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NUCLEAR PROPULSION - BACKGROUND
AND ENVIRONMENTAL CONDITIONS

Livermore, California

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Preface

This report was presented at a symposium on Ceramics in Aerospace Technology held at UCLA on January 29, 1963. It is to be incorporated into the book Ceramics in Aerospace Technology to be published by John Wiley & Sons, Inc. References given in this report to other chapters or sections are to chapters or sections in the book. ✓

NUCLEAR PROPULSION - BACKGROUND AND ENVIRONMENTAL CONDITIONS*

A. J. Rothman

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Livermore, California

May 1, 1963

I. Foreword

Our interest in this symposium is primarily in ceramic materials. We are confronted with an overwhelming number of possible systems and concepts to discuss. So little is known about many of these that few things can be said, about materials. In other cases, the systems are developing and materials problems have become real. Yet one wishes to have some feeling about where the developing systems fit into the overall picture. I have tried to steer a middle ground and describe in general terms requirements of space propulsion engines, and on the other hand to consider several developing systems, in which ceramic environments and demands are more clearly understood.

Nuclear reactors are discussed throughout this paper. A brief exposition of nuclear reactors is given in the appendix. However, detailed principles are considered beyond the scope of this presentation, and the reader is referred to references such as 1 and 2.

II. Introduction

We can immediately consider two regimes for aerospace nuclear propulsion:

(1) That region which is close to the earth's surface, well within the atmosphere. This is the region where one is vitally concerned with atmospheric drag, and is currently of interest largely for military purposes. We shall call this the "atmospheric propulsion" region.

(2) All other space regions, from above the dense atmosphere to truly outer space. This includes missions for satellite placement, for lunar

*This work performed under the auspices of the U. S. Atomic Energy Commission.

and interplanetary probing, and to the stars beyond. We shall simply talk about rockets of all types for these regions.

This breakdown is arbitrary, but helps us to distinguish between the two major areas of activity.

A. Atmospheric Nuclear Propulsion

For missions from point to point close to or at the earth's surface we of course have conventional aircraft or missiles using chemical energy. The corresponding nuclear propulsion devices would be the nuclear propelled aircraft or nuclear missiles. The former would be exemplified by the late deceased Aircraft Nuclear Propulsion (ANP) project, while the latter still exists as Project Pluto, a nuclear ramjet air-breathing system.

I choose to concentrate largely on Pluto since it is under serious consideration. First, however, I should give a brief history of Aircraft Nuclear Propulsion.

Nuclear reactors as a means of propelling aircraft had been considered almost simultaneously with civilian stationary reactors. And so in 1946 the Nuclear Energy for Propulsion of Aircraft (NEPA) project was begun at Fairchild to determine feasibility. Feasibility was also studied at MIT (Lexington Project) beginning in 1948. In 1950-51 the prospect appeared promising enough so that the Air Force and AEC funded General Electric to begin the Aircraft Nuclear Propulsion (ANP) project. Work began in 1951 to develop a direct cycle flying power plant, and after several perturbations expanded steadily after 1954. The work encompassed a series of test assemblies, known as the HTRE's (Heat Transfer Reactor Experiments). During the course of the work, metallic fuel elements were developed, and progress was made in the development of a BeO ceramic-core reactor (Advanced Core Test or ACT). At the same time, Pratt and Whitney was funded to study indirect cycle (liquid metal heat transfer agent). However, the ANP program was cancelled in mid-1961. Unlike Pluto, the ANP reactor was to have operated for long flight times and was to be part of a manned aircraft. Thus important parts of their program were extensive in-pile tests of components for hundreds of hours and the development of means to prevent fission fragment escape from fuel elements.

Another milestone in the development of gas-cooled ceramic reactors was the Daniels File project.⁽³⁾ This project sprang from a concept

of Professor Farrington Daniels and also began just after World War II. The project was to demonstrate the feasibility of a BeO-moderated, gas-cooled power reactor. Although it was cancelled after 1-1/2 or 2 years, it helped call attention to some problems to be solved in the application of BeO to the nuclear reactors, viz., the need to improve mechanical properties, to overcome thermal stresses, and to protect against water vapor corrosion. During the project, studies were begun on these matters as well as on radiation damage, etc.

III. Nuclear Ramjet (Pluto)

Project Pluto began as such in 1957 at the Lawrence Radiation Laboratory (LRL) of the University of California. Early materials work was also done at Atomics International. The purpose was to develop a nuclear engine for air-breathing missile application. This missile in turn was to be the agent of the SLAM system, which was of military interest because of the missile's ability to fly close to sea level at high speed. Thus the missiles would be difficult to detect without very costly radar or similar detection defenses. Another advantage is the flexibility in mission profile, as compared to ballistic type missiles, which are the only comparable competitors for intercontinental delivery. The SLAM concept would utilize map-matching guidance for precision and self-correction. In an ICBM system, once the button is pushed, no corrective action can be taken to retract the move, except drastic interception or destruction. In the SLAM system, on the contrary, one may change the direction long after take-off. I must interpolate here a major problem, viz., that details of the solutions and implementation of Pluto problems are classified as restricted data, as is the hardware. Thus, while I may define, indicate, and give early results, I am not at liberty to divulge demonstrated performance and details.

Let us now examine the environment and characteristics of the Pluto reactor and missile.

A nuclear ramjet engine (Figure 1) is very simple in principle. It is essentially a flying heat exchanger which "breathes" air, converts the velocity of air to pressure, permits the air to pass through holes in a nuclear reactor which is the heat exchanger, and finally exhausts the hot air from a nozzle to give the vehicle its thrust.

Fig. 1. Conceptual nuclear ramjet.

There are obviously a number of requirements — aerodynamic, nuclear, mechanical, chemical, etc., which must be met for the device to be successful (Table I).

The aerodynamic requirement is fundamental and sets the stage for the other problems. A systems study indicates the type of behavior shown in Figure 2. Here we show net thrust (relative only) as a function of heat exchanger wall temperature. Two matters are clear at once:

1. The net thrust coefficient maximizes near Mach 3, i. e., the reactor "wants to fly" there. Thus we must provide continuous thrust to maintain that speed, or else the operation becomes unstable and the ramjet will fail to fly.

2. Net thrust increases as wall temperatures rise. Thus we must reach a certain minimum wall temperature, and preferably should achieve much higher values than the minimum to allow an adequate margin of safety, compensate for design errors and performance perturbations, and also to permit savings, e. g., by decreasing size and so saving fuel and structure costs.

The stagnation conditions for Mach-3 air at sea level are 500 psi and 1000°F. This air pressure, in conjunction with high reactor temperatures, causes problems in strength and deformation and results in a highly reactive, oxidizing environment which materials must withstand. Pressure drop across the reactor, for example, will give a steady compressive stress and creep may result. Also the high oxygen partial pressures may enhance oxidation rates.

At this point, consider the materials choices. Economics dictates a moderated rather than a fast reactor, since the former requires much less fuel for a critical configuration, and enriched uranium becomes an important part of the cost of the reactor. Thus, a solid refractory moderator is required, which will then also be an important part of the structure. Since we must limit our choice to the lighter atoms as moderators, we may consider elements of mass number up to carbon. Some typically good moderators are shown in Table II. Melting points and high neutron absorption cross sections practically eliminate all choices except some hydrides, beryllium compounds, and graphite. Oxidation problems further limit us to BeO and metallic

Table I. Major programmatic problems - Pluto.

<u>Aerothermodynamic:</u>	Heat transfer, fluid flow, stability.
<u>Mechanical:</u>	Thermal stress, creep of reactor fuel elements; supporting structure.
<u>Nuclear:</u>	Accurate critical calculations, especially at high temperature. Radiation effects.
<u>Control:</u>	Rapid, reliable control system with reactor control rods operable at high temperatures.
<u>Chemical:</u>	High temperature oxidation of reactor fuel and supports. Corrosion of BeO by water vapor.
<u>Test Reactors:</u>	Provide critical assembly, blowdown facilities, and remote disassembly facilities for "hot" reactor.
<u>Fabrication:</u>	Several hundred thousand brittle fuel elements produced to close tolerances.

