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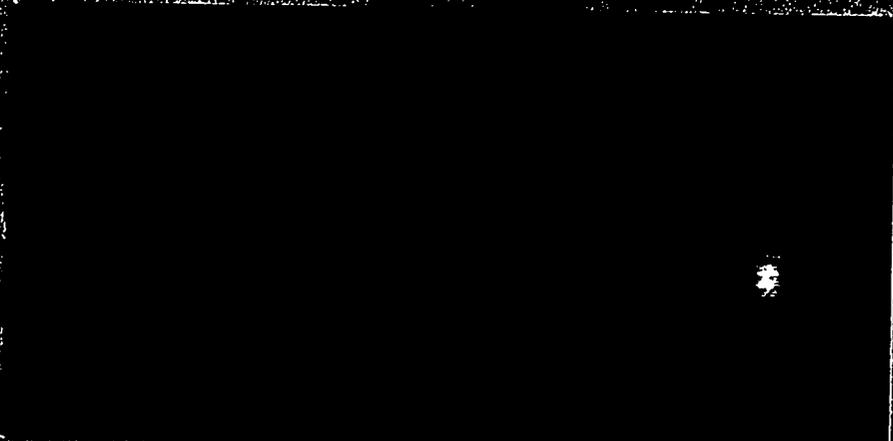
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NONWEAPONS ACTIVITIES OF

LOS ALAMOS SCIENTIFIC LABORATORY

Part II - Nuclear Rocket Propulsion

by

Samuel Glasstone

NUCLEAR PROPULSION

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NUCLEAR PROPULSION

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FOREWORD

In the early part of 1954, there were issued two reports (LA-1632 and 1633) summarizing the weapons activities at Los Alamos Scientific Laboratory. These reports were intended primarily for the information of new staff members of the Laboratory and for interested representatives of the Armed Forces and the Atomic Energy Commission. During the past two or three years, the work of the Laboratory has greatly expanded into applications of nuclear energy which are significant for national defense and security, but are not directly connected with weapons development. It was felt, therefore, that a description of the nonweapons activities of the Laboratory would serve a useful purpose at this time.

For classification reasons, it has been necessary to issue the report in three parts; the first is concerned with controlled thermonuclear reactions, the second with nuclear propulsion, and the third with power reactor experiments. As with the reports on weapons activities, the present reports are not intended to discuss the various topics in great detail, but rather to describe the underlying principles. Their basic purpose is to present a general background of the subject and to indicate the lines along which work is in progress in the Laboratory. It is in hope that the material contained in them will prove useful to new staff members and to others concerned with the activities of the Laboratory that these reports have been prepared.

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Norris E. Bradbury
Director

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ACKNOWLEDGMENT

I wish to take this occasion to express my sincere thanks to the many members of the Laboratory who helped, in one way or another, in the preparation of this report. Their generous and wholehearted cooperation not only greatly simplified my task, but made it a pleasure and a privilege. I would also like to thank the Director of the Laboratory and his staff for giving me the unique opportunity to write this series of reports, and for providing the facilities which made the work possible.

Samuel Glasstone

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Introduction

The nuclear propulsion activities of the Laboratory are mainly concerned with the design and development of nuclear fission reactors and associated equipment for the propulsion of guided missiles. Although the rocket motor is not necessarily the best method for utilizing fission energy in missile propulsion, it appears to be the form which is most capable of realization within a reasonable time. Consequently, the work of the Laboratory is being restricted, for the present at least, to nuclear rockets, although the results will probably be adaptable to other modes of propulsion if required.

Basically, a nuclear rocket missile would have the form shown in Fig. 1.

NOZZLE   REACTOR   PUMP   PROPELLANT   DEADLOAD
(GUIDANCE & WARHEAD)
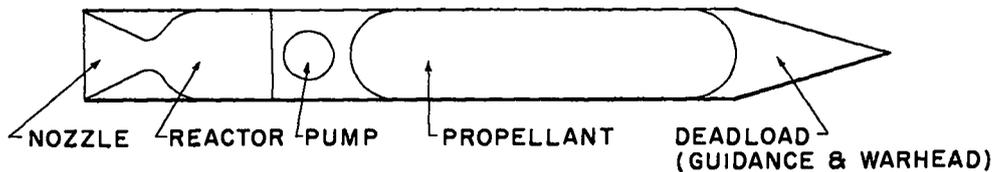
Fig. 1

The propellant, consisting of a readily volatile liquid, e.g., a normally liquid hydrocarbon of low boiling point, or a liquefied gas, e.g., liquid hydrogen, ammonia, or methane, is pumped through the reactor where it is vaporized and its temperature raised by taking up the heat from uranium-235 undergoing fission. The hot gases are then expelled through a convergent-divergent nozzle. Upon expulsion, the gases expand almost adiabatically, and so produce the thrust required to drive the vehicle. Although simple in principle, it will be seen that the design of a nuclear rocket motor involves many difficult problems.

Nuclear and Chemical Rockets

In a conventional (or chemically propelled) rocket, the thrust is supplied by expulsion through the nozzle of hot gases produced in a chemical reaction, such as the combustion of a liquid hydrocarbon fuel with oxygen. Since no special device, other than a burner, is required

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to permit the reaction to proceed once it has started, there is an appreciable saving in weight over a nuclear rocket, as far as the motor is concerned. For a power of 2,750 megawatts, for example, the reactor weight would probably be some 5,500 pounds, whereas a chemical combustion chamber of equal power would weigh a few hundred pounds at most. It is necessary, therefore, to examine the conditions under which a nuclear rocket would have advantages over a conventional rocket.

Consider a rocket, no matter what the source of its propulsive power, moving in force-free space, i.e., without a gravitational field (Fig. 2). Let M be the mass of the rocket and v its velocity at any instant.
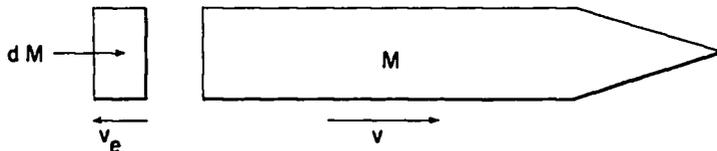
Fig. 2

Suppose the mass dM of propellant gas is expelled through the nozzle during a short time interval dt, and let v_e be the exhaust velocity of this gas. The force, i.e., the rate of change of momentum, of the expelled gas relative to the rocket must be equal to the force on the rocket, so that

$$-v_e \frac{dM}{dt} = M \frac{dv}{dt},$$

if the exhaust velocity is assumed to be constant. Upon integration this gives

$$v = v_e \ln \frac{M_o}{M},$$ (1)

where M_o is the initial mass of the rocket vehicle, and M and v are the mass and velocity, respectively, at any subsequent time. At burn-out time, i.e., when the propellant is exhausted, the mass is M_b and the velocity v_b, so that equation (1) becomes

$$v_b = v_e \ln \frac{M_o}{M_b} . \tag{2}$$

The value of the burn-out velocity is important because it essentially determines the range of the rocket-propelled vehicle. It should be noted that the difference between M_o and M_b is equal to the total mass of propellant carried by the vehicle.

Although this result applies strictly to a force-free field, the effect of gravitation does not significantly change the conclusions to be drawn from equation (2). It may, therefore, be used as a basis for comparing rockets of different types. Due to the weight of the reactor, M_b for a nuclear rocket will usually be greater than for a chemical rocket; hence, in general,

$$M_b \text{ (nuclear)} > M_b \text{ (chemical)}.$$

This means that if v_e and M_o are the same in both cases, the burn-out velocity (and hence the range) will be less for the nuclear device than for the chemical rocket. There are two possible ways in which the weight disadvantage of the nuclear system may be overcome: (1) by increasing M_o, the initial total mass, in proportion to the increase in M_b by the use of additional propellant, and (2) by increasing the exhaust gas velocity, v_e.

An increase in the amount of propellant, which will be used less efficiently, is not a satisfactory solution to the problem. Consequently, attention may be given to the possibility of increasing the exhaust gas velocity. For a fixed nozzle geometry (or efficiency),

$$v_e \propto \sqrt{\frac{T}{\overline{MW}}} ,$$

where T is the absolute temperature of the hot gas prior to expansion at the nozzle, and $\overline{MW}$ is the mean molecular weight. The ratio of the specific heats of the gas at constant pressure and at constant volume also determines the exhaust gas temperature, but it may be taken to be constant.

Because of limitations in the properties of materials at high temperatures, the gas temperatures will not be very different in nuclear and chemical rockets. In fact, the advantage, if any, will lie with the latter. However, the important point is that nuclear rockets can operate with propellants of low molecular weight, e.g., as low as 2.0 for hydrogen, although substances of somewhat higher molecular weight may prove to be more practical for some missions. In any event, average gas molecular weights in the range from 2 to 10 can easily be obtained in nuclear rockets, whereas values from 18 to 60 are more common for chemical rockets. Thus, the exhaust gas velocities in a nuclear rocket can be 2 or 3 times as great

as for a conventional rocket operating at the same temperature. Such a significant possible increase in v_e for a nuclear rocket can more than overcome the increase in the value of M_b.

For ballistic missiles of low or intermediate range, e.g., about 1,500 miles for the IRBM, the burn-out velocity does not have to be very large, namely, 18,000 ft/sec. In these circumstances, there is probably little to be gained from the use of a nuclear rocket in place of a chemically propelled vehicle, especially in view of the high cost of the fissionable material (uranium-235). But for an intercontinental ballistic missile (ICBM), such as the Atlas with a design range of 5,500 nautical miles (N. M.) and a nose cone capable of carrying a warhead weighing 1,500 pounds (or more), the situation may be very different.

Propulsive Efficiency

Various definitions have been proposed for the propulsive efficiency of a rocket, but basically it may be regarded as the fraction of the total energy expended that is converted into useful (kinetic) energy of the final load. This fraction varies with time during flight, but its value is a maximum when the vehicle velocity, v, is equal to the exhaust velocity, v_e, i.e., when $v/v_e = 1$. At both lower and higher velocities, a smaller proportion of the total energy is available to the final load. The reason for this may be understood from the following argument. Since the exhaust gases are traveling through space with the rocket, their velocity relative to the earth is always $v_e - v$. The rate at which kinetic energy is supplied to the exhaust gases is then proportional to the value of $(v_e-v)^2$, at any instant. It follows, therefore, that only when $v = v_e$, i.e., when $v/v_e = 1$, no kinetic energy goes to the exhaust gases, and the energy of the propellant is all utilized in the propulsion of the vehicle. At all other values of v/v_e, a certain proportion of the kinetic energy is wasted in the exhaust gases, so that the propulsive efficiency is less than when the vehicle and exhaust gas velocities are equal.

For the accomplishment of the Atlas mission, a burn-out velocity of 23,500 ft/sec is necessary. Although an exhaust gas velocity of some 13,000 ft/sec is theoretically possible in a chemical rocket using liquid hydrogen and oxygen as fuel, a reasonable value for v in a chemical rocket is probably 7,000 to 9,000 ft/sec. For much of the powered portion of its range, therefore, a chemically-propelled Atlas would be operating with a relatively low propulsive efficiency. However, for rockets of shorter range, requiring a burn-out velocity in the region of 10,000 ft/sec, chemical propulsion should be quite efficient.

In a nuclear rocket, on the other hand, the situation is to a large extent reversed. Reasonable values for the exhaust velocity are 27,000 ft/sec with hydrogen as propellant and 13,500 ft/sec with ammonia. In these cases, and especially in the former, least efficient use of the propellant is made in the early stages, i.e., at low vehicle velocities. But at high

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velocities, such as must be attained in an ICBM as burn-out is approached, the propulsive efficiency is high in a nuclear rocket.

The optimum over-all efficiency of a rocket system will depend upon the parameter which it is desired to optimize, e.g., size, weight, cost, state of readiness, etc. Consequently, the particular condition for optimum over-all efficiency may be only indirectly related to that of maximum propulsive efficiency. However, from what has been stated above, it would appear, in general, that there might be some benefit to be gained from the use of a two-stage system for a long-range rocket; the first stage would be chemically propelled, whereas the second stage would have a nuclear propellant. A further advantage of such a two-stage system is that contamination of the launching site by radioactive exhaust gases would be avoided. But, apart from the contamination problem, if a one-stage rocket is to be used, nuclear propulsion, with the possibilities of high exhaust gas velocities, should be preferable to chemical rockets for long-range missions.

