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NUCLEAR-POWERED FLIGHT

A Report to
The Atomic Energy Commission

by

The Lexington Project

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ERRATA

Correction to Appendix D of Lexington Project Report on "Nuclear-Powered Flight" dated September 30, 1948:

Certain material in Appendix D dealing with North American Aviation was incorrectly included under Project Rand. The following correction should be made:

On Pg. D-4, the first full paragraph should be transferred to Pg. D-3 at the end of the section dealing with "North American Aviation, Inc."

On Pgs. D-7 and D-8, under "Project Rand," items 1 through 11 and 16 through 18 should be credited to North American Aviation rather than to Project Rand.

AEC, Oak Ridge, Tenn., 2-7-49--253-A1790

NUCLEAR-POWERED FLIGHT

A Report to

THE ATOMIC ENERGY COMMISSION

by

THE LEXINGTON PROJECT

September 30, 1948

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Most of the charts and a few of the long tables appear at the end of the chapter to which they apply.

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Chapter I

NUCLEAR-POWERED FLIGHT : A SUMMARY

The Lexington Project, a group assembled by the Massachusetts Institute of Technology to evaluate the possibilities of nuclear-powered flight for the Atomic Energy Commission, has spent the summer of 1948 in reviewing existing knowledge of the subject and in appraising the problems to be solved and the prospects for success.

This chapter presents a summary of its report; subsequent chapters deal in more detail with the need and use for nuclear-powered aircraft, with the power plant, with the reactor and shield, with the airplane, with some problems of full-scale testing, and with performance possibilities.

The conclusions and recommendations resulting from this work are as follows:

CONCLUSIONS

1. Although success cannot be guaranteed, there is a strong probability that some version of nuclear-powered flight can be achieved if adequate resources and competent manpower are put into the development.

2. The resulting nuclear-powered plane would be specifically a military instrument for use in bombing and reconnaissance of targets which are beyond the reach of conventionally powered aircraft. The following characteristics of the plane seem achievable in the light of present information, although it is quite conceivable that they can be improved by developments now under way or not yet foreseen:

(a) An extreme range, e.g., around the world.

(b) A speed approaching the velocity of sound -- probably 600 miles per hour.

(c) An operating altitude of at least 30,000 feet, but probably not much above 50,000 feet.

(d) A content of fissionable material in its power plant of not over 200 pounds and possibly as low as 20 pounds (calculated as uranium 235). Because of the scarcity of fissionable material it is unlikely that many nuclear-powered planes will be built.

3. This airplane may take one of the following forms:

(a) A manned nuclear-powered plane. Such a plane would have to be very large, perhaps larger than anything yet constructed, in order to carry the heavy shield which is needed to protect

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the crew from the radiations of the nuclear reactor. Its size may be prohibitive, but if much more efficient shields can be developed it might well fall in the range of very heavy bombers.

(b) An uninhabited nuclear-powered plane, which tows at some distance behind it a small plane carrying the crew, the payload and the controls -- designated as a "tug-tow" arrangement. The shield in the nuclear-powered tug would be much lighter than that needed for the reactor of a manned plane, and the size of the tug would fall within the range of existing bombers. The towed plane which contains the crew would have a conventionally fueled power plant to permit independent operation at take-off and landing and, if desired, for short periods in flight. The tug-tow combination currently appears to be the most promising prospect.

(c) A pilotless aircraft, i.e., an uninhabited plane which is controlled only by a long-range guidance system. It is questionable whether such a pilotless nuclear-powered aircraft, if it could be developed, would show sufficient advantage over a similar device with chemical fuel to be of interest.

4. Several types of power plant seem promising for nuclear-powered airplanes. The best appear to be variations of the turbojet. The reactor may transfer its heat to air, to a liquid metal, to high pressure helium or possibly to some other fluid. It would be premature at this time to decide that one of these combinations of power plant and working fluid is necessarily the best and to "freeze" on it as the basis for further developments. There is some doubt that the air-cooled thermal reactor, which has received most attention to date, will prove to be the most suitable. The ramjet and the rocket are more remote possibilities which do not call for intensive study at present.

5. Development of the reactor is the outstanding problem because of the rigorous requirements for small size, high power, high temperature and reasonably small content of fissionable material. No materials which will assure that a reactor can be built to meet these requirements are yet available. Reactor materials development, combined with ingenuity in reactor design, is therefore the most critical need of the program.

6. There is a considerable margin of uncertainty as to the weight of shield which is required for protection. This uncertainty needs to be resolved because of the dominant significance of shield weights.

7. Many power plant components will require extensive development work, but the problems do not seem insuperable. The airframe itself will be a comparatively straightforward problem unless a decision to build manned planes entails going to tremendous gross weights.

8. Full-scale testing of the nuclear power plant and the testing and operation of the final airplane will be unusually hazardous and will require isolated sites.

9. A really intensive effort will be needed if a nuclear-powered airplane is to fly within fifteen years. The complete development is likely to cost well over a billion dollars, with the major expense coming in the later years.

10. Integration of work on reactors, power plants, materials and other components is essential for efficient progress towards the goal.

11. A vigorous and realistic aircraft reactor development program during the next few years should contribute to and benefit from other aspects of the Reactor Development program of the Atomic Energy Commission.

RECOMMENDATIONS

If it is decided that as a national policy the high cost in technical manpower, fissionable material and money can be justified, a strong development program on nuclear-powered flight should be undertaken.

The immediate objective of the program should be to establish a sound basis for making major design choices and for determining feasibility. The initial program might involve government expenditure of 150 to 200 million dollars over a three to five-year period. This would include both construction costs for new reactors and associated facilities to provide information needed to design a prototype aircraft reactor, and annual operating expenditures by the government agencies involved of perhaps 20 million dollars. Much of this sum could fairly be regarded as a part of the overall costs of nuclear reactor development.

After an appropriate interval -- perhaps three years -- and prior to heavy commitments which would be required for full-scale test construction, progress should be thoroughly evaluated to determine whether feasibility and military interest then are sufficient to warrant continuance.

The initial program should include the following:

1. Research on materials, which should be given high priority. This should emphasize determination of pertinent properties of the more promising materials for structures, moderators, coatings, fuels, shields and controls under high temperatures, large temperature gradients, chemical attack and ultimately under high radiation fluxes. A major part of the effort would be the development of new materials or combinations which are superior to those now available.

2. Experimental and theoretical studies on shielding, to determine the minimum thickness and weight for an effective shield and thereby to reduce the uncertainties to a point where they no longer introduce large variations in the estimates of required airplane size. The shielding program should include more refined measurements of the radiations transmitted by present shields, expansion of the current studies on new

shielding materials, use of unique facilities for shielding research at reactors now under construction, further study of the effects of ducts and perforations through shields, and more investigation of the sensitivity of components such as electronic equipment. It should be regarded as a basic problem in the development of mobile power units, which can be reasonably well advanced within the next few years.

3. A series of studies on reactors, power plants, the airplane and full-scale testing problems, as listed below. These studies should cover a wide enough range of types and conditions to ensure that they include the optimum ones, and should aim to narrow this range as rapidly as can be done with assurance.

(a) Preliminary design studies of aircraft-type reactors, which would call for some new physical measurements on materials and would lead towards criticality experiments.

(b) Analytical studies of power cycles and of types of propulsion.

(c) Exploratory airplane designs, to the extent needed to ensure that the direction of effort on reactors and power plants is consistent with aircraft potentialities.

(d) An early study of the major problems to be encountered in testing the full-scale power plant and in testing, operating and repairing nuclear-powered aircraft.

4. An appraisal of the military usefulness of the tug-tow scheme, which seems to offer such a major advantage by reducing weight and size that its acceptability would greatly enhance the prospects of flying with nuclear power. If the results of the scrutiny are favorable, an experimental investigation of its operational feasibility should be instituted at once, using existing planes.

The initial program should make every practicable use of existing reactors for materials studies and shielding experiments. The major requirement of such work is adequate technical manpower to plan and carry through programs which must be coordinated with the primary functions of these reactors.

As part of the Reactor Development program of the Atomic Energy Commission, one or more high-flux reactors should be constructed as quickly as possible. The objective should be a reactor which is capable of simulating the fluxes and temperatures which are anticipated in subsequent power units. Such a reactor should be designed and operated for the specific purpose of developing general technology and experience with power-type units and for more advanced studies of materials and structures which have been indicated to be promising. Its design and operation should not be handicapped by including other objectives which are not pertinent to power

development. It may well prove desirable to start with a low-temperature high-flux reactor which has provision for controlled high temperature at test locations, because such a reactor should be easier to design, build and operate than a high-temperature one and should be a more flexible instrument. A truly high-temperature high-flux reactor should come along as soon as possible. These reactors will be more useful if they do not have to be built at isolated sites.

- - - - - - - - - - - - -

INTRODUCTION

Scope of the Project

Under the date of May 21, 1948, the Atomic Energy Commission and the Massachusetts Institute of Technology entered into a contract by which the latter agreed to make an evaluation of the possibilities of the nuclear propulsion of aircraft. The scope of the study is given by the following quotation from the contract:

> "The Contractor hereby agrees to use its best efforts to furnish or obtain the necessary personnel and facilities and do all things necessary to perform with the utmost dispatch the following:
>
> "(a) Conduct a review of work which has been done or which is being done in the United States relevant to the propulsion of aircraft by nuclear energy
>
> "(b) Evaluate the possibilities of nuclear propulsion of aircraft from a broad scientific and technical standpoint
>
> "(c) Identify the important elements which should constitute a program for accomplishing the application of nuclear energy to the purpose of aircraft propulsion
>
> "(d) Estimate a time scale or schedule for accomplishment of the application of nuclear energy to the purpose of aircraft propulsion
>
> "(e) Make such other studies and reviews regarding nuclear propulsion of aircraft as may be agreed upon by the parties from time to time
>
> "(f) Prepare and submit to the Commission in the shortest possible time after performance of the studies and investigations called for by this contract a report summarizing the Contractor's activities, findings and conclusions."

This report has been prepared to fulfill the obligation undertaken under the last paragraph.

The Organization

The organization set up to carry out this assignment was known as the Lexington Project and consisted of forty-five professional members. Thirteen of these members were consultants; over half devoted their entire time to the Project during the summer of 1948. Twenty-two were Institute

staff members; the rest were drawn from various universities, institutes, and commercial organizations. A majority had had previous experience with the Atomic Energy Commission or the Manhattan District. A complete list is given in Appendix A.

Sources of Information

In carrying out the work, visits were made by one or more of the group to a number of locations, as specified in Appendix B. In many cases persons believed to have information which would be helpful have visited the Project at the request of the group. Appendix B includes a list of these visits.

Most of those who contributed information were interested in components of the problem or in related problems rather than in nuclear-powered flight itself. However, NEPA was set up specifically for studies in nuclear-powered flight and several other agencies, notably RAND, North American Aviation, the National Advisory Committee for Aeronautics and the Applied Physics Laboratory of the Johns Hopkins University, have considered this problem specifically. In no case, however, has a practical design been developed which could now be built.

The institutions and persons consulted have been generous in their time and energy in making available the information needed by the Project and their invaluable assistance is gratefully acknowledged.

Besides the program of visits, a large number of reports from groups both inside and outside of the AEC have been consulted.

The group has not confined itself to the review of the work of others but has carried out on its own account a fairly extensive program of study and computation.

Limitations of the Study

So far as the survey itself is concerned, an attempt has been made to determine as accurately as possible the prospects of realizing each type of nuclear-powered aircraft, with no restriction as to civilian or military, inhabited or guided, slow or fast. One exception to this rule refers to lighter-than-air craft; no interest in this field was uncovered, and no known studies are under way.

It is not considered the function of the Project to assess the relative merits of nuclear-powered aircraft as against other military programs. An attempt has been made to furnish the basis for a decision by the armed forces, by indicating as well as possible the probable characteristics of nuclear-powered aircraft, the cost in time and money, and the probability of success.

Further, while an important element of the task of the Project is to recommend programs of work, no recommendations have been made as to specific organizations for carrying out the programs. It is evident,

however, that the administrative coordination and the competence and enthusiasm of the people who are selected to do this job must be of distinctly superior quality.

Arrangement of the Report

The remainder of this chapter presents a general survey of the Project's findings and conclusions; subsequent chapters give supporting details for the various parts of the investigation.

THE AIRPLANE

All the aircraft considered in this study (except rockets, which are discussed later) are airplanes. The airplane is considered at length in Chapter V; a short summary of certain airplane characteristics is given at this point in order to lay a foundation for the subsequent analysis.

The Relation of Weight and Thrust

An important relationship connects the thrust of the power plant with the speed of the plane and the total weight which will be sustained in the air. This relation is very different below and above the speed of sound (which is about 760 miles per hour at sea level and drops to about 660 miles per hour at 37,000 feet and above). Below the speed of sound each pound of thrust will support, in a well-designed plane, from 15 to 20 pounds of weight, and this is true almost regardless of altitude. The figure can be raised somewhat by reducing protuberances, making the plane as nearly as possible all wing, increasing the wing span, and similar devices, but the best existing airplanes lie in the above range. The future may bring developments which will make a higher ratio practicable.

Above the speed of sound a pound of thrust will not support nearly so much weight; hence the chance for realizing a successful nuclear-powered plane is much less. There are only a very few planes as yet which fly above the speed of sound, and in these only about three pounds are supported per pound of thrust. In this new field relatively rapid improvements are being made, but the figures for subsonic flight probably will never be matched in supersonic flight.

Ratio of Structural to Total Weight

Another important relationship is the fraction of the total loaded weight of an airplane devoted to the structure (exclusive of power plant, fuel, crew, armament, payload, and accessories). If chemically powered (subsonic) airplanes are examined, this fraction will be found to lie in a rather narrow range around 1/3. At very low design speeds or very high design altitudes the fraction tends to increase. It is expected that nuclear-powered planes would follow this relationship.

The Effect of Altitude

The chief effect of high altitude on the airframe design is to increase the wing area which is required to carry a given weight. The structural weight by no means increases in proportion, but it does increase somewhat. High altitude also affects power plants materially, as will be brought out later.

Range

The great deficiency in conventionally powered airplanes, from a military point of view, is the impossibility of combining speed with extreme range. The only aircraft which could bomb the interior of Russia from bases in the United States and return would fly so slowly that they would have little chance against the defenses which they would meet. Very fast planes are available, but their fuel consumption is so high that they can not approach the extreme range desired.

It is this fact that makes the nuclear-powered aircraft interesting. The energy in the nuclear fuel is so great that the weight of the fuel consumed is negligible; consequently if a nuclear-powered plane can fly at all, it can be expected to have a range greater than the longest for which there is any evident military use.

Landing Speeds and Airstrip Length

Up to one-half of the total weight of a conventionally powered airplane is fuel. Consequently, when the plane is landed its weight is much less than when it took off. With the nuclear-powered airplane this is not the case, because the weight of nuclear fuel consumed is negligible compared with the airplane weight. Landing speeds with nuclear-powered aircraft will therefore have to be relatively high; this will increase the hazard of landing. (From this point of view an airplane designed for high altitudes would have an advantage, because such an airplane would have a large wing area, and therefore at low altitudes would support the weight at lower speeds.)

The high landing speeds will call for long airstrips. Extra thrust must be provided at take-off (jet or other assist) to keep down the airstrip length requirement at take-off to the same value as that required for landing.

Size Limitations

The largest military airplane in regular use, the B-36, weighs about 300,000 pounds when fully loaded. The B-29, as another example, weighs about 150,000 pounds in its heaviest form. It is quite certain that the limit on airplane size has not been reached. It is the cost of construction that has deterred the building of much larger planes, particularly as the advantage in still larger chemically powered airplanes is small. The ultimate limit is not known; but it seems reasonable that airplanes weighing over 600,000 pounds fully loaded could be built and flown.

Airframes for Nuclear-Powered Aircraft

The characteristics of an airplane are not greatly affected by using nuclear power. It is true that in such an airplane the load is more concentrated and that landing loads and speeds are increased, but the effect of these changes on the structure does not present any problems that cannot be solved through more or less conventional aircraft engineering procedures. Hence the general characteristics of a chemically powered airplane are applicable to the nuclear-powered airplane. Aircraft development is proceeding actively, particularly in the supersonic region, quite independently of atomic energy work. Hence if a decision is reached to proceed with nuclear-propelled aircraft, no early work specifically directed toward the development of airframes for nuclear power need be undertaken. This does not mean that the final design of the airframe would necessarily be an easy task, but it does mean that if and when the nuclear power plant has been developed to a point where its dimensions, weight, and thrust can be definitely specified, design of the airframe could be started with assurance that it will not be a bottleneck.

However, in the development of other elements, the power plant in particular, the ultimate use in an airplane must be kept in mind, so that something compatible with a possible and efficient airframe configuration and structure results. For this reason a development program must include, from the beginning, studies of complete airplanes as they would be built up from the element under development. The intensity of these studies would increase when a clear idea of the design of the power plant was available.

SHIELDING

By far the heaviest element to be carried by the nuclear-powered airplane is the shield which surrounds the nuclear reactor to protect personnel and other parts of the unit from the intense radiation which it emits. (The reactor or "pile" is the structure containing uranium or other fissionable material, in which the nuclear energy is released.) This heavy shield is a fundamental element which affects the whole picture.

What the Shield Protects Against

Of the numerous types of damaging radiation emitted by a pile in operation, the fast neutrons and the penetrating ("hard") gamma rays are the ones which have to be considered in shielding, since they are the most penetrating. The intensities of these components will vary with the energy output of the pile, and various pile designs will also have somewhat different distributions among the various radiations. In general, however, the problem can be simplified for the purpose of estimating required shield thickness by assuming that an aircraft reactor will liberate about 100,000 kilowatts of heat and that there will be some average distribution of the dangerous radiations for which shielding must be provided.

Composition and Thickness of Shield

It is not possible at present to give a satisfactory solution to the problem of the least weight of shield which will reduce the radiation of a given pile by a specified amount. This problem is not important in stationary piles, and hence has not received much attention. It is not an easy problem. The best materials for stopping neutrons are not the most efficient ones for hard gamma rays; and neutrons, on being stopped, generally produce gamma rays, which have to be stopped in turn. The best shield will therefore be composed of more than one chemical element. The fast-neutron-stopping properties of various elements are not fully known. Thus at the present time the composition and efficiency of an optimum shield cannot be stated.

Shielding the Manned Airplane

A thick shield is required for the manned airplane. For thick shields the density is of considerable importance. In the case of manned aircraft it may pay to go to extremes, even using scarce and expensive materials to attain high density. Keeping the size of the reactor small is also highly important when thick shields are used.

The most sensitive large element in the manned airplane is the crew (photographic film is more sensitive, but can be separately shielded). The tolerance of man is understood well enough for present purposes; the major uncertainty is really a policy matter as to the amount of injury permissible. Even so, the uncertainty in the amount of radiation which the crew may be permitted to receive is small when expressed in terms of shield thickness. There is also some uncertainty as to how far the crew can be placed from the pile, but the assumption of 33 feet which is made in this study seems reasonable and is not critical. In this case the reduction (attenuation) in the pile radiation must be about 100,000,000-fold. At specific gravity 6 this attenuation can be obtained with a known shield at a thickness of 4 feet, and if the same or equivalent atomic composition could be obtained at specific gravity 10, the thickness could be reduced to 2.4 feet. Further study may reduce the value of the product of density and thickness which is needed.

A simple calculation shows that the nuclear-powered manned airplane will be large. Assuming that the nuclear reactor is a sphere four feet in diameter, the known shield of specific gravity 6 would weigh 320,000 pounds. If the specific gravity could be increased to 10, and the atomic composition were unchanged, the shield weight would be 195,000 pounds. The first of these shields implies an extremely large aircraft, but the second might be flyable in a plane which is in the range of very heavy bombers. Since the reactor assumed here probably is larger than necessary, and since the product of density and thickness may be decreased by future work, a manned aircraft of reasonable size may become feasible.

The importance of reactor size may be seen by computing the weight of shield at specific gravity 10 but for a larger reacting sphere. For a five-foot sphere the weight is 255,000 pounds.

It might be thought that the shield need be applied only to the side of the pile toward the crew. However, radiations from the pile, particularly the neutrons, do not travel in straight lines, but are deflected and scattered by the air surrounding the plane. While some decrease in total shield weight can be obtained by making the shield thicker on one side, it is unlikely that the saving will exceed 10%; and ducts in the shield are likely to cause a greater increase in weight.

Shielding the Tug-Tow Combination

There would be great advantages in some arrangement by which the shield could be kept light and a crew could be carried with the aircraft. Great savings in shield weight can be obtained by moving the components which are sensitive to radiation to a considerable distance from the reactor. This can be achieved by putting the power plant into one airplane (the tug) and towing the crew and other sensitive components behind it in another airplane (the tow). The tug plane is flown by the crew in the towed plane with remote control.

A towing cable some 3,000 feet long might be used, and calculations indicate that its drag is not excessive.

At a 3,000-foot separation of tug and tow, the thickness of shield required is less than half and probably about a third of that required for the manned aircraft. The estimates for the tug-tow or for any other system requiring thin shields only are not so accurate as those for the manned aircraft. The results are more sensitive to errors in the fundamental data, to details in the methods of calculation, and also to modifications due to shadow shielding and ducting problems and to possible expedients which would be of no importance for the manned plane.

The reduction in shield weight for the tug-tow system is very considerable even under pessimistic assumptions. At half the thickness needed for the manned plane, and again assuming a four-foot sphere as the reactor, the shield weight for material of density 6 is 85,000 pounds. This weight is a more pessimistic estimate for the tug-tow than the corresponding estimate of 320,000 pounds for the manned plane.

Shielding the Pilotless Plane

Shielding can not be eliminated from a pilotless plane. The estimates of required shield thickness are the least certain for this type of aircraft. In addition to the general uncertainty about thin shields, further uncertainty arises here in the evaluation of the radiation tolerances of the sensitive components, payload, guidance, and electronic control equipment. Indeed the possibility of finding less sensitive components to substitute for those considered has hardly been explored.

The most probable figures indicate that somewhat more shielding is required in the pilotless plane than in the tug-tow. This extra shielding requirement arises because the sensitive components cannot be far removed from the reactor in the pilotless plane.

General Comment on Shielding

This discussion of shielding brings out four points:

(a) For thick shields the density should be as high as possible for a given atomic composition.

(b) It is highly important to keep the volume of the nuclear reactor to a minimum, in order to reduce shield weight.

(c) A thorough study of shielding, to resolve the outstanding uncertainty, is essential to a full understanding of the feasibility of the project and of the scale of operations which it implies. Fortunately, reasonable progress could be expected as the result of a planned program in a few years.

(d) Very serious consideration should be given to the tug-tow arrangement. The next section gives some further comments on this point.

PILOTLESS AND TUG-TOW AIRCRAFT -- FURTHER DISCUSSION

The Pilotless Aircraft

An intensive development of long-range guidance is under way at a number of centers in this country. Though workers in this field are optimistic, no such system is now available, or even clearly in sight. If and when an adequate system is developed, it will doubtless be equally applicable to chemically powered aircraft. For one-way missions such aircraft should have adequate range at high speed, and the main advantage of nuclear-powered aircraft, their great range, would not be needed. Such one-way bombing might, however, still depend upon reconnaissance by piloted two-way flights of nuclear-propelled aircraft.

The Tug-Tow Airplane

There are many possible variations in the general tug-tow scheme, but the following appears to be a suitable form.

The towed airplane would have conventional engines (probably jet engines) and a limited supply of fuel. It would contain the crew, the payload (atomic bomb or reconnaissance equipment) and radar. The nuclear-powered tug plane would take off separately under radio guidance from the ground, trailing the cable. The small plane would take off under its own power, hook on, take over the guidance of the tug plane and shut down its own engines. Over the target the towed plane might employ some fuel to increase the speed or altitude of the combination. On landing, the procedure on take-off would be reversed; or in view of the great range of the tug plane, it might not even be landed, a second towed plane with a new crew and payload being attached after the first had been detached.

The proposed towed plane might weigh 40,000 pounds. Of this, however, a large part is accounted for by the crew and payload, which would have to be carried in any case. On the whole, the increased power required by the combination over that of the pilotless version is moderate. The towed plane would be kept some hundreds of feet below the tug plane to keep out of its wake.

It should be noted that the elements of this proposal have already been employed in practice. Remote control of planes has been widely used -- for example, at Bikini. Towing of gliders is standard, and refueling in flight is being practised. It is further of importance that the scheme can be studied rather fully with the aid of conventionally powered planes, so that little time need be lost in developing a towing system after a nuclear power plant is available.

In view of the very great saving in plane size, and (as will be noted later) the greater feasibility, the early study and flight testing of the towing operation seems important to a program looking to the development of nuclear-powered aircraft.

NUCLEAR REACTORS FOR AIRCRAFT

In present airplanes the thrust required for propulsion comes from the heat produced through combustion of the fuel in air. In the propeller engine this heat is converted into mechanical work which drives the propeller; the propeller in turn accelerates a gas (air) rearward of the airplane and so produces the forward thrust. In the jet engine the combustion gases themselves are accelerated rearward through the engine nozzle and so produce the forward thrust.

In the present study all proposals which have been uncovered for applying nuclear energy to aircraft work on these same general principles, i.e., the nuclear energy produces heat which is either converted into mechanical work to drive a propeller or is added to a gas (air as in the jet engine or, say, hydrogen as in the rocket engine) to accelerate gas rearward of the airplane. All practicable proposals for producing work from nuclear energy involve, therefore, the production of heat and the transfer of this heat to some working fluid in one of the various forms of heat engine.

Thus the problem of a pile for an airplane involves the development of a scheme for removing heat at a high temperature from a small volume (to keep down shield weight) with a minimum weight of machinery, by means of a gas or a liquid. It will also be necessary to keep the products of nuclear fission from escaping from the pile.

Desirable Reactor Qualities from Heat-Transfer Considerations

In order to remove large amounts of heat from a reactor one would ask for a great number of passages of small diameter, to secure the greatest possible heat-transfer area. There should be a high temperature differential between the wall and the coolant fluid. A high-pressure gas is a better heat remover than a low pressure gas, if the

pressure drop is to be kept small with passages of a given diameter and length. Liquids are still better in this respect, with the liquid metals best of all.

