundary layers near the end walls will not sustain the radial pressure gradients generated by the vortex. Hence, there will be a radially inward secondary flow of about 5 percent on each end wall. This hydrogen

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blends with the 0.1 percent diffusion flow and moves along the axis to the discharge nozzle.

The Lewis Research Center's coaxial jet reactor is illustrated in figure 7. The central core of uranium gas would be injected at a much slower speed than the coaxially flowing hydrogen. Hopefully, the mixing processes can be tailored to minimize the uranium loss rate. A hydrogen buffer layer would be added with an intermediate velocity profile between the uranium and outer hydrogen layer to serve this purpose. The hydrogen to uranium flow ratio should be 50 to 100 or higher for reasonable fuel conservation.

Though hidden by the vortex flow, the United Aircraft scheme (fig. 6) must have axial velocities and axial velocity ratios comparable with those of the coaxial jet. Hence, the differences in the axial velocities between the fuel and the hydrogen would produce the same kind of mixing losses of uranium as for the coaxial jet.

On the other hand, the axial velocity component can be eliminated on the vortex reactor by using tangential entry and exit of the fluid. A schematic of this arrangement is shown in figure 8. In all of these reactors, the principal heat-transfer mechanism to the hydrogen is by radiation, and the hydrogen must be continuously seeded with graphite powders and other materials to absorb the radiant heat before it reaches the containing walls.

There are, of course, grave questions concerning the stability of the boundary between the uranium and hydrogen flows. These questions would be complicated by feasible nuclear criticality interactions due to expansions, contractions, or distortions of the uranium gas regions. There are also extreme heat-transfer problems in the exhaust nozzle that would require solution before such cavity reactors are practical.

The next scheme to be discussed is the thermonuclear rocket. The approach (not yet accomplished in the laboratory) is to heat a plasma of light elements to a temperature on the order of 1 billion degrees Kelvin. At such temperatures, a portion of the ions are moving at sufficient speeds to cause fusion upon collision, accompanied by the release of large energies.

Four frequently considered fusion reactions are shown as follows:

$$D + D \rightarrow He^3(0.8\ \text{Mev}) + n(2.4\ \text{Mev})$$

$$D + D \rightarrow T(1.0\ \text{Mev}) + p(3.0\ \text{Mev})$$

$$D + T \rightarrow He^4(3.5\ \text{Mev}) + n(14.1\ \text{Mev})$$

$$D + He^3 \rightarrow He^4(3.6\ \text{Mev}) + p(14.7\ \text{Mev})$$

The amount of energy liberated to each particle is given in million-electron-volt units. One electron volt is equivalent to 11,605° K. The first two reactions occur with equal probability and are between deuterium ions. The third and fourth are between deuterium and either tritium or helium 3, respectively. The

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difficulty with using deuterium-deuterium and deuterium-tritium reactions is that a large fraction of the energy appears as high-velocity neutrons.

At the temperature ranges of interest, only magnetic fields offer promise as a means of confinement. The neutrons are unaffected by magnetic fields and are thus lost from the reaction zone. Recovery of this energy in a cooled shield would only complicate a thermonuclear space propulsion system. Hence, reactions liberating charged particles that can be trapped by magnetic fields are preferred.

Deuterium and helium 3 might be provided as the fuel utilizing the fourth reaction. If the reactor temperature is held at a sufficiently high value, the probability of a deuterium-helium 3 reaction is much greater than the deuterium-deuterium reaction so that only about 5 percent of the energy would be liberated as neutrons.

The reacting plasma would be contained in a magnetic bottle as shown in figure 9. The charged particles are reflected back toward the reactor interior by the stronger fields on the ends. The plasma pressures of more than 1000 pounds per square inch suggest confining field strengths of over 100 kilogauss. These fields would be provided by superconducting magnets to minimize the power losses associated with containment. The field on one end of the reactor would be weaker than on the other end, which would allow propellant to flow through the magnetic nozzle to space to produce thrust.

The cryogenic magnet must, of course, be cooled to low temperatures with a liquid-helium system. To minimize the heat load on the magnet due to bremsstrahlung and neutron radiation, shields are provided, as shown in figure 10. The thermal capacity of the hydrogen cools the cryoplant and the neutron shield (secondary). This hydrogen is ejected by the reactor-exit jet. Additional cooling through a radiator system is required for the primary bremsstrahlung shield.