General Reactor Considerations

Since essentially nothing is known of the practical aspects of nuclear rocket design, the Laboratory is not yet concerned with the exact manner in which such a nuclear rocket motor might be used to propel a missile to best advantage. The immediate objective is to design and build a reactor of a type that could be used for rocket propulsion, although not necessarily capable of achieving the Atlas mission in one stage. This reactor will then be run on a stationary test stand, at the Nevada Test Site, under conditions that simulate, as closely as possible, actual operation in a rocket. Extensive instrumentation should provide a great variety of data that, it is hoped, will permit the design of a nuclear rocket motor of the same (or greater) power for use in an actual missile.

The first, basic stage in this program was to choose a specific reactor design and propellant that, in combination, appeared to offer the greatest prospects for success within a reasonable time, and that would provide information for future developments. The various factors which had to be taken into consideration before a decision could be reached are outlined below.

A nuclear reactor is essentially a furnace in which heat is liberated by the continuous fission of uranium-235 (or other fissionable material) that acts as the "fuel". As a source of energy, the reactor is unique in the sense that there is no limit, in principle, to its power (or rate of heat production). However, there is a maximum permissible operating temperature, set by the physical and mechanical properties of the materials present in the reactor. The actual operating power is then dependent upon the rate of heat removal by an appropriate coolant.

In a nuclear rocket, the propellant is the heat-removing medium, and the rate at which it takes up heat from the reactor is related to its mass flow rate, its specific heat, and the initial and final temperatures, i.e., entering and leaving the reactor, respectively. Suppose these temperatures to be more or less fixed, the former by the nature of the propellant and the latter by the maximum permissible reactor temperature. Then it is the specific heat and flow rate of the propellant that largely determine the power developed by the nuclear rocket motor.

In practice, the mass flow rate of the propellant in a reactor, as in other flow systems, is limited by the permissible pressure drop, and this is related to certain physical properties of the propellant and to the dimensions of the flow channels in the reactor. The ability of a specified flow rate to maintain a desired maximum temperature in the reactor will depend upon the heat-transfer properties of the system, and this is also bound up with the nature of the propellant as well as with the heat-transfer areas of the reactor. Thus, various reactor parameters, such as the dimensions of heat-transfer surfaces and flow channels, enter into a determination of the actual operating power. Increasing the size of the reactor would permit certain design changes that decrease the pressure drop and increase the heat-transfer rate for a given propellant flow rate.

Even if the increase in reactor size could be tolerated in a missile, the total increase in weight might prove a serious handicap. If the total weight of a nuclear rocket motor is plotted against the specific power expressed in megawatts per pound (Mw/lb) of reactor weight, the result, for a given payload (or deadload), is of the form shown in Fig. 3.

SPECIFIC POWER (Mw/LB)
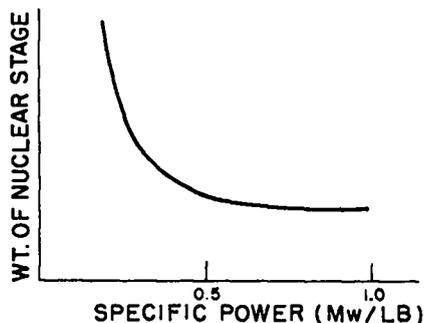
Fig. 3

The actual weight of the nuclear stage depends on the value of the deadload and the range, but the general character of the curve is always the same. The weight of the nuclear stage decreases rapidly at first with increasing specific power of the reactor, but beyond about

0.5 Mw/lb there is only a relatively small further decrease in the total weight.

It follows, therefore, that although there is little (or no) incentive to achieve higher specific powers than 0.5 Mw/lb for a nuclear rocket reactor, it is very desirable that the actual value should not be appreciably smaller. Thus, there is a design limit set upon the weight (and size) of a rocket reactor to operate at a given power. It may be remarked that this design limitation is extremely severe in comparison with the specifications for reactors intended for central-station use or even for the propulsion of ships or aircraft.

From the foregoing remarks, it is apparent that there are several factors to be taken into account in the design of a reactor for use in a rocket vehicle. In addition to those already mentioned, there are also the neutron characteristics of the nuclear fission system to be superimposed on to the other requirements. The minimum mass of fuel, e.g., uranium-235, needed to sustain the fission chain reaction and to permit continued operation of the reactor, i.e., the critical mass, is related to the size and also to the nature and quantity of other materials present. It is evident, therefore, that the design of a nuclear rocket motor involves consideration of a number of variables, which must be balanced properly, one against the other.

Selection of Rocket Reactor Design

In selecting the detailed design characteristics of the first full-scale nuclear rocket test reactor to be built in the Laboratory, many factors had to be considered. Some of these will be discussed in turn below, although in practice several had to be examined concurrently because of their close interrelationship. However, it should be mentioned that, although the basic decisions have been made, there are certain difficult problems still awaiting solution. A few of these will be referred to later in this report.

Neutron Energy

The energy (or speed) of the neutrons causing the majority of fissions in a reactor is related to the composition of the system. If the proportion of extraneous materials is small and they consist entirely (or mainly) of elements of large (or moderately large) mass number, there will be little chance for the high-energy (fast) neutrons produced in fission to be slowed down. Most of the fissions in the reactor will then be caused by fast neutrons. Such a "fast reactor", as it is called, has the advantage of small size and total mass. However, the critical mass of fissionable material required for a fast reactor is relatively large, and the small over-all size presents difficulties in connection with heat transfer to the propellant. In addition, the choice of a fuel material for a fast reactor operating at high temperatures would not be a simple matter.

By including a moderator, i.e., a substance of low mass number in the reactor, the

neutrons will be slowed down, so that more fissions are due to neutrons having less energy than the fast neutrons. With sufficient moderator, the great majority of the fissions will be caused by thermal neutrons, i.e., neutrons having the same kinetic energy as the molecules (or atoms) of their environment at the existing temperature. Such a "thermal reactor" has the advantage of low critical mass, reasonable size, and flexibility of design.

Because of the necessity for operating at high temperatures, graphite is a suitable moderator material for a nuclear rocket reactor. But a thermal reactor with this moderator would be larger, and require more graphite, than is desirable for its intended purpose. Furthermore, the large change in fission (and other) cross sections for neutrons in the thermal region, accompanying the considerable temperature changes that will occur in the reactor materials, would introduce serious control problems.

As a compromise, for nuclear rocket application, an "intermediate reactor" has been chosen. In a reactor of this type the majority of the fissions are produced by neutrons which have been partially slowed down, so that their energies are in the intermediate (or epithermal) range between fast and thermal. Such a reactor has a reasonable critical mass and over-all size, it can be designed for high-temperature operation, with graphite as moderator, and the neutron cross sections are not greatly affected by temperature changes.

Reactor Shape

A nuclear reactor may have any desired shape, but if other circumstances, e.g., composition of materials, are equal, the critical mass of fissionable material will be least for a sphere.* For rocket propulsion a spherical reactor has a number of practical drawbacks. The natural shape for use in a guided missile is obviously a cylinder, and since this happens to be advantageous from the heat-transfer standpoint and has a critical mass not much greater than for·a sphere, a cylindrical reactor was chosen.

Reactor Power and Weight

The operating power of the test rocket reactor will be between 1,500 and 3,000 Mw. The lower value is considered to be the smallest power output compatible with the performance of useful rocket-vehicle propulsion functions. Alone, a 1,500 Mw nuclear rocket, with ammonia as propellant, could carry a 5,000 lb deadload a net horizontal distance of 1,800 miles, while in conjunction with chemical boosters it should be capable of fulfilling the Atlas mission. On the other hand, the upper limit of the power given above will permit the design of a nuclear rocket having the Atlas range without boosting.

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*This is strictly true either for bare reactors, without reflector, or for those having essentially infinite reflectors.
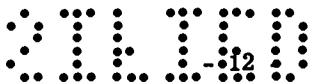
It has been found that, for several reasons, it is easier to design a rocket reactor of large power than of small power. On the other hand, the problems and cost of testing increase with the reactor power. If a specific power of 0.5 Mw/lb mentioned earlier, is desired, it appears that a total rated power of about 2,500 Mw is an approximately lower limit. Provisionally, therefore, a design power of about 2,700 Mw has been chosen for the first test reactor, weighing some 5,500 lb.

Fuel Elements

In order to achieve high exhaust gas temperatures (and velocities), it is desirable that the maximum temperature in the reactor be as high as constructional and fuel materials permit. Since uranium metal has a moderately low melting point ($1133^\circ C$ or $2070^\circ F$), it is necessary to use the fuel in the form of a compound, such as uranium carbide (UC_2) or uranium dioxide (UO_2). Because these compounds have poor mechanical properties and are poor conductors of heat, it is necessary to combine them with other materials in the fabrication of solid "fuel elements". The spaces between or within the fuel elements, which may be flat plates, cylinders, disks, tubes, etc., then provide the flow channels for the propellant (coolant), and their external surfaces constitute the heat-transfer areas. The heat of fission is liberated within the fuel elements and is transferred to the propellant passing over their surfaces.

Among the materials that might be used for fuel elements, two appear to be most promising for high-temperature applications; these are graphite and tungsten. Various carbides, nitrides, oxides, and other ceramic compounds of high melting point have been considered, but they are either too brittle or have too low a thermal conductivity (or both) to be of real value. In any event, their high-temperature properties are too uncertain and too variable for these substances to merit serious consideration. Graphite and tungsten, on the other hand, have adequate strength at high temperatures and are relatively good conductors of heat. The temperature drop from the interior to the exterior (heat transfer) surface of a fuel element is inversely related to the thermal conductivity of the material. With a good heat conductor there will, consequently, be less thermal stress within the fuel element and a greater over-all heat-transfer rate, for a given interior temperature.

It is known that the strength of graphite increases up to about $2800^\circ C$ ($5,000^\circ F$), but falls off very sharply at higher temperatures. The limited available data indicate that the inclusion of up to about 0.5 gram of uranium (as UC_2) per cm^3 of graphite does not affect the strength at $5,000^\circ F$. Below this temperature the strength of the uranium-loaded graphite is less dependent upon temperature, although above $5,000^\circ F$ the marked decrease in strength is observed, just as for graphite alone. Consequently, the maximum permissible operating temperature in the interior of a graphite-uranium fuel element would appear to be about $5,000^\circ F$.

Incidentally, at $5,000^{\circ}$F graphite is a moderately good heat conductor, with a thermal conductivity of 6 to 12 Btu/(hr)(ft^2)($^{\circ}$F/ft). The effect of loading with uranium is not definitely known. If, as is at present contemplated, the fuel is a mixture of uranium and zirconium carbides, which will be described later, the thermal conductivity may be higher than that of graphite alone.

Although the strength of tungsten at $5,000^{\circ}$F is uncertain, the metal is believed to be somewhat stronger than graphite at this temperature. On the other hand, powder compacts of UO_2 and tungsten metal, which is probably the form in which the fuel elements would have to be fabricated, are relatively weak. But the mechanical properties could undoubtedly be improved by swaging, working, or other treatment. The thermal conductivity of tungsten metal is appreciably greater than that of graphite at $5,000^{\circ}$F, but the addition of UO_2 would probably decrease the value.

Two other aspects of the fuel-element material remain to be considered, namely, (1) corrosion (or erosion) by the propellant and (2) the capture of neutrons. It will be seen shortly that corrosion (rather than erosion) of graphite by certain propellants at temperatures even below $5,000^{\circ}$F might be a serious problem. It is possible that tungsten may be more resistant under these conditions. As far as neutron capture is concerned, the advantage is decidedly with graphite. The absorption cross section of the latter for thermal neutrons (at ordinary temperatures) is 0.0045 barn, whereas that of tungsten is about 19 barns, with the probability that it is even higher in the energy range just above thermal. Consequently, tungsten is a definite "poison" and its presence in a thermal (or intermediate) reactor would have to be compensated for by the inclusion of additional amounts of uranium-235 to maintain criticality.