Fissionable Material Considerations

In the computations made in this study, the fissionable material (fuel) assumed is uranium 235, enriched to at least 50% concentration. Natural uranium, with only 0.7% of uranium 235, becomes chain-reacting only in piles which are too bulky for an airplane. If uranium 233 becomes available later, it can presumably be used and somewhat smaller amounts may be required. If plutonium should come into the picture, its use also could reduce the amount of fuel required for certain types of reactors, perhaps to a considerable degree. It does not appear that either of the alternatives would change the situation substantially, other than as to fuel inventory required. Studies on plutonium would be difficult because of the scarcity and highly poisonous nature of the material. The use of plutonium would require knowledge of the physical and chemical properties of its compounds at high temperature, particularly compounds with oxygen or carbon. Not much has been done in this field so far, but it is to be expected that the normal course of studies on breeding and power plant economics in general will involve such studies, and therefore no special program directed at the airplane problem is called for. For uranium 233 and 235 any chemical information of this sort not now available can be obtained from studies on ordinary uranium.

In order to keep down the inventory of nuclear fuel there are several desirable reactor characteristics. In the first place, the materials employed in a thermal reactor should be of low molecular weight and have little tendency to absorb neutrons; in addition, they must have the necessary mechanical strength at the operating temperature. The most desirable materials are, in order of nuclear effectiveness: highly hydrogenous materials such as hydrides; beryllium metal, oxide, or carbide; and graphite. If other materials must be employed, they should be selected from among the elements with low neutron absorption. High density materials are preferable to voids filled with gas; and for this reason liquid-cooled piles have an advantage. In the case of thermal reactors, for each selection of materials and percentage of voids there is a certain reactor size corresponding to minimum fuel. This size for minimum fuel inventory is quite often larger than the reactor size which would be needed for heat transfer, so that as the reactor becomes larger, the fuel inventory decreases. A reflector around the reactor, made of low molecular weight material but containing no fuel, also decreases the fuel inventory but increases the shield size and weight.

If suitable materials of low absorption can not be found, it may still be possible to produce a successful "fast" reactor, which is much less sensitive to the nuclear properties of materials. In this case there is no minimum in the size-inventory curve, and it will be important to keep the size as low as heat-transfer requirements will allow if the fuel inventory is not to be excessive.

In addition to the fuel which must be present to make the chain reaction initially self-sustaining, there must be incorporated in the pile sufficient extra fuel to maintain reactivity during operation as the fuel is partially consumed and neutron-absorbing fission products accumulate. This amounts to less than one pound per day at full power for the aircraft reactors. It is not likely that the number of missions to be carried out by one aircraft before changing power plants will be very large. Accordingly, the fuel consumed will be a small part of the inventory.

OPEN-CYCLE POWER PLANTS

Power plants in which the reactor heat is removed by air drawn from the surroundings, passed through the reactor, and rejected to the atmosphere will be referred to as open-cycle power plants. In other types of power plants the heat-removing fluid is recycled to the reactor; they will be called closed-cycle power plants. Each type includes forms employing propellers, jets and ramjets.

The Turbojet in General

Most of those who have studied the nuclear propulsion of aircraft, particularly NEPA, have chosen the open-cycle turbojet as the power plant for intensive study. The choice was based chiefly on the simplicity and low vulnerability of this type, and the prospect of extending turbojet development to the ramjet and to the rocket. The reasons, it should be noted, have little to do with the feasibility of attaining a successful power plant, and it is not likely that this type will be easy to realize. In this discussion the open cycle will be considered first because of its simplicity. Much of what follows is applicable to other cycles as well.

A schematic diagram of the working of the ordinary turbojet for chemical fuels is shown in Figure I-1. Air enters the compressor and after compression to about four times its original pressure is heated in the combustion chamber by the injection of liquid fuel. The hot gases enter the turbine, where enough power is extracted to operate the compressor, which is on the same shaft as the turbine. From the turbine the gases, still highly heated, are ejected to the rear, and so furnish the thrust to drive the aircraft.

The Compressor

So far as the diagram goes, the turbojet becomes nuclear powered by replacing the combustion chamber with a reactor, through which the compressed air is passed to be heated. A very preliminary study shows, however, that a compression pressure ratio of four to one is far too low for the nuclear-power case, since the volume of air to be heated could not be forced through passages containing the required heat-transfer area without either an excessive power consumption or an excessive reactor size. While it is not possible to state an absolute lower limit, a compression pressure ratio of 20 to 1 is near the minimum.

Airplane-type compressors of the desired pressure ratio do not exist at the present time, though some work on the development of such compressors is now in progress. The development of suitable compressors of the

necessary high efficiency would undoubtedly be costly, but there seems to be no reason to question the ultimate success of such a program.

The high compression ratio which is required normally calls for a division into two units with an intercooler between. This will be true for the nuclear-powered turbojet, except possibly for pressure ratios near the minimum. The design of the intercooler so that it does not introduce excessive weight and can be installed in a practicable plane without introducing excessive air resistance will not be easy. Studies of this matter indicate that it is not impossible.

The Turbine

The efficiency with which heat is converted into work will depend primarily on the temperature of the air entering the turbine of the turbojet. Studies of the open cycle indicate that for a successful aircraft design the lower limit for this temperature will be between 1500 and 1600°F, the highest temperature which is usable in jet turbines now in operation. Such turbines are not in general use, but their feasibility has been established. There is an intensive program in the aircraft and other industries for the development of materials for still higher gas temperatures, and it is to be expected that within a few years temperatures 100 to 300°F higher may be allowable. Besides the new materials, designs which permit the cooling of the blades are under study. Thus the availability of a suitable turbine for the nuclear turbojet may be regarded as assured.

The above statements are based on a short life of the turbine, e.g., 100 hours. Longer times would call for lower temperatures; but the completion of a few missions (of 24 hours each) is regarded as a sufficient goal for the early stages of the development.

The Reactor

The reactor for the open-cycle turbojet must deliver air at a temperature of at least 1500°F. In order to transfer the heat effectively, the temperature of the surfaces at which the transfer of heat to the air takes place must be higher. Calculation shows that the temperature difference must be 100 to 200°F if the reactor is to be of reasonable size and the pressure drop kept within bounds.

By the use of a large number of sufficiently small channels, it would theoretically be possible to transfer the heat to the air in a reactor of very small size -- so small that it would have to be composed almost of pure fuel. Such a reactor, however, besides requiring excessive amounts of fuel, could not satisfy the other requirements for a successful design. In particular, it is absolutely essential that the products of fission of the fuel be retained in the reactor; if they escape into the air stream their radiation will nullify the effect of shielding. In the manned airplane, calculation indicates that the escape of over one part per million of the fission products cannot be tolerated, and even for the tug-tow case the amount cannot be over 50 parts per million.

A practical design will probably have to be based on the use of one of the moderators mentioned in a preceding section. No hydride is known to be sufficiently stable. The other substances are stable, and do not melt even at higher temperatures than are under discussion. Carbon and beryllium, however, are attacked by oxygen, and so probably is beryllium carbide. For this reason, and to retain fission products, the moderator must be coated with some impervious layer, or possibly the air can be confined in metal tubes, as in a fire-tube boiler. In any case, the coatings or tubes would be neutron absorbers to some degree, and therefore would increase the fuel inventory.

The high rates of heat release imply either materials of very good conductivity, or a large temperature gradient, to allow for the escape of the heat generated. Large temperature gradients will result in differential expansion in the conducting material; and the resulting stresses may be a limiting factor in the use of brittle materials of low conductivity.

A further point is that certain substances are known to change in mechanical strength and thermal conductivity when exposed to large numbers of neutrons, and still more when bombarded by products of fission. This effect is still incompletely understood and must be evaluated before a material can be considered satisfactory for use.

At the present time, no coatings or tubes with good mechanical and nuclear properties are known which will withstand temperatures of 1700-2000°F and be absolutely impervious, mechanically stable, not attacked by air or moisture and not eroded by high-velocity air. The development of such materials must be expected to be a difficult task. The materials problem is the critical one for the reactor, and in fact for the whole project, and an intensive effort on materials research and development will be required before nuclear-powered flight is attained.

Ducting Through the Shield

The necessity for passing air through the reactor complicates the shielding problem. It is evident that, in the manned plane especially, straight channels are out of the question, as they would furnish paths for the escape of far more radiation than can be tolerated. The channels must therefore be tortuous, and the total weight and complication of shielding must be somewhat increased.

The Complete Aircraft with Open-Cycle Power Plant

In view of the numerous uncertainties as to the attainable material properties, component efficiencies and other factors, it is impossible to present any single coherent design which will represent the open-cycle nuclear-powered aircraft of the future. A study of the effect of many of the factors appears in Chapter VII. All that can be done here is to present some general results, based on the assumption that the problems discussed in previous paragraphs have been adequately solved.

In the open-cycle turbojet, the size of reactor required from heat-transfer considerations is large enough so that fuel inventory is not usually a controlling factor. Quantities from 50 to 200 pounds of uranium 235 may be expected. These are somewhat arbitrary figures, however, since they imply that suitable materials of construction will be found which are not strong neutron absorbers.

The following comments on the aircraft with open-cycle power plant apply also to the plane with a closed-cycle power plant.

In the first place a manned airplane will necessarily be very large. The size will be greatly affected by the eventual success in reducing shield thickness, but in the most optimistic cases the aircraft will lie in the upper size range of existing planes. With less favorable assumptions the size rises to millions of pounds, or no flyable airplane whatever can be designed.

The unmanned and tug-tow planes are, on reasonably favorable assumptions, in the range of existing large planes.

In the second place, speeds above the velocity of sound appear very remote. It is certainly not possible to say that they will never be attained by nuclear-powered aircraft, but even the tug-tow plane, in large sizes, and on favorable assumptions, will require a lift per pound of thrust much greater than that now available, though perhaps not beyond a reasonable goal.

Effect of Altitude

Most of the computations on which these statements have been based assume a design altitude of 30,000 feet. As the design altitude is raised, the machinery becomes bulkier and heavier, the wing area increases, and it becomes more difficult to force the less dense air through the reactor. Hence the feasibility declines rather rapidly. It is not possible to define the maximum possible altitude, but it is unlikely to be above 50,000 feet for the open cycle.

The Ramjet

The ramjet resembles the turbojet, except that there is no compressor, the aircraft speed being so great that the necessary compression occurs by impact. The compressor development, therefore, drops out, but because of the high temperatures required (3000°F) to get satisfactory performance, the critical materials problem becomes more severe; and the high speed reduces the lift-thrust ratio. It seems practically certain that the nuclear turbojet is easier to realize than the nuclear ramjet, and accordingly no work on ramjets (except perhaps a small amount of paper study) appears warranted at the present time.

CLOSED-CYCLE POWER PLANTS

In all the closed-cycle power plants the reactor coolant is retained in the system, being alternately heated and cooled. One example

of such a system is the ordinary condensing steam power plant. In a nuclear-powered plane we can imagine an analogous system. The reactor would correspond to the boiler. There would have to be an air-cooled condenser on the plane. The power produced might be used to turn propellers to drive the plane.

Though work on high-speed propellers is being carried out at the present time, the efficiency of present-day propellers falls off as the velocity rises above about 2/3 of the velocity of sound, and an arrangement more closely resembling the turbojet may be found preferable. How this might be worked out will be shown in two examples.

The Bismuth Turbojet

Figure I-2 illustrates one arrangement, which in effect is simply a turbojet in which the nuclear reactor heats the air by means of an intermediate liquid bismuth cycle. As the figure shows, a compressor and turbine are enclosed in a duct. Between them lies a heat exchanger. Molten bismuth circulates by means of a pump between the heat exchanger and the nuclear reactor.

The main points which distinguish this cycle from the open cycle are as follows:

(a) The heat transfer in the reactor is improved, because of the high density and heat conductivity of the fluid. This will make possible a smaller reactor; but the reduction would tend to increase the fuel inventory necessary. This increase is offset to some extent by the fact that the heat-transfer passages are not voids but are filled with a heavy substance of low neutron absorption. The net result is a smaller reactor, limited in size by the fuel inventory permissible.

(b) Calculation shows that a lower reactor temperature is feasible, possibly as low as 1500°F or conceivably even 1300°F. This should ease the materials problem, affording a wider choice.

(c) The reactor structure has to withstand molten bismuth, but not hot air.

(d) The compressor will have a compression ratio of about 6 to 1, in the same range as current turbojet compressors, so that this element of the power plant will require comparatively little development work.

(e) A very small amount of radioactivity is produced in the bismuth, but not enough to affect even the manned plane. Impurities in the bismuth may become intensely radioactive, however, and radiation from the heat exchanger would then affect the pilot, so that for the manned plane the bismuth must be very pure. The escape of fission products also must be prevented. This is not necessarily true for the tug-tow plane,

where the tolerable radioactivity is more than 10,000-fold greater. (Shielding of the heat exchanger would raise the weight of the plane so much that it appears to be out of the question.)

(f) The heat-transfer passages through the shield will be filled with heavy material; the absence of voids makes for a simpler and lighter shield.

(g) The weight of the heat exchanger and its cross section are obvious disadvantages of all closed cycles. Here, however, they are compensated by lower shield weight. Moreover, in a subsonic aircraft of suitable design, calculations indicate that the exchanger can be carried in the wings without contributing extra drag. The heat exchanger is a vulnerable part of the aircraft, since if the bismuth were lost by puncturing the system the plane would be destroyed.

It is evident that the bismuth might be replaced by a variety of other fluids, to give a cycle of essentially the same sort. Other liquid metals may be used, but bismuth and lead are almost the only ones which show low enough neutron absorption and would also not produce dangerous radiations in the heat exchanger. (Actually, bismuth becomes radioactive and quite poisonous, but the radiation would not penetrate the heat exchanger walls.) For the tug-tow arrangement other fluids might be considered.

The Complete Aircraft with Bismuth Turbojet

The various differences between the open-cycle and the bismuth turbojet result in somewhat lower gross aircraft weights with the latter on what appear to be about equally optimistic assumptions in each case. On the whole, it appears that the nuclear advantages of the bismuth turbojet make it superior to the open-cycle air turbojet from the standpoint of feasibility.

The Helium Gas-Turbine Compressor-Jet

One of the few fluids which does not become radioactive in the pile is helium. A cycle analogous to the bismuth cycle could be employed using high-pressure helium, which has fairly good heat-transfer properties. Another cycle, however, appears about as promising. It is illustrated in Figure I-3.

In this case helium goes through a closed gas-turbine cycle, being compressed, then forced through the reactor where it is heated, then expanded in a turbine, and then cooled in a heat exchanger. The helium turbine drives both the helium compressor and an air compressor which forces the propulsive air through the heat exchanger. The exact helium pressure will depend on the design, but it will probably be in the range of 500 to 3000 pounds per square inch.

In the helium cycle the chemical effect of the fluid disappears, since helium is inert. In most respects this cycle lies somewhere between the open-cycle turbojet and the bismuth turbojet. The reactor temperature will be intermediate. There will be some ducts in the shield but because of the high pressure they will be smaller than for the open cycle. The gross weights of aircraft should also be intermediate.

The helium turbine and helium compressor will require a considerable amount of development, though there is no real doubt that such components can be provided. Perhaps the most difficult problem peculiar to this cycle will be to secure absolute tightness against high pressure.

Vapor Cycles and Other Possibilities

Some studies have been made on cycles similar to the last two, but employing a condensable vapor such as steam or mercury. These cycles are operable at still lower reactor temperatures. It is felt, however, that the use of a boiling liquid directly in the reactor is too dangerous, because sudden changes in the proportion of liquid to vapor will tend to occur, and these would change the multiplication factor in a way which would make control very difficult. Hence, either a high pressure must be used, so as to prevent any vaporization in the reactor, or a separate fluid must be used in the reactor, as for example, a bismuth-mercury combination. The extra heat exchanger thus introduced complicates the power plant. Nevertheless, the vapor cycles constitute another possible avenue to a nuclear aircraft power plant which cannot be wholly disregarded.

Further propeller development may bring efficient high-speed propellers and therby permit a series of new possibilities, in which various power plants similar to the types described are used to drive such propellers.

Effects of Altitude in Closed-Cycle Power Plants

The effect of altitude is the same in direction for the closed as for the open cycle but the magnitude of the effect is somewhat less, because the reactor itself, operated in closed cycle, is not affected as it is in the open cycle. The attainable design altitude is therefore somewhat higher, though probably less than 60,000 feet.

Open Versus Closed-Cycle Power Plants

The power plant, including the reactor, is the element most in question in any nuclear-powered aircraft development. The problem of materials is not only difficult, but the solution of the problem depends on the discovery of new materials or new ways of using old materials, so that there is no guarantee of success. In this situation, the existence of numerous possible cycles, each offering a possible solution and each making different demands on materials, increases the ultimate likelihood of success.

Besides the materials question, which appears easier with the closed cycles, there are important development problems peculiar to the various

cycles. In the open cycle, the high-pressure-ratio compressor is the most important. In the closed cycles, the heat exchangers would not be of ordinary type but would have to be developed, in material suitable for extreme lightness and absolute tightness as well as for heat transfer. The helium compressor and turbines would also present a problem, probably somewhat less acute. Here again the possibility of choosing the most advantageous of several cycles makes ultimate success more probable even though some one problem should turn out (contrary to expectations) to be insoluble.

The cost in money and manpower of each of these developments, however, would make a parallel development of all the possible avenues extravagant. So far as careful analysis and preliminary study can clarify relative merits and probable feasibility of the various choices, such studies should be carried out as an important measure of economy. Even after the best judgment has been exercised, it may be found expedient to abandon one route and try another.

In connection with the probable difficulties to be encountered, it is worth noting, as an important consideration in favor of the tug-tow arrangement, that any of the possible difficulties could be eased at the expense of an increase in airplane gross weight. For example, the turbine inlet temperature could be lowered, or the spread between reactor and turbine temperature lowered, or a less efficient compressor or a heavier heat exchanger accepted, if the airplane gross weight is suitably raised. Thus the tug-tow arrangement, which if everything is successful will permit an airplane of relatively moderate weight, offers, so to speak, a reserve which can be employed as needed to retain feasibility even though specifications of some element are not met. The manned airplane, which is already so large that further increases will bring additional difficulties, is under more of a handicap from this point of view.

The great importance of determining minimum shielding is apparent. The shield which is now thought to be surely adequate is small enough so that a flyable plane can be obtained if sufficiently favorable component characteristics can be obtained. But any reduction in shield weight will furnish additional "reserve" to be applied as needed and will ease the problem of development where this will save the most work and time.

MISCELLANEOUS POINTS

The Rocket Power Plant

In a rocket the cooling fluid is not recycled, as in the closed cycle, nor is it drawn from the surroundings, as in the open cycle. It is carried with the craft and ejected during the operation. A study of the situation brings out several points:

(a) The working fluid must be of low molecular weight. Hydrogen would be by far the best, in spite of its bulkiness, if it could be carried. The most advantageous form for carrying would be as liquid hydrogen, which would probably be made at the launching site.

(b) To achieve any performance of value, particularly any performance superior to chemically powered rockets, reactor wall temperatures such as 4000°F are required -- much higher even than for the ramjet.

Nuclear-powered rockets would be very large, and the materials problem would be even more difficult than for the ramjet. A reasonable evaluation is that the rocket, like the ramjet, should not be attempted until after the development of one of the closed or open-cycle types.

Problems of Control

Reliable control of the entire complex system involving the reactor, power plant and airplane is essential. Both analytical and experimental work are needed.

Control of the power level of the reactor is only one item in the entire problem, but because of its novelty and hazardous nature may be discussed briefly. It is effected in principle by changing the multiplication factor, so that the neutron concentration builds up or falls off as this factor is above or below unity by a small amount. The change may be effected by introducing or removing rods containing neutron-absorbing materials, by removing or displacing reflector or shield to increase neutron leakage, or by altering the pile dimensions by relative motion of the parts.

The critical nature of the controls is illustrated by the fact that if the pile is just steady (multiplication factor of exactly 1) then an increase to 1.008 will make the pile almost uncontrollable. Yet the difference in potential multiplication factor between ambient temperature and operating temperature may by 10%. Furthermore, because of the effect of poisons in the fission products, the reaction of the pile to changes in operating level is quite complicated. Also, piles with high concentrations of fissionable material are less susceptible to the action of control rods, because the neutrons in such piles tend to be predominately fast neutrons for which no very good absorbing materials exist.

The above considerations are important mainly because they aggravate the mechanical problem connected with control. The control mechanism must operate very rapidly and accurately at high temperature and probably under considerable pressure, and must do so in response to some very accurate and reliable measurements over a very wide range. The problems involved do not appear insoluble, but they are difficult, and a careful consideration of control problems must accompany any reactor study, just as control design must go along with reactor design.

Problems of Reactor Mechanical Design

The actual mechanical design of a reactor from given materials will in itself require a great deal of skill and ingenuity. The reactor must be able to expand as it becomes hot; yet it must not be a loose structure,

since it must withstand the shocks associated with flight. The rather brittle character of the most desirable materials adds to the difficulty.

Problems of Full-Scale Testing

A whole series of problems arises in connection with the latter phases of the development program itself. One of these is the location of full-scale test-stand equipment. Assuming that a point has been reached at which the construction of a test-stand prototype of the power plant is justified, a location must be found. Because of its novel character, the intense concentration of high power and the more extreme temperatures and stresses involved, such a unit will be far more liable to loss of control than any existing pile. Yet loss of control implies a high probability of local lethal doses of radiation and a small area in the immediate vicinity may be expected to be uninhabitable for many years. The location must therefore be remote and yet at this location there must be adequate cooling water, a rather large supply of power, and housing for personnel.

A similar set of problems arises if an actual airplane is to be tested. It is hardly to be expected that a completely new (and large) airplane with a totally new power plant can be developed directly. There should be an intermediate chemically powered plane, as similar to the nuclear-powered as possible, for experience in test flying. The design and construction of this plane will in itself be expensive. The launching site for the final nuclear-powered plane will probably have to be even more remote than the test stand. It is to be expected that crashes may occur, and the site of a crash will be uninhabitable. Matters like the development of remote yet very large airfields and devices for assisted take-off must be contemplated.

All these problems appear capable of solution, given sufficient time and money; but in estimating cost and time scale they will be found in the aggregate to be large.

This remark applies as well to an additional series of problems associated with the handling of the aircraft after a mission. One point is the provision for landing gear. On the very large airplanes which are contemplated the development of conventional landing gear will be a big task, and the gear will add seriously to the plane weight. One solution would be landing on skids; but this involves greater risk to the reactor.

Another problem arises from the fact that heat production in a reactor does not cease when the chain reaction is stopped. The radioactive materials produced by the process generate heat, ten seconds after shutting down, equal to 3% of full power and 1% after an hour. The reactor after one flight will require cooling for a considerable period and this cooling cannot be provided from the aircraft power plant if it is to be serviced. Here again solutions are possible, but must be developed.

Recovery of Unconsumed Fuel

Another question requiring consideration is the recovery of the fuel. At best, the severe conditions of use make long life of the power plant improbable and only a small part of the initial inventory of fuel will have actually been used up.

The recovery of the remaining fuel is a problem analogous to that of recovering the uranium from the chemical plant at Hanford. This problem has not yet been solved; but active work to solve it is under way, and it does not appear necessary to inaugurate a separate research program for the recovery of the fuel in aircraft.

In any case, a quick turnover of the fuel in the aircraft could be obtained only at tremendous expense, and this should be kept in mind in evaluating the strategic value of such aircraft.

Power from Polonium

It is pertinent to mention that polonium has been suggested as a fuel for aircraft because it would require very little shielding. This proposal may be technically feasible but appears to have serious economic disadvantages. Some further study is warranted. The subject is considered in a separate memorandum.

DISCUSSION

The program which is indicated to achieve nuclear-powered flight has already been described at the start of this chapter under the headings of Conclusions and Recommendations. Before embarking on a consideration of its possible success, it is pertinent to outline the specifications which appear to be within reason and of probable interest by repeating points 2 and 3 of the Conclusions.

"2. The resulting nuclear-powered plane would be specifically a military instrument for use in bombing and reconnaissance of targets which are beyond the reach of conventionally powered aircraft. The following characteristics of the plane seem achievable in the light of present information, although it is quite conceivable that they can be improved by developments now under way or not yet foreseen:

"(a) An extreme range, e.g., around the world.

"(b) A speed approaching the velocity of sound -- probably 600 miles per hour

"(c) An operating altitude of at least 30,000 feet, but probably not much above 50,000 feet.

"(d) A content of fissionable material in its power plant of not over 200 pounds and possibly as low as 20 pounds (calculated

as uranium 235). Because of the scarcity of fissionable material it is unlikely that many nuclear-powered planes will be built.

"3. This airplane may take one of the following forms:

"(a) A manned nuclear-powered plane. Such a plane would have to be very large, perhaps larger than anything yet constructed, in order to carry the heavy shield which is needed to protect the crew from the radiations of the nuclear reactor. Its size may be prohibitive, but if much more efficient shields can be developed it might well fall in the range of very heavy bombers.

"(b) An uninhabited nuclear-powered plane, which tows at some distance behind it a small plane carrying the crew, the payload and the controls -- designated as a "tug-tow" arrangement. The shield in the nuclear-powered tug would be much lighter than that needed for the reactor of a manned plane, and the size of the tug would fall within the range of existing bombers. The towed plane which contains the crew would have a conventionally fueled power plant to permit independent operation at take-off and landing and, if desired, for short periods in flight. The tug-tow combination currently appears to be the most promising prospect.

"(c) A pilotless aircraft, i.e., an uninhabited plane which is controlled only by a long-range guidance system. It is questionable whether such a pilotless nuclear-powered aircraft, if it could be developed, would show sufficient advantage over a similar device with chemical fuel to be of interest."

Probability of Attaining the Goal

In assessing probability, the manned plane must be distinguished from the tug-tow combination; as already explained, the former will be successful only if quite good solutions are attained for every problem, while the latter allows for the possibility of some disappointing results and is decidedly more promising.

The most serious threat to feasibility is undoubtedly the reactor materials problem in its whole scope, including the fabrication into a stable workable pile with adequate controls. This problem is greatly eased by the fact that several quite different alternates on the power plant look promising. On the whole, the chances of success in the tug-tow case are high, and they may be assessed at better than even for the manned plane.