The performance of such a thermonuclear rocket is pretty spectacular. Thrust to engine weight ratios as high as 0.01 are feasible and correspond to about 1 or 2 kilowatts of jet power per pound of engine weight. The specific impulse would be on the order of 10,000 seconds. The performance of such a system would therefore be about an order of magnitude better than that predicted for a nuclear fission electric propulsion system.

The Lewis Research Center is investigating the feasibility of using ion cyclotron resonance as the means of kindling the thermonuclear reactor. For this scheme to be effective, magnetic field strengths in the 100-kilogauss range are required. Hence, the Center also has a modest effort devoted to the production of intense magnetic fields; several water-cooled magnets (fig. 11 shows one of these) in the 100-kilogauss range have been tested. Lewis also has the liquid-helium production capacity to support major projects with superconducting magnets. However, no one has yet sustained a controlled thermonuclear reaction in the laboratory either here or elsewhere. Hence, it is far from timely to get enthusiastic about the mission capabilities of thermonuclear systems - the understatement of the conference.

The final idea to be discussed is the radioisotope balloon shown in figure 12. Concentric conductors are arranged in either spherical or cylindrical geometry from the electrodes of a radioisotope battery. The inner shell would be coated with either an α-emitting or a β-emitting radioisotope. The high-energy charged particles that are ejected through radioactive decay generate the electrode potential to form a nuclear battery.

Because of the potential difference, there will be an attraction between the inner and outer electrodes. This attractive force must be canceled by rotating the outer balloon. If spherical geometries are used, the poles have no centrifugal forces. Hence, a yoke is also required.

The voltage output for a radioisotope battery is high - on the order of 1/2 million to 1 million volts setting a minimum size for the battery of a few feet in diameter. This high voltage would be used to accelerate colloidal charged particles to speeds corresponding to specific impulses in the thousands.

The overall weight of the system including powerplant and accelerator is estimated at a few pounds per jet kilowatt. The mission potential of such a lightweight system is, of course, very good. However, the optimism toward using such a system is not very high.

The problem is traceable to a combination of properties of the isotopes. The α-emitters are desired but are in very short supply. They also must be carefully contained, for they are toxic and are deposited in the bone marrow of animals, where the radioactivity does irreparable damage. The β-emitters are much more plentiful, but the high-speed electrons generate X-rays on impact with materials. Hence, 3 to 6 inches of lead would be required at the launch pad to shield the crew. Also, the heat generation of the radioisotope decay process cannot be shut off. Hence, a major cooling problem might be faced during launch on any high-power radioisotope system.

Research is being conducted on the radioisotope systems as well as on the other Wild Blue Yonder propulsion schemes discussed in this paper. None has progressed to the point of sufficient confidence to justify a development program, and some may never reach that point. The chemical rocket still looks like the work horse for missions to be flown in the next 10 years.