In reviewing the properties of graphite and tungsten, with a view to their possible use as the basic material for the fuel elements of a nuclear rocket reactor, the conclusion was reached that, at least for the present, graphite is to be preferred. Both its strength and thermal conductivity, although not outstanding, are believed to be satisfactory at $5,000^{\circ}$F, even when loaded with the necessary amount of uranium (as carbide). Furthermore, the graphite acts as a moderator, with a resultant decrease in the critical mass of fissionable material, in a homogeneous core. This effect is almost completely absent with tungsten, so that a separate moderator must be included in the reactor, giving a more complex (hetero-geneous) system, as described below. The main drawback to graphite is the possibility of corrosion by the propellant and various methods, to be described later, are being investigated for overcoming this difficulty.

Core Geometry and Moderator

There are two general ways in which the fuel can be disposed within the fission region (or "core") of a reactor. These two types of geometrical arrangement are called homogeneous and heterogeneous, respectively. In a homogeneous rocket reactor, the fuel would be distributed more or less (although not precisely) uniformly throughout the whole of the core. In a heterogeneous system, on the other hand, there are isolated lumps or "islands" of fuel arranged within a matrix of another material, the moderator.

The choice of the core geometry in a rocket reactor is bound up, to a great extent, with the choice of moderator for slowing down the neutrons. The neutrons liberated in the fission process, and which serve to maintain the fission chain, are fast. In a thermal or intermediate reactor, these neutrons must be slowed down and the material used for this purpose, i.e., the moderator, must have a low mass number, as stated earlier.

Graphite, consisting of carbon, mass number 12, is a fairly good moderator, and if it is to be used for the fuel elements, the design of the reactor is greatly simplified if a homogeneous core geometry is adopted with graphite as the moderator. In this case, the fissionable material is distributed throughout the whole of the graphite in the reactor core. A heterogeneous system with a graphite matrix, in which is arranged a pattern (or lattice) of fuel elements could be designed, but it would seem to have little advantage over the simple homogeneous core geometry.

It appears that the main argument for a heterogeneous core is that, when used in conjunction with a moderator containing a large proportion of hydrogen, an appreciable reduction can be realized in the critical mass of uranium-235. The propellant for a nuclear rocket would invariably be either hydrogen or a hydrogenous compound, and so the liquid propellant could also function as the moderator. Because of the high temperatures that must be attained in the fuel, a homogeneous reactor with a hydrogenous solid or liquid as moderator could not be realized. But in a heterogeneous system, no heat is generated directly within the moderator, except from the absorption of gamma rays and the slowing down of neutrons escaping from the fuel elements. Hence, the temperature in the moderator could be kept low enough for the hydrogenous material to remain stable.

The fuel elements in a heterogeneous rocket reactor may consist of graphite or tungsten loaded with uranium. If tungsten is chosen as the fuel-element base, then a heterogeneous geometry is almost mandatory due to the relatively high absorption cross section of this element for neutrons in the intermediate energy range, referred to as the "resonance" region. The lattice arrangement of the fuel in a matrix of moderator has the effect of decreasing the capture of neutrons in the resonance region.

Although a homogeneous reactor, with graphite as moderator, has been selected for the first nuclear rocket test reactor, consideration is being given to several heterogeneous (and other homogeneous) systems for future development. These include designs in which the fuel elements are either graphite (loaded with UC_2) or tungsten (with UO_2). The fuel elements may be arranged in a matrix of a hydrogenous moderator, which might be either a solid hydrocarbon, e.g., polyethylene or polystyrene, a liquid hydrocarbon which could also be the propellant, or a solid metal hydride, e.g., zirconium hydride stabilized with cerium, which is stable up to about 2,000°F under an excess pressure of a hydrogenous gas.

Fuel Distribution in Core

To obtain maximum volumetric performance from a reactor, it is desirable that all the fuel elements operate at their maximum allowable temperature. In order to insure such uniformity of temperature distribution throughout the reactor core, it is necessary to balance the coolant (propellant) flow and the heat (or fuel) distribution throughout the reactor. For neutrons of a given velocity, the fission (or heat generation) rate per unit volume, i.e., the fission density, at any point in the core is dependent upon the concentration of fissionable (fuel) material and the neutron density. If the fuel were distributed uniformly, the heat generation rate would be nonuniform, due to the nonuniformity of the neutron density. This is so because neutrons tend to escape from the outer regions of a reactor, with the result that the neutron density is highest in the interior and falls off gradually toward the outer surfaces.

One way of compensating for the greater fission density (or rate of heat generation) in certain regions of the reactor, is to adjust the propellant flow so that it varies in a parallel manner. Although this is possible in principle, it would be very difficult to realize in practice, especially in a rocket reactor, in view of the large number of coolant channels and their relatively small cross-sectional area. The alternative, more practical solution, is to maintain a uniform flow of the propellant coolant throughout all parts of the core, but to distribute the fuel in such a way as to maintain a uniform fission density in certain directions.

Perhaps the simplest design for achieving optimum heat transfer is based on a cylindrical reactor core with a uniform flow of coolant (propellant) parallel to the axis of the cylinder (Fig. 4). It is then required, ideally, that the fuel distribution be such as to give a uniform fission density in the radial direction. This density will not be the same at every axial location, but it should be uniform (radially) at each location.

The required radial uniformity (or "flatness") of fission density is achieved by increasing the fuel loading with increasing radial distance from the axis of the cylinder. Since the neutron density tends to decrease at the same time, the product of the neutron density and the fuel concentration, which determines the fission density and rate of heat generation, can
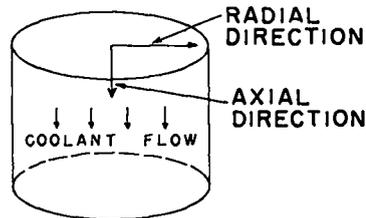
Fig. 4

be maintained essentially constant. Less variation in fuel loading is needed if the cylindrical core surface is surrounded by a neutron reflector. The characteristics of such reflectors will be considered shortly.

Reflectors at the top and bottom of the core, in addition, help to establish the axial fission density desired for maximum heat-transfer performance. It appears that nearly optimum conditions can be attained by having a sine-wave axial variation in the fission density along the first two-thirds of the core length, followed by an exponentially decreasing density over the final one-third of the length (Fig. 5). The required sine-wave variation is achieved
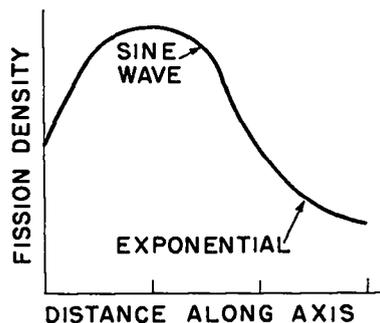
Fig. 5

if the fuel loading of the core, with a thin reflector, is uniform in the axial direction. This means, very simply, that in this region all the fuel layers are the same. The final one-third is then lightly loaded to provide the required exponential decrease in fission density.

Consideration has been given to both axial and radial flow patterns with various heterogeneous reactor systems. But it has been concluded that, in every case, a very much greater

complexity in fuel loading, than that described above, would be required in order to obtain equivalent heat-transfer performance. In this respect, therefore, the homogeneous cylindrical core geometry with axial flow appears to be most advantageous for the proposed nuclear rocket test reactor.

Reflector

A neutron reflector, which surrounds the core, can serve two important functions in reactor design. First, because it slows down and scatters back into the core neutrons which have escaped, the reflector can bring about a considerable decrease in mass of fissionable material needed to make the reactor critical. Second, by minimizing the net loss of neutrons from the outer layers of the reactor core, the presence of a reflector means less variation in the neutron density and, consequently, less variation in the fuel loading distribution to achieve optimum heat-transfer performance, as indicated above.

For an intermediate (epithermal) reactor, the reflector must have the same essential properties as a moderator, so that only materials of low mass number can be considered. In principle, hydrogenous materials might be expected to be good reflectors, but hydrogen has a moderately large cross section for neutron capture. Further, although the reflector temperature will be considerably less than that of the core, it would be sufficient to cause vaporization or decomposition of most hydrogenous liquids and many solids. This same objection applies to deuterium compounds which would, otherwise, be excellent reflectors. The possible reflector materials, at present, are therefore beryllium metal, beryllium oxide, and graphite.

Beryllium oxide is somewhat brittle, especially at moderate and high temperatures, and it is also a poor conductor of heat. Consequently, the choice lies between beryllium and graphite. Of these two, beryllium metal is preferred, in spite of its much greater cost, because it requires about half as much of this element to be as effective a reflector as a given thickness of graphite. Since the density of beryllium is only slightly greater than that of graphite, the use of beryllium metal as the reflector means a significant saving in both over-all weight and size of the complete rocket reactor.

Propellant

In selecting a propellant for a nuclear rocket motor, two properties should be considered, namely, the mean molecular weight of the exhaust gases and the heat capacity of the liquid. A low molecular weight of the exhaust gas means a high exhaust velocity, which is an advantage, as seen earlier. A high specific heat of the liquid is desirable since this will make possible a higher power output of the reactor for a given weight of propellant, assuming other conditions, such as mass flow rate and exhaust gas temperature, are unchanged.

-18-

It should be understood that, as far as the exhaust gas velocity is concerned, the molecular weight of the original propellant material is not important. Many substances of initially high molecular weight, such as liquid hydrocarbons and alcohols, are extensively dissociated at high temperatures to yield gases of fairly low average molecular weight.

From several points of view, liquid hydrogen is an excellent propellant; its molecular weight of 2.0 is the lowest possible and its heat capacity of 4 Btu/(lb) ($^{\circ}$F) or 4 cal/(g) ($^{\circ}$C) is large. Somewhat overriding the advantage of the high heat capacity is the low density, about 0.07 g/cm^3, so that a large volume of propellant, and hence large tankage weight, is required. The great drawback at present to the use of hydrogen as a propellant, however, is its low boiling point of -253°C (-423°F) at 1 atm pressure. The storage of liquid hydrogen, to be available as a nuclear rocket propellant, would thus require an extensive cryogenic system.

The main reason why liquid hydrogen was not chosen as the propellant for the first Los Alamos rocket test was the belief that a logistically simple system, which does not involve cryogeny, represented the most urgent need for ICBM applications. In view of the many problems to be solved in connection with the nuclear rocket system, it was considered undesirable at the present time to add the additional cryogenic engineering problems of storing large volumes of liquid hydrogen. It is highly probable, however, that this propellant will play an important part in future nuclear rocket developments.

Liquid helium, molecular weight 4.0, comes next to hydrogen in the matter of high exhaust gas velocity. It has the advantage of being noncorrosive to graphite even at high temperatures, whereas there is some evidence that in the temperature region of about 1000 to 1500°C (1800 to 2700°F) corrosion by hydrogen begins to be significant. However, the cryogenic problems associated with helium are much worse than for hydrogen. It appears that the production, handling, and storage of liquid helium, in quantities sufficient for rocket propulsion, are too difficult to warrant its consideration as a propellant.

The next possible propellant to examine is methane, which boils at -126°C (-259°F) at 1 atm pressure. The average molecular weight of the dissociated gases at about 5,000°F has been estimated to be 5.7. Like hydrogen, however, liquid methane is not a "ready" propellant, since it could not be stored in the rocket tank any appreciable time in advance of a mission. Another drawback to methane, indicated by preliminary experiments, is the fact that as a result of its decomposition at high temperatures it leaves a deposit of carbon. The behavior under rapid flow and high pressure conditions, such as would exist in a rocket reactor, is not known. But there is a possibility that the deposition of carbon would occur on the fuel-element surfaces, thus causing partial (or complete) blockage of the flow channels.

Because of dissociation at high temperatures, many hydrocarbons which are either easily liquefied and stored, such as propane and butane ("LP" gases), or are normally liquid at ordinary temperatures, such as those used for combustion engines or jet fuels, have an average molecular weight of 6 to 7 at 5,000°F. Such hydrocarbons would be quite attractive as propellants were it not for the fact that, like methane, they deposit carbon on surfaces at high temperatures.