Although component problems, such as the turbo-compressor for the open cycle and the heat exchanger for the closed cycle, are apparently less serious than the problem of reactor materials, it may yet prove impracticable to develop one or another component to meet the specifications required for flight. This possibility of failure on a specific

item need not greatly reduce the feasibility estimate for the tug-tow combination but is more serious for the manned airplane.

To the above must be added unknown or unconsidered factors favorable or unfavorable. Summing up, however, it would seem that the overall chance of developing a nuclear-powered plane of the stated characteristics by a fairly intensive effort appears reasonably good.

Probability of Improving on the Goal

Since it has been indicated that the probability of attaining the specified goal is quite high, it follows that there is also a good probability that something beyond the bare minimum can be attained.

As has already been indicated, improvements in any component can in principle be transferred, by suitable design, to any of several of the specifications. However, since the speed has already been set just below the velocity of sound, any substantial improvement in speed would require a very great improvement in supersonic aerodynamics because of the low weight-thrust ratio in the supersonic region. Such a great improvement appears very difficult, and supersonic velocities do not appear to be probable for a long time.

Altitude (up to 50,000 feet or so), size of plane, and fuel inventory are more or less interchangeable, and it would be a matter of choice with the designers of the future as to which one should be improved.

Time Scale

As has already been indicated, the initial program discussed at the start of the chapter should occupy some three to five years. The subsequent development cannot be clearly foreseen. An estimate of fifteen years elapsed before the first successful test flight would seem reasonable if an intensive peace-time effort is employed.

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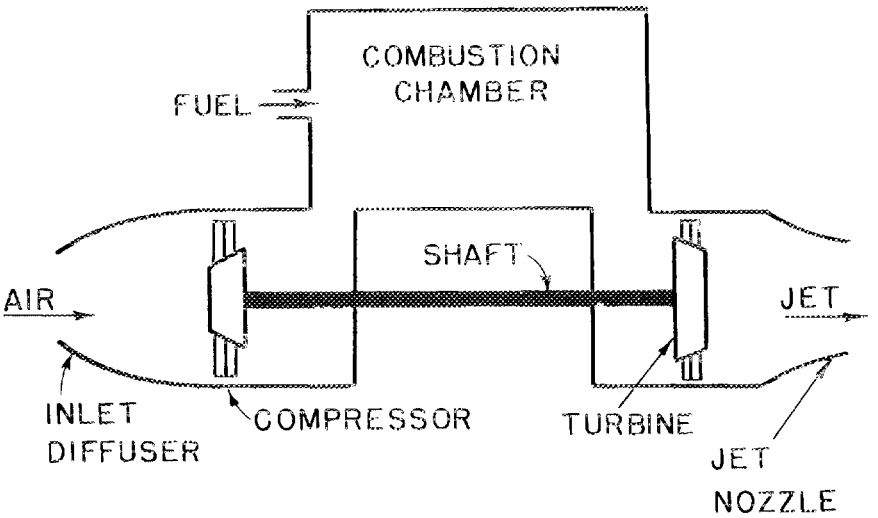
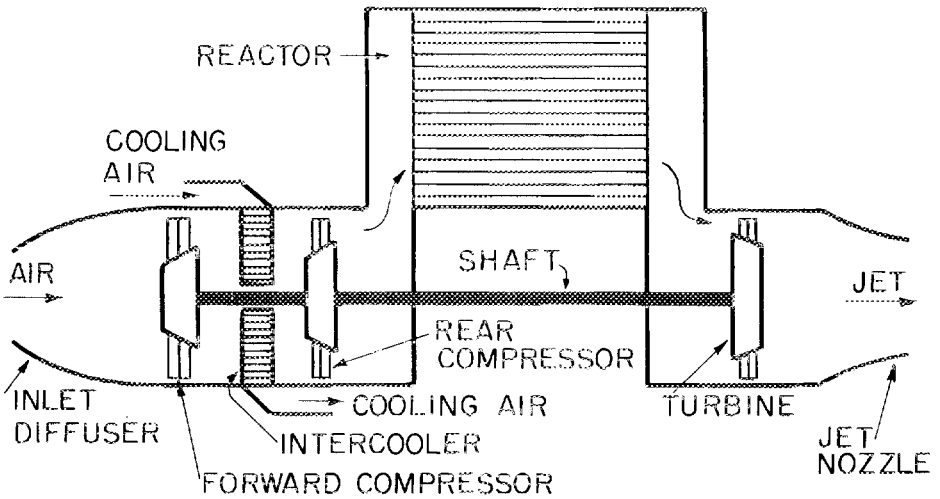


CHEMICAL TURBOJET



OPEN CYCLE AIR TURBOJET

FIGURE I-1

[redacted]

[redacted]

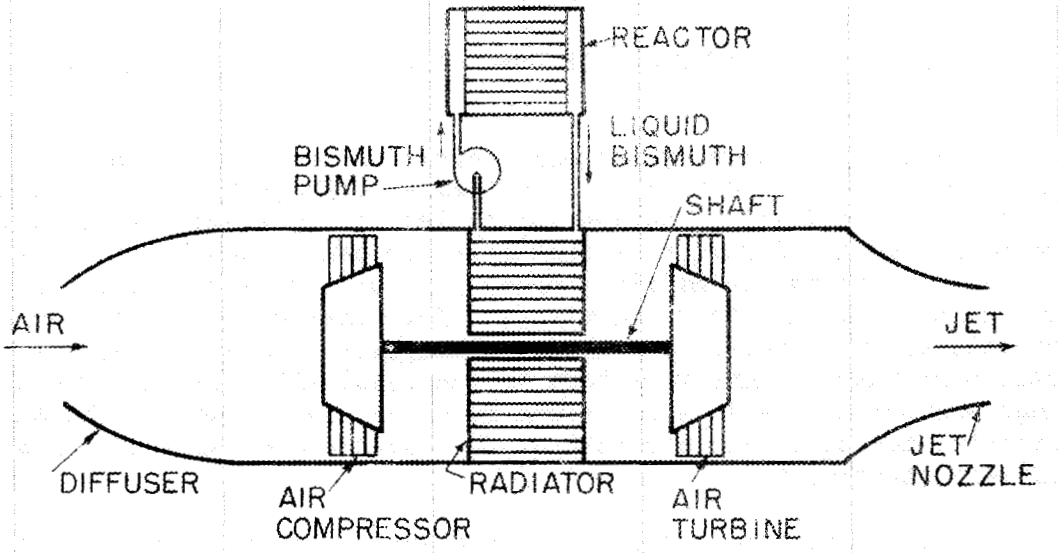


FIGURE I-2

BISMUTH TURBOJET

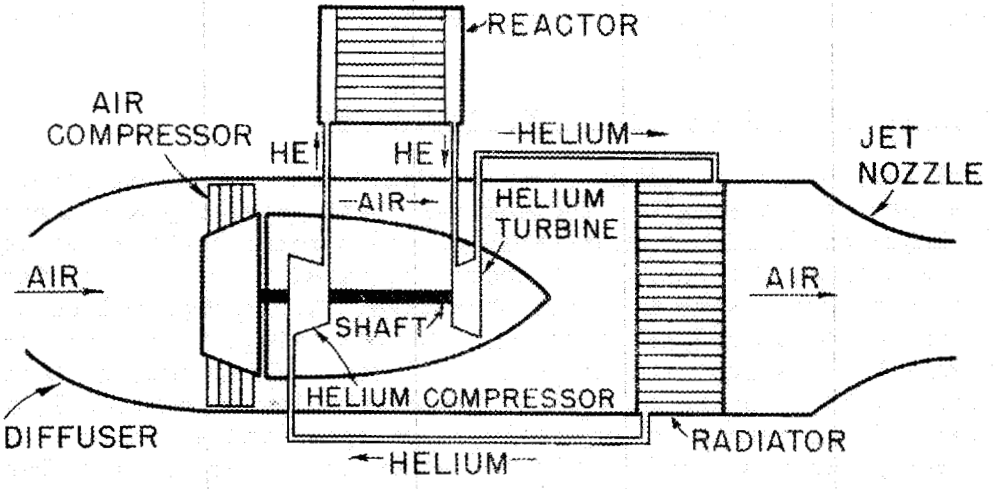


FIGURE I-3

HELIUM GAS-TURBINE COMPRESSOR-JET

[redacted]

Chapter II

THE NEED FOR AND PROBABLE USES OF NUCLEAR-POWERED AIRCRAFT

A decision to develop nuclear propulsion for aircraft must be made on the basis of military need. The Lexington Project has evaluated the technical feasibility of nuclear-propelled aircraft and has estimated the magnitude and duration of such a development. But it has no responsibility for judging the military necessity for such aircraft nor for describing possible military operations. Nevertheless, some consideration has been given to possible military employment in order to select realistic performance characteristics in the feasibility evaluations and to judge whether the various conceived types of planes such as the manned aircraft, the pilotless aircraft or the tug-tow are realistic from a military standpoint.

Furthermore, some study has been given to the performance potentialities of chemically propelled aircraft to determine whether there are military missions which these cannot perform but which nuclear-powered aircraft can accomplish.

Representatives of the Air Force, of the Bureau of Aeronautics and of many other organizations, who have given serious thought to the problems of military employment of aircraft, have been consulted on these matters of strategy, military requirements and the potentialities of chemically propelled aircraft. The primary objective of this chapter is to set forth the results of these consultations and studies as they have influenced the work of the Lexington Project on the feasibility of nuclear-propelled aircraft.

The Military Uses of Nuclear-Propelled Aircraft

Military usage of combat aircraft falls into two categories: the tactical and the strategic. Tactical aircraft generally are heavily armed, high-speed, highly maneuverable, short-range machines designed for defensive action, for support of ground or naval forces or for the attainment of some short-range military objective. Nuclear-propelled aircraft do not appear to be adaptable for use in this category because the relatively heavy weight and large size of the nuclear engine, together with its shield, require a large airplane with low maneuverability. Also there is no reason to expect that the speed and altitude performance of nuclear-powered aircraft would be better over short ranges than that of chemically propelled aircraft.

Strategic aircraft, on the other hand, are designed for offensive action deep within the territory of the enemy, so they must have long-range capabilities. Also they need good speed and altitude performance to make them less vulnerable to enemy defensive measures. In this category the unlimited-range capabilities of nuclear-propelled aircraft offer unique advantages over those with chemical propulsion. Furthermore, the possibility of attaining considerable advantage in strategic bombing with a small number of operations employing atomic bombs carried by nuclear-

powered aircraft may offset the disadvantages of the limited supply of nuclear fuel and the exceedingly complex operations required by these planes.

The military requirements for long-range bombers and reconnaissance airplanes, both of which are in the strategic category, must be based on an assumption concerning the strategic problems facing the Air Force and on an estimate of the enemy's technical and military defense capabilities. The strategic needs determine the range required, and the enemy's defensive capabilities determine the speed, altitude and protective armament required by the plane to penetrate the defenses successfully.

One basic strategic assumption is that Russia is the enemy. If war comes, a requirement of strategic aircraft will be to drop atomic bombs on well-protected industrial centers, deep in the heart of the Eurasian continent. These industrial centers are within range of chemically propelled aircraft currently available to the Air Force only if the aircraft are operating from bases in Europe, North Africa or Asia. However, a situation must be assumed where the U. S. Air Force cannot operate from such bases. Thus, the Air Force believes it vital to prepare to operate against Russia from bases within the United States. This requires aircraft with extremely long-range capabilities, up to 15,000 statute miles for the round trip plus the extra range required for emergencies. The last 1000 or 2000 miles to and from the target must be flown over enemy territory, thus requiring high-speed performance for considerable lengths of time.

The Air Force bases its estimate of Russian defensive capabilities on meager intelligence data from Russia and on information of the professional manpower and facilities taken from Germany after World War II. It is expected that five years hence Russia will have: effective early-warning radar networks; radar ground control for aircraft as good as the microwave early-warning and V-beam equipment presently employed by the United States; and efficient high-speed and well-armed jet fighters operational to at least 50,000 feet and manned by well-trained pilots. Intelligence sources indicate, in addition, that captured German data, manpower and facilities are being employed in the development of high-altitude (60,000 feet), high-speed (2000 m.p.h.), target-seeking anti-aircraft missiles with proximity fuses.

The Lexington Project requested the U. S. Air Force to specify the range, speed, and altitude performance which it felt was needed by long-range bomber and reconnaissance planes to carry out its strategic missions five, ten and twenty years hence. The Air Force, with general agreement by the U. S. Navy's Bureau of Aeronautics, gave the following requirements for bombers: (In each case the bomber payload is an atomic bomb or weapons of comparable or greater damage power).

	5 years	10 years	20 years
Radius of Action -- Nautical Miles	5,000	6,000	6,000
Statute Miles	5,700	6,800	6,800
Range -- Nautical Miles	11,000	13,000	13,000
Statute Miles	12,500	15,000	15,000
Altitude -- Feet	50,000	60,000	60,000+
Speed -- Knots	500	570	850
Miles per hour	570	660	1,000
Mach number	0.87	1.0	1.5

Successful strategic bombing by aircraft meeting the above specifications presupposes accurate knowledge of the location of the targets. The United States does not now have that necessary information with respect to Russian targets. For this target location, extremely long-range reconnaissance aircraft will be required -- aircraft carrying high-altitude photographic equipment and radar-mapping equipment, capable of making an aerial survey of large areas of central Eurasia and returning to base. These aircraft will not be small. The U. S. Air Force estimates that the photographic and electronic gear required to be carried aboard such aircraft would be roughly equivalent in weight to the atomic bomb itself. A considerable crew, apart from the flying personnel in the aircraft, would be needed to handle equipment.

In load-carrying capacity the long-range reconnaissance plane must therefore be equivalent to the long-range bomber. In addition, reconnaissance requires a greater range because the aircraft must make an aerial survey of an area, whereas the bomber needs only to fly to its target and return. In general, both bomber and reconnaissance planes are designed for the best possible speed and altitude performance.

Although it may be conceded that, other things being equal, the advantage in modern air warfare lies slightly with the attacker, the U. S. Air Force aircraft must still face a challenging enemy defense. However, the Air Force believes that the defense could be penetrated without unreasonable losses by aircraft meeting the speed and altitude characteristics quoted here.

The Ability of Chemically Propelled Aircraft to Meet the Military Requirements

The principal argument advanced for the development of nuclear-propelled aircraft is that chemically propelled aircraft cannot meet the above military requirements. The Lexington Project has not made a detailed technical evaluation of possible improvements in the performance of chemically propelled aircraft. However, several conferences dealt with the subject in one way or another. These discussions were with representatives of the U. S. Air Force Headquarters, the U. S. Air Force Air Materiel Command at Wright Field, Project RAND of Douglas Aircraft Company, North American Aviation's Aerophysics Laboratory, the Lewis Flight

Propulsion Laboratory of the NACA, Boeing Aircraft Company, the Glenn L. Martin Company, and the NEPA Division of the Fairchild Engine and Aircraft Corporation. (1) (2) (3) (4) (5) (6) (7) (8) (9) (10) Some discussion of conclusions to be drawn from these conferences is of interest.

All of the chemically fueled engines such as the reciprocating engine, the turbojet, the turboprop, the ramjet and the rocket are under intensive development at the present time. Many improvements are being made and will continue to be made, although the region of diminishing returns is rapidly being reached in some of these developments.

Several organizations including the Lewis Flight Propulsion Laboratory, Project RAND, Boeing Aircraft Company, and Wright Field, have made many analyses to indicate expected future performances of long-range aircraft employing these chemically fueled power plants. The results differ somewhat, depending on the nature of the assumptions. In Table II-1, there are listed some typical figures taken from the NACA study along this line. (11) These figures are based on a lift-to-drag ratio of 18, which requires quite good design for attainment, but are otherwise based on conservative assumptions.

Analyses made by others indicate that the figures quoted above, if interpreted as ultimate performance figures, are low. For example, the B-52 which is now being engineered by the Boeing Aircraft Company for the Air Force is expected to have, with turboprops for power, a range in the region of 9000 miles at speeds up to 500 miles an hour. Such a performance is better than those listed in Table II-1.

Figure II-1 is a curve drawn from results of a preliminary analysis by Project RAND, which is based on less conservative assumptions, but which RAND thinks are realistic. It is a plot of range versus Mach number for chemically propelled aircraft using gasoline. It shows strikingly how maximum range decreases as the design speeds of chemically powered aircraft are increased.

Even based on the more optimistic analyses, it does not seem likely that chemically propelled aircraft can meet the range requirements at the high speeds needed to penetrate the enemy's defenses. By adopting certain expedients such as refueling in flight, discarding landing gear, ditching in the ocean near an advance submarine on the return trip, etc., some of which are well along in development, it seems likely that the range of chemically propelled aircraft might be extended to meet the requirements of 5 years hence. In fact, the Air Force intends to do just that with the B-52. One plan is for the B-52 to be refueled in flight about 3000 nautical miles from base on the way to the target, thus extending its range to a point where it would meet the requirements of 5 years hence.

However, it now appears unlikely that chemically propelled aircraft can meet the requirements of 10 and 20 years hence, even employing all known expedients. Furthermore, refueling is not regarded as a really satisfactory solution of the range problem.

[redacted]

POTENTENTIALITIES OF CHEMICALLY FUELED AIRCRAFT EMPLOYING GASOLINE

TABLE II-1

	SPEED (M.P.H.)	ALTITUDE (FT.)	RANGE (STATUTE MILES)
RECIPROCATING ENGINE	200	UP TO 30,000	9,000 TO 10,000
	500	UP TO 50,000	4,000 TO 5,000
PROPJET	200	UP TO 30,000	9,000
	500	UP TO 30,000	4,000 TO 6,000
TURBOJET	400	15,000	5,000
	550	UP TO 50,000	5,000 TO 6,000
TURBOJET WITH AFTER BURNING	500	30,000	4,000
	550	30,000	4,000
RAMJET	2,000	30,000 TO 100,000	1,000 TO 4,000

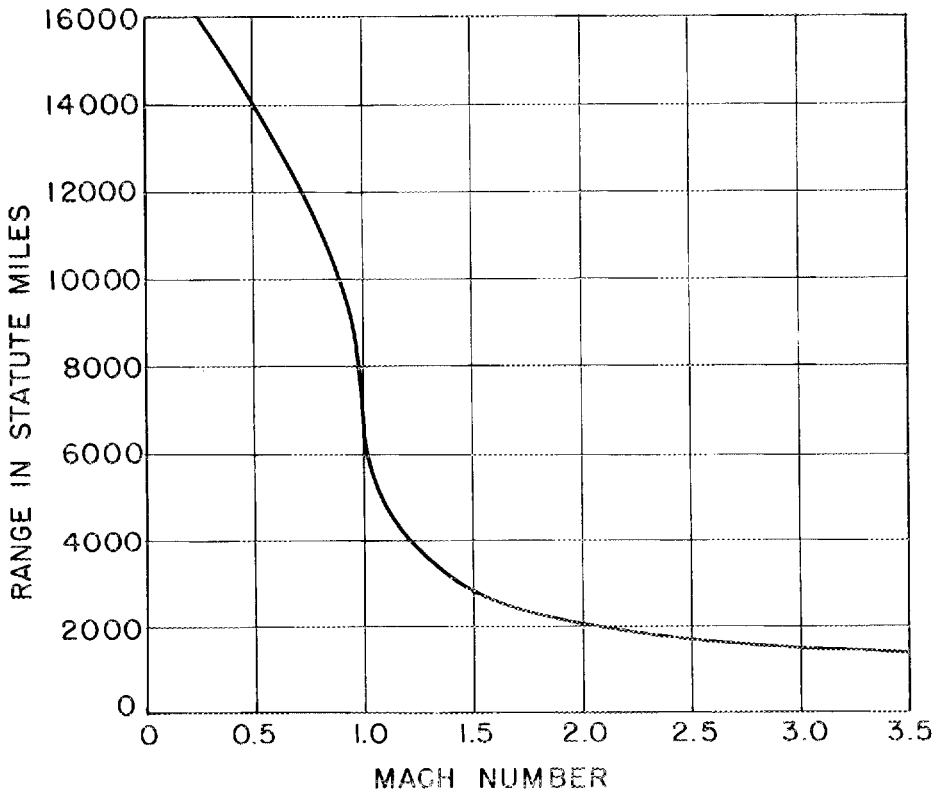


FIGURE II-1

[redacted]

The Use of Pilotless Nuclear-Propelled Aircraft

Formidable difficulties in the realization of nuclear-propelled aircraft stem, directly or indirectly, from the necessity of carrying a heavy shield around the reactor. In the case of manned nuclear-powered aircraft, the weight of the shield which is necessary to protect the crew from radiation becomes so heavy that it forces many features of design to the limits of feasibility.

It is inevitable, therefore, that the question should be raised as to whether difficulties arising from heavy shield weight could be alleviated by nuclear propulsion in a pilotless aircraft -- "a guided missile".

The elimination of a crew would indeed remove the element most sensitive to radiation, but would by no means completely eliminate the need for shielding. It would still be necessary to protect the guidance devices, electronic gear, lubricants, and the atomic bomb payload. Preliminary estimates (see Chapter V) of the radiation tolerances of these elements and of the shield weight necessary to protect them lead to the conclusion that such pilotless nuclear-propelled aircraft, though indeed lighter and smaller than piloted nuclear-propelled aircraft, are still in the weight and size category of modern heavy bombers. In general, shield thicknesses for pilotless nuclear-powered aircraft are about half those required by piloted nuclear-powered aircraft.

Without effective guidance equipment, any pilotless long-range bomber is useless, regardless of its other features. To be usable in the strategic situation herein considered, a guidance scheme must have sufficient accuracy for atomic bombing (circular error of about one mile) at distances up to 6000 nautical miles. Such requirements rule out most types of guidance schemes. The only long-range guidance systems which appear applicable at the present are the inertial and celestial-inertial systems. These long-range guidance mechanisms may be assisted at the terminus of the mission by homing equipment of the radar map-matching, infrared or television type.

No such guidance equipment is available today. However, more than a dozen contractors to the armed services are developing as rapidly as possible components which can be used in these guidance systems. The Lexington Project has not attempted to evaluate these developments nor to place a time scale on the efforts. It can only report that considerable optimism for early success exists among those working on the problem, but that estimates of the time required for successful development vary from a few years to a few decades.

It should be noted that long-range guidance equipment, if successfully developed, will be available for chemically propelled aircraft at least as early as it will for nuclear-propelled aircraft and probably earlier. The data in Table II-1 and Figure II-1 indicate that pilotless chemically fueled aircraft would be able to meet bomber requirements if the missions were one-way missions. If guidance developments are successful, pilotless

chemically propelled aircraft will probably be used in preference to nuclear-powered aircraft to deliver atomic bombs. However, such bombing missions could not be effective without reconnaissance to seek out targets and establish their map coordinates for use by the bombing plane guidance device, as well as to survey results after bombing. Such reconnaissance would still necessitate two-way missions, preferably with a crew. As was shown earlier, chemically propelled aircraft might not be able to perform these missions; hence atomic bombing by means of pilotless, guided, single-flight, chemically powered aircraft might still depend upon reconnaissance by piloted, two-way flights of nuclear-propelled aircraft.

Tug-Tow Nuclear-Propelled Aircraft

Nuclear-propelled aircraft, manned or pilotless, would operate in a manner quite similar to that of chemically powered aircraft, manned or pilotless. A novel type of nuclear-powered aircraft has been proposed, however, which would have some unique operational features. This is the tug-tow.

This composite device employs a tug airplane which is driven by a partially shielded nuclear power plant. It tows a more or less conventional plane attached by a cable perhaps 3000 feet long. The towed plane would carry the crew, payload, electronic equipment, photographic equipment and other radiation-sensitive components. The towed plane could be provided with chemically powered engines with sufficient fuel for a flight of short duration for use during take-off and landing, as well as for a short duration detached flight during the mission.

One of the questionable features of tug-tow aircraft is vulnerability. The composite device is probably more vulnerable; just how much no one has ventured to estimate. The extra vulnerability might be mitigated by turning on the chemical power of the towed plane in the target area to achieve greater speed or altitude or by detaching the towed plane temporarily from the tug for a high speed bombing run over the defended target area, during which run the tug would fly on a preset pilotless course, and reestablishing contact later.

While the Lexington Project has not intended to evaluate various operational tactics to be performed by nuclear-propelled aircraft, it has had preliminary talks with Air Force operational experts, with NACA personnel and with aircraft designers, to see if the tug-tow idea is operationally unsound. (12) (13) (14) No such negative opinion has been forthcoming. On the contrary, considerable enthusiasm for the idea has been expressed by many.

Civilian Use of Nuclear-Propelled Aircraft

It is difficult to predict the uses to which any new technical achievement would be put. Change in any of a large number of factors may completely alter the concept of employment. Even with the possibility of a change in one of several factors, it is still generally believed that civilian transport and freight use of nuclear-propelled aircraft is not to be expected. There are three factors which are influential here: first, the

operational hazards of everyday flight from, over, and to densely populated areas are immense; second, the uranium investment and consumption would probably be economically and militarily unjustified with the present limited supply of fissionable material; and third, the aircraft might be uneconomical for commercial handling, because of many special features of nuclear-powered flight, such as long runway requirements, difficulty in servicing engines, etc. It is conceivable that the second factor could be eliminated or decreased in importance with an improvement in the fissionable material supply picture or a lessening of the military importance or need of fissionable material. Also, the third factor might be overcome by sufficient technical development. The hazard factor, however, looms so large as to discourage the prospect of civilian flight by nuclear-powered aircraft.

Summary

Nuclear-powered aircraft, with practically unlimited range capabilities at design speeds, could probably meet the long-range reconnaissance and bombing requirements of the U. S. Air Force better than could chemically powered aircraft, if the missions are to be accomplished with a crew.

If satisfactory long-range guidance systems are developed, the bombing requirements can probably be met by pilotless chemically powered aircraft flying one-way missions. Nuclear-powered aircraft might still be needed to perform the necessary reconnaissance.

There do not appear to be any non-military uses which would justify nuclear-powered aircraft.

[redacted]

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(12) LP-119 Meeting with E. P. Manganiello and A. Silverstein on Aircraft Design and Power Plant Analysis; W. C. Cooley, August 10, 1948.