Table II. Neutronic properties of moderators.*

	Density, g/cm ³	Thermal Absorption Cross Section, cm ⁻¹	Slowing Down Power	Moderating Ratio
H ₂ O	1.0	0.0195	1.28	66
D ₂ O (99.75%)	1.1	0.000085	0.18	2100
Be	1.84	0.00109	0.15	141
BeO	3.01	0.000725	0.15	208
Graphite	1.6	0.000342	0.0612	180

*Data from Refs. 36 and 2, modified by author.

Fig. 2. Performance curves - nuclear ramjet.

beryllides as moderator choice. These latter have questionable to poor oxidation resistance at intermediate temperatures and poorly developed technology, so our obvious choice had to be BeO.

Fuel for the reactor must also resist severe oxidation conditions at high temperatures. This naturally leads to UO_2 , suitably stabilized. The phase diagram for $\text{UO}_2\text{-O}_2$ is complex and a number of metastable and questionable phases have been reported.⁽⁴⁾ Figure 3 shows a highly simplified phase diagram of the system, primarily exhibiting the lower oxidation cubic phases (UO_2 and U_4O_9) and the higher oxidation expanded phases (U_3O_8 , orthorhombic or hexagonal).

The volume change upon oxidation from UO_2 to U_3O_8 is over 30% and so tends to cause cracking of fuel elements in which an appreciable volume of UO_2 is dispersed in BeO. Furthermore, if one attempts to fabricate BeO- U_3O_8 fuel elements, one has to sinter in an oxidizing atmosphere, and so contend with the high vapor pressure of UO_3 gas over U_3O_8 (Figure 4). Similarly, this high vapor pressure militates against the use of unstabilized UO_2 or U_3O_8 at high temperatures shown to be desirable for reactor performance. To stabilize the cubic fluorite phase and reduce the UO_3 vapor pressure, compounds or solid solutions of urania may be used. Examples are lanthana and yttria, and the reduction of volatility is shown in Figure 4. The former two oxides presumably stabilize UO_2 by valence compensation,⁽⁵⁾ so that even though oxidation of the $\text{UO}_2\text{+M}_2\text{O}_3$ occurs, the oxygen fits into the anion-deficient lattice. Indeed, the volume change is negative upon oxidation.

Consider the oxide Y_2O_3 , with which UO_2 forms a series of solid solutions. A fuel element may be prepared by dispersing such a solid solution in BeO. The urania volatility is markedly reduced in this manner, but new problems are created. Yttrium has a higher thermal absorption cross section (1.2 barns) than Be (0.01 barn) and also displaces BeO moderator from a fixed volume reactor. Thus, additional fuel is required to compensate for it. Furthermore, there is a eutectic between BeO and Y_2O_3 at 1580°C which places an upper temperature limit on the fuel element.

At the high velocity near sea level, thrust produced by the reactor is largely needed to overcome air resistance (viscous drag) rather than to keep the missile aloft. As a corollary, it is obvious that the cross sectional area of the reactor and missile should be as small as possible, consistent with

Fig. 3. Simplified phase diagrams: $\text{UO}_2\text{-O}_2$.

Fig. 4. Vapor pressure of UO₃ over various compounds.

fuel cost, to reduce drag. This in turn forces high power generation per unit volume and large heat transfer rates. Thus thermal stress in the brittle matrix is an important problem to be overcome. Several approaches are possible, and indeed it is reasonable to utilize all of them to some extent. One may use as favorable a geometry as possible. The thermal stress due to internal heat generation in a thin-walled cylinder has a strong dependence on wall thickness. Thus, one should minimize the wall thickness of the cylinder consistent with practicability of fabrication. As an aside, note that for constant flow area, as wall thickness decreases, surface area exposed to the air stream increases and so a greater volume of coating is required for a given degree of BeO corrosion protection (see below) which then displaces BeO moderator and "costs" more fuel.

Another approach in the elastic behavior temperature region is to prestress the ceramic and so generate a stress reverse to that produced under operating conditions.

A materials approach is to improve the strength of BeO at intermediate temperatures (say below 1000°C) and to encourage creep relaxation at 1000°C or above to relieve thermal stresses.

The form of the thermal stress parameter versus temperature for BeO is given in the chapter on BeO. Because of the large decrease in thermal conductivity with temperature, elastic thermal stress becomes increasingly severe as temperature increases, up to temperatures at which relaxation may occur to relieve thermal stress.

Increase in elastic strength and decrease in relaxation temperature may be accomplished by utilizing fine-grained BeO. This is further treated in another chapter.

Still another approach is to use additives to enhance deformation (e.g., those which increase diffusion in the BeO). However, one must avoid excessive compressive creep at higher temperatures. This problem, too, will be considered in the chapter on BeO.

Having selected BeO as the moderator and fuel matrix, we must be prepared to cope with the volatilization of BeO in water vapor. The reaction becomes especially important at temperatures appreciably above 1200°C for the modest periods of time and high air velocities appropriate to planned Pluto flights. Operationally, the missile may fly high over humid areas to mitigate the problem. For example, a typical humidity may be reduced from 1% at sea

level to $< 0.05\%$ at 25,000, to $< 0.001\%$ at 50,000 feet.⁽⁶⁾ However, the advantage of flying low and the benefits to be gained by very high reactor temperatures forces one to consider protective coatings for BeO. A coating should:

1. Not be reactive with BeO at the use temperature.
2. Preferably be protective even for small thicknesses so as not to detract from reactor performance.
3. Have good thermal expansion match with BeO.
4. Have good thermal conductivity to avoid serious temperature gradients and thermal stresses.
5. Of course, be easy (i. e., cheap) to apply.

Compromises, of course, are called for. Coatings of various refractory oxides are possible, e. g., Al_2O_3 applied by slip casting, or by deposition from the vapor phase.⁽⁷⁾

Among other problems is that of fabricating high performance ceramic fuel elements reproducibly, and to close dimensional tolerances. Several hundred thousand such components must be provided for a reactor. This requires close control over raw materials, and reproducibility of mixing, extrusion, and sintering.

It is interesting to note that in compensation for severe oxidation and stress conditions, some of the customary nuclear reactor problems are avoided in the flying reactor. For example, radiation damage is less of a problem in Pluto than in civilian power reactors since the life span is moderately short and the temperatures are high, tending to anneal out much of the fission product and fast neutron damage, which would normally occur at low temperatures and long times. Another major requirement in civilian reactors is that of containment of fission products by the fuel elements. Even a small rate of release may cause deposits of high activity to build up and make maintenance of machinery difficult or exceed biological tolerances nearby. The small amount of fission products emitted from the ramjet, on the other hand, is diluted in many miles of atmosphere. This gives rise to activity which is minute compared to even the smallest atomic weapon. For example, consider the following illustration taken from Ref. 8. A 500-mega-watt reactor may produce about 100 grams of fission products in 5 hours of flight, during which it covers about 10,000 miles. Assume 1 gram is released to the atmosphere, which is a reasonable figure. It is clear that 1 gram of

fissioned material spread over 10,000 miles of path is negligible compared to the 1000 grams of fissionable material "burned" locally by a 20-kiloton fission bomb. Thus one need not design a fuel element with complete retention of fission activity.

Project Pluto is now under active development at LRL. The first of a series of reactors, Tory II-A, is shown diagrammatically in Figure 5. Figure 6 shows the stacking of the fuel elements in the reactor. Obviously precautions in the form of poison rods were necessary to prevent the core from going critical during assembly. Some design parameters are given in Table III for this reactor. Tory II-A was successfully operated in Nevada in 1961. Several runs were made to demonstrate engineering feasibility.

Table III. Tory II-A reactor parameters.

Power	155 MW
Air flow rate	708 lb/sec
Max fuel element wall temp	2250°F (1232°C)
Exit gas temp	1975°F (1080°C)
Core diameter	32 in.
Core length	48 in.

A more advanced reactor, Tory II-C, is now under construction and is intended for operation in Nevada late in 1963. The project will hopefully continue after suitability of a reactor has been demonstrated in ground tests to prove that a flyable reactor is also feasible.