Somewhat in the same category as the liquid hydrocarbons, at least physically, are the alcohols. These are inexpensive and could be stored easily in a rocket missile. They have fairly low average molecular weights (8 to 10) at high temperatures. However, very little is yet known of their corrosive properties, and these may be quite significant in view of the presence of oxygen in the alcohol molecule. The alcohols are being studied with a view to their possible future use as rocket propellants.

At the present time, the propellant which is being favored for the first test reactor is ammonia. Liquid ammonia is a fairly common article of commerce, and as far as storage is concerned, the problems are not greatly different from those for LP gas. The liquid boils at -33°C (-28°F) at 1 atm pressure; the critical temperature is 132°C (270°F) and the critical pressure is 111.5 atm (about 1650 psi). If two-phase flow in the reactor coolant channels is to be avoided, which is desirable, the operating pressure will have to exceed the critical pressure. Although this pressure, for ammonia, is high, it is not impossibly high. The specific heat of liquid ammonia is 0.48 cal/(g) (°C), which is considerably less than that of hydrogen, but the higher density of ammonia (0.7 g/cm^3), gives it a larger volumetric heat capacity. Hence, for a specified reactor power and propellant-flow rate, the weight of ammonia required will be greater but the volume will be less than that of hydrogen.

Ammonia dissociates essentially completely, into nitrogen and hydrogen, below 3,000°F at reasonable pressures, and the average molecular weight of the dissociated gas is 8.5. Since the exhaust gas velocity is inversely related to the square root of the molecular weight, ammonia as a propellant is not greatly inferior to the hydrocarbons, although it is by no means as good as hydrogen.

By means of calculations of the type given at the end of this chapter, it is possible to relate the gross weight of the rocket vehicle to the burn-out velocity for various propellants, assuming a given deadload, i.e., total weight of warhead and guidance equipment. The results for the propellants, hydrogen, JP-4 (jet fuel), and ammonia, based on a deadload of 5,000 lb, are shown in Fig. 6. The burn-out velocities for three ranges of special interest are indicated. It is seen that, for a given burn-out velocity (or range), the gross vehicle weight increases from hydrogen to ammonia. However, the size (or volume) of the missile depends
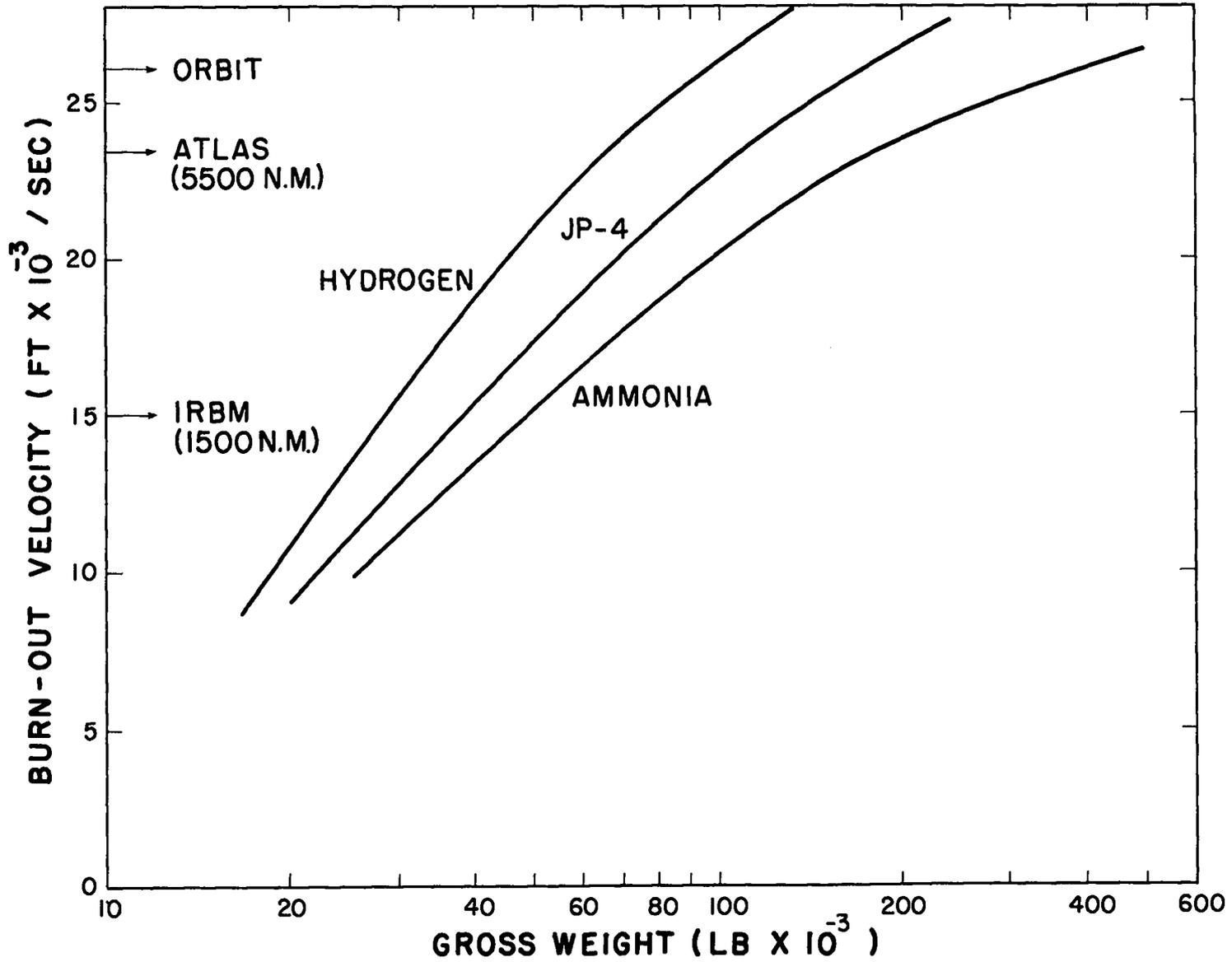
Fig. 6   Burn-out Velocity versus Gross Weight for Various Propellants

also on the density of the liquid propellant.

Basically, it is the exhaust gas velocity which determines the relationship between the gross weight and the burn-out velocity. Since, as a general rule, the liquid density and molecular weight increase while the attainable exhaust velocity decreases, the effect is to cause an initial decrease in missile size with increasing molecular weight of the exhaust gases. After passing through a minimum, the missile size subsequently increases.

The particular value of the average molecular weight at which the missile size is a minimum depends on the range and payload. It happens that for a typical Atlas (or somewhat less demanding) mission, liquid ammonia and the normally liquid hydrocarbons are the propellants for which the missile would have the smallest size. However, for increased payload or range requirements, the minimum occurs with propellants of lower molecular weight. In these circumstances it would probably be desirable to use liquid hydrogen (or possibly methane) as the propellant for a nuclear rocket reactor.

The main difficulty associated with the use of ammonia as coolant in a reactor with graphite fuel elements is the indication that, at high temperatures, corrosion is at least as bad as that due to hydrogen, mentioned above. Under similar conditions of temperature and high flow rate, helium gas does not affect the graphite to any appreciable extent, and so it appears that the attack by ammonia is true corrosion, rather than erosion. It is believed that this problem can be solved by coating the graphite surface with a resistant compound, such as zirconium carbide or niobium carbide (or a mixture), as will be discussed later.

Another interesting possibility which is being studied is to use a propellant consisting of a mixture of ammonia and a liquid hydrocarbon in suitable proportions. At high temperatures, the former substance corrodes away the graphite whereas the latter causes a deposition of carbon. It may prove feasible to find such conditions that the two effects compensate each other and there is no net change in the dimensions of the graphite fuel element or of the coolant flow channels between them.

The Nuclear Rocket Reactor

The design of the first Los Alamos test reactor, to be known as KIWI I, will not be established in detail for another year. Present indications are that it will generally follow the design concepts of "Old Black Joe", which is shown in Fig. 7 and described below. The fuel elements consist of graphite plates loaded with uranium-235 (about 0.1 gram U per cm^3). The plate thickness will be about 0.14 in. with a separation of 0.06 in. to provide flow channels for the propellant. The fuel plates will probably be in sections, as shown, for ease of fabrication, and also to provide a number of plenum chambers in which there will be some mixing of the

NUCLEAR ROCKET MOTOR TEST DEVICE

CONTROL ROD THIMBLE — — CONTROL ROD COOLANT

NUCLEAR FUEL PLATES

PRESSURE SHELL

BERYLLIUM REFLECTOR

40.5"

PROPELLANT INLET

NOZZLE
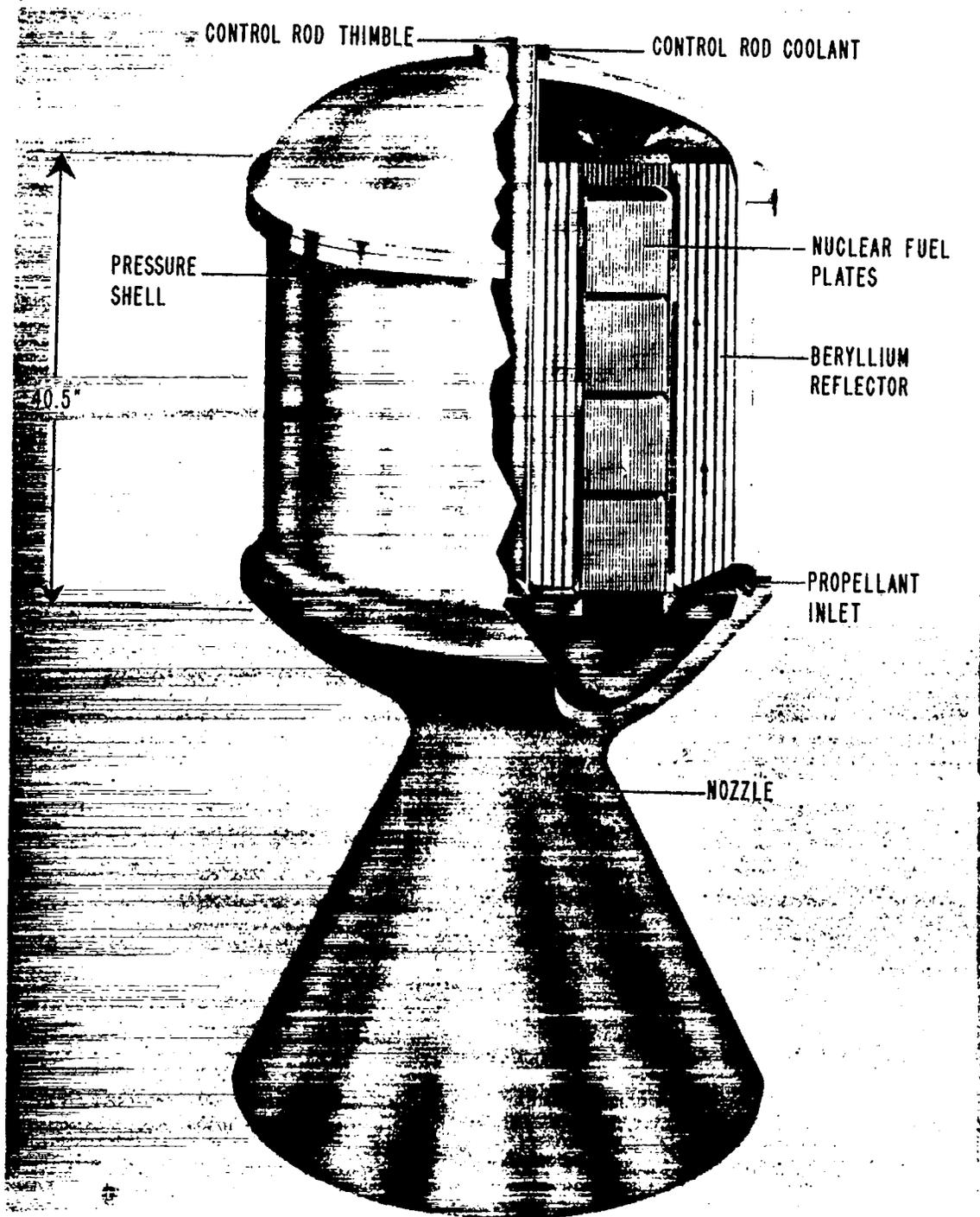
Fig. 7

propellant passing through different channels. To prevent vibrational instability (or flutter) which may result from the very rapid flow of coolant, the plates may be curved or be provided with spacer ribs (or both).