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Chapter III

POWER PLANTS

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SUMMARY

The Lexington Project made exploratory studies of a number of propulsion systems which seem to offer a reasonable prospect of being suitable for nuclear-powered flight. These studies, because of time limitations, were far from exhaustive in scope, but were broad enough to permit preliminary evaluations and comparisons.

Primary attention was directed towards flight at high subsonic speeds and at altitudes of 30,000 feet and over. Supersonic nuclear-powered flight was considered only briefly because present information on supersonic aerodynamics indicates that significant advances must be made before a supersonic nuclear-powered aircraft is feasible.

In the power plant studies presented in this chapter, the aircraft, the materials, and the nuclear considerations were used to delineate the range of variables considered. The integration of the airplane, material, and nuclear aspects with the power plant aspects of the complete nuclear-powered aircraft is presented in Chapter VII.

Special consideration was given to the three types of machines which appear to be of most promise for high subsonic flight speeds, namely, the open-cycle turbojet (Fig.III-1), the helium gas-turbine compressor-jet (Fig.III-4), and the bismuth turbojet (Fig.III-6). Less extensive studies were made of the ramjet, the rocket, the open-cycle turboprop, and of machines employing steam or mercury vapor cycles.

Performance characteristics of the propulsion systems were calculated for a range of such operating parameters as design altitude, maximum reactor-wall temperature, temperature at turbine inlet, and component efficiencies. In the case of such design variables as the pressure ratios of the air and helium compressors, the pressure drops across the reactor and heat exchanger, etc., systematic calculations were made in an attempt to determine optimum values, but the extraordinary number of parameters permitted only approximate optimizations within the time available for the study.

The characteristics of the propulsion systems under the various operating conditions are reported in terms of the power plant weight, power plant frontal area and reactor size, all as functions of the power plant thrust.

The chief conclusions are as follows:

(a) The open-cycle turbojet, the helium gas-turbine compressor-jet, and the bismuth turbojet at present offer most promise for subsonic nuclear-powered flight.

(b) For each of these machines the main obstacle to successful operation is likely to be the development of high temperature materials for the reactor with suitable mechanical, nuclear, and chemical properties. In this connection, the

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lower limits of reactor temperatures which will lead to reactors of reasonable size for each of these machines probably lie in the following ranges:

Open-Cycle Air Turbojet	1700-2000°F(927-1093C)
Helium Gas-Turbine Compressor-Jet	1500-1800°F(815-982C)
Bismuth Turbojet	1300-1500°F(704-815C)

(c) The open-cycle machine has the relative advantages of greater simplicity and less vulnerability to service failures and to attack; while the closed-cycle machines have the relative advantages of simpler material requirements, smaller shield weights, and higher maximum design altitude.

The chief recommendations are as follows:

(a) There should be continued an extensive analytical and design program for studying various possible power plants and their integration with the airframe, the materials, and the nuclear requirements.

(b) Attention should be focussed on the various types of closed-cycle turbojets and compressor-jets and on the open-cycle turbojet.

(c) The development of high-temperature reactor materials which have good mechanical properties and which are resistant to chemical attack and leakage of fission products is essential to the development of power plants suitable for nuclear aircraft, and such a program should be vigorously prosecuted.

INTRODUCTION

Function of Propulsion Systems

An aircraft propulsion system has as its chief function the production of a thrust which is used both for accelerating the aircraft and for overcoming the drag exerted by the air through which the aircraft moves.

In the case of a nuclear-powered aircraft, the propulsion system must by some means convert nuclear energy resulting from fission into thrust.

No schemes have yet been seriously proposed for obtaining thrust from nuclear energy in direct fashion, that is, either by making use of the momentum of fission fragments or by forming a propulsive jet of the reactive material itself. At present the only methods which have been proposed for employing nuclear energy and which have some prospect of feasibility involve the use of a nuclear reactor as a source of heat. The attention of the Lexington Project has accordingly been focussed on propulsion systems of this type.

Types of Propulsion Systems

If nuclear energy is used merely as a source of heat, it is clear that from a thermodynamic standpoint alone a nuclear power plant is similar to a conventional chemical power plant, the main difference being that in one case the working fluid is heated by chemical combustion while in the other case it is heated by passing through a nuclear reactor.

This similarity suggests that chemical-type power plants be considered for nuclear propulsion. Excluding piston engines because of obvious difficulties in introducing the nuclear fuel, there remain the turbojet, the turboprop, the ramjet and the rocket. When the propulsive air of these engines is heated in the reactor, the cycle is here termed an "open cycle", since the reactor coolant fluid passes through the reactor only once and is not recirculated.

Although chemical power plants resemble nuclear power plants in a thermodynamic sense, the criteria by which the two are judged as aircraft propulsion systems are quite different. For chemical engines the specific fuel consumption, and to a lesser degree the specific weight of the engine, are the important characteristics. For nuclear engineers, on the other hand, the size of the reactor is of prime importance, for it influences the weight of the shield which must be carried, and the shield weight in turn proves to be a crucial item in evaluating the practicability of nuclear-powered flight. Reductions in the reactor size may be realized only through corresponding increases in the rate of heat transfer. This suggests that in the nuclear engine it might be desirable to use fluids other than air for cooling the reactor. Such a process is here called a "closed cycle", since the reactor coolant, which might be either a gas at high pressure or a liquid metal, is recirculated through a closed system. Although the closed-cycle machines include a heat exchanger in addition to the reactor, the size of the heat exchanger is, from a weight standpoint, not so critical as the size of the reactor because the exchanger does not require shielding.

Several different types of machines are included in the open-cycle and closed-cycle categories. These types are classified and described briefly in the following sections.

Open Cycles

Open-Cycle Air Turbojet (Figure III-1). This machine is similar to the conventional chemical-fuel turbojet except that the air is heated in a nuclear reactor. Referring to the schematic layout of Figure III-1, air is scooped into the inlet diffuser, and, after being decelerated to moderate speed, is compressed in the forward compressor. Then, after being cooled in the intercooler, it is further compressed in the rear compressor. The air is then heated in the passages of the reactor and expands through the turbine where it delivers just enough energy to drive the compressors. The remaining energy in the air is used for expanding through the jet nozzle. A net thrust is obtained if the exit jet velocity exceeds the incoming air velocity.

As compared to conventional chemical turbojets, this cycle in the nuclear power plant requires a higher compression ratio and probably an inter-cooler. As will be discussed later in more detail, high compression ratios are employed to increase the air density in the reactor and thus to reduce the volume of the reactor.

Open-Cycle Air Turboprop (Figure III-2). This machine resembles the turbojet in its components, but the turbine produces more power than is required by the compressors. The excess shaft power is used for driving the propeller. Although it is usually desirable to retain a small amount of jet thrust with this arrangement, the major part of the thrust is obtained from the propeller.

Open-Cycle Ramjet (Figure III-3). At high aircraft speeds, the turbojet and the turboprop become less attrative, and a simpler type of propulsion system, the ramjet, is a more logical type of power plant. There is little or no machinery in the ramjet, compression being effected through ram action of the air. In a general way, and as will be explained in more detail later, it might be said that the ramjet is preferable to the turbojet only at flight Mach numbers in excess of about 1.5 or 2, and that the maximum temperatures required in the ramjet are considerably higher than in the turbojet or turboprop.

Since the ramjet appears to involve even more severe development problems than either the open-cycle turbojet or the closed-cycle machines discussed later, only a limited amount of attention was devoted to it by the Lexington Project.

Closed Cycles

The several types of closed cycles may be divided into two classes. In one class there is a closed-cycle power plant (e.g., a steam plant or a gas turbine) whose product is shaft work. In the second class the reactor coolant serves merely as a conveyor of heat from the reactor to the propulsive air passing through a turbojet or turboprop. Examples of these types are given below.

Helium Gas-Turbine Compressor-Jet (Figure III-4). In this machine helium is the working fluid of a gas-turbine cycle, receiving heat at high temperature in a nuclear reactor and rejecting heat at low temperature in a radiator. The power of the helium turbine exceeds the power required for the helium compressor, and the excess is used to drive the air compressor. Air is scooped into the propulsive duct, and, after being decelerated in the inlet diffuser, is compressed in the air compressor. The air is then heated in the radiator and expands to ambient pressure in the jet nozzle. Thrust is produced by virtue of the difference between the outgoing and incoming air velocities.

In the turbojet the net input to the propulsive air is heat. In the machine of Figure III-4, the net input to the propulsive air includes both heat and shaft work, and the machine is accordingly here described as a "compressor-jet".

A more conventional way of employing the net work of the helium gas-turbine (and the method considered by NEPA) is in the operation of a propeller, in which case the helium radiator would separately be cooled by ram air. This arrangement is subject, however, not only to the speed limitations of propellers but to the even more serious objection that the frontal area of the radiator is so large that it might not be possible to carry it on an airplane. In the compressor-jet the frontal area of the radiator is, by comparison, much smaller; first, because the air density in the radiator is greater, and second, because larger fractional pressure drops can be accepted in the radiator when the air is first compressed in a compressor.

The gas-turbine cycle of Figure III-4 can, of course, employ a wide variety of gases as the working fluid. However, as will be discussed in more detail later, helium seems to be the best choice from the combined viewpoints of materials, heat transfer and nuclear properties.

Bismuth Turbojet (Figure III-5). This machine is a turbojet in which heat is transferred from the reactor to the propulsive air by means of a secondary fluid, e.g., liquid bismuth. The small amount of power required by the bismuth pump is supplied by the hot-air turbine.

It was previously noted that with the open-cycle air turbojet, high pressure ratios are probably required for the machinery if the reactor size is to be kept within reasonable limits. This is not the case with closed-cycle turbojets, since the reactor size is independent of the air density. Consequently the air machinery in the closed-cycle turbojets will probably be similar to that in present-day turbojets.

Helium Turbojet (Figure III-6). This is also a closed-cycle turbojet, but in this case heat is transferred from the reactor to the propulsive air by means of a recirculating gas (in this case, helium) under high pressure.

There are several different ways of supplying the power for circulating the helium, all of which should lead to nearly identical results as regards overall performance of the power plant. In the system investigated by the Lexington Project and shown in Figure III-6, the helium is circulated by a small helium gas-turbine which delivers no net power.

As with the binary bismuth turbojet, the air machinery will probably be similar to that used in existing turbojets.

Ternary Bismuth Vapor-Cycle Compressor-Jet (Figure III-7). In this machine there is a vapor cycle, most probably employing steam or mercury as the working fluid, which receives heat at high temperature and rejects heat at low temperature to the propulsive air in a radiator. For reasons given more fully later, it is probably not feasible to heat the steam or mercury directly in the nuclear reactor; hence a liquid metal, e.g., liquid bismuth, is used as a ternary fluid for convecting heat from the reactor to the steam or mercury boiler. The compressor-jet in this system operates like the compressor-jet in the helium gas-turbine compressor-jet system of Figure III-4.

Other Closed Cycles

There are many variations of the closed cycles described above, depending on the choice of working fluids and on the combinations of the different possible components into various systems. The studies possible within the time available to the Lexington Project indicate that the machines described above offer most promise for nuclear propulsion of aircraft, but it is conceivable that further studies will disclose other promising methods.

Rocket (Figure III-8)

The rocket does not conveniently fit into either the open-cycle or the closed-cycle category.

In the nuclear rocket the working fluid is stored in the fuselage, preferably as a liquid so as to conserve volume. During operation the working fluid is pumped to the reactor under high pressure and is there evaporated and heated to a very high temperature, following which it passes through the nozzle and leaves as a high-speed jet.

For reasons given more fully later, the most probable choice of working fluid is hydrogen.

In a general way, the rocket is most suited to high supersonic flight speeds and requires reactor wall temperatures of the order of 5000°F (2760°C). As with the ramjet, the developmental problems of the rocket appear to be much greater than the problems of the other cycles mentioned previously, and hence the nuclear rocket was only briefly investigated.

Relation of Propulsion System to Other Component Problems

It is generally true that engineering structures involve compromises among the various component parts so as to lead to an integrated design which is both workable and efficient, but in the case of a nuclear-powered aircraft this is true in superlative measure.

In the first place, an aircraft can scarcely be separated from its power plant, since the size, weight and air flow requirements of the latter are intimately connected with the size, configuration, and aerodynamics of the entire vehicle. Thus, from one point of view, the power plant may be judged on its weight, frontal area, bulk, and requirements as to location within the airframe. From another point of view, however, the airframe design determines the weight of airplane which a power plant with a given thrust will support and also determines the fraction of the aircraft gross weight which is available for the power plant. For steady, horizontal flight, the aircraft weight is equal to the aerodynamic lift, and the net power plant thrust is equal to the aerodynamic drag on the parts of the airplane exterior to the propulsion system. It follows, therefore, that the ratio of power plant weight (in this case, including

the shield) to power plant thrust must lie within a limit given by

$$\frac{\text{Power plant weight}}{\text{Power plant thrust}} \leq \frac{\text{Power plant weight}}{\text{Gross weight}} \times \frac{\text{Lift}}{\text{Drag}}$$

As an example, for an aircraft at 40,000 feet altitude flying at Mach number 0.9, it appears that a lift-drag ratio of about 15 may be obtained with a structure which permits 72.5% of the gross weight to be used for the power plant and shield. In this case the power plant, including shield, can not weigh more than about 11 pounds per pound of thrust developed, otherwise the airplane will not be able to fly.

The design altitude and speed of the aircraft influence the performance, size and weight of the different types of power plants and hence play a part in the choice of the propulsion system.

The performance characteristics of the power plant are sensitive to the temperature of the working fluid. Furthermore, the size of the reactor may be reduced through the use of high reactor wall temperatures. But the maximum allowable temperatures are determined by the development of suitable materials, and consequently the performance which may be expected of a certain type of power plant depends on the degree to which structural materials are developed. In addition, the choice of working fluid for the power plant is closely related to the materials problems involved.

On the nuclear side of the picture, there are a variety of links to the power plant. The nature of the reactor coolant, in particular whether or not it becomes radioactive, in part determines whether any equipment besides the reactor must be shielded. This is a vital point because if the machinery, ducting and radiator required appreciable shielding the closed cycle would probably not be worthy of consideration. A number of factors connected with the power plant relate to the required investiment of uranium, e.g., the working fluid, the size and configuration of the reactor heat transfer passages, and the reactor materials necessary for the high temperatures which are desirable.

It is clear from the previous discussion that the choices of power plant and of working fluid are limited by the aircraft, materials, shielding, and nuclear requirements. These considerations underlie the selection by the Lexington Project of the power plants and working fluids which were thought to be most worthy of intensive study.

Methods of Appraising Power Plant

The factors which influence the performance characteristics, size and weight of a given type of power plant may be divided into three categories, as follows:

Operating Specifications, such as aircraft speed and altitude. Information on existing designs of supersonic aircraft indicates that the chances of developing a successful supersonic nuclear-propelled aircraft

are remote unless very substantial improvements in the lift-to-drag ratio of supersonic aircraft are made, or unless new methods of employing nuclear power are developed. Therefore, virtually all the studies of the Lexington Project were confined to subsonic flight. In Chapter II it was pointed out that nuclear-powered flight would in the foreseeable future have none but military applications. Corresponding to this argument, the analyses were made for speeds in the vicinity of the velocity of sound and for altitudes of from 30,000 feet to 50,000 feet.

Factors Limited by Engineering Practice, such as the efficiencies of component parts (diffuser, compressor, turbines, nozzles) and the maximum temperatures allowable in the reactor and at the turbine inlet. While for good performance the designer would prefer to have high efficiencies and high temperatures, he is limited by what is actually achievable along this line. The general procedure for the present studies has been to estimate the performance characteristics for a number of values of the parameters, ranging from values within present engineering practice to values which are possibly achievable within the next few years.

Variables Within the Control of the Designer, such as the pressure ratios of compressors and turbines, and the pressure drops across the reactor or radiator. The selection of variables of this type should be made so as to lead to best overall performance of the power plant in the airplane. In the studies described here, variables of this type were varied in an attempt to obtain optimum values, but time permitted only an approximate optimization procedure because of the many variables involved.

The characteristics considered to be most significant for assessing the suitability of the power plant for nuclear-powered aircraft are as follows:

(a) Turbine inlet temperature

(b) Maximum reactor wall temperature

(c) Size of the reactor core as a function of thrust

(d) Weight of the power plant (excluding shield) as a function of thrust.

(e) Frontal area of the power plant as a function of thrust.

Although it is clearly not possible to reach final conclusions relating to the power plant of a nuclear aircraft on the basis of these characteristics alone, for the sake of separating various components insofar as possible it was thought desirable to present results of this type in this chapter, since such results are independent of all but power plant considerations. Conclusions regarding the power plant must finally rest on an integrated survey of materials, aircraft, shielding, nuclear and power plants problems. Such a survey is presented in Chapter VII.

It should be emphasized that the estimates of power plant performance presented in this chapter do not, within the assumed values for component efficiencies, maximum reactor wall temperature, turbine inlet temperature, etc., represent the best possible designs. This is so because in the time available it was not possible to optimize fully all the parameters within the control of the designer, such as pressure ratios of the air and helium machinery, pressure drops in the reactor and radiator, ratio of helium or bismuth flow to air flow, and so on.

Finally, all the estimates of power plant performance were based on cruise conditions. Time did not permit studies of take-off or landing performance, or of the problem of operation off the design point.

OPEN-CYCLE TURBOJET AND TURBOPROP

Introduction

The open-cycle gas-turbine system is widely recognized as one of the best means of propelling a chemical-fueled airplane at high subsonic speeds. Inasmuch as the major elements of such a system can be incorporated in the power plant of a nuclear aircraft, it was decided to examine the potential performance of a nuclear-powered airplane utilizing an open-cycle gas-turbine propulsion system. The method of operation of such a system is discussed in the Introduction to this chapter and is illustrated in Figures III-1 and III-2.

Previous Work

Various organizations, in addition to the Lexington Project, have investigated the possibilities of the open-cycle turbojet for application to nuclear propulsion of aircraft. These groups include NEPA (1), RAND (3), North American Aviation (2)(4), the Applied Physics Laboratory of the Johns Hopkins University (5), and NACA (6)(7). None of these studies can be considered to be complete.

NEPA and the combined RAND-North American groups have carried out the most comprehensive investigations. NEPA is now engaged in a thorough analysis of the open-cycle turbojet. For its overall aircraft program, NEPA has made a tentative decision to concentrate on the open-cycle turbojet, but has not frozen on a specific design.

North American Aviation (2), under subcontract with RAND, has also investigated the mechanical components of the open-cycle turbojet in some detail. Apparently, much of the analysis already completed by this group overlaps projected NEPA work. RAND and North American have not attempted to integrate their studies in a complete final design, inasmuch as reactor and shielding problems have not been intensively analyzed.

The Applied Physics Laboratory (5) conducted a cursory examination of nuclear-powered flight, but its work was based upon unclassified nuclear data. No attempt was made to examine the engineering aspects in any detail, but an appraisal of the probable characteristics of nuclear-powered aircraft was carried out. It was concluded that the engineering and

materials problems had not been investigated sufficiently to demonstrate feasibility.

The National Advisory Committee for Aeronautics (6) (7), had completed some studies of possible power plants for nuclear-powered flight (exclusive of the reactor) prior to the initiation of the Lexington Project. These calculations were extended to the open-cycle turbojet, and the results (8) were used as supplementary material by the Lexington Project.

In order to make an intelligent appraisal of the possibilities of the open-cycle turbojet, the Lexington Project undertook additional calculations of the performance of an airplane powered by an open-cycle system. These data have been used to formulate the subsequent discussion of the effects of variations in the design variables on the performance of this type of power plant.

Evaluation Criteria

The evaluation of a nuclear power plant for aircraft logically can be based upon the size and weight of the power plant machinery and reactor per unit of thrust developed. The size of the nuclear reactor is of paramount importance, since the required shield thickness is in any case large, and is almost independent of reactor size.

Small size machinery is very desirable, since the equipment should fit into the airplane without introducing a significant increase in the drag. Perhaps the best single measure of the size of the machinery is the maximum frontal area, since large frontal areas may impose significant aerodynamic restrictions by requiring a large fraction of the airplane volume to be devoted to air ducts and machinery and by introducing additional external drag.

The weight of the air-cooled reactor (exclusive of the shield) is rarely of importance when compared with the machinery weight. The weight of the mechanical components is seldom as large as the necessary shield weight, but at high altitudes the machinery weight may become so large that the airplane cannot carry a shield of sufficient size.

Effect of Component Characteristics and Design Variables

The size and weight of the machinery, assuming good mechanical design, is fixed by the altitude and speed of the aircraft, the thermodynamic and mechanical efficiencies of the components, the temperature of the fluid entering the power-producing turbine, the pressure ratio of the compressor, the reactor pressure drop, and the thrust required. The size and weight of the reactor are determined by nuclear, heat-transfer, fluid flow and materials considerations. Calculations have been carried out to determine the effect of each of these factors on the power plant performance. A detailed description of the calculations procedure is presented in Appendices III-A and III-E.

Design Altitude and Velocity. The design altitude and airplane velocity are primarily set by tactical considerations. The speed and altitude largely determine the temperature, pressure and density of the air entering the power plant. In this study, the effect of variations in altitude between 30,000 and 60,000 feet and in design speed, between M = 0.5, and M = 1.5 have been examined, but the major attention has been given to altitudes between 30,000 and 50,000 feet with a flight Mach number of 0.9.

Inlet Diffuser. The design of the intake diffuser presents no particular problems other than those associated with a conventional airplane power plant installation. The ram efficiency cannot be estimated accurately until the details of the particular installation are known. Since a nuclear-powered airplane will be quite large, a reasonably good intake should be possible. Since the diffusion process is the initial stage of compression and will affect considerably the overall pressure ratio of the power plant, a sacrifice in other directions would be justified in order to improve the diffuser. A ram (or diffusion) efficiency of 90% has therefore been assumed in all cases.

Ram efficiency as here used is defined as the ratio of isentropic to actual increase of enthalpy for diffusion from atmospheric (static) pressure and temperature at the duct entrance to the actual total pressure at the duct exit.

Compressor. The need for small-sized reactors indicates that compressor pressure ratios of from about 20:1 to 60:1 may be required. The pressure ratios of compressors used in present day turbojets and turboprops range from 3 to 8. At much higher pressure ratios the temperature rise becomes so great that intercooling would normally be used to reduce the work of compression. If such intercooling is provided, the problems of installation are considerable; but there is no insurmountable difficulty in designing for high pressure ratios, since this would amount essentially to connecting two low-pressure compressors in series, with an intercooler between. The design and installation of the intercooler presents some problems. The cooling air flow is roughly the same as the compressor air flow, and although it appears that the intercooler frontal area can be made about equivalent to the compressor frontal area, to do so will require ingenious design.

The efficiency of compression is ordinarily defined as the ratio of the isentropic power to the actual power required for the same pressure limits. This efficiency is inherently lower for a group of stages in series than for one a single stage since the temperature increases in successive stages. Thus, while single-stage efficiencies of 88% or more have frequently been reported for the axial-flow type of compressor, it cannot be expected that the same values will be obtained for a compressor consisting of a group of these stages.

It is notoriously difficult to measure accurately the efficiency of such compressors, especially when the power cannot be measured directly, but must be inferred from temperature measurements. However, the following table, showing the efficiencies of a number of compressors, as measured by the NACA laboratories, is probably as representative of current

values as any such compilation could be. The units listed include the compressors used in most of the turbojets and turboprops currently installed in United States military airplanes, as well as some experimental units. In every case the compressor tested was a complete operating unit, and not merely a single stage of a multistage unit, although in some cases the compressor was not assembled with the turbine to form a complete power plant.

In the studies made by the Lexington Project, the present-day compressor efficiency was taken as 78%, and the efficiency which might be obtained with high pressure ratio units in five to ten years as 82%. In most cases, an overall compressor pressure ratio of 40 was assumed. This pressure rise is produced in two groups of stages, separated by an intercooler in which there is 5% pressure loss, with a pressure ratio of approximately 6.5 across each group. Discussions with L. S. Hobbs of United Aircraft Corporation (10), P. B. Taylor of Sanderson and Porter (11), and N. C. Price, formerly with Menasco Manufacturing Company (12), indicate that compressors of the contemplated efficiencies and pressure ratios could eventually be manufactured.

In all calculations, one stage of intercooling having an effectiveness of 0.5 and an internal stagnation pressure drop of 5% was used. The effects of changing the number, effectiveness or pressure drop of the intercoolers have not been investigated by Lexington Project. The compressor pressure ratio was varied between 20 and 80, and the compressor efficiency between 78% and 100%.

Reactor Inlet Ducts. The air leaving the compressor is carried by ducts through the shield and into the reactor. The duct geometry essentially determines the pressure drop experienced by the air in its flow through the shield. Based on NACA data (15), one dynamic head (based on the conditions at the reactor tube entrance) was taken to be a representative pressure loss. Although, because of radiation from the pile, some cooling of the shield is necessary, this requirement has not been considered in the preliminary analysis.