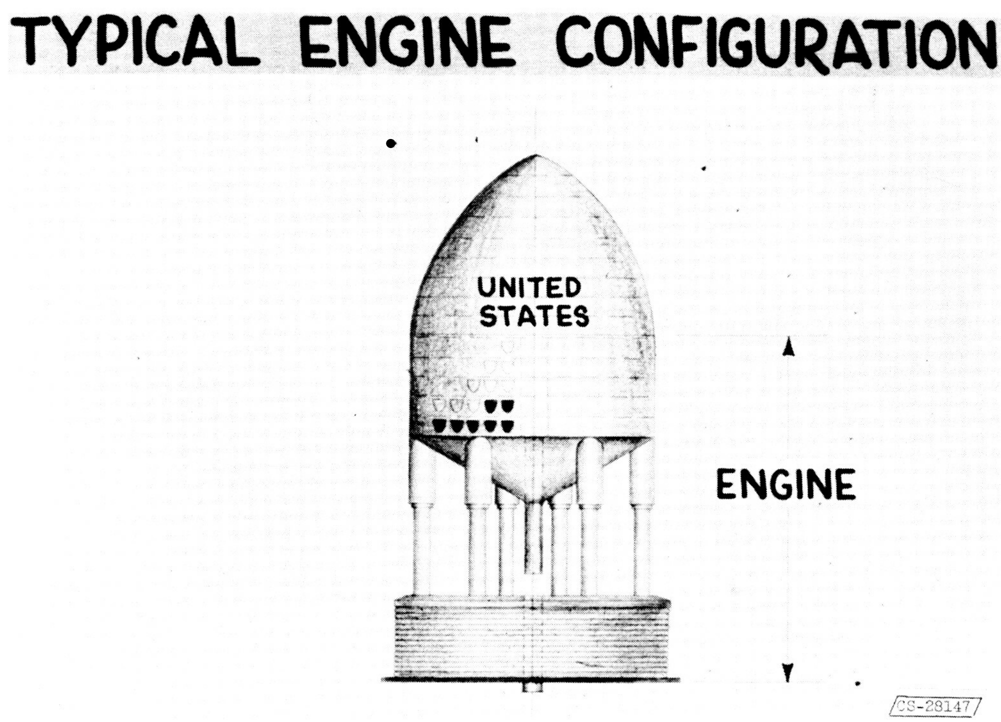

Figure 1

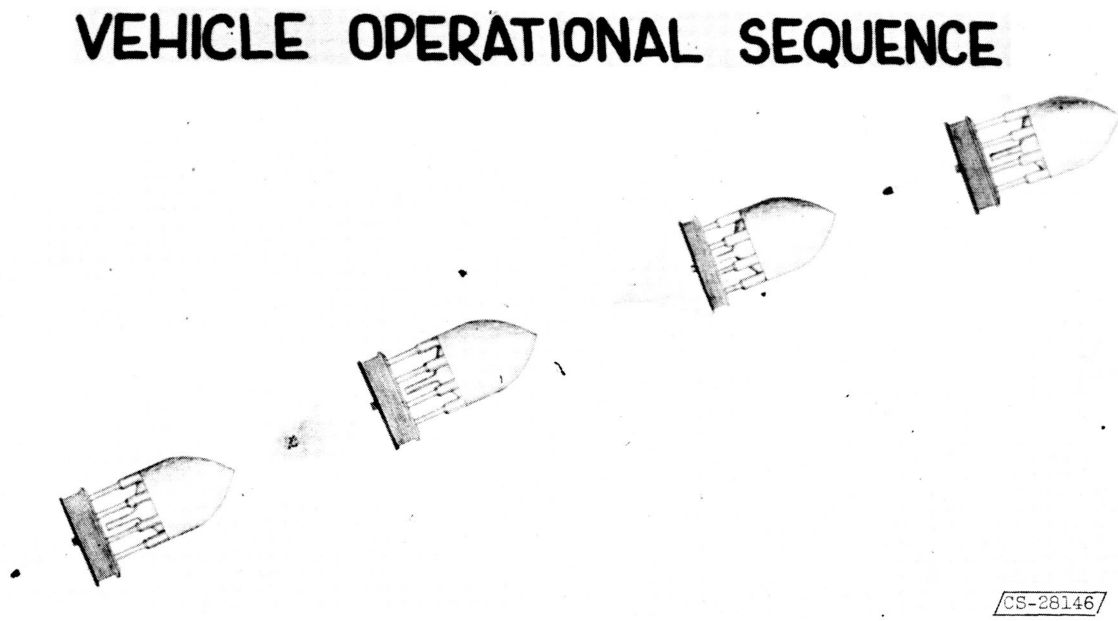

Figure 2

SATURN BOOSTED

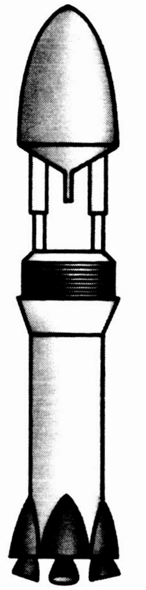

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Figure 3

VORTEX GAS CORE REACTOR