Around and on top of the fuel region will be layers of beryllium, about 4 in. thick, to act as the side and top relectors. The side beryllium reflector will be separated from the core by layers of 1 in.-thick carbon, which is a relatively poor conductor of heat. This carbon will thus act as a heat insulator, so that only a small proportion of the heat generated in the core will be lost to the reflector by conduction. However, heat will be generated in the reflector by the slowing down of neutrons and the absorption of gamma rays escaping from the core. The bottom reflector will be a thick graphite plate, which also serves as a support for the core and reflector structure.

The ammonia propellant, at a pressure of about 1,500 psi, will enter the reflector region at the bottom and flow upward through the side beryllium relector, thus removing heat from it. After entering a plenum chamber at the top of the core, the propellant will flow down, through the top beryllium reflector, and then between the fuel elements in the core. For a power of about 2700 Mev, the weight flow rate will be 550 to 600 lb/sec. The flow will be highly turbulent, with a Reynolds number of the order of 100,000, thus assuring flow stability. Such a large Reynolds number means a relatively high pressure drop in the core (about 250 psi) and a large temperature difference between the fuel element surface and the coolant. The design temperature of the exit gas is expected to be 4,500°F and the maximum fuel element surface temperature, near the exit, will then be 4,800°F, since above this temperature the strength of the graphite decreases significantly. The maximum allowable load stress used for the design of the graphite components is 1,000 psi.

At a C/U^{235} ratio of about 350 or 400 to 1, such as is at present envisaged for the fuel elements, the critical mass of uranium-235 in the reactor is about 40 kg with 4 to 4-1/2 in. beryllium reflector. It is of interest to mention, however, that the quantity of uranium-235 that will be consumed during operation is negligible. The total operating time of the reactor, either on the test stand or in flight, will be about 5 min and at a power of 2700 Mw, the total energy production will be somewhat more than 9 Mw-days. The fission of 1 gram of uranium-235 releases roughly 1 Mw-day of energy. Consequently, only a little more than 9 grams of fissionable material will actually be consumed in the operation of the nuclear rocket reactor. This is an insignificant fraction of the total that must be present for criticality, i.e., for continued power-level operation.

In order to permit rapid start-up, the reactor core must contain more than the critical mass of fissionable material, so that the system is supercritical. To compensate for this,

a control rod, consisting of refractory material that is a good neutron absorber, e.g., gado- linium and samarium oxides, will be inserted in an axial thimble in the central island of the reflector (see Fig. 7). If the control rod is inserted, the reactor is subcritical and will pro- duce very little power. To start up the reactor the rod is withdrawn at a suitable rate which must be determined by calculation and experiment. The withdrawal must be sufficiently rapid to permit attainment of full power within about 1 min or so. However, if it is too rapid, the reactor may go completely out of control and suffer damage. When the desired power is at- tained, the control rods must be positioned so as to check the power increase and maintain the reactor at the desired level.

The problem of control of a nuclear rocket reactor is excessively complex as so many factors are involved and the operation must occur within a very short time interval. Reactor control is bound up very closely with the magnitude and sign of the temperature coefficient of reactivity, i.e., the change relative to criticality resulting from a change in temperature. Further consideration to this topic will be given below.

In the preceding sections of this report, several of the factors leading to the conceptual design of Old Black Joe have been reviewed. Some of the involved problems arising in the implementation and engineering of this broad design will now be examined.

Fabrication of Fuel Elements

Regardless of the shape of the fuel elements, e.g., plates in Old Black Joe or tubes of various kinds for future designs, there are three main aspects of their fabrication. These are (1) production of graphite of adequate strength, (2) loading with uranium, and (3) coating with a protective layer to prevent corrosion by the propellant. In some cases, the first two aspects may be combined.

Production of Graphite

Reactor-grade graphite, almost completely free from poisons that capture neutrons, is obtainable commercially. One way in which it can be made is to mix lamp black and fine graphite powder with pitch, to act as a binder, and extrude the mixture at about $120^{\circ}C$ ($250^{\circ}F$) and 1,500 psi, into the required final shape. The so-called "green" material is first "coked" (or "baked") by heating at about $1,000^{\circ}C$ ($1,830^{\circ}F$) for a day or so to carbonize the pitch, and then at 2,700 to $3,000^{\circ}C$ (4,860 to $5,430^{\circ}F$) for 5 or 10 minutes, or longer depending on the size, to produce graphitization. If the mixture after coking is impregnated with warm pitch and then re-coked before the graphitization stage, the strength and density of the graphite are improved. As it is relatively soft, the graphitized material can be machined to a desired shape or size.

Work is proceeding in the Laboratory on the production of graphite along the lines indicated above. In addition to forming the required shapes by extrusion, both hot and cold pressing are being investigated. In the mixture employed in the pressing operation, a furane resin is used in place of pitch as the binder. There are possibilities that greater mechanical strength after graphitization may be obtained by pressing as compared with extrusion.

Loading of Graphite

Loading of graphite may be carried out in three general ways: (1) after preparation of the required shape, the graphite can be impregnated with uranium from a solution; (2) the uranium in the form of UO_2 or UC_2 may be incorporated in the mixture used to make the graphite; and (3) the uranium, as UC_2, is mixed with the material to be applied as a protective coating on the graphite surface (see below). Apart from other considerations, the impregnation or coating method might be preferred if it is desired - as it is in some applications - to have the uranium loading near the surface of the fuel element. On the other hand, incorporation of the uranium in the graphite mix would give uniform distribution throughout the fuel element.

In the impregnation method, the outgassed graphite is immersed in a uranyl nitrate solution. In one procedure, which has been found satisfactory, a solution of $UO_2(NO_3)_2 \cdot 2H_2O$ in tertiary butyl alcohol at $70^\circ C$ ($160^\circ F$) is used. After draining off excess solution, the graphite is cooled to room temperature, when solidification of the absorbed solution occurs within the pores. The butyl alcohol is removed in a vacuum, leaving uranyl nitrate within the graphite. Slow heating in a vacuum at about $300^\circ C$ ($570^\circ F$) converts the nitrate to the oxide UO_3, and further heating to $2,000^\circ C$ ($3,630^\circ F$) leads to the formation of uranium carbide, as a result of interaction with the graphite (carbon). The product may be reimpregnated with the uranium solution and the treatment repeated in order to increase the uranium loading.

An alternative impregnation process, using an aqueous solution of $UO_2(NO_3)_2 \cdot 6H_2O$, shows promise of success. After soaking up the solution, the graphite is heated in an oven at a temperature of 64 to $100^\circ C$ (150 to $212^\circ F$), i.e., just above the melting point of the hexahydrate ($60^\circ C$ or $140^\circ F$), to remove water. Subsequently, the absorbed uranyl nitrate is converted to oxide and carbide, as described above.

Uranium, as UO_2 or UC_2 in the required proportion, can be added to the mixture of lamp black, powdered graphite, and binder before extrusion (or pressing). The extruded (or pressed) piece is then coked and graphitized in the usual manner. Although simple in principle, this loading procedure frequently leads to poor mechanical properties of the resulting graphite. This may be due to gas accumulation around the particles of UC_2, either present initially or produced by the interaction of UO_2 with carbon. Better results may perhaps be

obtained by carrying out coking and graphitization in a vacuum.

No matter which loading procedure is used, the uranium is ultimately in the form of UC_2 dispersed in the graphite. For fuel-element application in a power reactor, this has one obvious disadvantage: the eutectic temperature of the UC_2-graphite system is about 2,310°C (4,200°F). Hence, in a fuel element in which the temperatures exceed 4,500°F, the UC_2 will be present as a liquid. Apart from the probable deleterious effect on the strength of the graphite, the exudation of drops from the surface of the fuel element would be very undesirable. It appears that this difficulty may be overcome by inclusion of zirconium carbide (ZrC) with the UC_2. A mixture of 20 w/o UC_2 to 80 w/o ZrC has been suggested; in graphite this system liquefies at about 2,800°C (over 5,000°F). Because of its small cross section (0.18 barn) for thermal neutron capture, zirconium will have a relatively minor effect on the critical mass. The problem of loading the graphite with the appropriate mixture of uranium and zirconium is being studied.

Coating of Graphite

To protect the graphite fuel elements from corrosion (and erosion) by the propellant, coating with a refractory carbide is being investigated. Only two carbides, namely those of zirconium and niobium (ZrC and NbC), have sufficiently high melting points and small neutron capture cross sections (1.1 barns for Nb) to be at all promising. Some preference has been expressed for a mixture of 20 w/o of NbC and 80 w/o of ZrC. In equilibrium with excess graphite, this mixture begins to liquefy at a temperature greater than 2,850°C (5,160°F), which is above the proposed maximum surface temperature of the reactor fuel elements.

Various methods for coating graphite with zirconium and niobium are being investigated. These include deposition from heated vapor of the chloride (or iodide), dipping the graphite into the molten metal, painting with a slurry of the metal oxide and heating, melting a thin foil of metal on the surface, and deposition of metal by electrophoresis from a colloidal solution. The electrophoretic method is proving to be of considerable interest because of its simplicity. The metal, e.g., niobium, is ground very finely in a ball mill with a suitable liquid, e.g., isopropyl alcohol, and possibly a stabilizer, until a colloidal solution is formed. In this sol, the small metal particles are electrically charged, the sign depending on the impurities, etc., present in the medium. Suppose the colloidal metal particles carry negative charges, as is frequently the case. The graphite is then made the anode, with an insoluble cathode, in the colloidal solution and a potential difference is applied. The metal particles migrate to the anode and are deposited on the graphite as a coherent layer.

After depositing a thin layer of the desired metal on the graphite, the surface is flash heated to melt the metal and form a continuous layer. Subsequently, the whole is heated to

produce a coating of carbide. If necessary, a further deposit of metal may be laid down on the carbide coating to insure complete coverage of the surface. Although it is hoped that the coating of ZrC, NbC, or a mixture of both, will protect the graphite against corrosion by ammonia, essentially nothing is yet known about the behavior of the nitrides which will probably be formed.

Testing of Loaded Graphite

The properties of graphite, which is a highly variable material, have been the subject of some study at high temperatures. But almost nothing is known about graphite loaded with uranium (and perhaps also zirconium) carbide in the region of 2,500°C (4,500°F). For the design of the rocket reactor, the properties of greatest interest are the tensile strength, creep, thermal conductivity, thermal expansion, in addition to corrosion resistance of the coated material. Studies for the purpose of obtaining further information are in hand, but are faced with tremendous difficulties in working at temperatures up to 2,500°C.

For the determination of the physical and mechanical properties, test pieces of graphite are generally used and they are heated electrically by virtue of their resistance. It should be noted, however, that many of the properties of graphite are highly anisotropic. That is to say, a property measured in one direction may be quite different from that observed in the same piece of graphite along another direction. This may introduce a problem in connection with the thermal conductivity. For the design of the reactor system, the conductivity in the thinnest direction of the fuel-element plate is the significant quantity, but it may prove to be difficult (or impossible) to measure. The thermal conductivities in the two directions at right angles may be quite different from that in the direction of interest, i.e., the one in which almost all the heat flows from the interior to the outside of the fuel element.

For corrosion studies with propellant, graphite tubes, about 0.06 in. i.d., are used; this is approximately the proposed distance between the fuel plates in Old Black Joe. These are heated electrically and the propellant is pumped through at velocities of the order of those to be expected in the actual reactor. It will be recalled that the operating time of the rocket reactor is expected to be about 5 min, so that similar time periods are used in corrosion testing. After the test, the graphite tube is removed, cut open, and examined. Because of electrical power supply limitations, the extent of this work is somewhat restricted. But steps are being taken, as will be seen shortly, to remedy the situation.