Nuclear Reactor. In the reactor, the air is heated, and as a result of acceleration and friction forces, undergoes a drop in pressure. This pressure drop is determined by the heat flux distribution within the reactor, the size of the flow passages, the dimensions of the reactor, the temperature rise of the air, and the maximum permissible wall temperature within the reactor. The magnitude of the pressure loss may strongly affect the performance of the power plant. On one hand, the thermodynamic efficiency of the power plant cycle is improved when the pressure losses are low. On the other hand, low pressure losses require large reactor and shield duct dimensions. These conflicting effects indicate that the pressure drop must be optimized in order to achieve a useful design. The stagnation pressure drop in the reactor and ducting between the compressor discharge and the turbine inlet was varied between 20% and 40% of the stagnation pressure of the air leaving the compressor. In the majority of the calculations made by the Lexington Project, the longitudinal pile heat flux distribution was taken to be sinusoidal, with the maximum power level at

TABLE III-1

EFFICIENCY OF VARIOUS COMPRESSORS AS MEASURED BY NACA

Unit	Type	Comp. Effic. %	Approx. Press. Ratio	Remarks	NACA Report No.
A	Centrifugal	72	Unknown, but less than 4	Less than rated speed	R.M. No. E8G02a
		69	3.95	Rated speed	
B	Centrifugal	70	4.0	Rated speed	R.M. No. SE8C15
		62	5.09	Max. press. ratio	
		75	1.6		
C	Ax. Flow	84	3.99	Approx. rated speed	R.M. No. E6E14
		79	4.95	With modifications to increase air flow	
D	Ax. Flow	78	3.2		Private com. from E.R.Sharp
		77	3.3	Max. press. ratio	
E	Ax. Flow	71	3.63		Private com. from E.R.Sharp
		65	4.15	Max. press. ratio	
F	Ax. Flow	86.5	3.0		Private com. from E.R.Sharp
G	Ax. Flow	77	5.0	Approx. rated speed	R.M. No. E7J20
H	Centrifugal	79	2.23		R.M. No. E7E05
I	Centrifugal	80	2.85	(Revised model of H)	R.M. No. E7G03a
		79	2.7		R.M. No. SE7L12
		81	1.9		R.M. No. SE8A22
		72	3.93	Max. press. ratio	
J	Mixed Flow	76	1.34	Max. Effic.	R.M. No. E7O05a
		75	2.4		
K	Supersonic	79	1.93		R.M. No. E7L10

the center of the reactor and the power level dropping off at each end to a value equal to 20% of the peak heat flux. The radial flux distribution was considered uniform, although it was realized that this ideal situation would only be approximated in practice. The effect of the tube diameter and reactor geometry, including the compounding of the flow passages, was also investigated. However, except where otherwise indicated, the reactor was assumed to be a single-pass right square cylinder. In computing the reactor dimensions, allowance was made for the headers required to admit the air into the reactor. The mass velocity in the headers was made the same as the mass velocity in the pile passages, since higher values of the header mass velocity would lead to excessive pressure losses. Further considerations are given in the section titled "Reactor Heat Transfer".

In this chapter are presented the combinations of reactor dimensions which satisfy the heat-transfer conditions. The combinations of reactor dimensions which satisfy the nuclear requirements are presented in Chapter IV. Finally, in Chapter VII, reactor sizes are selected which conform both to heat-transfer and nuclear requirements.

In calculating the weight of the reactor core, it was assumed that the matrix has a specific gravity of 2, corresponding roughly to that of graphite. Inasmuch as the reactor core weight is virtually negligible, the results are not sensitive to this assumption.

Reactor Outlet Ducts. The air leaving the reactor passes through ducts in the shield before entering the turbine. The exit duct air pressure drop was assumed to be one dynamic head, based on the air density and velocity at the exit of the heated section of the reactor.

Turbine. The power generated in the turbine is determined by the turbine pressure ratio, efficiency and inlet temperature. If a turbojet unit is to be employed, the turbine pressure ratio is adjusted so that the power produced by the turbine is just great enough to drive the compressor and accessories. On the other hand, if a ducted fan or propeller system is to be employed, the air passing through the turbine is expanded to nearly atmospheric pressure in order to produce the additional power required to drive these units.

A high pressure ratio across the compressor implies a high pressure ratio across the turbine. The turbine pressure ratio will be considerably greater for a turboprop than for a turbojet, since in the latter a large fraction of the total pressure drop occurs across the propulsion nozzle. In either case a multistage turbine would probably be desirable from a performance standpoint.

Multistage turbines are not commonly used in turbojets and turboprops, and while the thermodynamic design presents no unusual problems, successful mechanical operation may require considerable development. As is the case with the compressor, the problems will be greatly magnified if the flow is large (high thrust per unit), since long blades will be required, and both centrifugal and vibratory stresses will probably be increased.

A relatively short life -- of the order of 100 hours -- might be considered satisfactory for a nuclear power plant, but it is difficult to design to a high degree of accuracy, and a considerable margin of safety would have to be allowed. At high temperatures a very small change in either stress or temperature will change the life by a large amount. Moreover, the temperature control system has not yet been designed, and the excess of temperature which may result from the inherent limitations of the control system is not known.

The overall turbine efficiency is defined as the ratio of actual turbine shaft-power output to the power available in an isentropic expansion from the given initial total pressure and temperature to the total pressure at the turbine exit. The turbine is thus not charged with the power represented by the kinetic energy and pressure head of the exhaust gases. Because of reheat effects, the overall turbine efficiency may be slightly higher than the individual stage efficiencies. This is in contrast to the compressor where the overall efficiency is always lower than the individual stage efficiencies. In addition, because of boundary layer effects, turbine-stage efficiencies are inherently greater than compressor-stage efficiencies. Consequently, overall turbine efficiencies are usually greater than corresponding compressor efficiencies. Present-day single-stage turbines exhibit efficiencies of 80 to 90% (References 13 and 14). A study conducted by United Aircraft Corporation (14) indicated that the efficiency of the turbine will be relatively independent of the design pressure ratio. Consequently, it appears that a turbine efficiency of 83% is quite feasible today and in five to ten years 87% is foreseeable. In most of the Lexington Project calculations, a value of 87% has been used, but when the effect of turbine efficiency on the power plant characteristics was studied the interval of 83% to 100% was covered.

Turbine inlet temperatures of 1300 to 2100°F (704 - 1149°C) have been considered. Present units operate at inlet temperatures up to about 1500 to 1600°F (815 - 871°C). Higher temperatures result in better theoretical performance, but impose more severe manufacturing problems. Lower temperatures drastically reduce the performance of the unit. It is to be expected that work now in progress in the fields of new materials and special cooling techniques will permit turbine temperatures to reach 1800°F (982°C) or higher in the next five to ten years. The materials problems of the power plant machinery are discussed more fully in a later section of this chapter.

Exhaust Nozzle. After leaving the turbine, the hot air is ejected from the airplane by means of an exhaust nozzle. The velocity of the air leaving the nozzle is determined by the temperature and pressure of the air leaving the turbine and by the nozzle efficiency.

The nozzle efficiency is defined as the ratio of the actual to the isentropic decrease of enthalpy for expansion from the total pressure and temperature at the turbine exit to ambient atmospheric pressure. The efficiency of the propulsion nozzle itself is usually high, but there is likely to be a significant loss in the duct between the turbine exit and the propulsion nozzle. However, it is considered that with careful

attention to details it should be possible to obtain an overall exhaust duct and nozzle efficiency of 90% in a large airplane, and this value has been used in the Lexington Project calculations.

Propeller. The turboprop differs from the turbojet in that the expansion of the gases in the turbine is carried nearly to atmospheric pressure, so that the shaft power delivered by the turbine is sufficient to drive not only the compressor but also a propeller which furnishes most of the propulsive thrust. At the present time it is not possible to maintain the efficiency of propellers at high flight speeds. For instance, the efficiency of the NACA 4-(5)(08)-03 propeller drops from 90% at a flight Mach number of 0.6 to 50% at a flight Mach number of 0.85 (Reference 20). Theoretical studies (16) (17) indicate that in the future this loss in efficiency can be significantly reduced, permitting propellers to be effectively used for airplane propulsion at flight speeds up to Mach number 1.5. However, the mechanical problems associated with the development of high-speed propellers are very great, and some authorities doubt that the theoretical predictions will be achieved.

Ducted Fan. In some cases it is advantageous to obtain a given thrust with a discharge velocity intermediate between that of a turbojet and turboprop, chiefly to improve the propulsive efficiency by obtaining a better ratio of jet velocity to airplane velocity. Intermediate values of velocity and flow can be obtained by the use of a ducted fan, which is essentially a multistage propeller enclosed in a duct. The possibilities of ducted fans have not been sufficiently explored to permit a reliable estimate of their utility. Chew (19) has presented some information. It is probable that the ducted fan will offer some advantages in the flight speed range near Mach number 1, but these may be offset by the more difficult installation problems.

Compressor Frontal Area. The magnitude of the compressor frontal area depends upon the value of the ratio of the inner to the outer radius of the first-stage compressor blades (often called the hub-tip ratio), and the volumetric flow rate of the air entering the compressor. In nearly all turbojets and turboprops designed up to the present time, the hub-tip ratio has been at least 0.65, and usually 0.7 or more. An increased flow capacity, and thus an increased thrust, can be obtained without increase of outer diameter by using smaller values of these ratios, but these ratios probably cannot be made much smaller without considerable loss of efficiency and perhaps excessive blade stresses. Therefore, an axial-flow compressor with a hub-tip ratio of 0.707 was assumed for these studies.

The outer diameter of a centrifugal compressor will of course be greater than that of an equivalent axial flow unit because of the location of the diffuser beyond the impeller. A multistage centrifugal compressor with intercooler, such as would be required for an overall pressure ratio of 40, would probably weigh about as much as a corresponding axial flow unit, and will present a more awkward installation problem.

The Mach number of the air entering the compressor affects the required frontal area. Low gas velocities require large areas, whereas high velocities introduce undesirable compressibility effects. An inlet

axial Mach number of 0.4 represents a reasonable compromise, and has been utilized in the Lexington Project open-cycle turbojet calculations.

Machinery Weights. Time was not available to permit the detailed component and installation studies needed to determine accurately the weights of the various power plants considered. The ratio of the weight of the complete unit to the rated sea level thrust was calculated for a number of turbojets for which these data were known (13), and it was found that the departures from the average value of approximately 0.5 pounds of weight per pound of thrust were quite small. The weights included the weight of the conventional combustion chamber, which would not be used in a nuclear power plant, but this is such a small fraction of the total weight of a unit that it can be disregarded.

For these same units the specific thrust (or specific impulse) was also found. The sea level static values for the individual units did not differ much from an average value of approximately 53 pounds of thrust per pound per second of air flow.

As a result of this review, a turbojet unit with a compression ratio of 4, exhibiting a specific weight of 0.5 pounds per pound of thrust, and a specific thrust of 53 at sea level was selected as a basis for the weight calculations. Corrections were applied to allow for the differences between the power plants considered in the Lexington Project studies and this basic turbojet model, according to the following arguments.

The weight of a turbojet unit is primarily a function of the volumetric air flow rate and the compressor pressure ratio. The flow rate determines the size of the ducts and the frontal area of the compressor and turbine, and consequently the weight of the unit is approximately proportional to the volumetric air flow. For a given thrust, the required volumetric air flow rate is inversely proportional to the product of the specific impulse and the density of the air entering the compressor.

The compressor pressure ratio determines the number of stages in both the compressor and turbine, and hence sets the length of these components. Since the pressure ratio per stage is nearly constant, the number of stages, and hence the axial length of the compressor and turbine is approximately proportional to the logarithm of the compressor pressure ratio.

In addition to these factors, 20 per cent allowance was included for the intercooler, extra shafting and ducting, and other uncertain factors which are indeterminate at a preliminary stage in the analysis. The equation which resulted from these considerations and which was utilized to calculate the weight of the turbojet machine is as follows:

$$\frac{W_{tj}}{F} = \left(\frac{53}{I}\right)\left(\frac{\rho s\, l}{\rho c I}\right)\left(0.36 + 0.4 \log_{10} R_c\right)$$

where W_{tj} denotes the power plant machinery weight, F the thrust, I the

specific impulse, ρ_{sl} the density of air at sea level, ρ_{cI} the density of the air entering the compressor under flight conditions, and R_c the compressor ratio. It is recognized that this relation is approximate. However, the general form agrees quite well with the trend predicted by Pavlecka (2) from his analysis of axial-flow compressor characteristics. Fortunately, the power plant weight is a small fraction of the necessary shield weight in most cases, becoming important only at high altitudes. Hence, while the design ceiling of the open-cycle nuclear-powered airplane cannot be accurately predicted at this time, the calculated performance at altitudes near 30,000 feet should not be grossly in error.

The specific weight of the turboprop power plant is greater than that of the turbojet. This is due to the propellers, gearing, greater turbine expansion ratio, and more complicated installation problems. A rough analysis of these factors indicates that a turboprop unit will weigh 50% more than a turbojet unit of the same thrust. This increase factor is in line with the data presented in Reference 3. The equation used to calculate the weight of the turboprop machinery is as follows:

$$\frac{W_{TP}}{F} = 1.5 \left(\frac{58}{I}\right) \left(\frac{\rho sL}{\rho cI}\right) (0.36 + 0.4 \log_{10} R_c)$$

Results

The quantitative results of the design calculations for the open-cycle nuclear power plant calculated by the Lexington Project according to the methods of Appendices III-A and III-E are presented in Figures III-9 to III-48. These plots show the effect on the reactor volume, reactor diameter, power plant weight (including reactor but omitting the shield), and compressor frontal area of the various factors which influence the performance of the power plant. Thrust has been selected as the independent variable inasmuch as it is directly related to the airplane gross weight.

It must be emphasized that the reactor dimensions have been based solely on heat-transfer considerations in this chapter. Although the plotted data are largely based on a reactor free-flow ratio of 0.4, this ratio is under the control of the designer. A very simple relation connects the diameter of the reactor and the free-flow ratio, namely,

$$D\sqrt{FFR} = \text{constant}$$

providing other conditions are unchanged. With this rule the charts of this chapter may be used for finding the reactor diameter and length corresponding to any free-flow ratio. As shown in Chapter VII, the actual selection of reactor length and diameter and of free-flow ratio is based both on heat transfer and nuclear considerations.

Although the total power plant weight and total compressor frontal area are presented in the charts, it is extremely unlikely that a single compressor or turbine would be employed in practice. Instead, a number of moderate sized units are envisaged. Such a setup would increase the flexibility and reliability of the power plant and decrease the cost and difficulties of manufacture. However, the installation difficulties would be increased, and a detailed study would be required to determine the optimum number of units. Due to the shield weight, only one reactor would be used.

An important consideration which has not been plotted in the charts is the intercooler size. The intercooler frontal area is significant, and unless the device is carefully designed, it may become excessive. Preliminary estimates for a single design condition indicate that it may be possible to reduce the intercooler frontal area to a value which is nearly the same as the compressor frontal area. If this is done, the intercooler will not be a major installation bottleneck.

Due to time limitations, it has not been possible to investigate the detailed interactions resulting from the systematic permutation of the pertinent variables. Furthermore, only design point operation has been considered. For instance, take-off conditions have not been investigated in detail. If extra thrust were required at take-off, it could be provided by JATO units or by burning a small amount of chemical fuel in the air leaving the reactor or turbine.

The study required for an actual power plant design must cover all of the aspects of the power plant in a much more comprehensive fashion than was possible here. However, it is felt that the Lexington Project analysis covers those phases of the open-cycle power plant which are critical to an assay of feasibility.

Comparison of the Turbojet and Turboprop Power Plants. Figures III-9, III-10, III-11, III-12 present a comparison of the open-cycle turbojet and turboprop power plants when designed for an altitude of 30,000 feet and a flight Mach number of 0.9. These calculations assume that a propeller efficiency of 0.8 may be obtained at the above operating conditions. Present-day propellers do not achieve these efficiencies, and a considerable effort will be required if a value of 0.8 is to be realized. Despite the optimistic propeller efficiency, the turboprop unit is only slightly better than the turbojet. At a given thrust, the larger specific impulse of the turboprop results in a decreased air flow through the machinery. Consequently, the reactor diameter and volume (set by heat transfer) and the compressor frontal area are smaller for the turboprop than for the turbojet. However, the machinery of the turboprop unit weighs more than the machinery of the turbojet. The performance of the turboprop unit at flight speeds in the vicinity of Mach number 1 is not significantly better than that of the turbojet, and in view of the difficulties associated with the installation and operation of efficient high speed propellers, the turbojet is to be preferred to the turboprop. This statement is not valid at low flight speeds where significant advantages can be obtained by the use of propellers. Since at Mach numbers

near unity the propeller did not seem desirable, it was felt that although the ducted fan might be better in some respects, it would not offer startling advantages and because of limitations was not given further consideration. This does not mean that the ducted fan should not be included in a more comprehensive survey than was possible here.

The Effect of Design Altitude. Figures III-13, III-14, III-15, III-16 illustrate the effect of airplane design altitude on the open-cycle turbojet power plant. Increased design altitudes increase the size and weight of the power plant and reactor. The thrust produced by a given power plant is directly proportional to the mass of air handled. As the design altitude of the operating aircraft is increased, the density of the air taken into the propulsive system decreases. Consequently, for the same thrust the volume of air swallowed by the power plant increases. The size of the machinery, ducts and reactor are all increased by an increase in the volumetric flow rate of the working fluid. As a result, when the design altitude is increased, a larger and larger fraction of the load-carrying capacity of the plane must be devoted to the power plant and reactor, and the flyable shield weight must be reduced. The results presented in Figures III-13, III-14, III-15, and III-16 indicate that a nuclear-powered airplane utilizing an open-cycle turbojet unit will have a maximum design altitude in the vicinity of 60,000 feet, even if no shield were required. This figure will be reduced to perhaps 50,000 feet when a shield of adequate thickness is included.

The Effect of Design Flight Speed. Figures III-17, III-18, III-19, and III-20 illustrate the effect of flight speed on the open-cycle turbojet power plant. The performance of a turbojet power plant is relatively independent of the flight speed of the airplane. This does not imply that a given power plant will operate equally effectively at all flight speeds, but that with proper design, turbojet design points can be varied over a wide range of speeds up to about Mach Number 2.

Increasing the flight speed decreases the required mechanical compressor ratio and decreases the specific impulse. At high flight Mach numbers, the ram pressure rise is significant. For 30,000 feet altitude, at $M \approx 0.5$, the ram pressure rise is approximately equivalent to a compressor pressure ratio of 1.17; at $M \approx 0.9$, it increases to 1.61; at $M \approx 1.5$ to 3.3; and at $M \approx 3$ to 29. Consequently, with the same compressor exit pressure, the mechanical compressor ratio and frontal area may be decreased as the flight Mach number is increased. This possibility has been utilized in the calculations leading to the results presented in Figures III-17 to III-20. The reduction in compressor pressure ratio results in a decrease in the machinery weight per pound of air handled. Above a flight Mach number of about 2, a mechanical compressor is no longer necessary, and the turbojet power plant logically simplifies to a ramjet.

The specific impulse decreases with increase in flight speed; this results in an increase in the air-flow rate and consequently in the size of the reactor required for a given thrust. The lower specific impulse also tends to counterbalance the decrease in machinery weight afforded by lower mechanical compressor pressure ratios.

With a turbojet power plant, the difficulties associated with supersonic flight speed are largely due to aerodynamic factors. At the present time, airplane configurations suitable for supersonic speeds have appreciably lower lift-to-drag ratios than do airplane configurations designed for subsonic speeds. Thus, for successful supersonic flight, either the performance of supersonic aircraft must be radically improved, or the performance of the turbojet power plant must be materially improved. Such an improvement would require substantially higher working fluid temperatures than those necessary for subsonic designs.

Effect of Component Efficiencies. Figures III-21, III-22, III-23, and III-24 illustrate the effect of component efficiencies on the open-cycle turbojet power plant. An increase in the component efficiencies of the turbojet unit always results in an increase in the power plant performance. It is believed that, given sufficient time and effort, it would be possible to design and build units exhibiting an inlet diffuser efficiency of 0.9, a compressor efficiency of 0.82 at a pressure ratio of 40 to 1, a turbine efficiency of 0.87, and an exhaust nozzle efficiency of 0.9. It is possible that efficiencies slightly higher than these could be obtained. However, as Figures III-21 to III-24 show, the foreseeable efficiencies are already so high that tremendous gains cannot be expected from component improvements. On the other hand, as is pointed out in more detail later, low efficiencies result in a serious loss in performance.

Effect of Turbine Inlet Temperature. Figures III-25 through III-32 illustrate the effect of turbine inlet temperature on the open-cycle turbojet power plant. At fixed component efficiencies and compressor pressure ratios, there is a minimum turbine inlet temperature at which the power plant will produce net useful work. With foreseeable component efficiencies ($\eta_D = 0.90$; $\eta_c = 0.82$; $\eta_T = 0.87$; $\eta_N = 0.90$), and at a compressor pressure ratio of 40, a turbine inlet temperature of less than about 1300°F (704°C) will not produce net thrust. If component efficiencies are increased or compressor pressure ratios decreased, the minimum turbine inlet temperature can be decreased. However, a decrease in the compressor pressure ratio adversely affects the reactor size, and it appears that a compressor pressure ratio of between 20 and 40 is near optimum.

If the reactor wall temperature were not a limiting factor, an increase in the possible turbine operating temperature would always be desirable. However, since the reactor wall temperature is probably controlling, there is an optimum turbine temperature. For instance, with a maximum wall temperature limited to 1832°F (1000°C), the optimum turbine inlet temperature is approximately 1650°F (899°C). With higher reactor wall temperatures, the optimum turbine temperature rises. Present operational equipment utilizes turbine inlet temperatures of the order of 1500°F (815°C), but new materials or special cooling techniques can be expected to raise this maximum allowable temperature to 1800°F (982°C) or higher.

Effect of Maximum Reactor Wall Temperature. Figures III-33, III-34, III-35, and III-36 illustrate the effect of maximum reactor wall temperature on the open-cycle turbojet power plant. An increase in the maximum

reactor wall temperature decreases the reactor volume and diameter. If the turbine temperature is fixed, the power plant machinery weight and the compressor frontal area are unaffected. In general, the maximum wall temperature should be at least 50°F (10°C) greater than the turbine inlet temperature. If heat-transfer considerations and turbine temperatures limit the design, increasing the difference between the turbine inlet and the maximum wall temperature is desirable since the size of the reactor may be decreased. However, no significant reduction in the reactor dimensions is obtained when the difference between the maximum wall temperature and the turbine inlet temperature is allowed to exceed 500°F (260°C). Maximum reactor wall temperatures less than about 1700 or 2000°F (927 - 1093°C) result in an unattractive open-cycle power plant.

Effect of Compressor Pressure Ratio. Figures III-37, III-38, III-39, III-40 illustrate the effect of compressor pressure ratio on the open-cycle turbojet power plant. An increase in the compressor pressure ratio decreases the required reactor volume and diameter, but increases the power plant weight and the compressor frontal area. The decrease in the size of the reactor is a result of the increased density of air flowing through the reactor. These data indicate that a compressor pressure ratio lying between perhaps 20:1 and 40:1 is a reasonable compromise between the conflicting requirements of small reactor size and low power plant weight. These high compressor pressure ratios imply the use of an intercooler.

The Effect of Reactor Pressure Drop. Figures III-41, III-42, III-43, and III-44 illustrate the effect of reactor pressure drop on the open-cycle turbojet power plant. The term "reactor pressure drop" includes the loss in the shield ducts as well as in the reactor. An increase in the reactor pressure drop results in a decrease in reactor size; offsetting this effect is an increase in the power plant weight and the compressor frontal area. The optimum reactor pressure drop appears to lie in the range between 20% and 40% of the pressure of the air leaving the compressor; but within this region the performance of the power plant is relatively insensitive to the design pressure loss.

The Effect of Reactor Geometry. Figures III-45, III-46, III-47, and III-48 illustrate the effect of reactor geometry on the open-cycle turbojet power plant. For a given air temperature rise and reactor pressure drop, the reactor geometry does not influence the power plant machinery. However, the pile configuration strongly affects the reactor dimensions.

All of the reactor dimensions presented in Figures III-45 to III-48 are based solely on heat-transfer requirements. Although the minimum pile sizes set by heat transfer are frequently too small from a nuclear standpoint, the necessary heat can always be extracted from a larger reactor. There are several possible methods of partially offsetting the disadvantages incurred when nuclear factors require larger reactors than the optimum dimensions set by heat transfer. The pressure drop across the reactor may be decreased, resulting in a higher specific impulse. The maximum wall temperature may be decreased even though the air temperature rise is unchanged. The free-flow ratio and the tube spacing may be varied, easing the structural and nuclear problems.

An increase in the power generated per unit volume of reactor is almost always accompanied by an increase in the thermal stresses. Unless the materials are carefully chosen and ingenious structural configurations devised, the thermal stresses resulting from high power output in small-size reactors may cause mechanical failure. The characteristics of the reactor as a heat-transfer system are discussed more fully in a later section of this chapter; the nuclear aspects are covered in Chapter IV.

Conclusions

1. Although extensive research and development would be required to utilize the open-air cycle power plant to propel a nuclear-powered plane, this power plant offers some promise of ultimate success.

2. The primary problem is the development of materials for use in the construction of the reactor. These materials of construction must not only meet the nuclear requirements, but also must:

(a) withstand temperatures of 1700°F (927°C), at the very minimum, and in all probability, high values.

(b) prevent the contamination of the coolant by fission fragments

(c) withstand chemical attack by hot air

(d) withstand the thermal stresses introduced by the high heat flux densities required

(e) retain satisfactory properties after being subjected to radiation damage.

3. Materials for the manufacture of power plant components other than the reactor do not appear to be a bottleneck. Better materials than those are now available, especially for turbine blades, would be desirable, but it is probable that such advances will be made.

4. The required component efficiencies appear attainable, although such problems as the development of efficient high pressure ratio compressors and turbines will involve a considerable effort.

5. Even if great strides are made in propeller development, the propeller or ducted-fan propulsive systems do not appear to offer significant advantages over the turbojet at flight Mach numbers above 0.8.

6. The maximum design altitude of a nuclear-powered aircraft propelled by an open-cycle engine will be in the vicinity of 60,000 feet, even if no shield were carried. With an adequate shield the maximum design altitude is not likely to exceed 50,000 feet.

7. The open-cycle turbojet is not limited to subsonic flight speeds *per se*, but the aerodynamic characteristics of present supersonic airplanes must be improved before supersonic nuclear flight is possible.

HELIUM GAS-TURBINE COMPRESSOR-JET

Introduction

The gas-turbine compressor-jet is one of the more promising of several possible closed-cycle propulsion systems.

Figure III-4 is a schematic diagram of the proposed system, which is described more fully in the Introduction to this chapter. In order to evaluate the applicability of the helium gas-turbine compressor-jet to nuclear-powered aircraft, it is desirable to estimate the weight of the power plant per unit thrust, the frontal area per unit thrust, the size of the reactor and shield, as well as to evaluate the practical problems.

In the only previous study of the helium cycle, that of NEPA (1), the conclusion was reached that this type of propulsion system was not feasible, primarily because of the size of the radiators which might be necessary. However, the basic assumptions of the study seemed pessimistic, and, in addition, the cumbersome arrangement of the cycle was more suited to stationary power plants than to aircraft power plants. In order to insure that no possible approach was overlooked, the Lexington Project re-evaluated this system on the basis of an improved arrangement.