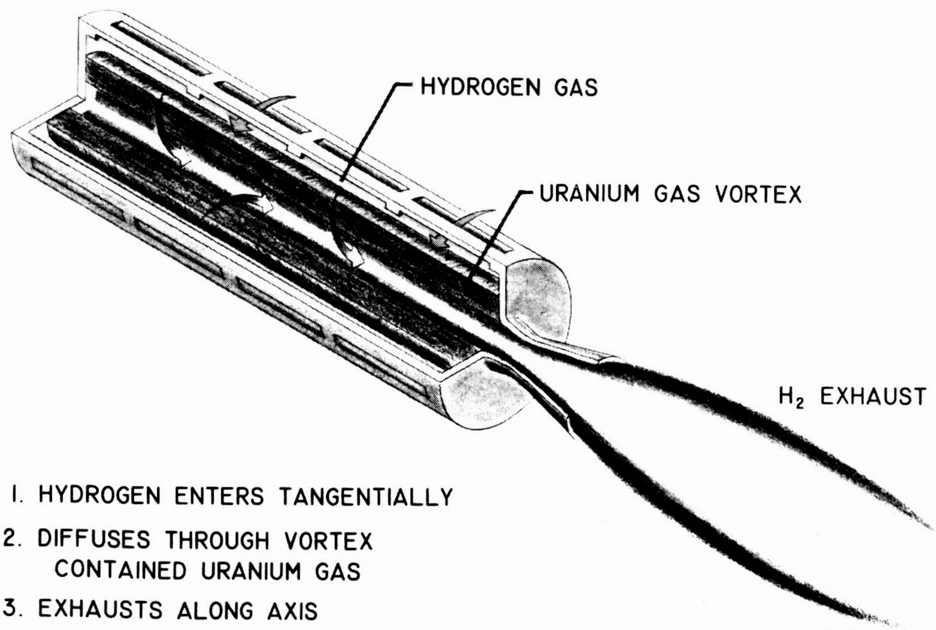

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Figure 4

VORTEX TUBE AND VORTEX MATRICES

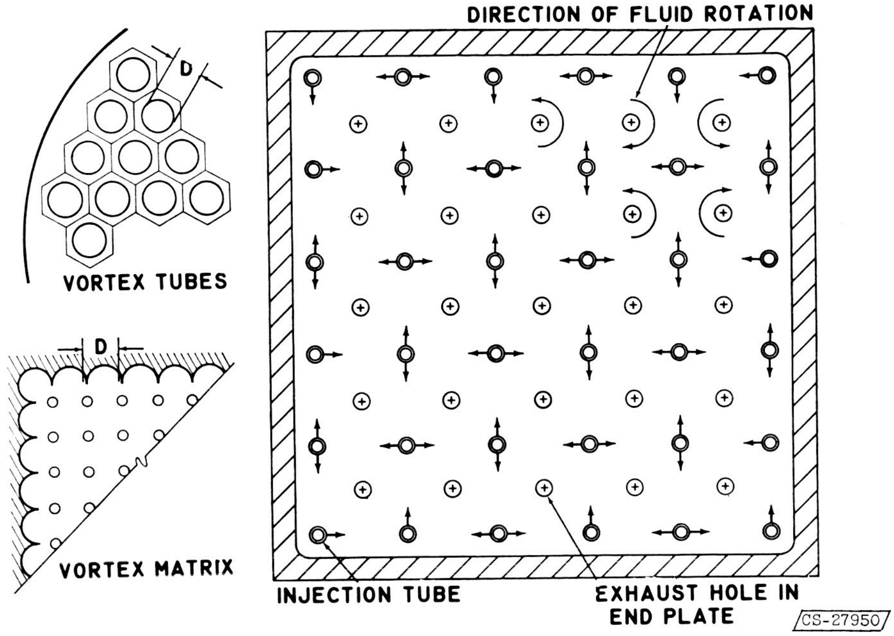


Figure 5

UNITED AIRCRAFT REACTOR

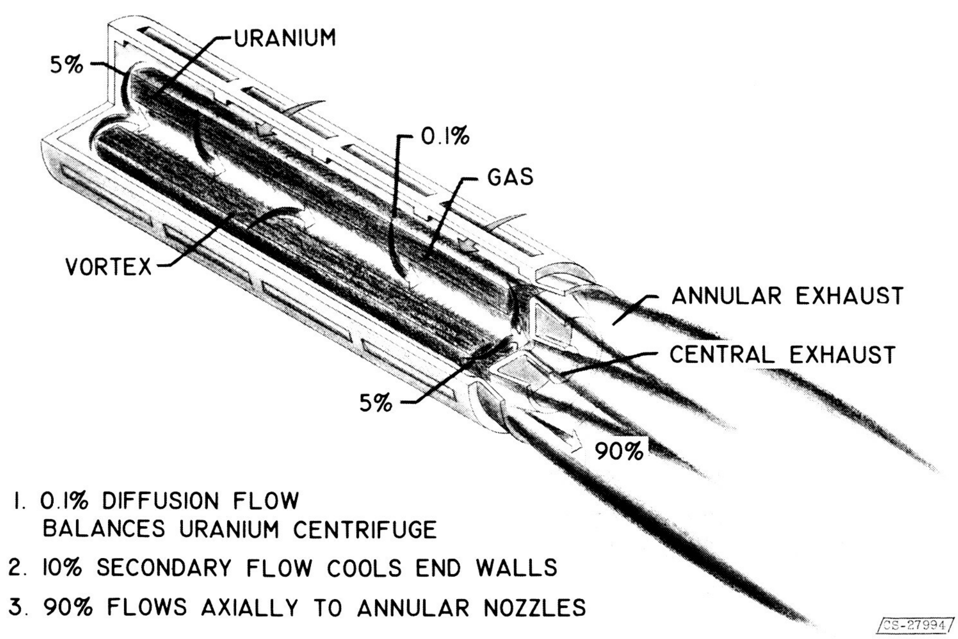
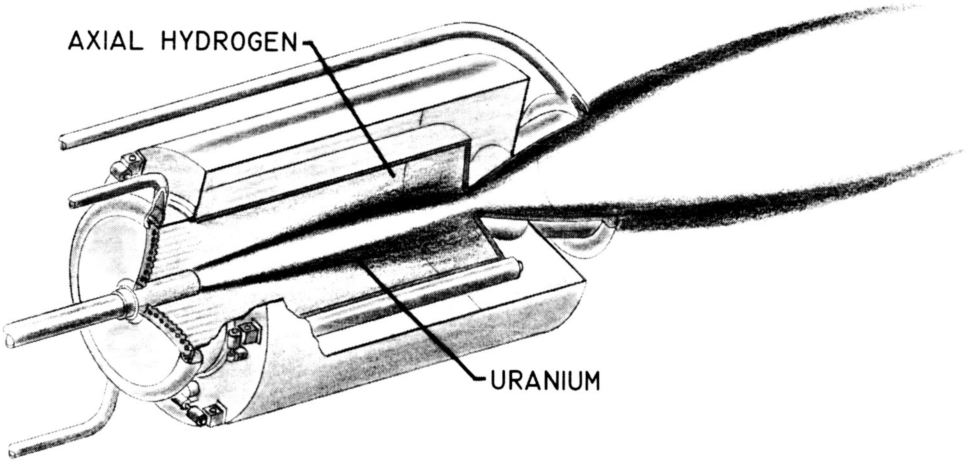