More details on the behavior of BeO under environmental conditions mentioned above are presented in the chapter on BeO.

IV. Space Nuclear Propulsion

A. Background

We now turn to consideration of vehicles for space regions beyond the immediate vicinity of the earth's surface, i. e., rockets.

Let us first examine the principal classes of space propulsion systems (Table IV). We can classify these by energy source or by the means of obtaining thrust (e. g., directly by heating the propellant, or indirectly by generating electricity). Electric power in turn may be used in various ways. Probably of most interest in this category are the ion rocket and electro-magnetic plasma rocket.

Fig. 5. Tory II-A.

Fig. 6. Stacking of fuel elements in Tory II-A.

Table IV. Space propulsion systems.

I. Energy SourceA. Transportable

1. Chemical
2. Nuclear
 - a. Fission
 - b. Fusion
 - c. Radioisotopes

B. Environmental

1. Solar
2. Wireless
3. Ion or radical recombination
4. Magnetic fields

II. Means of Obtaining Thrust

- A. Direct heat transfer to gaseous propellant (thermal rockets, e.g., Rover)

- B. Conversion of primary source to electricity

1. Generation of electricity

- a. Dynamic (turboelectric generator, plasma power generator)
- b. Static
batteries, fuel cells, thermocouples, thermionic converters

2. Utilization of electricity

- a. Thermal: heating of propellant by electric arc (arc plasma)
- b. Acceleration of charged particles through electrostatic field (ions, colloids, dusts, droplets, etc.)
- c. Electromagnetic (magnetically accelerated plasma)

- C. Impulse (nuclear explosions, e.g., Orion)
-

A good deal more may be said (and has been — see bibliography) about comparisons among the various devices, especially since sufficient data to evaluate them in detail are mostly lacking.

Although it is not within the scope of this paper to discuss rocket principles in detail, a brief indication of why one must consider propulsion by nuclear means is in order.

From momentum conservation, we obtain the basic rocket equation expressed in various forms,^(9, 10) for launching from "free space," i. e., where gravity and drag may be neglected. One may apply a correction reducing v_B if take-off occurs from the earth, but this is incidental to our considerations. The mass ratio, m_L/m_B is:

$$\frac{m_L}{m_B} = \exp \left(\frac{v_B}{V_E} \right) = \exp \left(\frac{v_B}{I_{sp} \cdot g_0} \right) \quad (1)$$

where

$$I_{sp} \cdot g_0 = V_E \quad (2)$$

by definition, and I_{sp} is the specific impulse, a figure of merit.

Thrust is related to specific impulse by

$$F = \frac{m_L - m_B}{t_B} \cdot I_{sp} \cdot g_0 \quad (3)$$

The specific impulse for the ideal expansion of an ideal gas is

$$I_{sp} \text{ (ideal)} = \frac{1}{g_0} \sqrt{\frac{2\gamma}{\gamma-1} \cdot \frac{RT}{M}} \quad (4)$$

Note that I_{sp} is dependent on the (gaseous) propellant used and its temperature. The power may be expressed as

$$P = \frac{1}{2} F \cdot I_{sp} \cdot g_0 \quad (5)$$

The definitions of the symbols used above are:*

m_L = mass of vehicle of launching,

m_B, v_B = mass of vehicle and velocity at burnout,

*The definitions and equations are consistent for the CGS system. Modification is required for the lb force-lb mass system common in engineering practice, but the details are not relevant to the present discussion.

V_E = velocity of exhaust gas,
 I_{sp} = specific impulse, seconds,
 F = thrust,
 t_B = burning time,
 R = universal gas constant,
 g_0 = standard gravitational acceleration,
 M = molecular weight,
 γ = heat capacity ratio = c_p/c_v ,
 T = absolute temperature,
 P = power.

The problem of space propulsion is to attempt to get high specific impulses and so favorable mass ratios (low propellant consumption) and at the same time keep the thrust-to-mass ratio high. Like El Dorado, this goal is the one sought, but practically not attained. Unfortunately, these two requirements have generally not been compatible, and one must make compromises. The really high specific impulse systems have low thrust.

One is interested in placing a useful payload in a given location, which will require a particular thrust for a specified trajectory. It is evident that we wish to do this at the lowest cost. In turn, this leads to the demand for low propellant usage and light auxiliary equipment, i.e., equipment necessary to lift the payload but of no particular value at the space location, such as pumps, propellant tanks, nuclear reactor, etc. The effect of propellant on the mass ratio appears in equations (1) and (2). In order for m_L/m_B to approach unity, clearly one wants a propellant of high specific impulse, or thrust produced per unit mass of propellant consumed per unit time. Of course, equation (5) reveals that for a given thrust, the power must also increase as I_{sp} does. Higher power generally means heavier equipment, which in turn increases the cost, and further increases the thrust required for the mission.

If a particular mission is chosen, including the trajectory, the rocket burnout velocity (v_B) is thereby determined. Table V⁽¹¹⁾ shows some typical situations.

Table VI reveals the behavior of chemical systems, for launching rockets from the earth's surface to escape velocity. These idealized conditions ignore the weight of supporting structure, etc., and are merely concerned with

Table V. Incremented vehicle velocity requirements for typical minimum energy missions.⁽¹¹⁾ (Courtesy of G. P. Sutton from Rocket Propulsion Elements (1956), Table 12-2, p. 392. Published by John Wiley & Sons, Inc.)

Mission	Ideal velocity, [*] ft/sec	Approximate actual velocity, ft/sec
Satellite orbit around earth (no return)	26,000 to 35,000	30,000 to 41,000
Escape from earth (no return)	36,700	42,500
Escape from moon	7,700	9,200
Earth to moon (landing on moon, no return)	44,200	53,000
Earth to Mars (no return)	57,500	66,000
Earth to Venus (no return)	73,000	85,000
Earth around moon and return to earth [†]	74,000	83,000
Earth to moon, landing on moon and return to earth [†]	88,000	105,000
Earth to Mars, landing on Mars, and return to earth [†]	112,000	130,000

* Neglects air resistance, navigational corrections, and gravitational losses.

† Assumes no savings by air braking within atmospheres.

Table VI. Mass ratios for various propellants^{*} (earth-launched to escape).^{**}

	V_E , ft/sec	I_{sp} , sec	m_L/m_B
Solid propellant	6,900	225	200
Liquid O ₂ -kerosene	8,800	275	65
Liquid O ₂ -hydrogen	12,800	400	17
Hydrogen (2730 °C)	31,000	950	3.2

* Adapted from Ref. 10 (Schreiber).

** $V_B = 36,700$ ft/sec. Atmospheric drag ignored.

the propellant.* We also ignore atmospheric drag. The best chemical systems give a mass ratio of about 17. Thus, ideally, only 1/17 or about 6% of the launch weight reaches escape velocity. For the less effective solid propellant, only 1/200 of the launch weight escapes. Note that hydrogen heated to 2730°C would permit escape of 1/3.2 or 30% of the launch weight.

Another virtue of high specific impulse is the reduction in mission time for planetary exploration. Figure 7 taken from Ref. 12 shows the total trip time for a projected 7-man Venus mission for various I_{sp} . The weight of the spacecraft launched from earth was 1.7 million pounds, and the curve applies only to high thrust systems where thrust-to-mass ratio exceeds 0.1, otherwise the trip time increases considerably.

Nuclear energy has greater potential than chemical energy since, in equation (4), I_{sp} may be increased by increasing T and decreasing M . Clearly in the nuclear propulsion scheme hydrogen may be used instead of the higher molecular weight compounds introduced during chemical reactions. T is limited in the chemical rocket case since chemical bonding energies are only of the order of kilocalories or volts. However in nuclear fissioning, one can, in principle at least, achieve orders of magnitude higher temperatures. In practice, unfortunately this is not so, as we shall see.

B. Classes of Nuclear Rockets

At this point, consider "nuclear rockets" as a number of classes, depending on just how the nuclear energy is applied to produce thrust. This is a more specialized repetition of part of our earlier list.