The helium gas-turbine compressor-jet, to a much greater extent than the open-cycle air turbojet, has a large number of design parameters which play interconnecting roles in determining the performance of the entire aircraft. Unless reasonably good combinations of these parameters are chosen, the resulting designs will not represent the true potentialities of the power plant. Hence an optimization procedure involving systematic variations in the significant parameters is essential to a fair appraisal of the power plant. Because of time limitations and the extraordinary number of parameters, the optimization for the present studies was carried out in only crude fashion. The general procedure was as follows:

First, the entire propulsion system was divided into three parts: the helium gas-turbine cycle, the propulsive duct (or compressor-jet), and the reactor.

Second, the general characteristics of the helium gas-turbine were studied separately. As a result of this study eight specific helium cycles were selected for possible incorporation into complete propulsion systems. Judgment was exercised as to the suitability of the specific helium cycles for the overall propulsion systems. This does not guarantee, however, that the optimum choices were made.

Third, the general characteristics of the propulsive duct for each of the eight specific helium cycles were explored. A specific design for the propulsive duct was then selected to go with each helium cycle. Here again the choice was not necessarily the optimum.

Fourth, for each specific combination of helium cycle and propulsive duct a number of possible reactor configurations were determined.

Finally, using the previously chosen designs for the helium cycle, propulsive duct and reactor, the characteristics of a number of complete helium gas-turbine compressor-jets were studied.

Working Fluid

The desirable characteristics of the working fluid are:

(a) Lack of chemical reactivity toward the materials used in the reactor and machinery.

(b) Lack of induced radioactivity.

(c) Low molecular weight.

Helium fulfills all of these requirements to a remarkable degree. Although hydrogen has a lower molecular weight, it does not satisfactorily meet the other criteria. There are gases which are perhaps satisfactory in respect to items 1 and 2, but they have relatively high molecular weights. For these reasons, helium was chosen for this study.

Helium Gas-Turbine Cycle

Parameters. The characteristics of a gas-turbine cycle depend on the working fluid, the efficiencies of the compressor and turbine, the temperature at the turbine inlet, the temperature at the compressor inlet, the compressor pressure ratio, and the turbine pressure ratio. In the present studies a compressor efficiency of 85% and a turbine efficiency of 88% were assumed for most of the calculations. These figures are somewhat higher than those now attainable with air compressors, but are perhaps justifiable because of the moderate pressure ratios required for the helium machinery and also because the Mach numbers in the blading will correspond virtually to incompressible flow. In order to determine the manner in which the selection of component efficiencies affects the performance, calculations were also made for a "pessimistic" case in which the compressor and turbine efficiencies were respectively 80% and 82%, and for an "optimistic" case in which these efficiencies were respectively 90% and 92%.

The choice of turbine inlet temperature may be dictated either by the maximum allowable temperatures for blading materials or by the need for maintaining a reasonable temperature difference in the reactor. A range of turbine inlet temperatures from 1340 to 1800°F (727-982°C) was used in these studies.

The temperature at the helium compressor inlet is a critical factor in the performance not only of the helium cycle but also of the propulsive duct. For best overall performance the selection of this temperature should be based on an optimization procedure. After some preliminary calculations, the range of variation for this parameter was narrowed down to limits between 160 and 240°F.

The pressure ratio of the helium compressor primarily influences the performance of the helium cycle. A range of pressure ratios from 2 to 6 was examined in these studies.

The size of the reactor, ducting and radiator can be reduced by permitting large pressure drops in these components, but only at the expense of a reduction in the efficiency of the helium cycle. The pressure drop should, therefore, be determined by an optimization procedure. For most of the calculations it was assumed that the turbine pressure ratio would be 0.9 of the compressor pressure ratio. Some auxiliary calculations were made to determine how corresponding values of 0.80 and 0.95 would affect the helium cycle separately.

Criteria of Performance. Three criteria of performance were chosen for assessing the suitability of the helium gas-turbine as a component of the complete propulsion system:

(a) The thermal efficiency, defined as the ratio of net shaft work to reactor heat input, is a measure of the principal output of the gas turbine. However, both the work output and the heat rejected from the helium cycle add to the performance of the propulsive duct.

(b) The work ratio, or the ratio of net shaft work to turbine work, is a measure of the sensitivity of the gas turbine to slight changes in component efficiency or temperature. Values of the work ratio below approximately 0.2 are probably undesirable.

(c) The specific helium flow, or pounds of helium per net horsepower-hour of shaft work, is a measure of the necessary rate of circulation of helium and consequently of the size of the reactor, helium machinery and helium ducts.

Assumptions and Method of Calculation. The characteristics of the helium cycle were explored for the three sets of assumptions shown in the following table:

	"Pessimistic" Assumptions	"Working" Assumptions	"Optimistic" Assumptions
Comp. Efficiency, %	80	85	90
Turb. Efficiency, %	82	88	92
Turb. Press. Ratio / Ratio Comp. Press.	0.80	0.90	0.95

The method of calculation is described in Appendix III-B.

Results. Figures III-49 to III-57 show the performance of the helium gas-turbine cycle alone. The following discussion refers principally to Figures III-49 to III-51, which are based on the working assumptions.

For each combination of turbine inlet temperature and compressor inlet temperature, it is desirable from one standpoint to work at the peak of the efficiency curve of Figure III-49. On the other hand, it is desirable to work at the minimum point on the curve of specific helium flow of Figure III-51. And, finally, points of low work ratio in Figure III-50 should be avoided. The best choice of compressor pressure ratio depends on a compromise between these more or less conflicting considerations.

The choice of working parameters was made somewhat arbitrarily and is indicated by the circled Roman numerals of Figures III-49, III-50 and III-51. The eight combinations of variables which were selected may be summarized as follows:

Case:	I	II	III	IV	V	VI	VII	VIII
Turb. Inlet Temp. °F	1540	1540	1540	1340	1800	1800	1800	1440
Comp. Inlet Temp. °F	200	200	160	200	240	160	200	200
Comp. Press. Ratio	3	4	3	3	3.65	4.25	4.0	3.0

A comparison of the charts for the different sets of assumptions shows that an improvement in component efficiency to the values listed as "optimistic" does not lead to startling improvements in performance. On the other hand, a reduction in component efficiency to the "pessimistic" values causes the cycle efficiency to drop to what is probably an impractical value.

Propulsive Duct

Parameters. For a given design flight speed and altitude, and a specific helium cycle (including compressor pressure ratio, turbine pressure ratio, compressor and turbine inlet temperatures, and compressor and turbine efficiencies), the performance of the propulsive duct depends on the efficiencies of the inlet diffuser, air compressor and exhaust nozzle, on the ratio of helium flow to air flow, and on the average Mach number through the radiator tubes.

The ratio of helium to air flow is of consequence in determining the amount of air flow for a given helium cycle and a given reactor power, and, therefore, influences the size of the air compressor and radiator.

The choice of a value for the ratio of helium-to-air flow is subject to two conflicting requirements. The ratio should be high in order to

keep frontal areas small and machinery weights low, but if the ratio is made too high the terminal temperature differences in the heat exchanger become too small. For a given group of helium conditions, a maximum value of the helium-to-air ratio, corresponding to zero temperature difference at one end of the radiator, can be calculated. This limiting ratio generally was found to be from about 0.15 to 0.20, and both the conflicting requirements can be satisfied rather well with a ratio about 0.03 below the limiting value. This would give reasonable radiator temperature differences as well as fairly low airflows.

Since the radiator cannot have a large frontal area if it is to be installed easily in an aircraft, the air velocity in the radiator tubes should be fairly high. On the other hand, an excessively high air velocity produces large pressure drops and thus reduces the efficiency of the propulsive duct. The selection of the proper air velocity therefore requires an optimization procedure. For the present calculations the mean Mach number in the radiator tubes was varied from 0.1 to 0.3.

Criteria of Performance. The following quantities were used for appraising the suitability of the different designs of the propulsive duct as part of a complete helium gas-turbine compressor-jet:

(a) The frontal area per unit of thrust is a measure of the difficulty associated with the installation of the machinery in the airplane. In some cases the frontal area is determined by the air compressor and in other cases the maximum frontal area is that required by the radiator. The figures reported for frontal areas in each case are based on the larger of the two values.

(b) The reactor power per unit of thrust is perhaps the best overall measure of the efficiency of the propulsion system.

(c) The weights of the various components per unit thrust are important in determining how much of the gross aircraft weight is available for carrying a shield.

Assumptions and Method of Calculation. For reasons given previously, most of the calculations were made at a flight Mach number of 0.9 and an altitude of 30,000 feet. Variations in altitude were included in some of the radiator calculations to show the effect of altitude on power plant performance.

The efficiency of the inlet diffuser was assumed to be 90%, in conformity with a similar assumption for the other machines considered in this chapter.

Since the air compressor requires a rather moderate pressure ratio, i.e., in the vicinity of 2, an efficiency of 85% was assumed for it, which is somewhat higher than the figure for existing trubojets. In estimating the frontal area of the air compressor it was assumed that the axial Mach number at the entrance to the annulus would be 0.5, and that the total frontal area would be 1.8 times the annulus area.

The exhaust nozzle was assumed to have an efficiency of 92%.

For purposes of calculation, the radiator was assumed to be of the shell-and-tube type with air flow inside the tubes and helium flow outside the tubes counter-current to the air. The heat exchanger dimensions were calculated on the assumption that the air-side heat-transfer coefficient was controlling. The tubes were taken to be of 1/8 inch inside diameter, with a wall thickness of 0.004 inch, and with a density of 490 pounds per cubic foot. The radiator weight was assumed to be twice the weight of the tubes, and the total frontal area was taken to be twice the total inside cross-sectional area of the tubes.

It is worth noting that while the radiator dimensions and weights were calculated for the geometry described above, the results are also approximately applicable to other types of radiator of good aerodynamic design -- such as a plate-type radiator.

The weight of the radiator was never a significant fraction of the gross weight of the airplane, and the radiator frontal area was frequently smaller than the compressor frontal area.

The weights of the air compressor and the helium machinery were based on data for present-day turbojets with rough corrections for altered operating conditions.

The weight of the helium machine (turbine and compressor) was taken as 2.5 pounds per cubic foot per second of volume flow at the compressor inlet, for a compressor pressure ratio of 4. For other pressure ratios it was assumed that the weight of the machine varies as the 0.6 power of the number of stages, with the stage pressure ratio taken as 1.05. To allow for gearing and ducting, the weight obtained above was multiplied by 1.25 for a reactor inlet pressure of 200 psi and by 2.00 for a reactor inlet pressure of 1000 psi, with logarithmic interpolation used for other pressure levels.

The weight of the air compressor and associated gearing and shafting was taken as 1 pound per cubic foot per second of volume flow at the compressor inlet for a pressure ratio of 2. For other pressure ratios, it was assumed that the weight varies as the 0.6 power of the number of stages, with a pressure ratio per stage of 1.15.

These methods of calculating machinery weights are consistent with the corresponding methods used with the open-cycle turbojet.

The calculation procedure, together with additional assumptions, is presented in Appendix III-B.

Results. The characteristics of the propulsive duct for each of the eight specific helium cycles are shown in Figures III-58 to III-89, inclusive.

To clarify the meaning of these charts, the results of helium cycle I (Fig. III-58 to III-61) will be referred to.

For a given ratio of helium-to-air flow an increase in the air speed through the radiator tubes tends to reduce the frontal area of the radiator. The increased pressure drop through the radiator, however, tends to reduce the thrust of the propulsive duct, so that in the range of interest the curve of radiator frontal area per unit thrust against reactor power per unit thrust tends to drop, while that of compressor frontal area tends to rise. The combined curve showing the maximum frontal area therefore goes through a minimum which investigation shows corresponds to the condition for which the two areas are equal. These ideas are illustrated in Figure III-58. Considering frontal area and reactor size, it is desirable to select conditions to the left of the minimum point on Fig. III-58. Figure III-58 also shows that there is an optimum value of the ratio of helium-to-air flow if it is desired to keep the frontal area to a minimum. If curves for a greater number of values of this ratio were drawn, the family of curves would have an envelope, and it would be desirable to select points on this envelope and to the left of the minimum. The final selection of the design point would depend on an optimization procedure. Within the time available for this study, it was necessary to make partially optimized selections of the helium-to-air flow ratio and of the average Mach number in the radiator tubes, using charts like Fig. III-58 as a guide.

Arguments like those given above for the frontal area apply as well to the radiator weight, as shown in Fig. III-59. Since, for altitudes up to about 50,000 feet, the frontal area of the radiator is more crucial than its weight, the area rather than the weight was made the basis for choosing the design of the propulsive duct.

Fig. III-61 shows that the air compressor weight is likely to be sensitive to the value of the helium-to-air flow ratio employed. However, reference to Fig. III-58 shows that helium-to-air flow ratios which are desirable from the standpoint of frontal area are also desirable from the standpoint of air compressor weight.

According to Fig. III-60 the weight of the helium machinery at a pressure level in the reactor of 1,000 psi is of the order of 0.1 to 0.2 pounds per pound of thrust, and, therefore, does not play a decisive part in the design of the helium cycle and the propulsive duct.

Selection of Propulsive Duct Design. Using the type of reasoning given above, several specific design points were chosen for incorporation into complete helium gas-turbine compressor-jets. These specific design points are shown by the circles on Figures III-58 to III-89.

Reactor

Parameters. The reactor design must fulfill both the nuclear and heat-transfer requirements. In this chapter are determined the combinations of reactor dimensions which satisfy the demands imposed by heat transfer. The combinations of reactor dimensions which satisfy nuclear conditions are presented in Chapter IV. Finally, in Chapter VII, reactor

sizes are selected which satisfy both heat-transfer and nuclear considerations.

The heat-transfer parameters which influence the reactor design are the pressure drop required to pump the helium through the reactor, the inlet and outlet temperatures of the helium, the reactor power per unit thrust, the heat flux distribution within the reactor, the reactor geometry, the maximum reactor wall temperature, and the stagnation pressure of the helium entering the reactor.

Assumptions and Methods of Calculation. The studies of the helium gas-turbine and the propulsive duct described previously allowed 5% of the inlet helium stagnation pressure to be expended in pumping the helium through the reactor.

These component studies indicated certain desirable combinations (Cases I through VIII) of the inlet and outlet temperatures of the helium flowing through the reactor and of the reactor power per unit thrust, combinations which were made use of for the reactor calculations. The values selected for these eight cases represent the results of a partial optimization procedure.

Other parameters influence only the reactor portion of the power plant. In these studies, the pile heat flux distribution was not varied. The longitudinal pile heat flux distribution was taken to be sinusoidal with the maximum power level at the center of the reactor and the power level decreasing at each end to a value equal to 20% of the peak heat flux. The radial flux distribution was considered uniform, although it was realized that this ideal situation would only be approximated in practice. Shield cooling was not taken into account. The reactor was assumed to be a single-pass right square cylinder. In computing the reactor dimensions, allowance was provided for headers. The reactor dimensions presented in this section are largely based upon a free-flow ratio of 0.4; however, with all other conditions remaining the same, the values corresponding to other ratios may be obtained by means of the relation:

$$D \sqrt{FFR} = \text{Constant}$$

Reactor sizes corresponding to ranges of maximum reactor wall temperatures between 1600 and 2500°F and helium pressures between 250 and 2000 psi were determined.

It is to be noted that high helium pressures may be used to decrease the maximum wall temperatures required for a given size reactor. Consequently, the final design will be set by balancing the problems imposed by high temperatures against the problems imposed by high pressures.

A general discussion of reactor heat transfer is presented in a later section of this chapter, and a numerical example of a compressible flow heat-transfer calculation is included in Appendix III-E.

Complete Propulsion System

Parameters. The successful synthesis of a complete helium gas-turbine compressor-jet requires the careful integration of each of the components. In this study, only partial optimization of the final design was achieved. Further work is necessary to explore the cycle more fully.

Flight altitude and Mach number are largely determined by tactical considerations. High values of both these variables are desirable, and it would be of interest to explore the variation in performance of the power plant as flight conditions are changed. The Lexington Project selected an altitude of 30,000 feet and a flight Mach number of 0.9 as the minimum values of tactical interest. Operation at higher altitudes (up to 50,000 feet) was examined, but higher speeds could not be investigated in the time available for the study.

Some parameters, such as the maximum reactor wall temperature, the stagnation pressure of the helium in the reactor, and the turbine inlet temperature, are limited by the state of the engineering art. In general, values of these quantities should be high, but not so large that the problem of finding suitable materials of construction is insurmountable. Consequently, the ranges of temperatures and pressures which were explored were confined to regions, below which the power plant performance appeared unattractive, and above which, the solution to fabrication difficulties seemed remote.

The remaining parameters, namely, the helium temperature at the compressor inlet, the helium compressor pressure ratio, the reactor pressure drop, the air compressor pressure ratio, the reactor geometry, and the heat flux distribution, are all largely under the control of the designer. Values for these quantities were selected which appeared to be reasonable and which the early calculations indicated would result in approximately optimum performance.

Criteria of Performance. The criteria of performance selected for assessing the potentialities of the helium gas-turbine compressor-jet as part of a nuclear-propelled aircraft include the size of the reactor, the maximum frontal area of the air-swallowing components (either the compressor frontal area or the heat exchanger frontal area), and the weight of the power plant machinery (including the reactor, but excluding the shield). Thrust was chosen as the independent variable.

The size of the reactor determines the shield weight required for a given amount of radiation attenuation.

The maximum frontal area is a measure of the power plant installation problems. Large frontal areas may impose significant aerodynamic and structural restrictions by requiring a large fraction of the airplane volume to be devoted to air ducts and machinery. In extreme cases, it may not be possible to install the machinery. The weight of the power plant is seldom as large as the necessary shield weight, but at high altitudes

the machinery weight may become so great that the airplane cannot carry a shield of appreciable size.

Assumptions and Methods of Calculation. Utilizing the assumptions discussed previously, operating conditions for the helium gas-turbine, propulsive duct and reactor were selected and combined into several complete power plants, and values of the reactor diameter, maximum frontal area, and power plant weight (excluding the shield) were computed as a function of the gross thrust. The detailed methods of calculation are presented in Appendix III-B.

Results

The results of the performance calculations of the integrated power plants are presented in Figures III-90 to III-99.

Effect of Design Altitude. Figures III-90 to III-93 present the effect of altitude on power plant performance. Altitude affects only the air-swallowing machinery of the helium gas-turbine compressor-jet power plant. As the flight altitude is increased, the density of the air entering the propulsive duct decreases. Since the thrust of the power plant is roughly proportional to the mass of air handled, for a given thrust the volumetric air-flow rate must be increased with increasing flight altitude. Consequently, the maximum frontal area and the total power plant weight increase. The maximum design altitude of the airplane utilizing the helium gas-turbine compressor-jet is expected to be greater than that of an airplane powered by an open-cycle nuclear turbojet. This follows from the fact that the size of the reactor of the helium machine is relatively independent of the flight altitude, whereas the dimensions of the open-cycle reactor must be increased at higher design altitudes.

Effect of Design Flight Speed. Although no calculation of the effect of flight speed on the helium gas-turbine compressor-jet was carried out, it is possible to predict, in a qualitative fashion at least, the influence of flight Mach number. In general, for flight speeds below a Mach number of 0.9, the performance will be slightly better than at Mach number 0.9. At supersonic speeds, however, a decrease in performance must be expected. This is due to the fact that supersonic speeds develop such high ram temperatures that the effectiveness of the radiator will be decreased. In addition, the higher intake velocity will result in a lower relative velocity increase in the air passing through the propulsive duct, and hence a lower specific impulse. As a result of these considerations, it is believed that the helium gas-turbine compressor-jet will perform satisfactorily up to flight Mach numbers of about 1.5, but will become unattractive at higher flight speeds. It is probable, therefore, that if supersonic aircraft lift-to-drag ratios can be improved to a point where they are comparable to subsonic values, low supersonic nuclear-powered flight could be achieved with a helium gas-turbine compressor-jet power plant.

Effect of Component Efficiencies. The effect of component efficiencies on the performance of the integrated power plant was not determined.

However, the results of three assumptions ("pessimistic", "working", and "optimistic") relating to the efficiencies of the components of the helium portion of the cycle have been presented in Figures III-49 through III-57. These data indicate that if the component efficiencies are restricted to the "pessimistic" assumptions, the power plant performance is likely to be unsatisfactory. When compared with the performance computed on the basis of "working" efficiencies, the advantages in performance obtained with the "optimistic" efficiencies are significant but not startling.

Effect of Turbine Inlet Temperature. The effect of turbine inlet temperature on the performance of the helium gas-turbine compressor-jet is presented in Figures III-94 to III-97. A high turbine inlet temperature reduces the sensitivity of the overall performance of the power plant to the effects of such factors as component efficiencies and flight speed. Furthermore, the maximum frontal area required is decreased by an increase in the turbine inlet temperature. The total power plant weight is not grossly affected by changes in turbine temperatures. Although these figures indicate that the performance of the unit remains remarkably uniform even when the turbine inlet temperature is lowered from 1800 to 1400°F, a slight decrease in component efficiencies will markedly reduce the low-temperature performance.

Effect of Maximum Reactor Wall Temperature. The effect of the maximum reactor wall temperature on the performance of the helium gas-turbine compressor-jet is presented in Figures III-98 and III-99. If the turbine inlet temperature and other cycle parameters are kept constant, the maximum reactor wall temperature affects only the dimensions of the reactor. An increase in the maximum reactor wall temperature decreases the reactor dimensions required to meet the demands of heat transfer. On the other hand, high reactor wall temperatures seriously increase the problems of finding suitable structural materials. Consequently, since the advantages of increasing wall temperature significantly above the turbine inlet temperature are not tremendous, it is probable that a temperature difference of the order of 300°F will prove to be satisfactory.

Significance of Results

In view of the extraordinary complexity of the design calculations for this power plant and the relatively small amount of time available for the exploratory studies, it was necessary to make a large number of arbitrary assumptions and to select working parameters which, in some cases, might be far from optimum. For these reasons it is worth assessing the limited results given here in the light of what might be disclosed by a more thorough study.

On some counts the results might be considered optimistic. In other respects they might be considered pessimistic.

As an example, in order to avoid the very lengthy calculations required to relate the radiator to both the propulsive duct and the helium gas-turbine, it was assumed that all the resistance to heat transfer is on the air

side of the radiator. From this standpoint, at least, the calculated radiator frontal areas and weights are too small.

However, there are compensating factors; in particular, the possibility of improvement through a more extensive optimization procedure. For instance, it is possible to increase the pressure drop on the helium side of the radiator at the expense of some reduction in helium-cycle efficiency; the error due to neglecting the helium-side resistance may thus be minimized. Furthermore, a better definition of the envelope curves on such charts as Fig. III-58 would lead both to a choice of smaller radiators and to a better overall efficiency. And finally, as Chapter VII indicates, it might well be desirable to have less helium pressure drop in the reactor and more in the radiator.

Similar arguments, pro and con, may be given concerning other parts of the analysis. It is believed, however, that the results of this study are correct as regards the order of magnitude. And, since the helium gas-turbine compressor-jet seems to offer considerable promise, it is recommended that this system be studied more extensively.

Conclusions

1. Although extensive research and development would be required to utilize the helium gas-turbine compressor-jet power plant for nuclear propulsion of aircraft, this system offers promise of ultimate success.

2. The primary problem is the development of materials for use in the construction of the reactor. These materials must not only meet the nuclear requirements, but also must:

(a) Successfully withstand temperatures of at least 1500°F, and in all probability, higher temperatures

(b) Successfully prevent contamination of the helium by fission fragments

(c) Successfully withstand the thermal stresses introduced by the high heat flux densities required

(d) Successfully withstand radiation damage.

3. Materials for the manufacture of power plant components, other than the reactor, do not appear to be a bottleneck. Better materials than those now available, especially for turbine blades, would be desirable, but it is probable that such advances will be made.

4. The successful construction of some of the power plant components, such as the heat exchanger, will be difficult and costly, but should not require the development of completely new materials and techniques.

5. The maximum design altitude of a nuclear-propelled aircraft propelled by a helium gas-turbine compressor-jet will probably be somewhat

higher than the maximum design altitude achieved by the use of an open-cycle nuclear-fueled engine.

6. The helium gas-turbine compressor-jet is probably not limited to subsonic flight speeds per se, but the aerodynamic characteristics of present supersonic airplanes must be radically improved before supersonic nuclear flight is possible.

7. The frontal area of the power plant at 30,000 feet is of the order of 5 to 10 square feet per 1000 pounds of thrust. This corresponds to about 2 to 3% of the wing planform area, so that it should be possible to submerge the power plant within the wing.

BISMUTH TURBOJET

Introduction

The bismuth turbojet is one of the more promising of several possible types of closed-cycle nuclear power plants for aircraft propulsion. Fig. III-5 is a schematic diagram of the proposed system, which is described more fully in the Introduction to this chapter.

In order to evaluate the applicability of this system to nuclear-propelled aircraft, it is desirable to estimate the weight of the power plant per unit thrust, the frontal area per unit thrust, the size of the reactor and shield, as well as to evaluate the practical problems.

As in the case of the helium gas-turbine compressor-jet, the bismuth turbojet has a large number of design parameters which play interconnecting roles in determining the performance of the entire aircraft. Unless reasonably good combinations of these parameters are chosen, the resulting designs will not represent the true potentialities of the power plant. Hence an optimization procedure involving systematic variations in the significant parameters is essential to a fair appraisal of the power plant. The bismuth turbojet is thermodynamically a simpler system than the helium gas-turbine compressor-jet, and in the bismuth cycle it was possible to analyze more carefully the interplay of the design variables. However, time did not permit a complete investigation, and, consequently, there may exist a margin of possible improvement over the results reported here. The general procedure was as follows:

First, the entire propulsion system was divided into two parts: the bismuth cycle (including the reactor), and the turbojet.

Second, the general characteristics of the bismuth cycle were investigated separately. As a result of this study, five specific bismuth cycles were selected for possible incorporation into complete propulsion systems. Judgment was exercised in the choice of the cycles, but this does not guarantee that the optimum conditions were selected.

Third, the general characteristics of the air turbojet portion of the

[redacted]

cycle were explored for each of the five specific bismuth cycles. A specific design of the turbojet was then selected for combination with each bismuth cycle.

Finally, using the previously chosen designs for the bismuth cycle and turbojet cycle, the characteristics of a number of integrated bismuth turbojets were studied. As noted previously, many design variables were chosen on the basis of engineering judgment, and hence the integrated designs are not optimized in many of the parameters.