Figure 6

COAXIAL JET REACTOR

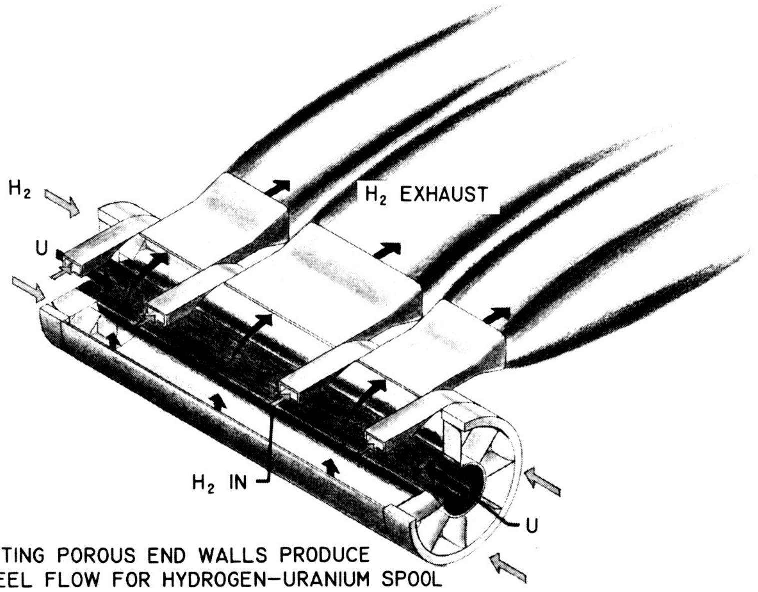


1. URANIUM AND HYDROGEN FLOW AXIALLY
2. URANIUM FLOWS MUCH SLOWER THAN HYDROGEN
3. TURBULENT MIXING GRADUALLY CONSUMES URANIUM
4. LOW SPEED HYDROGEN BUFFER LAYER MINIMIZES URANIUM LOSS