- (1) Heat exchanger (Rover)
- (2) Gaseous core reactor
- (3) Impulse or explosion (Orion)
- (4) Recoil (fission fragments or other particles from nuclear reactions)
- (5) Thermonuclear (fusion)
- (6) Nuclear - Electric (ion or plasma)

*If we assume with Schreiber⁽¹⁰⁾ that about 10% of the launch weight is in the form of structure, tanks, etc., then m_L/m_B must always be less than 10 or else the payload becomes zero. This requirement would exclude chemical rockets from earth surface to escape, unless multistaging is used (as is the case at present).

Fig. 7. Effect of specific impulse on mission time.⁽¹²⁾ (Taken from F. E. Rom and R. G. Ragsdale, Proceedings of the NASA-University Conference on the Science and Technology of Space Exploration, Volume 2, (NASA SP-11) Nov. 1962, p. 64, Figure 42-3.)

Consider the heat exchanger rocket (Figure 8). It can be demonstrated⁽¹⁴⁾ that heating by radioisotopes cannot seriously be considered for space power (except for small auxiliary units as in SNAP-1), primarily because of their lack of availability and the high cost of separating them from radioactive wastes. Thus, let us confine our attention to reactor fissioning, as exemplified by Project Rover.

Unfortunately, we are once again temperature limited, as we were with the chemical rockets, since we must employ a solid matrix for the fuel elements. Once again chemical bond energies are limiting. In fact, since heat must be transferred from the fuel elements to the gas, propellant temperatures will be lower than those achieved in chemical rockets, where the containment walls may be cooled below combustion temperatures. However, we may still make a substantial gain by using a low molecular weight propellant, hydrogen. The ideal specific impulse of hydrogen at 3000°K is 950 seconds, as calculated from equation (4) (also Table VI). Even at 2500°K, I_{sp} is 870 seconds.

Thus, we have a potential specific impulse of over twice the best chemical system. Similarly, at escape velocity, the idealized mass ratio m_L/m_B from equation (1) or Table VI will be near 3, and we can now have a payload, structure, core, etc., of 30% the launch weight, as opposed to only 6% which we calculated earlier for the chemical case. Although conceivably the performance could improve somewhat by even more refractory materials, better heat transfer techniques, etc., we could not expect the specific impulse to markedly increase, e.g., double, since the highest melting point known is near 4000°K. Rom⁽¹²⁾ estimates a ten-fold reduction in weight for nuclear in interplanetary spacecraft when compared to chemical rockets.

We shall return later to more detailed considerations of the heat transfer nuclear rocket.

Gaseous core reactors (Figure 9) would offer much higher temperature potential. In this case, we would require a gaseous fissionable fuel in a chamber. A propellant gas is heated passing through it and exhausts from the rocket. The containing chamber would be the moderator and reflector, and sufficient fissionable material would have to be present at any one time to permit the reactor assembly to remain nuclear-critical.

GLL-638-1752

Fig. 8. Rocket with solid fission heat source.⁽¹³⁾ (By permission from Canadian Aeronautical Journal, Oct. 1959 issue, "Propulsion Systems for Space Travel," p. 340, by H. S. Ribner.)

Fig. 9. Schematic diagram of a gaseous nuclear rocket motor.

Fissionable materials are costly (90% U-235 costs about \$15,000-\$20,000 per kg). One may assume that at least 2 kg of U-235 are required in the chamber to maintain criticality, and may calculate^(15,9) that about 2000 throughputs of propellant are necessary to reach escape velocity. Thus the fuel cost is prohibitive unless we separate the fissionable material from the exhaust gas so well that only a small fraction of the former is lost. Finally, cooling of the chamber walls must be effective so that we do not approach temperatures of 4000°K where solids melt or vaporize because of weakened binding forces. So we have several formidable problems to overcome, which are related more to engineering techniques than to the nature of the materials employed. However, the fact that we could reach very high temperatures and specific impulses makes this concept appealing for future study.

An interesting analysis⁽¹⁶⁾ of the turbulent vortex gaseous fission rocket indicates that it has a problem in common with other "promising" advanced propulsion concepts, i. e., one pays for the high specific impulse by a low ratio of thrust to weight.

The other items on our list also have rather difficult problems to solve. The explosive nuclear rocket (Figure 10) which would be propelled by a series of nuclear explosions is being studied as Project Orion at General Dynamics. In principle, many small bombs (0.01 to 0.1 kilotons) are dropped at ~ 1-second intervals and exploded at 100-1000 feet away. The bombs are loaded with low molecular weight propellants (e. g., plastics), and part of the energy reaches a pusher plate which propels the vehicle.⁽¹⁷⁾ Project Orion is still active, although there is the practical question of the high cost of the fissionable material in such bombs without recovery. An interesting material problem presumably would be the pusher platform which must withstand thermal and impact shocks.

Fission recoil rockets (Figure 11) require a plane reactor which also acts as a sail. The rocket is impelled by fission fragments recoiling aft. Because the range of fission fragments in solids is so short, the reactor must be thin. However, this shape inherently conflicts with the compact shape optimum for criticality in reactors. Again, preventing excessive reactor temperatures would be a serious problem, since we may cool only by radiating to space. If instead, a radioisotope source of alpha particles is

Fig. 10. Orion conceptual design has pusher plate made of ablative material to absorb shock of nuclear explosion. Plate transmits impulse to payload through water-cooled springs.⁽¹⁷⁾ (By permission from April, 1961 issue of Nucleonics, 19, p. 100.)

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Fig. 11. Conceptual drawing of a thin plane reactor for the production of an anisotropic flux of fission fragments. The thick reflector on one side absorbs the fission fragments over one hemisphere.⁽¹⁴⁾ (By permission from Propulsion Systems for Space Flight, by W. R. Corliss (1960) Figure 7-1, p. 230, McGraw-Hill, New York.)

used, we avoid the problem of criticality, but create other serious ones. It is concluded⁽¹⁴⁾ that this technique cannot seriously compete in the foreseeable future.

In thermonuclear reactions, which are the opposite of nuclear fissioning, several nuclei "fuse" to form others and release large energies. A very energetic one is the deuterium-tritium reaction:

The 17.5 MeV thus produced represents an energy of 4×10^7 kW-hr/lb of reactants compared to 10^7 kW-hr released by complete fissioning of 1 lb of fissile material (U-235). This process would be most exciting for space use. Although it can be made to occur in the thermonuclear bomb, a sustained thermonuclear reaction has not yet been achieved in the laboratory. If and when it is, it offers the possibility of very high specific impulses and much better thrust-to-weight ratio than the electric rocket.⁽¹⁸⁾ Although we must reluctantly drop thermonuclear rockets at the present time, we may point out that an important materials problem connected with them is the use of superconducting magnets to minimize the weight. A concomitant problem is the requirement for refrigeration equipment for heat dissipation. Magnetic fields are required to contain the plasma.

We finally come to the nuclear electric rockets, i. e., those which use nuclear energy to generate and utilize electricity for propulsion. The electricity may be used thermally to heat a propellant gas which then expands through a nozzle. Heating of hydrogen gas may be carried out by electrical resistance elements of tungsten, or an arc-plasma may be generated to heat the gas. The former is very simple and may give I_{sp} of ~1000 seconds,⁽¹⁹⁾ but erosion of the elements restricts this device to short mission times. The latter, or arc-jet system, has undergone laboratory tests at $I_{sp} = 1100$ seconds, and a 1-kW unit is ready for flight test. Somewhat larger units (30 kW) are also under development. The arc-jet system will probably be limited to modest missions, e. g., orbit-to-orbit transfer, etc.⁽¹⁹⁾

Alternatively, a plasma may be formed and accelerated electromagnetically.⁽²⁰⁾ Many such systems have been considered.⁽²¹⁻²³⁾ Lifetimes of plasma engines tend to be short because of erosion of chamber and nozzle walls.

The electrostatic engines utilize an electric field to accelerate charged particles. Of most interest among all electric propulsion systems is the ion rocket (Figure 12). The ion rocket consists of an electrostatic source which ionizes cesium vapor after it contacts a hot surface (e.g., tungsten at 1100°C).⁽²⁴⁾ These ions are accelerated in a high voltage field. The positive ion beam must be neutralized before it leaves the rocket. Ion rockets are capable of up to 20,000-second specific impulses. Several problems are erosion of the ionizer and corrosion of metal containers and duct by the propellant before ionization. However, this latter problem is not severe if the cesium is pure.