Working Fluid

The excellent heat-transfer properties of the liquid metals make them attractive as pile coolants.

In addition, the high density exhibited by many liquid metals simplifies the nuclear problems. The ducts required to convey a liquid metal through the reactor shield probably will be small, and a liquid metal inside the inlet and exit ducts will act as a reasonably effective shield. Furthermore, a liquid metal in the reactor tubes does not reduce the average pile density as severely as in the case of a gaseous coolant.

Most of the liquid metals which exhibit suitable melting and boiling points are subject to excessive induced radioactivity. The nuclear behavior of possible pile coolants is discussed in detail in Chapter IV. There it is shown that pure liquid bismuth is attractive as a coolant. The induced gamma activity of bismuth is expected to be negligible, but neutron bombardment forms a small amount of polonium, which is an alpha emitter. However, the bismuth itself and the walls of the ducts should provide adequate shielding in this respect.

The chief disadvantage of bismuth is its relatively high melting point, namely, 520°F. On the other hand, it may well be possible to use a mixture of bismuth with lead. This will result in a much lower melting point and will probably not result in excessive induced radioactivity. The boiling point of bismuth, approximately 2640°F, is high enough to avoid troubles from vaporization at the probable operating temperature.

It is realized that other liquid metals may be suitable for the reactor coolant, and hence the results presented for bismuth should be regarded as an illustration of the potentialities of a liquid metal system rather than as a final decision in regard to a liquid metal coolant.

Bismuth Cycle (Including Reactor)

Parameters. The important bismuth cycle parameters are the temperatures of the bismuth entering and leaving the reactor, the maximum reactor wall temperature, the power required to pump the bismuth through the reactor, the reactor geometry, the heat flux distribution within the reactor, and the heat output of the reactor. These quantities determine the size of the reactor needed to satisfy heat-transfer requirements and, in addition, strongly influence the performance of the turbojet.

[redacted]

Criteria of Performance. The following characteristics of the bismuth-cooled reactor were used for judging the suitability of different possible designs for incorporation into a complete bismuth turbojet:

1. The maximum reactor wall temperature is a measure of the difficulties which might be encountered in the use of materials.

2. The temperature of the bismuth leaving the reactor is important to the efficiency of the turbojet, and also influences the size of the bismuth-air heat exchanger.

3. The temperature of the bismuth at the reactor inlet is important in determining the size of the heat exchanger. In addition, it is connected with the rate of bismuth flow, and, consequently, with the size of the reactor.

4. The reactor heat output must be matched to the heat requirements of the turbojet.

5. The reactor diameter influences directly the weight of shield of a given thickness required to surround the reactor.

6. The power required to pump the bismuth through the reactor must be supplied by the air turbine and, therefore, affects the efficiency of the complete propulsion system.

The reactor must fulfill both the nuclear and heat-transfer requirements. In this chapter are presented certain combinations of reactor dimensions which satisfy the demands imposed by heat transfer. The combinations of reactor dimensions which satisfy nuclear conditions are presented in Chapter IV. Finally, in Chapter VII, reactor sizes are selected which satisfy both heat-transfer and nuclear considerations.

Assumptions and Method of Calculations. Assumptions similar to those used with the other power plant studies of this chapter were made for the sake of consistency. The longitudinal heat flux distribution was taken to be sinusoidal, with the maximum power level at the center of the reactor, and with the power levels at the ends equal to 20% of the maximum power. The lateral heat flux distribution was taken to be uniform, although it is recognized that this ideal situation could be only approximated in practice. No account was taken of the need for cooling the shield.

The reactor was assumed to be a single-pass right square cylinder, with the bismuth flowing longitudinally through straight smooth tubes. In computing the reactor dimensions, allowances for headers were provided in such a way that the bismuth velocity in the header does not exceed the velocity in the tubes.

The reactor dimensions presented in this section are all based upon a free-flow ratio of 0.4. However, with all other conditions remaining constant, the dimensions corresponding to other free-flow ratios will be obtained with good accuracy by means of the relation:

$$D\sqrt{FFR} = \text{constant}$$

In computing the small amount of power required to pump the bismuth, the efficiency of the bismuth pump was taken to be 100%. Pressure losses of the bismuth outside the reactor were neglected, but header losses were included.

A general discussion of reactor heat transfer is presented in a later section of this chapter. In addition, an analysis of reactor heat transfer, together with a numerical example for the bismuth reactor, is given in Appendix III-E.

Results. In order to conserve time, five specific sets of temperatures were chosen for calculating dimensions of the reactor. These five conditions are as follows:

Condition	Maximum Wall Temperature	Inlet Bismuth Temperature	Outlet Bismuth Temperature
I	1840°F	540°F	1740°F
II	1840	896	1740
III	1840	1180	1656
IV	1840	1340	1656
V	1370	1015	1270

Conditions I to IV are for a single maximum wall temperature, and the variations of inlet and outlet bismuth temperature were chosen so as to permit at least a crude optimization of these temperatures. It might also be noted that conditions I to IV represent increasing flow rates of bismuth through the reactor. Condition V was chosen for use in later calculations to determine whether the bismuth turbojet might be practical at a very low turbine inlet temperature.

The heat-transfer characteristics of the reactor for these five sets of conditions are presented in Figures III-100 to III-104. On each of these charts the reactor diameter is plotted against reactor heat output for constant values of the reactor tube diameter. It should be noted that along a curve for a given tube diameter the pumping power increases as the reactor heat output increases.

Information concerning the pumping power is presented in Figures III-117 and III-118. It should be noted that these charts are for certain specific designs of the propulsive duct which were chosen as described below under the heading "Air Cycle".

Figure III-117 shows the reactor diameter versus reactor heat output for several constant ratios of pumping power to thrust power, and refers specifically to the conditions tabulated adjacent to the chart. Figure III-117 in conjunction with Figure III-102 gives a complete picture of the possible reactor designs which might be used with a specific design

of the complete bismuth turbojet.

For simplicity in calculations, it was assumed that the pumping power could be ignored if it were less than 3% of the thrust power. In Figure III-118 are shown, therefore, calculated values of reactor diameter as a function of reactor heat output for the specific air cycle designs indicated, and for a ratio of pumping power to thrust power of 0.03. This chart may, of course, be used with Figures III-100 to III-104 to yield other details of the reactor.

Finally, it is worth remarking that the power allotted for overcoming pressure drop in the reactor should, for a careful design, be selected by means of an optimization procedure. However, it is evident that little may be gained by making the pumping power less than 1 to 3% of the thrust power. From the standpoint of the overall aircraft design, it is shown in Chapter VII that little is to be gained by making the pumping power much in excess of 3% of the thrust power. Therefore, while the figure of 3% is somewhat arbitrary at this point, it has been chosen for bismuth turbojet studies of this chapter.

Air Cycle (Turbojet)

Parameters. For a given design flight speed and altitude, bismuth temperatures entering and leaving the reactor, and bismuth pumping power, the performance of the turbojet depends on the efficiencies of the inlet diffuser, air compressor, air turbine, and exhaust nozzle, on the air compressor pressure ratio, and on the average Mach number in the radiator tubes.

The air compressor pressure ratio has the same influence on the bismuth turbojet performance as it has on conventional chemical turbojets. There is generally an optimum pressure ratio whose value is a function of such factors as component efficiencies, temperatures, etc.

Since the heat exchanger cannot have a large frontal area if it is to be installed easily in an aircraft, the air velocity in the tubes should be fairly high. On the other hand, an excessively high air velocity produces large pressure drops and thus reduces the efficiency of the air cycle. An optimization procedure is therefore necessary for selecting the air velocity. In the present study, calculations were made for a range of average Mach numbers in the radiator tubes between 0.1 and 0.3.

Criteria of Performance. The following quantities were used for appraising the suitability of the different possible designs of the turbojet as part of a complete bismuth turbojet power plant.

(a) The frontal area per unit thrust is a measure of the difficulty associated with the installation of the air machinery in an aircraft. Under some conditions the maximum frontal area was determined by the air compressor, and under other conditions the maximum frontal area was that required by the radiator. The figures reported for frontal areas are based on the

condition where the frontal areas are equal, which investigation showed to correspond always to the minimum value for the maximum frontal area of the turbojet.

(b) The weights of the heat exchanger, bismuth, and air machinery, all per unit thrust, are important in determining how much of the gross aircraft weight is available for carrying a shield.

(c) The reactor power per unit thrust is perhaps the best measure of the efficiency of the propulsion system.

(d) The specific impulse (pounds of thrust per pound per second of air flow) is a measure of the amount of air required to produce a given thrust and therefore is a rough gauge of the size and weight of the air-swallowing components.

Assumptions and Method of Calculation. For reasons given previously, all calculations were made for a flight Mach number of 0.9 and an altitude of 30,000 feet.

The efficiency of the inlet diffuser was assumed to be 90%, in conformity with similar assumptions for the other power plants considered in this chapter.

Since the air compressor requires a pressure ratio of the same order as in existing turbojets, i.e., of the order of 6, it was assumed to have an efficiency of 83.5%, which is about the best efficiency exhibited by compressors in turbojets now under development. In estimating the frontal area of the air compressor, the axial Mach number at the entrance to the first-stage annulus was taken to be 0.5, and the total frontal area was assumed to be 1.8 times the annulus area.

An air turbine efficiency of 87% was assumed. This figure is believed to be consistent with the assumed value for the compressor efficiency.

The exhaust nozzle was assumed to have an efficiency of 90%.

In those cases where an intercooler was provided between the forward and rear components of the compressor, it was assumed that the intercooler effectiveness was 0.50 and that the pressure loss of the propulsive air flowing through the intercooler was 5% of the pressure entering the intercooler. In addition, the forward and rear compressors were assumed to have equal pressure ratios.

For purposes of calculation, the bismuth-air heat exchanger was assumed to be of the shell-and-tube type, with air flow inside the tubes and bismuth flow outside the tubes. The heat exchanger dimensions were calculated on the assumption that the resistance to heat transfer on the bismuth side is negligible. It was further assumed that the tubes are of 1/8 inch diameter with a wall thickness of 0.004 inches and with a density of 490 pounds per cubic foot.

The total radiator weight was taken to be twice the weight of the tubes, and the total frontal area was taken to be twice the total inside cross-sectional area of the tubes.

Although the radiator dimensions and weight were calculated for the geometry described above, the results are also approximately applicable to other types of good aerodynamic design, such as a plate-type heat exchanger.

For estimating the weight of the bismuth, it was assumed that the total volume of the liquid metal was one-fourth of the total volume within the tubes of the bismuth-air heat exchanger, and this figure was increased by 20% to allow for the amount of bismuth in the reactor and ducts.

In estimating the weight of the air compressor, air turbines, shafting, gearing and auxiliaries, the combined weight was assumed to be two pounds per cubic foot per second of air flow at the compressor inlet, for a compressor pressure ratio of 4. Weight estimates for other pressure ratios were made by assuming that the average pressure ratio per stage was 1.15 and that the total weight varies as the 0.6 power of the number of stages. This method of calculating weights is based on data for present-day turbojets, and is consistent with corresponding assumptions used with the other power plants considered in this chapter.

The methods of calculation, together with a numerical example, are given in Appendix III-C.

Results. Some of the characteristics of the air cycle are shown in Figures III-105 through III-110.

For a given air compressor pressure ratio an increase in the air speed through the tubes tends to reduce the frontal area of the radiator. The correspondingly increased pressure drop, however, also tends to reduce the thrust of the propulsive duct, so that, in the region of interest, the heat exchanger frontal area per unit thrust drops sharply with increasing reactor power per unit thrust. On the other hand, the compressor frontal area increases steadily as the turbojet efficiency decreases. The point where the heat exchanger and compressor frontal areas are equal corresponds to the condition where the maximum frontal area of the turbojet is at its lowest value, as shown for example in Figure III-109. Considering frontal area and reactor size, it is desirable to select conditions to the left of the minimum points on the curves of Figure III-107. Figure III-107 shows also that the compressor pressure ratio plays a significant part in determining the maximum frontal area. However, it will be noted that the maximum frontal area is of the order of two square feet per thousand pounds of thrust; this corresponds roughly to only about one per cent of the wing planform area, which in turn means that there should be no difficulty in installation of the air machinery and radiator. Furthermore, the frontal area need not play a major role in selection of the design, provided that this area is not increased too much beyond the figures mentioned above.

Arguments like those given above for the frontal area apply equally to the radiator weight, as shown in Figure III-108. It may be noted that

the weight of the heat exchanger is of the order of 0.5 pounds per pound of thrust; this weight is therefore not critical, and should not seriously influence the selection of the design.

The results mentioned above indicate that the frontal area of the radiator is probably more critical than its weight. Consequently, the mean Mach number through the radiator tubes was, for each compressor pressure ratio, selected at the minimum point on curves like those of Figure III-107.

Using heat exchanger designs based on this specification, Figure III-105 shows the air specific impulse for the propulsive duct as a function of compressor pressure ratio. Between a pressure ratio of 4 and 20 the specific impulse is roughly constant. Above pressure ratios of about 11 the specific impulse drops off rapidly unless an intercooler is used. Taking into account the difficulties of building machines with high pressure ratios, Figure III-105 indicates that a pressure ratio in the neighborhood of perhaps 4 to 8 is desirable. With a turbine inlet temperature of 1500°F the specific impulse is then about 40 pounds of thrust per pound per second of air flow. This figure is roughly comparable to the specific impulse of existing chemical turbojets.

The weights of the turbojet units corresponding to the designs of Figure III-105 are presented in Figure III-106. For compressor pressure ratios in the neighborhood of 6, Figure III-106 shows that the weight of air machinery would be of the order of two pounds per pound of thrust. While not critical, this figure is large enough to dictate careful design of the machinery, particularly at high altitudes.

For reasons given previously, a compressor pressure ratio of 6 was chosen for further calculations. Figure III-109 shows the maximum frontal area as a function of reactor power for this compressor pressure ratio, for several temperatures at the turbine inlet, and for bismuth temperatures of 1656°F and 1180°F, respectively, at the inlet and outlet of the heat exchanger. It is also clear from this chart that the frontal area increases rapidly as the turbine inlet temperature is reduced.

Using heat exchanger designs based on the minimum points of Figure III-109, there is plotted in Figure III-110 the air specific impulse against turbine inlet temperature for the fixed compressor pressure ratio and bismuth temperatures mentioned above. It is evident that the machine becomes unattractive for turbine inlet temperatures below about 1000°F to 1200°F.

Complete Power Plant

With the performance of the bismuth-cooled reactor and of the turbojet approximately established, the final step of the procedure is to match the two systems and determine overall weights, sizes and frontal areas as functions of the thrust. An adequate appraisal of the potentialities of the bismuth turbojet would require a far more involved optimization procedure.

than was possible at this time. However, the results given here are probably satisfactory as regards order of magnitude.

Parameters. The parameters for the complete power plant are the same as for the individual components, except that they are closely interrelated as regards the complete power plant.

Assumptions and Method of Calculation. The principal assumptions have been given previously.

In analyzing the complete power plant, a "standard" case was selected as a starting point, and a few variations from the standard were systematically explored. The specifications of the "standard" case are as follows:

Compressor Pressure Ratio:	6.0
Turbine Inlet Temperature:	1500°F
Maximum Reactor Wall Temperature:	1840°F
Temperature of Bismuth Leaving Reactor:	1656°F
Temperature of Bismuth Entering Reactor:	1180°F

An analysis of the bismuth turbojet, together with a complete numerical example, is given in Appendix III-C.

Results. Figures III-111 to III-116, inclusive, show some illustrative results for the "standard" bismuth turbojet, and, in addition, show the effects of variations in turbine inlet temperature and in altitude.

Figure III-111 shows variation of maximum frontal area with turbine inlet temperature and with thrust. With a turbine inlet temperature of 1500°F, the frontal area is of the order of 2 square feet per thousand pounds of thrust; this corresponds to a frontal area of only about 1% of the wing planform area, which is about the same percentage as for existing chemical turbojets. If the turbine inlet temperature is reduced below about 1100°F, the maximum frontal area increases very rapidly and soon becomes large enough to make the installation of the machine difficult.

From Figure III-112 it may be seen that the total power plant weight (excluding shield) for a turbine inlet temperature of 1500°F is of the order of 2 pounds per pound of thrust. This figure is not at all critical at an altitude of 30,000 feet, but, as will be seen later, at higher altitudes the increase in this weight ultimately becomes a limiting factor. As the turbine inlet temperature is reduced below about 1100°F, the power plant weight grows rapidly and reduces the available shield weight even more rapidly.

Figure III-113 shows the reactor diameter (and length) as a function

of thrust for turbine inlet temperatures of 900 and 1500°F. For each of these temperatures the maximum reactor wall temperature was made 340°F greater than the turbine inlet temperature. It is important to note that the reactor diameters given in this chart are merely illustrative and apply specifically for a reactor free-flow ratio of 0.4 and for a reactor tube diameter of 3/8 inch. As shown in Chapter VII, considerations of fuel investment generally suggest the use of different free-flow ratios; if this is so, the reactor diameter (and length) may be estimated from the approximate rule that the diameter varies inversely with the square root of the free-flow ratio, all other conditions remaining unchanged. The choice of a reactor tube diameter of 3/8 inch in Figure III-113 is also illustrative. For the range of thrusts shown, this tube diameter yields a bismuth pumping power which is negligible compared to the thrust power. However, it is certainly worth while to make the bismuth pumping power at least one or two percent of the thrust power; and an optimization procedure might suggest the use of even greater pumping powers. Thus, the reactor tube diameter employed in an actual design might be less than 3/8 inch, and this in turn would yield reactor diameters (and lengths) less than those shown in Figure III-113. Conclusions of this sort were used in selecting the integrated designs described in Chapter VII.

Figure III-114 shows how the maximum frontal area varies with altitude. As an approximate rule, the frontal area varies inversely with the ambient air density. However, if the wing area is determined by cruising rather than take-off and landing conditions, the wing area also will vary inversely with density. Hence, for altitudes greater than about 30,000 feet, the ratio of frontal area to wing area will remain approximately constant, and the problems of installation will remain unchanged.

According to Figure III-115, the power plant weight approximately doubles as the altitude is increased from 30,000 to 50,000 feet. This is a serious matter, since, at 50,000 feet, for the conditions assumed, the power plant weight (excluding shield) is nearly 5 pounds per pound of thrust. This indicates that at an altitude of perhaps 60,000 feet it would not be possible to carry any shield. It might be noted parenthetically that weight estimates of machinery are not accurate until detailed designs have been completed. The error in power plant weight is not important for low altitudes because the power plant weight is then small. At high altitudes, however, any error in estimated power plant weight greatly magnifies the error in the estimate of allowable shield weight, so that the calculations of overall aircraft performance given in Chapter VII are very approximate at altitudes near 50,000 ft.

One of the very desirable features of the closed-cycle power plant is illustrated in Figure III-116, which shows that the size of the reactor is almost independent of flight altitude.

Significance of Results

As with the helium gas-turbine compressor-jet, the limited time available for studying the bismuth turbojet required that a large number of assumptions be made somewhat arbitrarily, and that certain working para-

meters be frozen at values which might not be optimum. It is, therefore, worth examining the limited results given here in the light of what might be disclosed by a more extensive study.

In some respects the analyses might be considered optimistic. On other counts they might be thought pessimistic, and many arguments, pro and con, might be offered at various points.

However, rough calculations indicate that where assumptions might be considered optimistic, a modification of those assumptions would not seriously affect the conclusions drawn in this chapter. In addition, it is important to keep in mind the possibility of improving the performance figures shown in this section through a more extensive optimization procedure.

In any case, it is believed that the results of this study are correct as regards order of magnitude, and since these results make the bismuth turbojet appear very attractive, it is recommended that this type of power plant be studied more extensively.

Problems of Development and Construction

The air compressor and air turbine either are within the range of present practice or require only a reasonable extension of present practice. No unusual difficulties are envisaged here.

The system through which the bismuth circulates is novel and will require considerable research, development and ingenuity. The problems here are chiefly those of obtaining completely satisfactory materials for containing the bismuth within the reactor, of starting up with a fluid which is solid at normal temperatures, and of developing such mechanical components as the heat exchanger and pump.

Conclusions

1. Although extensive research and development would be required to utilize the bismuth turbojet for nuclear-powered aircraft, the promise of ultimate success seems reasonably good.

2. The primary problem is the development of materials for use in the construction of the reactor. These materials must not only meet the nuclear requirements, but also must successfully withstand temperatures of at least 1300 or 1400°F, and in all probability higher temperatures. In addition, the materials must withstand the thermal stresses introduced by extraordinary heat flux densities and must have satisfactory properties after being subjected to radiation damage. Furthermore, coatings or cannings must be developed to prevent contamination of the bismuth by fission fragments.

3. Materials for the manufacture of power plant components, other than the reactor, do not appear to be a bottleneck.

4. The successful development and construction of the heat exchanger will be difficult and costly but should not require the use of completely new materials or techniques.

5. The maximum design altitude of a nuclear-powered aircraft propelled by a bismuth turbojet will, in all probability, be somewhat higher than the corresponding figure for the open-cycle turbojet. It seems likely that the maximum design altitude of the bismuth turbojet when carrying an adequate shield will be about 50,000 feet and almost certainly not greater than 60,000 feet.

6. The bismuth turbojet is probably not limited to subsonic flight speeds per se, but the aerodynamic characteristics of supersonic airplanes must be radically improved before supersonic nuclear-powered flight is possible.

7. The frontal area of the power plant for altitudes of 30,000 feet and greater is of the order of 1% of the wing planform area. Consequently, there should be no great problems involved in submerging the power plant within the wing.

HELIUM TURBOJET

Introduction

Figure III-6 is a sketch of a closed-cycle propulsion system which has been designated the helium turbojet. In principle it is similar to the bismuth turbojet, except that the pile coolant is gaseous helium instead of liquid bismuth.

In the arrangement considered here the power required by the helium compressor is supplied by a helium turbine. Thus, pumping of the helium is accomplished by means of a small helium gas-turbine set which has zero net-work output. Alternatively, the pumping power for the helium might be supplied directly by the air turbine. It seems likely that the overall performance of the system for the two pumping arrangements would be virtually identical so long as the helium turbine and air turbine had approximately the same efficiency.

A number of calculations for this power plant indicated that it embodies about the same characteristics as the helium gas-turbine compressor-jet with respect to reactor size and machinery weight, and has a frontal area slightly larger than the bismuth turbojet. In view of the fact that the helium turbojet is so similar to the other power plants mentioned above, the analysis of the helium turbojet was not carried beyond the preliminary calculations.

As in the case of the other power plants, the helium turbojet has a large number of design parameters which play interconnecting roles in determining the performance of the entire aircraft. Because of time limitations, however, an extensive optimization procedure was dispensed with in favor of the following procedure:

First, the entire propulsion system was divided into three parts: the helium gas-turbine cycle, the air turbojet, and the reactor.

Second, the general characteristics of the helium gas-turbine set were investigated. As a result of this study, a specific design of the helium gas-turbine set was chosen for incorporation into the complete propulsion system.

Third, the general characteristics of the air turbojet cycle were explored with reference to the specific helium design previously selected. From this study a specific design of the air turbojet was selected.

Finally, the previously chosen designs for the helium and air systems were combined with reactor designs to show the performance of the complete power plant.

Working Fluid

Since a gaseous coolant is used, the considerations underlying the choice of the gas will be the same as those described in the section entitled "Helium Gas-Turbine Compressor-Jet". The reasons given in that section indicate why helium was chosen for the present study.

Assumptions and Method of Analysis

The analysis of the helium gas-turbine cycle previously employed for the helium gas-turbine compressor-jet is equally applicable to the helium turbojet when the net power output of the cycle is made zero. Furthermore, the reactor calculations for the helium turbojet are similar to those for the helium gas-turbine compressor-jet. Finally, the air turbojet calculations previously made for the bismuth turbojet are directly applicable to the helium turbojet. As a consequence, it was possible to narrow down the range of variables studied in connection with the helium turbojet. One complete design point was calculated.

From the analysis of the helium gas-turbine cycle the pressure ratios of the helium compressor and turbine were selected as 1.3 and 1.17 respectively.

The analysis of the turbojet, with reference also to the characteristics of the helium cycle, led to the choice of helium temperatures at the heat exchanger inlet and heat exchanger outlet of 1680 and 470°F, respectively. Then, when the effect of the ratio of helium flow to air flow had been determined, an air compressor pressure ratio of 6.0 was selected, using arguments similar to those employed in connection with the helium gas-turbine compressor-jet. The choice of the average Mach number in the radiator tubes, and consequently of the frontal area of the radiator, was made after a series of calculations similar to those described in conjunction with the helium gas-turbine compressor-jet.

The assumptions made in connection with the heat exchanger are identical with those given for the helium gas-turbine compressor-jet.

Additional assumptions are as follows:

Altitude	30,000 feet 40,000 feet 50,000 feet
Flight Mach No.	0.9
Helium compressor efficiency	85%
Helium turbine efficiency	88%
Efficiency of inlet diffuser	90%
Air compressor efficiency	85%
Air turbine efficiency	88%
Efficiency of exhaust nozzle	90%
Helium pressure at reactor inlet	1000 psi
Helium temperature at reactor inlet	590°F
Helium temperature at reactor outlet and turbine inlet	1800°F
Air turbine inlet temperature	1448°F

The methods of calculation are identical with those previously discussed for the helium gas-turbine compressor-jet and the bismuth turbojet.

Results

Only the results for the complete power plant are presented here.

Figure III-119 shows the frontal area of the power plant as a function of thrust for altitudes between 30,000 feet and 50,000 feet. It will be seen that the frontal area at 30,000 feet is about 3 square feet per thousand pounds of thrust, which is about half the frontal area per unit thrust for the helium gas-turbine compressor-jet under similar assumptions of efficiency, temperature, etc., and slightly higher than the corresponding value for the bismuth turbojet. At higher altitudes, the increase in frontal area is approximately proportional to the increase in aircraft size, and hence is not significant.

Figure III-120 shows the power plant weight (excluding shield) as a function of thrust for altitudes between 30,000 feet and 50,000 feet. At 30,000 feet the power plant weight is of the order of 2.5 pounds per pound of thrust, which is about 15 or 20% greater than the corresponding figure for the helium gas-turbine compressor-jet. At higher altitudes the power plant weight increases rapidly and ultimately limits the maximum design altitude.