Heat-Transfer Studies

Another field in which information is completely lacking but is urgently needed is that of heat transfer to gases at very high weight flow rates and large Reynolds numbers, and at

high temperatures. Measurements under these extreme conditions involve many difficulties as well as novel techniques. Until corrosion-resistant coatings are available, the heat-transfer studies are being made with helium or nitrogen as coolant in tubes similar to those employed in the corrosion tests. The tubes are heated electrically and temperature measurements are made with optical pyrometers. The heat transferred to the flowing gas from the heated graphite is determined in a specially designed calorimetric system.

The preliminary heat-transfer work is being done with single tubes at a power of about 150 kw. Work is proceeding on facilities which will permit increase in the available electrical power in stages to 500 kw, 1 Mw, and ultimately to 10 Mw. This will permit the testing of larger and larger flow systems until ultimately heat-transfer measurements are made on actual graphite plates of the type to be used as fuel elements in the reactor. These test plates will have a protective coating, if necessary, and will use the actual propellant as coolant. If they are loaded with uranium, then ordinary uranium (or depleted uranium), but not uranium-235, will be used.

Criticality and Related Neutronics Studies

With the aid of the electronic computing machines available in the Laboratory, critical masses can now be calculated theoretically with a considerable degree of accuracy, provided all the basic information is supplied. It is desirable, however, to make experimental determinations of critical mass and fission density distribution under a variety of conditions. Preliminary work for the nuclear rocket reactor is being done with a mock-up called Honeycomb. This consists of several frames of aluminum in which the cells of the "honeycomb" are each about 1 ft long and have a square cross section of about 3 x 3 in. In each of these cells are placed graphite slabs separated by oralloy (93.5 percent uranium-235) strips, to represent loaded fuel elements of the proper composition. The ammonia propellant is simulated by strips made from a mixture of melamine plastic and polyethylene, which has the composition $NH_3 + C_{1.5}$. Blocks of beryllium reflector surround the pseudocylindrical core to complete the reactor mock-up (Fig. 8).

In addition to making determinations of critical mass, Honeycomb is used to study the fission density distribution throughout the system. This is referred to as a "fission traverse". Two methods can be used for making such a traverse. In one procedure, thin aluminum foils, called "fission catchers" are located at a number of points in the core in contact with the oralloy strips. Fission products, in amounts proportional to the fission densities at the various locations, are collected on the aluminum foils. After the test of the mock-up, at very low power, the fission foils are removed and their beta-activities are determined. This provides a measure of the fission density distribution. In the second method, which is the one
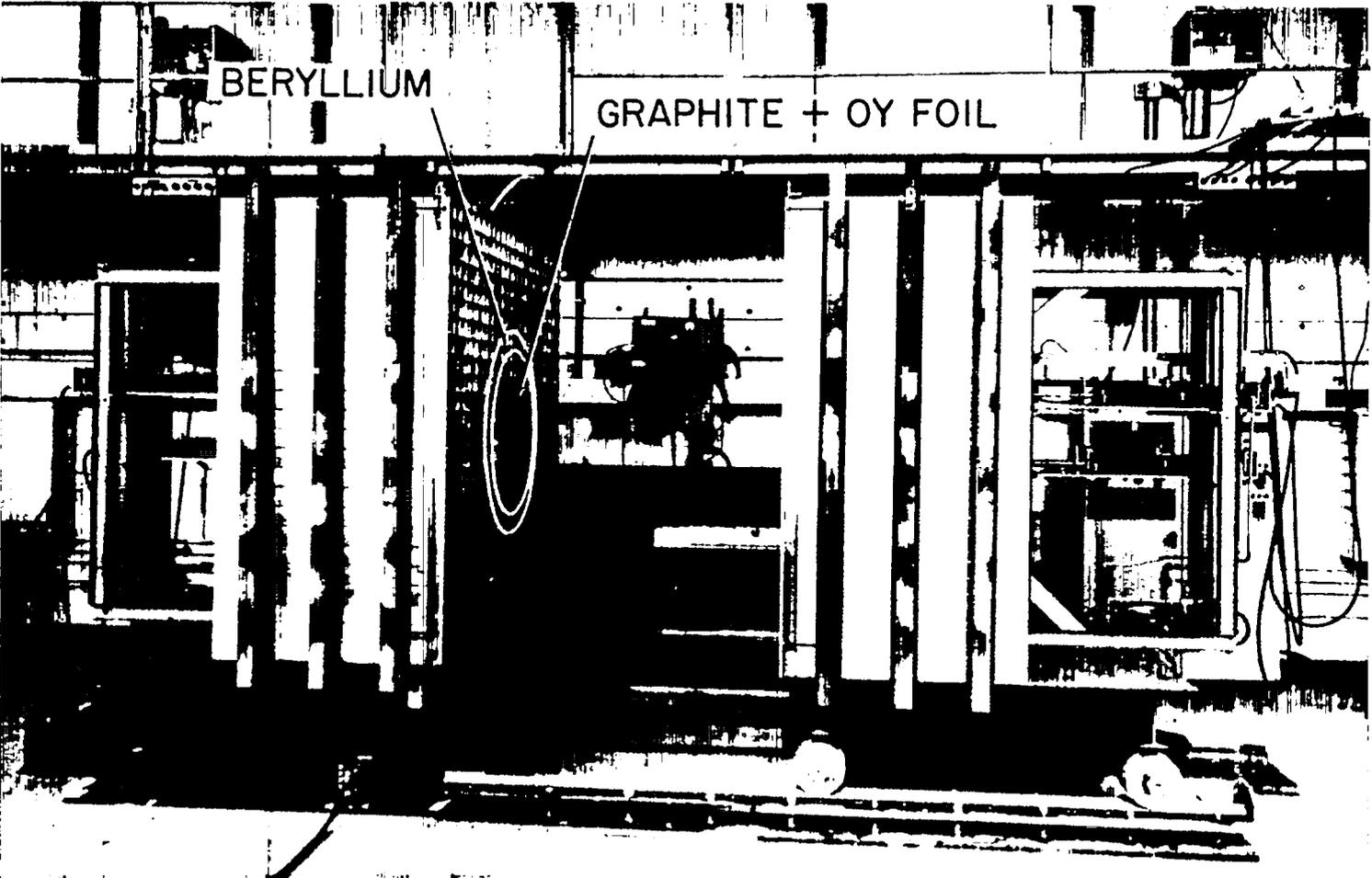
Fig. 8

mainly used, the gamma activities of the fission products in the oralloy strips themselves are observed. The activity is, of course, proportional to the fission density. The value of the fission traverse lies in the fact that it can be used to determine the fuel distribution required to give an essentially flat radial fission density distribution. As seen earlier, such a distribution is highly desirable in order to achieve optimum heat-transfer conditions from fuel element to propellant.

Another application of Honeycomb is to find the poisoning effects of various materials, such as coatings on graphite, that may have to be included in the reactor. In somewhat the same category is the study of control rod effectiveness. Such information is necessary for the design of the reactor control system.

Plans are being made for construction of Zepo (zero power), a more exact mock-up of the nuclear rocket test reactor. It is hoped that at least 10 percent of the fuel elements will be loaded with uranium-235, whereas the others will be simulated with graphite plates and oralloy strips. With Zepo, the fission density distribution will more closely resemble that to be expected in the completed reactor. Finally, the actual test device will be assembled at Los Alamos and its nuclear (fission) properties determined at very low power, before it is shipped to Nevada for the full-scale test.

Reactor Control

There are a number of peculiar features related to reactor control in general which make the problem of controlling a rocket reactor extremely difficult. In conventional systems there is usually a relationship between the position of a control effector, e.g., a throttle, and the amount of power being produced. This is not the case for a nuclear reactor. The position of the control rod (or rods) determines the rate at which the power is being increased (during start-up) or decreased (during shut-down). But when the reactor is producing power at a steady rate, the rod will be essentially in the same position irrespective of the power output.

To start up a reactor, the neutron-absorbing control rod is moved out, so that the reactor becomes supercritical. The neutron density, fission rate, and power production, which are related to one another, then increase at a definite rate. If n_o is the neutron density at the time of start-up, then at any subsequent time t, the neutron density, n, may be represented by

$$n = n_o e^{t/T} \quad ,$$

where T is called the "reactor period". The period (or e-folding time) is the time in which

the neutron density (or power) is increased by a factor e, i.e., 2.71. The shorter the reactor period, the more rapidly will the power output increase. However, if the period is very short, the neutron density may increase so rapidly that the design power of the reactor can be greatly exceeded before the mechanically operated control rods can become effective in reducing the power. This would result in permanent damage to the reactor.

In start-up, which should be completed within a minute or less for a rocket reactor, strict period control is necessary. This is achieved by means of servomechanisms associated with "period meters" which are operated by neutron detectors. As the desired operating power is approached, the period must be increased, to slow down the rate of power increase. In other words, the control rod is pushed into the reactor to make it less supercritical. Control now passes to a neutron level indicator which, in combination with a servosystem, will adjust the control rod so as to maintain the neutron density at the value necessary to produce the required power from the reactor. Thus, the purpose of the control rod is now to maintain the neutron density (and power) essentially constant at the level reached in the start-up stage.

In the start-up and operating stages, another peculiarity of the system will become apparent. This is the effect of neutron moderators and of neutron absorbers (or poisons). Normally, the propellant, ammonia, might be expected to act as a poison, in some degree, because both the nitrogen and hydrogen capture neutrons to an appreciable extent. However, due to the considerable moderating effect of the hydrogen, the reactivity of the intermediate reactor is actually increased, rather than decreased. On the other hand, some of the fission products formed during operation are very strong poisons. To what extent moderation and poisoning will affect the control of a nuclear rocket reactor cannot be precisely known until measurements have been made with an actual test reactor.

When the operating power of the reactor is reached, control may be retained by the neutron level indicator. However, it is generally preferable to transfer the control to a system which actually measures the useful power output. In the case of a rocket reactor, for example, the thrust is the significant quantity and this depends upon the mass flow rate of the propellant and the exhaust gas velocity, the latter being determined by the reactor exit temperature, as seen earlier. To achieve maximum thrust, this temperature should be as high as possible. Thus, a control system may be designed to maintain the temperature close to the permissible maximum and to adjust the flow rate of the propellant to give the required thrust.

In this connection, a third unusual property of a nuclear reactor system may be turned to advantage. If, as is highly desirable but not always attainable, the reactor has a negative

temperature coefficient, the degree of criticality (or reactivity) will decrease with increasing temperature, and vice versa. Then, if there is, for example, an increase in the flow rate of the propellant, so that more heat is removed from the fuel elements, there will be a tendency for the reactor temperature to drop. As a result, the reactivity will increase and there will be an accompanying increase in fission rate and heat production, which will serve to raise the temperature to its initial value. Thus, a reactor with a negative temperature coefficient lends itself to automatic self-control based on the power demand. This type of control has a rapid response, since it does not involve mechanical actuation of rods.

A simplified outline of the control system for a nuclear rocket reactor may be given with the aid of the block diagram in Fig. 9. Not shown in the diagram is a plug which may have to be placed in the nozzle during reactor start-up to help establish the proper flow conditions of the exhaust gas and also to build up the propellant pressure in the reactor. In the start-up phase, the control rods will be actuated by signals from neutron indicators. Control will then pass to the gas temperature and flow rate meters as mentioned above. Through an automatic programmer, the signals received will determine the turbine and pump speeds and the control rod motion (if required).

The design of a control system that is accurate and stable and has a rapid response involves numerous problems. It is being carried out somewhat along the following lines. First, an analysis is made of the various components of the system, e.g., pipe lines, pump and turbine, reactor (kinetic behavior), heat transfer to propellant, control rod actuator, and nozzle. The time behavior is then expressed by means of a series of differential equations which may have to be based on simplified models.