In all electric rockets, the amount of equipment required to generate electricity, accelerate the plasma or ions, and radiate excess heat to space is large; thus they have very low thrust-to-mass ratios (ca. 10^{-4} to 10^{-5}). The best among them have high specific impulses, not temperature limited, of about 5000 to perhaps 20,000 seconds. The ion drives seem to have the edge with respect to thrust and specific impulse. Obviously, they must operate over long periods to permit the required velocities to build up since they accelerate so slowly. This magnifies the problem of reliability over time intervals, and means a relatively long-lived reactor compared, for example, with the minutes-to-hours life required for Rover reactors. Their roles appear to be as interplanetary probes, where the long distances favor their high specific impulse and the long times permit the low thrusts they offer. It is clear that they cannot be self-boosted from the earth because of their low thrust/mass ratios.

NASA Space Electric Rocket Tests (SERT) have begun this year and are expected to continue until 1970 to qualify electric thrusters (primarily in rockets) for mission application.⁽²⁵⁾

A summary of the capabilities of the major propulsion system in terms of thrust/weight and specific impulse is given in Figure 13.

C. Electric Power Supplies for Nuclear Rockets

It is often conceded that the development of efficient and compact power supplies for the electric rocket is behind that of the propulsion systems.^(19,26) The most critical requirement is that of reducing the weight of electric generators, although reliability for long periods and high efficiency

Fig. 12. Ion-propulsion system.⁽¹³⁾ (By permission from Canadian Aeronautical Journal, Oct. 1959 issue, "Propulsion Systems for Space Travel," p. 341, by H. S. Ribner.)

Fig. 13. Summary chart showing the major space propulsion systems and their approximate performance limits. The dashed lines indicate the approximate performance ranges needed for several classes of space missions.⁽¹⁴⁾ (By permission from *Propulsion Systems for Space Flight* by W. R. Corliss (1960), Figure 10-1, p. 265, McGraw-Hill, New York.)

are also desired. Power required will range from kilowatts for modest missions to perhaps 40 megawatts for manned planetary missions.

We will consider first the problem of nuclear reactors for turboelectric power supplies, and then some promising methods of generating electricity without the usual turboelectric generators.

Nuclear Reactors for Electric Generation

Nuclear reactors for electric rockets have certain inherent problems compared to nuclear heat exchanger rockets. Because the former must use a closed cycle, radiators are required, which are heavy enough to dominate the system at existing feasible reactor temperatures.⁽²⁷⁾ Also, a low thrust system requires a higher velocity increment and thus more energy to accomplish a similar mission. Both of the above reasons result in much more severe radiation damage problems in the electric rocket reactor. Such reactors would most likely utilize liquid metal (e. g., Na) in a Rankine cycle to drive a turboelectric generator. It is conceivable, although perhaps less likely, that a gas coolant would be used in conjunction with a gas turbine, since these require higher specific masses (weight rocket per kW developed).⁽¹⁴⁾ As things now stand, since turbine materials severely limit the usable cycle temperature, there is little point in taking advantage of all-ceramic reactors as opposed to the lower temperature metal or metal-clad reactors.

Of course, a real virtue of higher temperature reactors is better Carnot efficiency. In particular, the difficulty of radiating heat to space from the working fluid leaving the turbine is reduced as the rejection temperature increases, since power radiates as T^4 . Higher cycle temperatures would permit higher rejection temperatures, which could reduce radiator surface and weight. (Incidentally, an important virtue of small radiator surface is lower vulnerability to meteorite penetration.) Weight savings are important because of the low thrust-to-mass ratios of the nuclear-electric rockets. Consequently, high power density reactors are desirable. This in turn requires materials capable of withstanding higher temperature and thermal stresses. By and large, containment materials for liquid metals will probably be refractory metal alloys (e. g., Nb or Ta), although ceramic materials will undoubtedly be used as fuels and moderators.

Work has been going on, chiefly in the classified SNAP projects, to develop a suitable reactor for generation of auxiliary electrical power. These

so far have been small compared to electrical requirements for propulsion, but they point in the proper direction. SNAP will be discussed in a subsequent chapter.

SPUR (Space Power Unit-Reactor), however, is now being developed with joint USAF-AEC interest. This project would develop a fractional megawatt range compact reactor, utilizing alkali metal coolant in a Rankine cycle. The goal is a 300-kW (electrical) unit operable for one year. Reportedly, this could be extended to 1 MW and 2-3 year operating life. Such reactors will not be available until late in the 'sixties.⁽²⁸⁾

It should of course be pointed out that many development problems must be solved for optimum space reactors. The boiling-condensing cycle (Rankine) is more efficient than the all-gas cycle (e.g., Brayton) but offers problems in condensing of vapors because of the absence of gravity in most space missions. Turbine blades must withstand creep and corrosion attack.

Direct Conversion

Because of the large weight of rotating machinery to convert thermal energy (e.g., from a nuclear reactor) to electricity, it is intriguing to attempt to generate electricity directly from thermal energy. Among the devices considered so far are the thermoelectric generators, the thermionic converter, and the plasma generator. The thermoelectric devices do not appear capable of generating currents required practically, and so we will not consider them further.

The thermionic converter consists essentially of a hot electron emitter such as the carbide of uranium or zirconium, and a cooler anode collector. The thermionic system may be compact, and its conversion efficiency good, up to 17%.⁽²⁹⁾ The principal problem is to increase current density by reducing the space charge, which limits current flow. This may be done by reducing the electrode spacing to a few mils, which of course makes large scale fabrication difficult. Also, maintaining this small clearance at high operating temperatures for long times is a problem. Another approach is to utilize cesium vapor, which ionizes to a plasma, and reduces both the space charge and the work function of the cathode, i.e., the energy and thus temperature required to remove electrons from the cathode. This combination of hot cathode, cesium plasma, and cold anode is called a plasma diode, or sometimes a plasma thermocouple by analogy to metallic thermocouples.

In nuclear use, the emitter may be a fuel element itself or be heated directly by a fuel element bonded to it, and thus the whole assembly becomes part of a reactor. Because of this, radiation-induced swelling may arise because of gas formation in the fuel elements, which would destroy the close tolerance assembly. Special precautions must be used such as multiple nucleation sites to precipitate extremely fine bubbles instead of permitting accumulation of large bubbles; or atoms in solid solution which reduce the mobility of the fission gas within the lattice. The collector must be cooled sufficiently or it too will emit. Cooling may be accomplished by circulating liquid metal.

An alternate system would be to have the thermionic emitter assembly removed from the reactor, and heated by cycling liquid metal. This avoids the radiation damage problem but reduces the emitter temperature and performance considerably.

The type of materials problems which arise in these systems, other than those mentioned are⁽³⁰⁾:

1. Requirement of excellent bonding of the fuel element-emitter, and collector-electrical insulator-coolant passages, to improve thermal contact. Thus one must match thermal expansions.
2. Liquid metal containment.
3. Prevention of cesium vapor corrosion of the components.
4. Chemical compatibility of the components and minimization of material transport at high temperatures for long times.

A final electrical system is the plasma generator. This device is essentially the reverse of the plasma accelerator. Conceptually, a plasma would be generated by heating a gas in a nuclear reactor and by injecting alkali ions into it. As it passes through a magnetic field, an electric field would be generated. This system has some promise in reducing weight per kilowatt⁽¹⁹⁾ although it is probably too young to assess properly.

D. Heat Transfer Nuclear Rocket (Rover)

We have spent some time reviewing a number of the competitive nuclear systems for space propulsion, some of them only conceptual. It is now appropriate to consider in more detail the most well developed of these systems, the solid-core nuclear heat exchanger. In the U. S., this work is centered around Project Rover, and is sponsored jointly by NASA and the AEC.