CS-27997

Figure 7

WHEEL FLOW REACTOR



1. ROTATING POROUS END WALLS PRODUCE WHEEL FLOW FOR HYDROGEN–URANIUM SPOOL
2. AXIAL VELOCITIES TAILORED TO GIVE MINIMUM SHEAR
3. PRINCIPAL HYDROGEN FLOW ENTERS AND LEAVES TANGENTIALLY

CS-27996

Figure 8

THERMONUCLEAR ROCKET

BASIC COMPONENTS

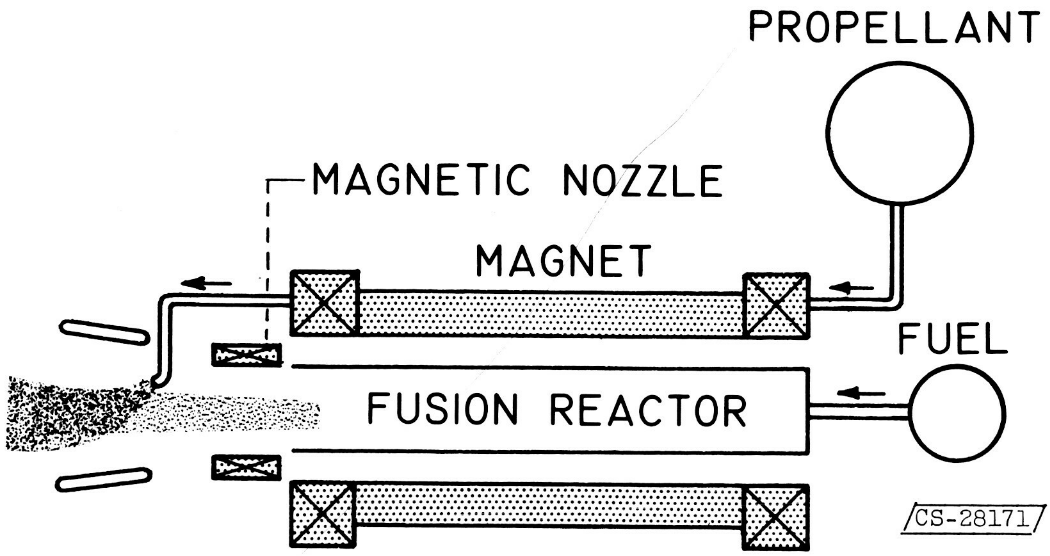


Figure 9