The next stage is to design and build an analogue computer which will simulate the behavior of the actual system, as determined by the differential equations obtained in the analysis. This computer will facilitate the design and testing of servomechanisms and also the general programming of the system operation. Then, as real system components become available, they will be tied into the simulator until eventually, in the test reactor, all the components will have replaced their analogues.

The control system of the nuclear rocket test reactor will actually be more involved than is really essential, because the objective of the test is to provide as much basic data as possible. The instrumentation to study the reactor behavior will be very complicated. In addition to observing the steady-state characteristics, various perturbations will be deliberately introduced to determine the response of the system to transient changes. The information obtained will then be used to design the simplest possible control systems for a rocket reactor that is intended to fly.
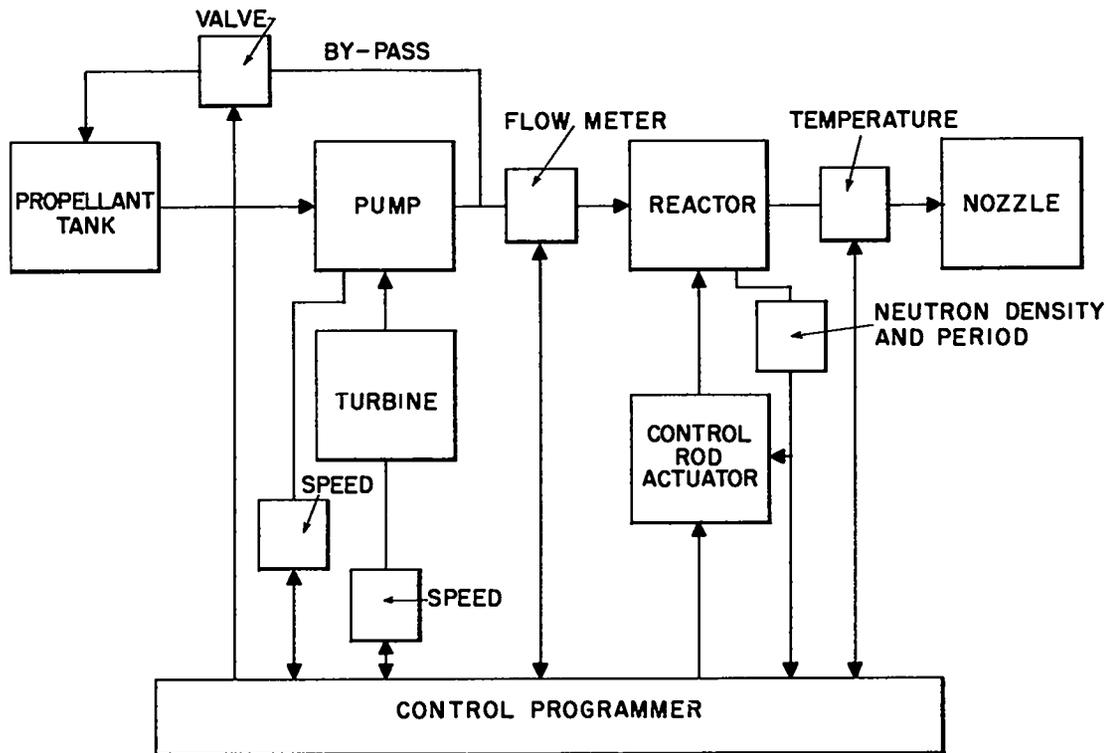
Fig. 9   Control System for a Nuclear Rocket Reactor

Nuclear Rocket Reactor Test

The testing of the first nuclear rocket test device is planned for the latter part of 1958. Because of the high radiation levels that will be attained, the probability of contamination from fission products in the propellant exhaust, and the general safety problems involved in an operation on the scale planned, the work will be done at the Nevada Test Site. Present plans call for two test cells, a tank farm, assembly and disassembly buildings, a "hot" storage area, possibly instrument bunkers, and, at some distance, a control room (Fig. 10). It may be mentioned, in the latter connection, that a single test, lasting about 5 min at full power, will consume about 180,000 pounds (about 30,000 gallons) of liquid ammonia. The test cell will house the test reactor itself on a thrust stand and will have shielded compartments for control and diagnostic equipment. It will also contain a room for the propellant pump, a turbine (about 6,000 h.p.), and their controls. The pump must be located close to the test reactor in order to provide the required high pressure and high flow rate. But the liquid ammonia tanks will be about 500 ft away and will be connected to the pump by 10-in. pipe.

In an actual nuclear rocket vehicle, the fate of the reactor when propellant flow ceases is immaterial. It will probably destroy itself, due to the large amount of heat produced by radioactive decay of the fission products formed during the period of reactor operation. An important aspect of the test will be a "post mortem" examination of the reactor core, and so steps must be taken to insure its survival when the propellant is exhausted. To shut the reactor down, the control rod will be inserted automatically and, at the same time, liquid nitrogen (or other coolant) will flow through the core to remove the fission-product heat. The device will then be removed and disassembled by remote manipulators, and interesting components will be returned to the Laboratory for detailed study.

There are four main objectives of the first Nevada test, namely (1) to demonstrate the feasibility of a nuclear rocket motor operating at a power (or thrust) of practical interest; (2) to study both the steady state and dynamic performance of the system, to provide data for the development of subsequent nuclear rocket motors; (3) to evaluate the behavior of materials in high radiation fields and at high temperatures; and (4) to learn something about the operation of a test facility for rocket reactors.

Proving the feasibility of the system will involve measurements of neutron density (or flux), flow rates, increase in temperature of propellant, operation time, total thrust, etc. From these such parameters as the specific thrust, the reactor power, and its specific power will be evaluated.

Performance analysis will include a study of the temperatures and propellant flow rate throughout the core and moderator. Apart from visual evidence of degeneration of the fuel
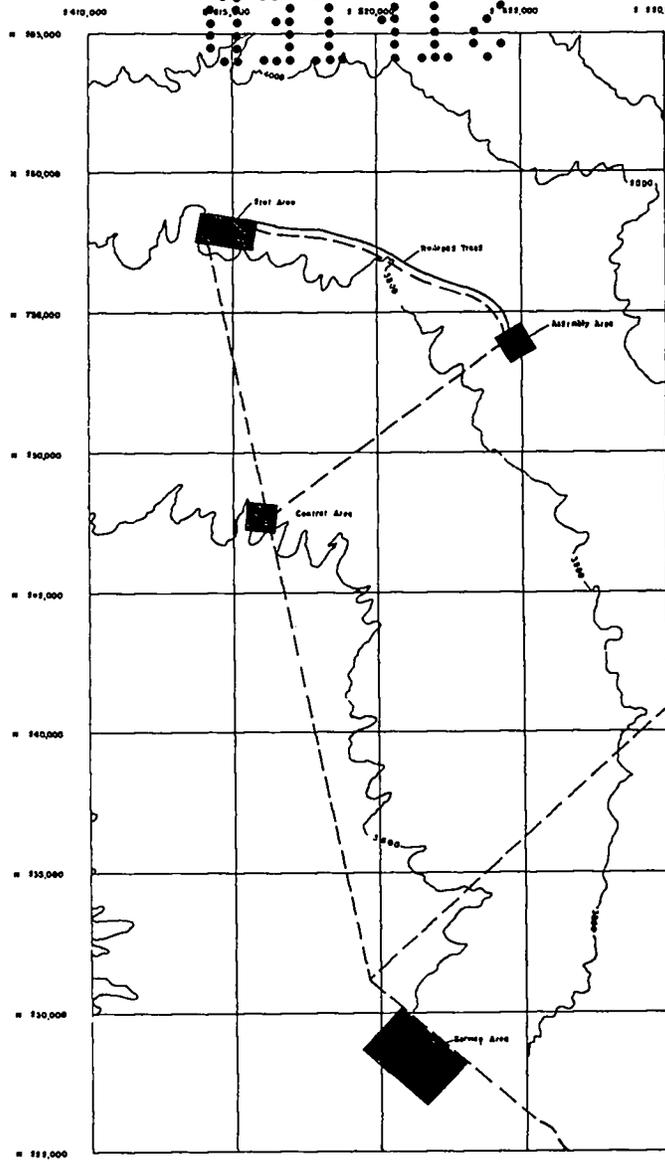
PROJECT 400 PROPOSAL

Fig. 10

elements, one of the most important measurements will be of the distribution of fission products, and hence the fission density, in the core. The dynamic response of the system to transient perturbations and the behavior of the controls will be studied. Structural stresses and vibrations during operation of the reactor will be observed, and a detailed examination will be made of the exhaust gases. A major problem here will be the installation of the complex instrumentation without interfering with the operation of the reactor at full power.

In a nuclear rocket vehicle, the propellant tank will lie between the reactor and the payload, containing the guidance and the warhead. The tank will then act as a partial radiation shield. Nevertheless, the rapid exhaustion of the propellant may mean that important components of the missile will be exposed to intense radiation from the reactor. One of the purposes of the test will be to map the radiation field to provide data concerning the environment of the guidance and warhead controls. The behavior of the reactor materials themselves, which are exposed to both radiation and high temperature, will also be investigated.

To obtain information useful for the future operation of rocket reactor test facilities, studies will be made of shielding, of activation of the surroundings by neutrons, and of contamination of the site by fission products escaping in the exhaust gas. In addition, experience will be gained in the operation of a nuclear rocket motor in both hot and cold states.

Metal Fuel Elements

A fact which must not be overlooked is the possibility that the exhaust gas temperatures of 2,500°C (4,500°F), which are essential to the success of the present nuclear rocket design, may not be attainable, due to failure of the fuel elements. A fair amount is known about ordinary graphite, but loading with uranium (and other) carbides introduces some uncertainties. In any event, it is unfortunate that graphite is a highly variable substance with pronounced directional properties. Two pieces made by what appears to be exactly the same procedure often have different characteristics, presumably because of differences in crystal orientation within the grains. With fuel elements operating under (or close to) limiting conditions, as is the case at temperatures in excess of 2,800°C (5,000°F), slight differences in the properties of graphite may mean the difference between success and failure.

Another uncertain factor is the degree of protection of the graphite that can be provided against corrosion by the propellant, by coating with certain refractory carbides. If the coefficient of expansion at high temperatures differs from that of the loaded graphite, the coating may crack and thus become ineffective.

In case the coated graphite fuel elements should prove to be ineffective at the high temperatures necessary for satisfactory nuclear rocket operation, studies are proceeding in the

Laboratory of fuel elements with a high melting point metal as the base. Three metals, namely, tantalum, tungsten, and niobium appear to be of interest because of their high melting points and strength at high temperatures. However, the first two have large cross sections for neutron capture. Consequently, possible binary alloy systems, e.g., molybdenum-tungsten and tantalum-niobium, each of which forms a continuous series of solid solutions, are being investigated.

It appears unlikely that uranium metal will be sufficiently soluble in the solid phases of these systems to make a useful fuel element for a rocket reactor. The probability is, therefore, that UO_2, with a melting point of about 2,850°C (5,150°F), will have to be incorporated by using the techniques of powder metallurgy. Heat treatment methods will have to be developed to provide adequate strength, and there is some evidence from the UO_2-Mo system that this can be done without great difficulty.

Fuel elements prepared in this manner may have superior mechanical and physical properties to those based on graphite, but the adoption of the metal-based type would involve a complete change in the design of the reactor. The two possibilities of interest appear to be (1) a heterogeneous system using a hydrogenous material, possibly the propellant, as moderator, or (2) an essentially nonmoderated system of simple design which would, however, have a fairly high critical mass. A further discussion of these matters is considered to be beyond the scope of the present report.

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APPENDIX*

Basic Rocket Calculations

The purpose of this appendix is to provide a general indication of how certain parameters of a rocket vehicle can be estimated. The thrust force, F_T, is equal to the rate of change of momentum of the exhaust gases, and if their velocity is constant, v_e, then

$$F_T = v_e \frac{dm}{dt} \, , \tag{1}$$

where dm/dt is the mass flow rate of the propellant. Over the powered part of the missile trajectory, the acceleration of gravity is essentially constant, so that the gravitational force, F_G, on the vehicle is given by

$$F_G = -Mg, \tag{2}$$

where M is the vehicle mass.