The development has been done chiefly by the Los Alamos Laboratory of the University of California, although some early work was also done in parallel by LRL. In this exposition, too, we are hampered by the fact that this project is security classified, so we have access to principles, concepts, and the results of some early work, rather than current technology. In the following discussion the term "HTNR" will be used as a shorthand equivalent of heat transfer nuclear rocket, to avoid confusion or conflict with the actual classified Rover project. Thus, the writer is not speaking directly in behalf of the Rover system.

First, reconsider the requirements of HTNR. We require a container to hold hydrogen, preferably in the liquid state, a heat exchanger-reactor consisting of fissionable fuel in a matrix, reflector and pressure shell around it, and finally an exhaust nozzle.

The previous equation for specific impulse indicated the requirement for high temperature. We cannot expect improvements in materials to produce marked breakthroughs in possible space missions using the HTNR since we still are limited to the chemical bond strengths in the solid materials and since I_{sp} is proportional only to $T^{1/2}$. However, we can expect measurable and entirely practicable improvements within the temperature limitations of solids. Many of these improvements will probably come from systematic improvements in fabrication tolerances, uniformity in fuel distribution, reduction in variability; in short, from better process and product control. Possibly, improvements in graphites to permit better high temperature properties, and coatings to reduce corrosion (see below) will also allow increased reactor temperatures.

We can systematize the probable materials requirements and problems in a HTNR as follows:

Fuel-Matrix (Moderator)

1. High melting point.
2. Low neutron absorption cross section or else consider for fast reactor.
3. Low vapor pressure.
4. Compatibility of fuel with matrix.
5. Good thermal conductivity and thermal stress resistance.

6. Easy fabricability to close tolerances.
7. Good mechanical properties.
8. Creep not excessive at high temperatures.
9. Corrosion resistant to propellant (H_2).
10. Radiation damage not excessive.
11. Fission product release not excessive.

Several Other Materials Problems

Structural materials in reactor

Control rods

Reflector

Nozzle

Liquid hydrogen container

What are the consequences of high core temperatures on materials problems? An immediate effect is the limitation of the fuel-matrix combination. The most economical reactor from a minimum U-235 fuel viewpoint is a moderated reactor. As we considered in the Pluto case, the only reasonable moderator atoms are those up to carbon in mass. Table VII shows the two main ones to be considered, BeO and graphite. Although a less efficient moderator requiring a larger volume reactor, graphite has a marked temperature advantage. Also BeO has a destructive phase transformation at 2050°C , further limiting its usage. Finally, graphite does have higher strengths above 1800°C .

Since high temperature is so important, we could also consider matrix materials of high melting point such as tungsten. (Table VII). Aside from problems of bulk fabrication of tungsten, its thermal absorption cross section and resonance absorption integral for epithermal and higher energy neutrons (see appendix) are so high that one must either use a fast reactor (which is costly in U-235), or choose a nonhomogeneous reactor where the moderating region is separated from the tungsten-fuel region, adding considerably to the difficulties. A fast reactor (one using highly energetic neutrons) has poorer neutron utilization and so the reactor costs more in terms of U-235. However, the absence of moderator may permit a more compact (lighter) reactor, and the weight savings may be attractive enough to offset the U-235 cost. Thus, there is interest in refractory metal-fueled materials for such reactors.⁽³¹⁾

Table VII. Properties of Some Nuclear Rocket Core Materials.*

Property	BeO	Graphite	Tungsten
Melting point	2570°C	3630°C (sublimes)	3380°C
Density, g/cc (20°C)	2.7	1.7	19.1
Tensile or bending strength, psi	13,000** at 1600°C	3000-6000 at 2500°C	5000-9000 at 2500°C
Macroscopic thermal absorption cross section, cm ⁻¹	0.0007	0.00034	1.17
Comments	Can be readily fabricated	Properties anisotropic, easily worked	Difficult to fabricate and expensive in bulk form (plates)

*Ref. Corliss, Ref. 14, p. 157. Modified by the present author.

**Hot pressed, 99.7% dense (Ref. 37).

Other obvious requirements are what we could loosely call chemical or physical compatibility of the fuel, matrix, and environment. The vapor pressure, especially of the fuel phase, must be reasonably low and the fuel and matrix phases must be compatible chemically. These requirements could perhaps be relaxed somewhat by the use of barriers separating the phases, if one can find a suitable barrier. An extremely important problem in this category is the corrosiveness of hydrogen to graphite at high temperatures to form volatile hydrocarbons. Figure 14 shows corrosion rate of graphite by hydrogen at high velocities.⁽³²⁾ The reaction becomes appreciable above 1500°C, and one would probably worry about problems of redeposition within the gas passages, which in turn would increase pressure drop, cause hot spots and engender loss of performance. Clearly at 2500°C uncoated graphite would be lost rapidly (0.1 inch in 5 minutes). Thus a diffusion resistant barrier is needed to coat the hydrogen passages.

One naturally turns to the carbides as coatings compatible with graphite. Carbides also tend to be quite refractory. However, note that since carbon is generally soluble in the carbides at high temperatures, the melting point of the coatings tends to be reduced with time. An example cited by MacMillan⁽³²⁾ is ZrC which has a melting point of 3500°C when pure and only 2850°C when saturated with carbon.

Fig. 14. Typical rapid increase of corrosion rate with temperature of commercial-grade graphite; hydrogen flow rate is 5×10^4 lb/ft²/hr.(32) (By permission from "High Temperature Materials for Rocket Reactors," April 1961 issue of Nucleonics, Fig. 5, p. 88, by D. P. MacMillan.)

Here we are concerned with the use of a coating thick enough to guarantee complete coverage and thick enough to prevent serious corrosion by being an effective diffusion retardant. Yet the coating must not be so thick that we magnify problems of differential thermal expansion between coating and matrix; introduce a large temperature drop in the coating, creating a thermal stress problem therein; and add too much to the mass of the reactor. The details of what one must do depend on optimizing performance based on allowable core temperature and other factors, some of which we have just indicated.

Even more than was true for Pluto, we require high power generation per unit volume of reactor, which in turn means high thermal shock on startup and high steady state thermal stresses. MacMillan⁽³²⁾ points out that the bulk power density for ground take-off must be about 100 MW/ft^3 or 3.5 kW/cm^3 . Compare this number with values below 0.1 kW/cc for stationary thermal power reactors. Then it is obvious that thermal stresses will play a large role. This calls for one or more of the following:

- a) high strength materials,
- b) an appropriate amount of creep to relieve thermal stress,
- c) good design and fabrication procedures to permit high surface to volume ratios in the core, e.g., by thin wall, multihole construction.

Graphite is highly anisotropic, having much higher thermal conductivity in the layer direction (a-axis) than perpendicular to it. One could conceivably use a stacked plate core with flow holes perpendicular to the layers. Thus we may take advantage of the high conductivity along layers.

Aside from steady state thermal stress, thermal shock on startup is another problem. One wants to reach temperature as soon as possible to avoid waste of precious hydrogen during the heat-up. The thermal stress parameter $Sk/E\alpha(1 - \mu)$ is of interest in the temperature region where graphite behaves elastically. (S = strength, k = thermal conductivity, E = Young's modulus, α = thermal expansion coefficient, and μ = Poisson's ratio.)

Consider the mechanical properties of the matrix, especially at high temperature. Rocket reactor life is only minutes and so creep rate and

time for creep rupture are of concern. Figure 15^(32, 33) shows the tensile creep rate of LDC graphite containing 0.25 gram of U per cc. Note that creep increases much faster than linearly with stress.

A valuable characteristic of graphite is that while most materials have lost a major part of their room temperature strength at 2000-2500°C, graphite reaches its peak strength at 2500-2600°C, where it is about twice as strong as its room temperature value. It is about as strong as tungsten, the most refractory metal, at this temperature. Although graphite becomes plastic above 2700°C, it still has residual strength up to almost 3000°C and remains solid up to ~3900°C, where it sublimates (at one atmosphere).

In connection with the problem of radiation damage, although the flux is high ($> 10^{16}$ neutrons/cm²-sec), the lifetime is short, so that integrated dose is small compared to usual reactors. Also, temperatures are so high that radiation damage tends to be annealed out to a large extent. Thus, this fortunately does not appear to be a major problem in the HTNR in short time applications.