THERMONUCLEAR ROCKET

INCORPORATION OF SHIELDED AND
CRYOGENICALLY COOLED MAGNET

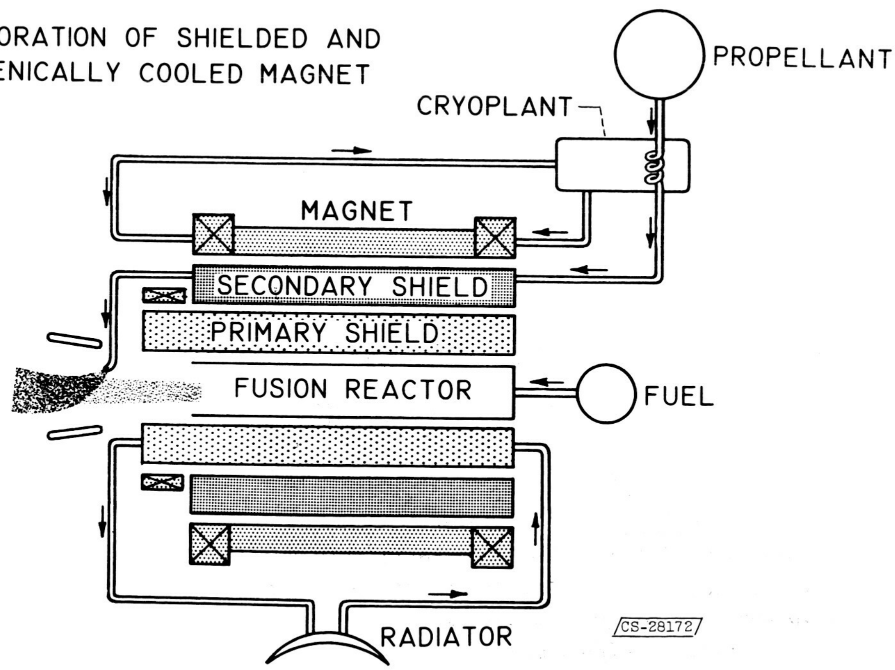


Figure 10

100,000-GAUSS WATER-COOLED ELECTROMAGNET

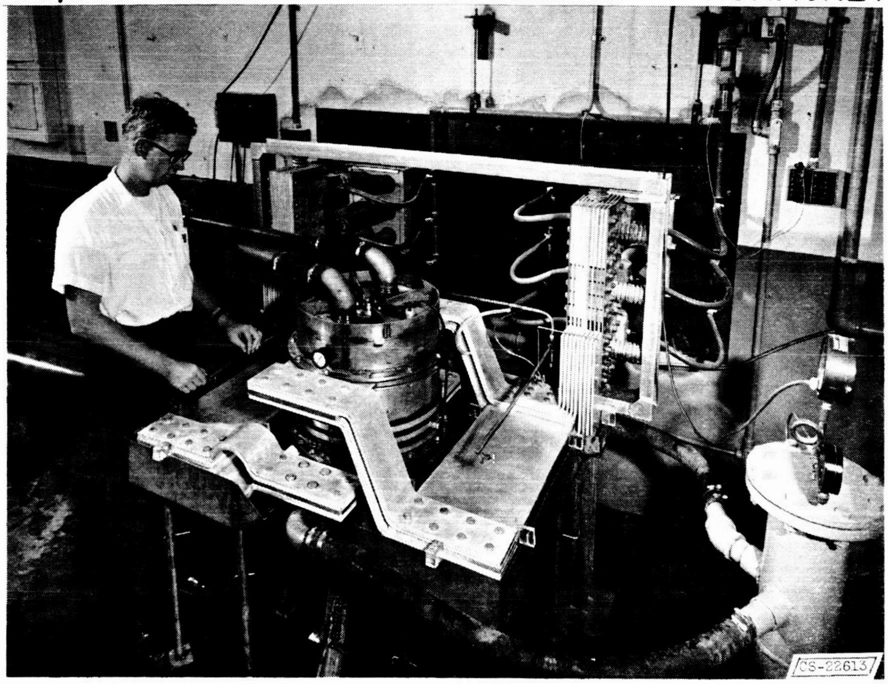


Figure 11

ELECTROSTATIC PROPULSION SYSTEM WITH DIRECT NUCLEAR ELECTROGENERATOR

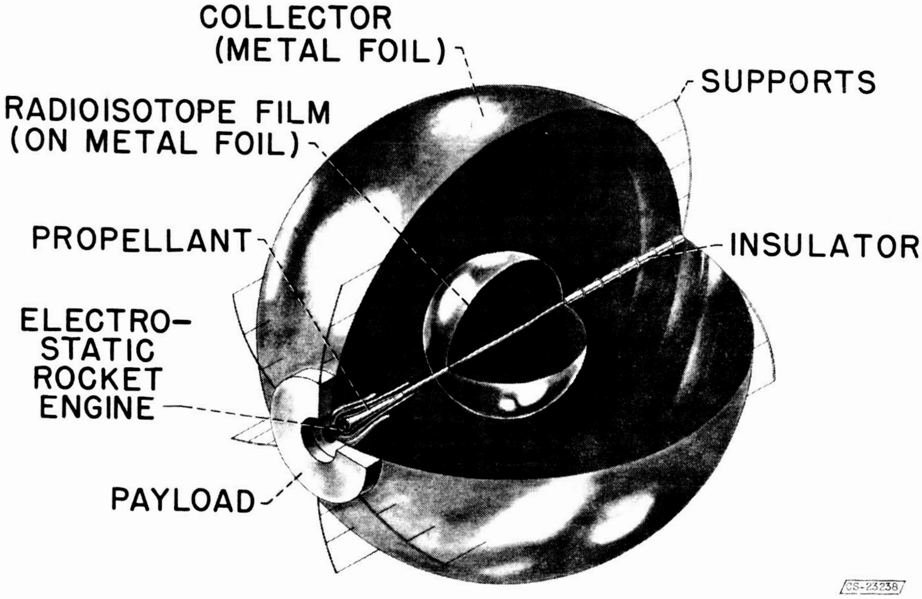


Figure 12