The sum of the forces on the missile is equal to its mass, M, multiplied by the acceleration, dv/dt. Hence, neglecting air resistance, which introduces an error of no more than a few per cent for a large missile, it follows from equations (1) and (2) that, for vertical flight,

$$M \frac{dv}{dt} = v_e \frac{dm}{dt} - Mg$$

or

$$dv = v_e \frac{dm}{M} - g \, dt. \tag{3}$$

Part of the propellant (about 6 per cent) is used to drive the turbopump which forces the propellant through the core. Hence, the decrease in mass is not all useful for imparting thrust; thus

$$dm = -\beta \, dM,$$

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*Material supplied by H. T. Gittings.

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where β is the fraction of the propellant producing thrust. Upon combining this expression with equation (3), the result is

$$dv = -\beta v_e \frac{dM}{M} - g\,dt .$$

Upon integration, this gives

$$v = \beta v_e \ln \frac{M_o}{M_o - \dot{M}t} - gt ,$$

where M_o is the initial mass of the missile and $\dot{M}$ is the time rate of change. If the trajectory is making an angle θ with the horizontal, the gravitational force term is decreased by $\sin \theta$, so that

$$v = \beta v_e \ln \frac{M_o}{M_o - \dot{M}t} - gt \sin \theta . \tag{4}$$

This is a general equation for the velocity of the missile as a function of time.

If t_b is the time of burn-out (or burning time), the total mass of propellant, M_p, is given by

$$M_p = \dot{M} t_b , \tag{5}$$

and the burn-out velocity, v_b, is obtained by replacing t in equation (4) by t_b or by its equivalent, $M_p/\dot{M}$. The result, for an average $\sin \theta$, i.e., $\overline{\sin \theta}$, is

$$v_b = \beta v_e \ln \frac{1}{1 - \frac{M_p}{M_o}} - gt_b \, \overline{\sin \theta} . \tag{6}$$

The quantity M_p/M_o is usually called the "mass ratio" of the rocket and will be represented here by the symbol h.

From equation (5), the burning time may be written as

$$t_b = \frac{M_p}{\dot{M}} = \frac{M_p}{M_o} \cdot \frac{M_o}{\dot{M}} = h\frac{M_o}{\dot{M}} . \tag{7}$$

If a_o is the initial total acceleration of the rocket missile, the thrust force may be represented by $M_o a_o$. Further, since dm/dt is numerically equivalent to $\beta dM/dt$, it follows from equation (1) that the force can also be represented by $\beta v_e \dot{M}$, i.e.,

$$M_o a_o = \beta v_e \dot{M}$$

or

$$\frac{M_o}{\dot{M}} = \frac{\beta v_e}{a_o} .$$

Upon combining this result with equation (7), it follows that

$$t_b = \frac{h \beta v_e}{a_o} , \qquad (8)$$

and if this is introduced into equation (6), with M_p/M_o replaced by h, the result is the basic relationship

$$v_b = \beta v_e \left(\ln \frac{1}{1-h} - \frac{h}{a_o/g} \overline{\sin \theta} \right) . \qquad (9)$$

It might appear that the burn-out velocity would be increased by making a_o/g, i.e., the initial acceleration, as large as possible. However, as will be seen below, the attainable mass ratio, h, also depends upon a_o/g.

The gross rocket weight, W_o, is made up of the sum of the weights of five important components. For a rocket with ammonia as propellant and a payload of 5,000 lb, these are given by the following relationships:

Propellant	$= w_p = hW_o$
Tankage	$= w_t = 0.02\, w_p$
Equipment (pump, turbine, etc.)	$= w_e = 50 + 0.007\, \dfrac{a_o}{g}\, W_o$
Motor	$w_m = 1750 + 0.018\, \dfrac{a_o}{g}\, W_o$
Payload	$w_l = 5000 ,$

so that

$$W_o = \frac{6800}{1 - 0.025 \, a_o/g - 1.02h} \text{ lb}.$$

(10)

For an exhaust gas temperature of 2,500°C, with ammonia as propellant, the gas velocity is found to be close to 13,300 ft/sec (see below). Using this value for v_e and taking β as 0.94, elimination of h between equations (9) and (10) leads to

$$v_b = 12,500 \left(\ln \frac{1}{0.02 + 0.025 \, a_o/g + 6800/W_o} \right.$$

$$\left. - \frac{0.98 - 0.025 \, a_o/g - 6800/W_o}{a_o/g} \sin \theta \right).$$

(11)

It can be shown from this equation that v_b has a maximum, for a given W_o, when a_o/g is about 2.3. However, more complete calculations indicate that the optimum value of a_o/g lies between 1.5 and 2.0.

The maximum theoretical exhaust velocity, v_m, is given by

$$v_m = \sqrt{\frac{2g\gamma}{\gamma-1} \cdot \frac{R}{MW} T_c} \quad ,$$

where γ is the ratio of the specific heats of the chamber gases at the (absolute) temperature T_c, and R is the universal gas constant, i.e., 1544 (ft)(lb)/(mole)(°R). For ammonia as propellant, taking T_c as 5,000°R, MW as 8.48, and γ as 1.33, it is found that

$$v_m = 15,300 \text{ ft/sec.}$$

The actual exhaust velocity, i.e., v_e, is related to v_m by

$$v_e = \sqrt{1 - \left(\frac{p_e}{p_c}\right)^{(\gamma-1)/\gamma}} \, (v_m \lambda),$$

where p_e/p_c is the ratio of exit gas to chamber pressures and λ is the nozzle efficiency. The value of p_e/p_c is calculated below and found to be 2.42×10^{-3}, and λ is generally taken as 0.985. Using these data, it is found that

$$v_e = 13,300 \text{ ft/sec },$$

which is the result used above.

The reactor (or rocket) power, P, in megawatts, is related to the maximum exhaust velocity and the mass flow rate of the propellant, $\dot{m}$, by

$$P(Mw) = 0.678 \ \beta\dot{m}\left(\frac{v_m}{1000}\right)^2 \ ,$$

and if β is 0.94, it is found that for P = 2,700 Mw, which is postulated for the present calculations,

$$\dot{m} = 18.1 \ \text{slugs/sec}$$

or for the weight flow rate, $\dot{w}$, the result is

$$\dot{w} = 18.1 \times 32.2 = 583 \ \text{lb/sec}.$$

The throat area, A_t, can now be determined from the equation

$$\dot{w} = A_t p_c g \ \gamma \cdot \frac{\sqrt{\left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/(\gamma-1)}}}{v_m \ \sqrt{\frac{\gamma-1}{2}}}$$

where p_c is the chamber pressure, which is expected to be 1,500 psi. Upon introducing this value, it is found that

$$A_t = 91.6 \ \text{in}^2 \ ,$$

which means that the throat radius is 5.4 in. It has been shown that with a chamber pressure of 1500 psi, a nozzle expansion ratio of 25 to 1 may be used without flow separation in the nozzle at sea level. Hence, if A_e is the area of the nozzle exit, it follows that

$$A_e = 25 \ A_t = 2280 \ \text{in}^2 \ .$$

The rocket thrust, F, is related to the throat area and the chamber pressure by

$$F = c_F \ A_t \ p_c \ , \tag{12}$$

where the thrust coefficient, c_F, is defined by

$$c_F = \sqrt{\frac{2\gamma^2}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/(\gamma-1)}\left[1 - \left(\frac{p_e}{p_c}\right)^{(\gamma-1)/\gamma}\right]} + \frac{p_e - p_a}{p_c} \cdot \frac{A_e}{A_t},$$

where p_e is the exit gas pressure and p_a is the ambient pressure. The pressure ratio, p_e/p_c, may be obtained from its relationship to the nozzle area ratio, A_t/A_e, by

$$\frac{A_t}{A_e} = \left(\frac{\gamma+1}{2}\right)^{1/(\gamma-1)}\left(\frac{p_e}{p_c}\right)^{1/\gamma}\sqrt{\frac{\gamma+1}{\gamma-1}\left[1 - \left(\frac{p_e}{p_c}\right)^{(\gamma-1)/\gamma}\right]}.$$

Using the values for A_t/A_e and of γ postulated above, numerical solution of this equation gives

$$\frac{p_e}{p_e} = 413$$

and since p_e is 1500 psi,

$$p_e = 3.63 \text{ psi.}$$

Hence,

$$c_F = 1.75 - 0.167\, p_a\,,$$

where p_a is expressed in psi. Substitution of this result, together with the known values of A_t and p_c, in equation (12) then leads to

$$F \text{ (lb)} = 241{,}000 - 2290\, p_a$$

for the thrust in pounds. At sea level, p_a is 14.7 psi, and so the sea-level thrust, F_o, is

$$F_o = 207{,}000 \text{ lb.}$$

It is of interest to note that the thrust may be calculated with a fair degree of accuracy by means of the semiempirical expression

$$P \text{ (watts)} \approx F \text{ (lb)} \times v_e \text{ (ft/sec)}.$$

In the present case, P is 2700×10^6 watts, v_e is 13,300 ft/sec, so that

$$F \approx \frac{2700 \times 10^6}{1330} \approx 200,000 \text{ lb},$$

in good agreement with the value given above.

Assuming the initial acceleration to be 1.5 times gravity, i.e., $a_0/g = 1.5$, then

$$W_0 = \frac{F}{a_0/g} = 138,000 \text{ lb}.$$

From equation (10), the mass ratio, h, is found to be

$$h = 0.895.$$

Using equation (9), and taking $\overline{\sin \theta}$ to be 0.75, the burn-out velocity is given by

$$v_b = 23,100 \text{ ft/sec},$$

but if allowance is made for 1.5 percent drag loss, the result is

$$v_b = 22,800 \text{ ft/sec}.$$

The relationship of the maximum ballistic (or nonpowered) range, R, in nautical miles to the burn-out velocity, for optimum flight angle θ_b at burn-out, is

$$R \text{ (N.M.)} = 120 \tan^{-1} \frac{\frac{v_b^2}{2R_0 g}}{\sqrt{1 - \frac{v_b^2}{R_0 g}}} \text{ for } \theta_b = \tan^{-1} \sqrt{1 - \frac{v_b^2}{R_0 g}},$$

where R_0 is the radius of the earth. For θ_b of 25° and v_b equal to 22,800 ft/sec, the ballistic range is found to be nearly 4,800 N.M. The powered range is roughly 200 miles, so the total range will then be close to 5,000 N.M.

The characteristics of the ammonia propelled nuclear rocket, with a power of 2,700 Mw, a payload of 5,000 lb, and an initial acceleration of 1.5g, assuming β to be 0.94, are then as follows:

Thrust	=	207,000	lb
Specific thrust (v_e/g)	=	400	sec^{-1}
Gross weight	=	138,000	lb
Propellant	=	123,000	lb
Tankage	=	2,600	lb
Equipment (pump, etc.)	=	1,600	lb
Motor	=	5,500	lb
Payload	=	5,000	lb
Burning time	=	211	sec
Propellant flow rate	=	584	lb/sec
Total range	=	5,000	N.M.

Reasonable dimensions for the complete vehicle would be a length of 75 ft and a diameter of about 9 ft. It may be mentioned that a hydrogen propelled rocket designed to achieve the same mission would be appreciably larger although it would weigh less.

Although the methods of calculation, which can be found in the standard texts, are not given, it is of interest to present the results in Fig. 11. These show the changes in various quantities during the powered portion of the flight of a 2,700 Mw nuclear rocket using ammonia as propellant. The postulated steering program is a vertical flight for 30 sec, followed by a powered turn to an angle of 80° with the horizontal with gravity operating for the remainder of the flight. The initial acceleration, i.e., a_o/g, is taken to be 1.5, so that the actual (net) vertical acceleration is 0.5 g, i.e., 16 ft/sec^2. The powered range, preceding the ballistic range, is seen to be about 10^6 ft, i.e., nearly 200 miles.

For reviews of nuclear rocket propulsion and related matters, the following references may be consulted:

NA-47-15	LAMS-1983
TID-2011, pp 79-170	LAMS-2021
LAMS-1870	LAMS-2036

Fig. 11