The release of fission products to the launching pad, to the atmosphere, to the sea in case of an immersion of a HTNR resulting in a (super-critical) nuclear excursion, and finally release of radioactivity during re-entry are questions which require answers. Tests of the Kiwi (see below) have shown very minor fallout during test runs, so that the contamination hazard does not appear severe.⁽³⁴⁾ Higher temperature reactors may present some problems at the launching pad, however. Upper atmosphere contamination should be entirely trivial compared to the existing burden from weapons tests before the Russian series.⁽³⁴⁾ The remaining problem, that of uncontrolled re-entry of a "hot" reactor, which has operated long enough to be appreciably radioactive, is a problem which must be considered.

Control rod materials are also important in a high power-high temperature reactor. The atoms in these materials must be good neutron absorbers, and elements of B, Cd, Gd, and Eu are of most interest. Of ceramic interest is the use of boron carbide, formed by powder techniques. Other possibilities are the oxides of Gd and Eu which may be clad in a refractory metal and swaged. A control rod and its channel must be very reliable, since jamming might result in total destruction of the core and its environment.

Fig. 15. Graphite strength increases with temperature to exceed strength of most refractory metals at high temperatures.⁽³²⁾ (By permission from "High Temperature Materials for Rocket Reactors," April 1961 issue of Nucleonics, Fig. 1, p. 86, by D. P. MacMillan.)

Several other materials-related problems which exist in the HTNR are:

- (1) High temperature structural materials within or adjacent to the reactor for support of the core.
- (2) Metal embrittlement due to the flow of liquid hydrogen required to cool the exhaust nozzle and regeneratively heat (boil) the hydrogen.
- (3) Keeping liquid hydrogen in storage tanks relatively free from meteorite punctures without adding excessive weight to the rocket.

Our failure to discuss these problems in detail does not imply that they are minor problems, but that they are beyond the scope of this presentation.

Environmental testing is a major and difficult problem in high power nuclear reactors. To test temperature effects, electric heating may be used. However, no test reactors such as MTR (Materials Testing Reactor) or ETR (Engineering Test Reactor) can reach the high dose typical of HTNR reactors. Ultimately, one must carry out materials tests in actual prototype reactors. These tests are costly and require remote disassembly and hot cell facilities. Also interpretation is not simple since a number of variables may be present, unlike simple laboratory tests. However, fuel element statistics are large, and many other characteristics of the system (neutronics, engineering design, control) may be simultaneously determined.

As a final word about HTNR, the following paraphrased argument by MacMillan⁽³²⁾ is worthwhile for illustrating the type of problems in materials selection for a rocket reactor.

Niobium carbide has a melting point of 3500°C, which appears attractive for fuel element use. Assume that the fuel element must be 1/8 in. thick and contains 20% UC. UC has a melting point of only 2350°C, and the solid solution UC-NbC of this composition has an estimated melting point of 3230°C. At a reasonable HTNR power density of 100 MW/ft³, the ΔT from surface to centerline may be estimated ($\Delta T = Q t^2 / 8k$). For a k of 8 Btu/hr-ft-°F (or 0.138 watts/cm-°C), we calculate that $\Delta T \approx 317^\circ\text{C}$, and so the surface temperature is 2915°C. Allowing 10% in temperature for variability in fuel loading and heat generation, this temperature is reduced to 2625°C. Then the gas temperature becomes ~2500°C, assuming that it is not practicable to bring the gas temperature closer than 100-150°C to the fuel element surface temperature. Thus we are forced to a gas temperature 1000°C below the

melting point of the matrix material. This is characteristic of the practical application of materials in reactors.

It is of interest that the Rover concept has been tested in a number of ground tests in Nevada,⁽³⁵⁾ the Kiwi A series (3 reactors), and the later generation Kiwi B series. In the latter, the use of liquid hydrogen as a propellant was tried, with conditional success. The government considered these feasibility studies to have been promising enough to warrant subsidy of larger programs. The NERVA system (Nuclear Engine for Rocket Vehicle Application) is now being developed by Aerojet and Westinghouse. Also RIFT (Reactor In-Flight Test) to be carried out by a number of aircraft contractors will determine the rocket capabilities under actual flight conditions.

E. Closing

Having reviewed a number of proposed propulsion methods, it seems worthwhile to summarize the range of application of the more promising nuclear propulsion schemes (Table VIII).⁽²⁷⁾ It is clear that none of them is ideal in all respects, as we mentioned earlier. Each of the two contenders which are closest to realization has its own particular range of application, the nuclear heat exchanger primarily for lunar exploration and some nearby planetary missions, and the nuclear-electric for longer-range interplanetary missions.

It is perhaps redundant to point out once again that materials will play an important, perhaps a limiting role in these ventures.

Acknowledgment

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Table VIII. Comparison of the more promising nuclear propulsion schemes.⁽²⁷⁾

	Nuclear Heat Exchanger	Electric	Gaseous	Nuclear Explosive
Specific impulse, sec	~800	> 1000	< 3000	?
Thrust/weight	~30	$\sim 10^{-4}$	?	1
Structure temperature	~2500°C	Metals, <2500°F	?	?
Hydrogen storage problem?	Yes	No	Yes	No
Number of fissions required	Low	Heat exchanger X50*	Low	Probably high
Radioactivity in engine	Low	High	Very low	Very low
Required operating life	1/2 Hour	10^4 Hours	?	Short
Probable application	Cis-lunar	Special cis-lunar, interplanetary	?	?
Probable availability	1968-1970	Special-1967, interplantetary-1975	?	?

* There are several reasons why a larger number of fissions are required for a nuclear-electric rocket than for a heat exchanger rocket: 1. The higher exhaust velocity means a higher kinetic energy for a particular change in momentum. 2. The low acceleration of low thrust systems requires a higher velocity increment for the same mission.⁽²⁷⁾ 3. The lower temperature nuclear reactors for electric propulsion have lower thermodynamic efficiency.

References

- ¹R. L. Murray, Introduction to Nuclear Engineering (Prentice Hall, New York, 1954).
- ²S. Glasstone, Principles of Nuclear Reactor Engineering (D. Van Nostrand, New York, 1955).
- ³C. R. McCullough, U. S. Government Report #Mon N-383 (Sept. 1947).
- ⁴H. R. Hoekstra in Uranium Dioxide: Properties and Nuclear Applications, ed. by J. Belle (USAEC, Washington, 1961), p. 299 ff.
- ⁵W. B. Wilson, C. A. Alexander, and A. F. Gerds, "Stabilization of UO_2 ," J. Inorg. Nucl. Chem. 20, 242 (1961).
- ⁶G. M. B. Dodson, Proc. Roy. Soc. A236, 187 (1956).
- ⁷G. R. Van Houten, J. F. White, and J. M. Botje, "Formation of Alumina Coatings on BeO by a Vapor Deposition Process," GE Report Apex-699 (June 1961).
- ⁸T. C. Merkle, "The Nuclear Ramjet Propulsion System," UCRL-5625, June 1959 or p. 34 in Advanced Propulsion Techniques, ed. by S. S. Penner, (Pergamon Press, New York, 1961).
- ⁹T. C. Merkle, "The Capabilities of Nuclear Rockets" – Paper delivered to American Nuclear Society (November 20, 1957).
- ¹⁰R. E. Schreiber, "The Potentialities of Nuclear Propulsion for Space Flight," AECU-4196 (May 1, 1959).
- ¹¹G. P. Sutton, Rocket Propulsion Elements (J. Wiley and Sons, New York, 1956).
- ¹²F. E. Rom and R. G. Ragsdale, Proceedings of the NASA – University Conference on the Science and Technology of Space Exploration, Chicago, Illinois (Nov. 1-3, 1962), Vol. 2, p. 64.
- ¹³H. S. Ribner, Canadian Aeronautical Journal (October 1959), p. 340-344.
- ¹⁴W. R. Corliss, Propulsion Systems for Space Flight(McGraw-Hill, New York, 1960).
- ¹⁵R. H. Fox, "A Study of the Nuclear Gaseous Reactor Rocket," UCRL-4996 (October 13, 1957).
- ¹⁶M. L. Rosenzweig, W. S. Llewellyn, and J. L. Kerrebrock, ARS Journal 31, 873 (1961).
- ¹⁷J. W. Eerkens, Nucleonics 19, 100 (1961).
- ¹⁸E. E. Callaghan, Proceedings of the NASA – University Conference on the Science and Technology of Space Exploration, Chicago, Illinois (Nov. 1-3, 1962), Vol. 2, p. 183.
- ¹⁹E. Stuhlinger, "Electric Propulsion," Astronautics, 7, 60 (1962).
- ²⁰W. H. Bostick, Proceedings of IX International Astronautical Congress, Amsterdam (1958), Vol II, p. 794.

References (Continued)

- ²¹G. R. Brewer, M. R. Currie, and R. C. Knechtli, Proceedings IRE 49, 1789 (1961).
- ²²S. T. Demetriades, Astronautics 7, 21 (March 1962) and 7, 40 (April 1962).
- ²³M. C. Gourdine, ARS Journal 31, 1670 (1961).
- ²⁴E. Stuhlinger and E. Urban, Astronautics 6, 38 (Dec. 1961).
- ²⁵J. H. Childs and R. J. Cybulski, "SERT and Early Electric Propulsion Systems," Astronautics and Aerospace Eng. 1, 112 (1963).
- ²⁶L. R. Shepherd, "A Survey of Possible Propulsion Systems," Physics and Medicine of the Atmosphere and Space, ed. by O. O. Benson and H. Strughold (John Wiley and Sons, New York, 1959).
- ²⁷R. Fox, "Nuclear Propulsion: Trends and Problems," UCRL-6736 (Jan. 1962).
- ²⁸Anon., Space/Aeronautics 37, 74 (1962).
- ²⁹A. E. Potter, Jr. and D. T. Bernatowicz, Proceedings of the NASA - University Conference on the Science and Technology of Space Exploration, Chicago, Illinois (Nov. 1-3, 1962), Vol. 2, p. 148.
- ³⁰S. C. Carniglia and R. L. McKisson, Transactions of Amer. Nucl. Soc., "Nuclear Materials for Space Application," National Topical Meeting, Cincinnati, Ohio (April 17-19, 1963).
- ³¹F. G. McGuire, Missiles and Rockets, p. 16 (Apr. 8, 1963).
- ³²D. P. MacMillan, "High Temperature Materials for Rocket Reactors," Nucleonics 19, 85 (1961).
- ³³P. Wagner et al., J. Applied Physics 30, 152 (1959).
- ³⁴H. R. Schmidt, "The Nuclear Rocket Engine and Flight Program," IRE Transactions on Nuclear Science - International Symposium on Aerospace Nuclear Propulsion, Las Vegas, Nevada (October 23-27, 1961).
- ³⁵R. E. Schreiber, Nucleonics 19, 77 (1961).
- ³⁶H. Etherington (Editor) Nuclear Engineering Handbook (McGraw-Hill, New York, 1958).
- ³⁷S. C. Carniglia, private communication.

Additional Bibliography

R. W. Bussard and R. D. DeLauer, Nuclear Rocket Propulsion (McGraw-Hill, New York, 1958).

R. H. Fox, "Realities of Low-Thrust versus High-Thrust Nuclear Propulsion," UCRL-6428 (April 24, 1961).

D. P. MacMillan, "Materials for Nuclear Rockets," Metals Engineering Quarterly, p. 5 (August 1962).

F. E. Rom, A. E. Lietzke, and P. G. Johnson, "Nuclear Rockets for Unmanned Missions," Nucleonics 20, 53 (Nov. 1962).

D. W. Gates, "Materials for Plasma Propulsion," Metals Engineering Quarterly, p. 17 (August 1962).

H. Todd et al., "Materials for Ion Propulsion," Ibid., p. 9.

M. J. Zucrow, "Space Propulsion Engines - Their Characteristics and Problems," Amer. Scientist 50, 409 (1962).

A. M. Rothrock, "Aircraft and Spacecraft Propulsion," Canadian Aeronautical Journal, p. 172 (May 1959).

R. Fox, "Nuclear Propulsion: Trends and Problems," UCRL-6736 (1962).

S. S. Penner, Editor, Advanced Propulsion Techniques (Pergamon Press, New York, 1961).

J. Grey, Astronautics 4, 23 (October 1959).

Anon., Space/Aeronautics "Gas Core Reactors," 38, 54 (Sept. 1962).

R. V. Meghreblian, Nucleonics 19, 95 (April 1961).

R. V. Meghreblian, ARS Journal 32, 13 (1962).

W. E. Moeckel, "Propulsion Methods in Astronautics," in Advances in Astronautical Sciences, Vol. II (Proceedings of the First International Congress in the Aeronautical Sciences, Madrid, September 8-13, 1958), p. 1078. Space/Aeronautics 37, April 1962 (whole issue).

K. F. Gantz, Ed., Nuclear Flight. (Duell Sloan and Pearce, New York, 1960). Astronautics and Aerospace Engineering, "Space Electrical Power," 1, May 1963 (whole issue).

Astronautics, "Nuclear Rocket Program," 7, Dec. 1962 (whole issue).

APPENDIX

The following brief and somewhat oversimplified account of a nuclear reactor is presented so that discussion involving reactors in this and subsequent chapters will be clearer. The reader is urged to consult more complete works on nuclear reactor principles.^(1,2)

"Fuel" consists of fissionable atoms, generally U-235. Uranium containing higher concentrations of U-235 than occur in nature (0.7%) is "enriched." In the fissioning process, a U-235 atom captures a neutron, and splits into smaller atoms (fission products). More neutrons are produced in this process, as well as various rays and particles (α , β , γ emission). The energy is released in the form of these energetic particles and rays as well as in the kinetic energy of the neutrons and fission products. In small masses of fuel, the surface-to-volume ratio is so high that too many neutrons escape from the surface to keep the fissioning process going. However, if the mass is made sufficiently large, enough neutrons to continue the process will be captured by uranium atoms before they escape, in which case we have a "critical" mass.

A moderator consists of atoms of low mass which slow down neutrons, since they more readily exchange energy with the neutrons upon collision than do larger atoms, which receive "blows" by the neutrons and remain relatively unmoved. A helpful analogy is the collision of a fast moving billiard ball with a stationary one of similar size. One may deduce that the moving ball yields a major part of its velocity (and hence energy) to the stationary one by considerations of momentum conservation. On the other hand, if a fast billiard ball collides with a much heavier body, e. g., a stone wall, it rebounds with almost all of its speed and energy unchanged.

It happens also that U-235 atoms will more readily capture slower neutrons than fast ones. Thus the use of a moderator reduces the mass of U-235 required for criticality: the more effective the moderator, the smaller the required U-235 mass.

Neutrons produced by the fissioning process may also be captured by atoms other than U-235. This is a wasteful process since the neutrons are not then available to continue the chain reaction. A measure of this is the "absorption cross section" of the atoms. This cross section depends on the neutron velocity, and is normally expressed for neutrons having velocities

typical of room temperature up to a few thousand degrees as "thermal absorption cross section." A measure of neutron absorption at much faster velocities is measured by the "resonance" absorption integrated cross section. Obviously, we wish to have a moderator of low absorption cross section. In fact, one measure of a moderator is the "slowing down power," or ability to reduce the neutron velocity during collision, and another is the "moderating ratio" or ratio of slowing down power to absorption cross section.

"Thermal" reactors are the usual ones which utilize moderators to slow neutrons down to velocities corresponding to reactor temperatures. "Fast" reactors utilize little or no moderator, and thus must employ more fuel for a critical mass. However, since absorption by extraneous atoms occurs chiefly in the thermal or slower neutron region, fast reactors may use materials having fairly high absorption cross sections without severe penalty. "Epithermal" reactors are less completely moderated than thermal ones, and so the neutron velocities are somewhat faster than thermal.

For control purposes, reactors must have control or poison rods, which absorb neutrons markedly. Thus by withdrawing and inserting these rods, the reactor may be rendered more and less critical, respectively, or completely shut down.

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