For the special case where the free-flow ratio is 0.4 and the pressure at the reactor inlet is 1000 psia, Figure III-121 shows the reactor diameter (and length) as a function of thrust. It will be noted first of all that the helium turbojet, like the other closed-cycle power plants, is not affected significantly by altitude as regards reactor size. If nuclear considerations require a free-flow ratio different from 0.4, the reactor diameter (and length) may be estimated from the relation that the diameter is inversely proportional to the square root of the free-flow ratio, all other things remaining unchanged. If mechanical considerations suggest the use of a pressure level different from 1000 psia, the reactor diameter may be estimated from the relation that the diameter is inversely proportional to the square root of the pressure level, all other things remaining unchanged.

Conclusions

As might be expected, the general characteristics of the helium turbojet are quite similar to those of the bismuth turbojet, since the power plants differ only with respect to the reactor coolant and its pumping system. The liquid metal coolant requires the use of reactor sizes smaller than those for gaseous coolants, but the nuclear and chemical inertness of helium is an advantage not possessed by the liquid metals thus far considered.

From the short study presented here of the helium turbojet, this system would appear to offer about the same degree of promise as the helium gas-turbine compressor-jet and the bismuth turbojet.

TERNARY BISMUTH VAPOR-CYCLE COMPRESSOR-JET

Characteristics of Vapor Cycles

If it is found impossible to obtain reactor materials suitable for operation at the temperatures envisaged for the cycles previously considered, it may be still possible to use a cycle for which lower temperatures would be satisfactory. For this reason, some attention has also been given to the possibilities of vapor cycles, which would hardly be considered otherwise because of the inherently large size of the power plant.

A vapor cycle, an example of which is shown in Figure III-7, differs from the conventional gas-turbine cycle chiefly in the amount of negative work required. In the gas-turbine cycle the working medium is in the gaseous state at all times; in the vapor cycle it is in the liquid form during the compression process. The work of compression, which is approximately two-thirds of the work available from expansion in the gas turbine cycle, is reduced to a very small fraction in the vapor cycle because of the small amount of energy required for a large increase in liquid pressure.

The most common example of a vapor cycle is the steam turbine power plant. A small amount of negative power is required to increase the pressure of the feed water from atmospheric or condenser pressure to boiler pressure, but this is negligible compared with the turbine output. For

all practical purposes the net output of the plant is given by the isentropic power of expansion multiplied by the turbine efficiency, and large outputs are possible even with relatively low initial temperatures. In a corresponding gas-turbine power plant, the net output is the difference between the power delivered by the turbine and the power required by the compressor, and even with reasonably good efficiencies of compression and expansion, the compression power required is such a large fraction of the expansion power that a fairly high temperature is needed merely to make the plant self-sustaining; a much higher temperature is necessary for a reasonable net output. Thus the initial temperature of the gas turbine plant is limited to a certain minimum value by considerations which do not enter the picture in the case of a vapor cycle.

In addition to its size and weight, there are other serious objections to an aircraft power plant using a vapor cycle, all of which must be carefully balanced against the advantages it offers with respect to operating temperatures. This power plant has all the disadvantages of complexity and vulnerability inherent in a closed cycle, and may require the carrying of large masses of binary and ternary fluids.

Working Fluid

Water (or steam) and mercury are the only two fluids with properties which appear at all attractive. Both have been used for stationary industrial power plants. Heretofore a mercury turbine has always been operated in conjunction with a steam turbine, the heat released in condensation of the mercury being used to generate steam, but there is no basic reason why the mercury turbine alone cannot be used. The chief reason for using mercury in industrial power plants is to obtain high initial temperatures (i.e., high, relative to corresponding steam temperatures) without excessively high pressures, but the cycle can also be used with initial temperatures which are low relative to those required for a gas turbine power plant.

Operation Without Intermediate Heat-Transfer Fluid

Vapor cycles conceivably might operate in two ways. The simplest method, and the one which requires least weight of power plant, is to vaporize the fluid in the reactor itself. The vapor then expands through a turbine which furnishes mechanical power, is condensed in a heat exchanger where the heat of vaporization is removed by atmospheric air flowing over the tubes, and is finally pumped back to the reactor to repeat the cycle.

Although this method seems very attractive, it has such disadvantages from a nuclear standpoint that it probably cannot be seriously considered. In the first place, any bubbles in the reactor passages will make proper control of the reactor exceedingly difficult, if not impossible, because of the large density changes. Bubbles might be avoided if the fluid is in a supercritical state, but there are at least two serious objections to the use of supercritical water, even assuming that it is practicable to insure a supercritical condition at all times in the reactor.

First, the pressure required would be very high -- several thousand pounds per square inch. Second, with a cosine power distribution along the reactor, even a small change of power level would result in a comparatively large change of density near the discharge end of the reactor tubes unless the operating temperature is sufficiently far from the critical value. Probably there would be no danger that the multiplication factor would get out of bounds, but maintenance of the desired constant temperature would be very difficult. With pressures which might be used, the operating temperature would have to be limited to a maximum of about 700°F or thereabouts, which would result in a relatively low cycle efficiency, low specific thrust, and high specific power plant weight.

Mercury absorbs neutrons so readily that it would not be a suitable reactor coolant even in the supercritical state. Water is not so objectionable in this respect, but its oxygen would become sufficiently radioactive to prohibit its use on a manned airplane.

Operation With Intermediate Heat-Transfer Fluid

The second method is to use an intermediate heat-transfer fluid without these nuclear disadvantages,-- such as helium or liquid bismuth,-- which would absorb heat in the reactor and transfer it to the water or mercury in a separate heat exchanger. This has the objectionable feature of requiring a greater difference between the turbine and reactor temperatures, as well as greater weight and complexity, but it seems to be the only practicable way in which a vapor cycle can be utilized. This method has been assumed in the studies here made.

Figure III-7 shows diagrammatically the operation of the cycle. Liquid bismuth has been taken as the heat-transfer fluid because of its relatively low melting point (520°F), its high coefficient of heat transfer, and its excellent nuclear properties. Bismuth itself is a good shielding material, moreover, since it is a liquid metal of high density, only relatively small passages will be required in the conventional shield to conduct it to and from the reactor, so that the additional thickness of shield required to compensate for such passages could be quite small. A pump is required to circulate the bismuth, but very little power is needed for this purpose.

Use of Propeller or Ducted Fan

The shaft power output of the turbine might be utilized to drive a propeller or ducted fan; this possibility has been investigated to some extent by NACA (6)(7) and by NEPA (1). In this case the heat transferred to the coolant air in the condenser is utilized to provide whatever additional thrust can be obtained by the rearward discharge of the air. However, the maximum air pressure rise is only that corresponding to the available ram, and the air temperature rise may not be sufficient for a net propulsive thrust; if this is the case, the resultant drag must be subtracted from the propeller thrust. Even when there is a positive thrust, it will be very small compared to the propeller thrust.

The NEPA (1) study concluded that a vapor cycle might be feasible, but this analysis was not carried very far. It was reported that this

cycle "seems to present immediate advantages for the application to inhabited aircraft, due mainly to the fact that the high heat-transfer coefficients which may be attained permit smaller reactors and consequently lower shielding weights. However, the development of an aircraft type steam or mercury turbine may be estimated to require, independently of the reactor development itself, a much longer period of time and much greater expenditure than the comparable open cycle development". It is believed that the view expressed here, as to the difficulties of turbine development, is perhaps too pessimistic.

The NACA (6) (7) devoted most of its attention to the condenser, since this was considered to be one of the most critical items, and no opinion was expressed as to the feasibility of the project as a whole, probably because of lack of knowledge of the reactor and shield weights required. The calculations were not carried to the point where all variables were optimized, but the following tabulation shows condenser values which may be considered as approaching the best that can be expected.

NACA Report:	RM No. E7J01	RM No. E8C23
Working fluid	Steam	Mercury
Altitude, feet	30,000	30,000
Airplane speed, miles per hour	500	500
Condenser frontal area, sq ft/1000 net thrust h.p.		
Condenser in nacelle	18	6.3
" " wing	-	6.0
Condenser weight, lb/1000 net thrust h.p.	280	260
Fraction of gross airplane weight available for reactor, working fluid, shield, and payload		
Condenser in nacelle	0.41	0.48
" " wing	0.48	0.49

Use of Compressor-Jet

Another possibility for this cycle is to utilize the turbine shaft output to drive the compressor in a compressor-jet arrangement, as shown in Figure III-7. Atmospheric air is taken on board at ram pressure, compressed slightly in the compressor (normally the pressure ratio across the compressor would be about 2 or less), and then discharged into the condenser, where it cools and condenses the circulating vapor. In the condenser the air is further heated, and of course loses some pressure, but is finally discharged from the propulsion nozzle at a velocity sufficiently high to furnish net propulsive thrust, this constituting the entire thrust output of the power plant. The cases which have been considered in this study assume the use of such a compressor-jet.

Outline of Calculations

Time was not available to study all possible variations of the vapor cycles, but enough cases were calculated for both water and mercury to give a reasonably comprehensive picture of the possibilities. The effect of altitude and of airplane velocity should be about the same as for the other closed cycles which have been considered. As in any conventional steam turbine operating on the Rankine cycle, the net output is approximately proportional to the turbine efficiency; there is no magnification of the effect of the component efficiencies as in the gas-turbine (Brayton) cycle.

The size and weight of the condenser evidently are limitations which may prove to be decisive. In general, the size can be reduced by increasing the difference between the temperature of the condensing fluid and the maximum temperature of the air in the compressor-jet duct, or by allowing an increased pressure drop of the air in the duct. In either case, the net thrust per unit mass of air flow will be decreased, and certain optimum values of these variables can be found which correspond to maximum thrust per unit area, or per unit weight, of the condenser tubes. By varying the turbine exhaust pressure for a given initial pressure and temperature, the ratio of the total available change of enthalpy which is utilized as shaft work to drive the compressor, and as heat to increase the temperature of the air after compression, can be changed, and an optimum ratio can be similarly determined. Other design values can likewise be varied to give a large number of different possible combinations. It must be borne in mind, however, that the basic advantage of the cycle is lost if too high a reactor wall temperature is used.

The calculations also require an estimate of the size and weight of the intermediate heat exchanger, or boiler. The greater the temperature difference allowed between the bismuth and the working fluid, the smaller will be the exchanger, but too great a difference will necessitate such a high reactor temperature as partially to defeat the purpose of using this cycle. In these studies a maximum temperature difference of about 300°F was assumed to be satisfactory.

Range of Variables

An altitude of 30,000 feet and a Mach number of 0.9 were assumed in all calculations. The following table shows the ranges of the other variables in the cases calculated for subcritical water and mercury.

Subcritical Water

Case:	1	2	3	4	5	6	7
Turb. inlet press., psia	1000	1000	1700	1700	1700	1700	1700
Turb. exh. press., psia	240	240	240	140	140	140	140
Turb. inlet temp., °F	1000	1000	860	1200	860	860	860
Condenser log mean temp. difference, °F	190	147	159	82	100	100	100
Mean Mach No. of condenser air	0.2	0.2	0.2	0.2	0.2	0.25	0.15

[redacted]

Mercury

Case:	1	2	3	4	5	6	7	8	9
Turb. inlet press., psia	85	85	85	85	85	85	85	85	85
Turb. exh. press., psia	3	3	3	3	3	15	0.4	3	3
Turb. inlet temp., °F	984	984	984	984	984	984	984	984	984
Condenser log mean temp. difference, °F	228	228	228	228	228	276	185	264	187
Mean Mach No. of condenser air	0.15	0.2	0.3	0.35	0.4	0.3	0.3	0.3	0.3

The ratio of inlet to exhaust pressure is a more significant variable than either of these pressures taken alone, and a reasonable picture of the possibilities can be obtained by varying only the exhaust pressure. For this reason the turbine inlet conditions were maintained constant in all the cases calculated for mercury. A lower inlet temperature might have been used, with a correspondingly lower reactor temperature, without affecting very much the condenser frontal area and weight per unit thrust.

Assumed Efficiencies

The following efficiencies were assumed in all the vapor cycle calculations:

Intake ram	0.90
Turbine	0.82
Compressor or compressor-jet	0.85
Duct and propulsion nozzle	0.92

The ram efficiency is the same as has been assumed in the calculation of all the other cycles of this chapter. While this efficiency cannot be accurately predicted until the details of installation have been determined, it is reasonable to expect that 0.90 can be obtained in a large airplane with careful design.

The turbine efficiency has been taken somewhat lower than the value used in the open-cycle calculations. This is because the kinetic energy of the exhaust cannot be utilized, and must be considered as part of the turbine loss. It is believed that 0.82 can be obtained with a turbine of the general capacity here in question, even though its size and weight would probably have to be considerably less than that of a stationary industrial turbine of the same capacity.

Because of its relatively low pressure ratio, the efficiency of the compressor in the compressor-jet arrangement has been assumed to be somewhat higher than the efficiency of the compressor used in the open-cycle turbojet. Furthermore, since the discharge from the condenser is likely to be more uniform than the discharge from a turbine, and there is no necessity for changing from an annular to a circular path, the duct and propulsion nozzle efficiency has also been taken somewhat higher.

[redacted]

Methods of Calculation

Conventional methods were used to calculate the shaft power output of the steam and mercury turbines, as well as the amount of heat released in condensation. The power required to pump the water, mercury and bismuth was neglected. The condenser and compressor-jet were calculated by the same method used to calculate the similar heat exchangers and compressor-jets of the other closed cycles, as given in Appendices III-B and III-C. The condenser is assumed to consist of a large number of 1/8 inch tubes of 0.004 inch wall thickness, through which the coolant atmospheric air flows, and around which the vapor condenses.

Estimated Weights

The total weight of the power plant was estimated to be about 5200 pounds per 1000 pounds thrust for steam, and about 6100 pounds per 1000 pounds of thrust for mercury. This includes the weights of the turbine, condenser, intermediate heat exchanger, compressor, pumps, gears, accessories, ducts, and working fluid, but not the weight of the reactor or shield. It is emphasized that these are only rough figures, which cannot be considered as having any substantial foundation in either a detailed study of the equipment and installation or in previous experience. It is particularly difficult to estimate the amount of bismuth and water or mercury which must be carried until the entire system is laid out in some detail.

Calculated Values

The following table summarizes the results for three of the cases calculated. None of these represents an optimum condition, but the first and third indicate what may be expected under favorable cycle operating conditions with subcritical water and mercury, respectively. The second case is for supercritical water, and since only a single set of calculations was made for this condition, the results cannot be considered as necessarily even approximating the best to be expected.

Some Calculated Results For Vapor Cycles

Case:	1	2	3
Working Fluid:	Water (subcritical)	Water (supercritical)	Mercury
Specified Conditions			
Turbine inlet press., psia	1700	3600	85
Turbine inlet temp., °F	860	1000	984
Turbine exh. press., psia	240	400	3
Log mean temp. diff. in condenser, °F	133	159	187
Mean Mach number of condenser air	0.2	0.2	0.3

Calculated Results

Spec. impulse, lb. sec/lb.	10.8	14.2	14.85
Frontal area of condenser sq ft/1000 lb. thrust	19.8	14.0	9.3
Weight of condenser tubes, lb./1000 lb. thrust	735	467	349
Length of condenser tubes, inches	14.2	12.7	14.3
Number of condenser tubes number/1000 lb. thrust	117,000	82,500	54,700
Reactor heat output, kw/1000 lb. thrust	8000	6660	7950
Reactor wall temp., °F	1120	-	1420

Condenser Frontal Area

If a vapor cycle is used, the total condenser frontal area is likely to be one of the major limiting factors of the installation. Assuming a wing loading of 60 lb/sq ft, and a lift-drag ratio of 15, the ratio of frontal area to wing area is found to be approximately 6 or 8 per cent of the wing planform area. With customary wing thickness ratios, this would amount to about 60 or 80 per cent of the area of the maximum section of the wing perpendicular to the direction of flight.

If supercritical water is used, the required condenser frontal area can be reduced. However, even with an intermediate heat exchanger to eliminate control difficulties, the high pressure required is still a serious objection to the use of water.

Use of mercury would still further reduce the frontal area, and in this case the pressure differences are very moderate, since the difficulties of maintaining a tight system should be diminished. Although the turbine exhaust pressure is 3 psia for the case given in the table, the results did not differ very much for an exhaust pressure of 15 psia, so that it would be possible to avoid too low absolute pressures if it is found that these are not desirable. The frontal area required in this case compares unfavorably, however, with the area required in other closed-cycle power plants.

Of course it might be possible to install condensers wholly inside the wings, even when the required frontal area is as great as that for the case of subcritical water given in the table, especially if different sections of the condenser are located at different distances from the leading edge of the wing. In any case, however, a totally submerged installation of a large heat exchanger is likely to offer great difficulties, and a detailed installation study will be necessary before the practicability of any proposed design can be ascertained. These exchangers also have the disadvantage of greater vulnerability and probably of less reliability than is usually permitted in an airplane installation, but these considerations are not of themselves decisive for the very specialized airplane in question here.

Power Plant Machinery

From a thermodynamic standpoint the mercury or steam turbine will be of a more or less conventional type, regardless of whether water or mercury is used. However, just as in all the other closed cycles, the difficulties of maintaining a tight system will be formidable, particularly if high pressures are used. The bismuth-water boiler and the bismuth-mercury boiler will require development, but do not appear to involve such novel features as to raise serious doubts as to their practicability. Considerable experience has already been accumulated with industrial mercury-water boilers, and marked improvements have been made in recent years. No doubt it would be possible to take full advantage of this experience in designing a mercury boiler for an aircraft application. The construction of a light reliable condenser of the required capacity will also be a major manufacturing problem.

It will be noted that the specific impulse values given in the table are very low compared with those of conventional turbojets and with those calculated for other cycles in this study. This is inherent in cycles using such low turbine and jet temperatures, and the advantages of low reactor temperatures cannot be obtained without a sacrifice of specific impulse. Consequently, the weight of power plant machinery will be relatively greater for a vapor cycle than for the other cycles considered in this chapter.

Conclusions

Relatively low reactor temperatures are possible if a vapor cycle is used, but for nuclear reasons an intermediate heat-transfer fluid is necessary. The resultant complications, as well as the large frontal areas and power plant weights required, make it inadvisable to consider this cycle unless it is found impossible to develop materials suitable for cycles operating at higher temperatures.

NUCLEAR RAMJETS

At first thought, the combination of a ramjet with a nuclear reactor may appear to be very promising for the propulsion of aircraft. Basically the ramjet consists merely of a duct through which air flows when the ramjet moves through the air, as illustrated by Figure III-3. The air entering the duct is compressed by deceleration relative to the duct, is heated, and then is expanded through a nozzle back to the ambient atmospheric pressure. Since the heated air leaves the duct at a velocity higher than its entering velocity, a thrust force results. In chemically powered ramjets, the compressed air is heated by the combustion of fuel, but a nuclear-powered ramjet can readily be imagined where the air is heated by a nuclear reactor. No moving parts would be required except those needed to control the reactor, and thus there would appear to be very little to wear out before the eventual exhausting of the fuel supply in the reactor itself.

There are several serious disadvantages to the ramjet-nuclear reactor combination, however. In the first place, the ramjet develops no static thrust, and in fact must be moving at high subsonic and in many cases at supersonic speeds before it develops sufficient thrust to sustain itself in flight. Some auxiliary propulsion device would, therefore, be required to launch a nuclear ramjet and to accelerate it to high speeds. Some idea of the magnitude of the problem may be gained from the fact that a major limitation at the present time in the size of chemically fueled supersonic ramjets is the problem of launching them.

Another characteristic of the ramjet which is particularly disadvantageous when feasibility is considered to be marginal is the relationship between thrust and drag as affected by flight speed. The thrust of a ramjet increases with flight speed approximately at the same rate as the drag. Thus, if a ramjet-propelled plane cannot be designed to work at one speed it is unlikely that it will work at another speed. A turbojet, on the other hand, produces a thrust that is substantially independent of flight speed; therefore, if a design of a turbojet-propelled plane appears marginal at one flight speed, a reduction in speed will usually insure successful operation.

A further serious disadvantage of the ramjet is the high wall temperature at which the reactor must run if it is to be competitive with the turbojet. If the air is heated to, say, 1500°F (815°C), a reasonable temperature at the turbine inlet for turbojets, rough calculations indicate that the required ramjet frontal area will be four or five times that required by a turbojet delivering the same thrust at a Mach number of 0.9. The size of the ramjet will then be excessively large and it most likely cannot be submerged in an airplane. If nacelles are used, the additional external drag will largely neutralize the thrust. As indicated by the preceding paragraph, this comparison will become more favorable for the turbojet at lower speeds and more favorable for the ramjet at higher speeds.

At a Mach number of 0.9 and reactor temperatures comparable to the turbojet, the ramjet not only has a frontal area which is probably excessive for an airplane, but the additional complication of a secondary heat-transfer fluid would almost certainly be necessary because of the otherwise excessively large reactor size. These difficulties at subsonic and low supersonic flight speeds can be overcome only by using temperatures far in excess of those required for the turbojet.

For the reasons given earlier, the ramjet becomes more attractive than the turbojet at flight Mach numbers greater than about 1.5 or 2.0. However, in order to keep the air flow of the ramjet (and consequently the frontal area and reactor size) within reasonable bounds, it seems necessary to use reactor wall temperatures of at least 3000°F (1650°C), and probably higher.

Calculations of the performance of nuclear ramjets for supersonic flight have been made by NEPA (1)(21), North American Aviation (4), and the Applied Physics Laboratory (5). A summary of the resulting supersonic missile designs is given in the following table.

		NEPA	North American	APL
Payload, lb.		12,000	8,500	12,000
Flight Mach Number		2.5	3	2
Altitude, ft.		40,000	36,000	50,000
Reactor Wall Temp.,	°F	3540	3540	3360
	°C	1949	1949	1849
Max. Diameter, ft.		8	5	8
Gross Weight, lb.		47,000	17,600	75,000

No consideration was given to shielding the reactor in any of the calculations presented by these groups. Since a minimum shield thickness of $1\frac{1}{2}$ to 2 feet would be probably required for protection of the payload and guidance system, these calculations represent something much below the minimum of size and performance that could be of any practical value. Without extensive calculations, it is difficult to predict whether a nuclear ramjet carrying a two-foot shield is at all feasible in an aircraft weighing less than a million pounds, even if the reactor materials problems could be solved.

From the above considerations, it is evident that the nuclear ramjet makes more difficult all of the critical problems of the nuclear turbojet, with the exception of the compressor and turbine development, and adds the additional major problem of providing an auxiliary means for launching and accelerating the aircraft to operating speed. Furthermore, the compressor and turbine development in the case of the turbojet, although possibly expensive, can be considered reasonably straightforward since it involves merely extending the principles already established for chemically fueled turbojets to larger sizes and greater pressure ratios; whereas, the major reactor problems are completely new and unique. The Lexington Project has, therefore, considered the nuclear ramjet only briefly, believing that the development of a nuclear turbojet and probably other types of propulsion systems have much more possibility of being feasible. In any case the development of other propulsion systems should be accomplished before ramjet development is pursued. Because of the improvement of ramjet performance with increasing speed and the limitations of other systems, it is recognized that the ramjet is probably the most promising means of nuclear propulsion at Mach numbers of 1.5 and above, so that eventually it must be given more careful consideration.

NUCLEAR ROCKETS

A rocket, like all the propulsion devices considered in this chapter, produces thrust by expelling a jet of high-velocity hot gases. It differs from all of the other devices considered in that it does not use atmospheric air as a propellant but instead uses propellants that are stored aboard the vehicle, and it is consequently independent of the earth's atmosphere. A chemically fueled rocket produces the hot jet gases by combustion or by chemical decomposition of the propellants; a nuclear rocket, on the other hand, would use nuclear energy to heat the propulsive gas. A schematic sketch of a nuclear rocket is shown in Figure III-8. A liquid

[redacted]

propellant is injected at fairly high pressures (perhaps 20 to 50 atmospheres) into the reactor where it vaporizes and is superheated to a high temperature. The hot gas is then expanded through a suitable exhaust nozzle where it is accelerated to a high velocity.

Since the nuclear rocket must carry its supply of propellant, its possible advantages over a chemical rocket are somewhat different than the advantages of other types of nuclear power plants over their chemical counterparts. In the other types, the nuclear power plants eliminate the fuel requirements (neglecting the small uranium consumption), and, therefore, have a practically limitless range. In the case of the nuclear rocket, however, the range will always be limited by the amount of propellant that can be carried. Therefore, the advantage of a nuclear rocket must lie in its ability to obtain more thrust per unit mass flow of propellant than does a chemical rocket. The improvement in thrust per unit mass flow must in fact be considerable in order to be of interest, because even a nuclear rocket must provide considerable shielding for instruments, payload, etc.

The thrust per unit mass of propellant is directly proportional to the exhaust velocity, and the exhaust velocity for a given pressure ratio is in turn proportional to $\sqrt{T_c/M}$, where T_c is the gas temperature entering the exhaust nozzle and M is the molecular weight of the gas. In principle, therefore, nuclear rockets could surpass chemical rockets by heating the propellant gas to higher temperatures and by the use of a propellant gas with a lower molecular weight.

Chemical rockets operate at gas temperatures between 4000 and 5000°F, although the motor walls are maintained at much lower temperatures by various cooling techniques. To reach a given gas temperature in a nuclear rocket, however, the temperature of the reactor walls must be higher than the gas temperature, so that to exceed the performance of a chemical rocket, with equal molecular weights, the reactor wall temperature of a nuclear rocket must exceed 5000°F. The structural and material problems posed by such high temperatures are so formidable that there is little immediate prospect of improving rocket performance by this method.

From these considerations, it is evident that the advantage of a nuclear rocket over a chemical rocket lies only in the possibility of using a propellant of lower molecular weight. The obvious choice of propellant would be hydrogen, which has the lowest possible molecular weight. For comparable temperatures and pressures, this would permit about a threefold increase in specific impulse over chemical rockets, although the difficulties of storing aboard the rocket liquid hydrogen with its relatively low density would reduce the improvement in performance of the vehicle as a whole.

The potential improvement in performance of a nuclear rocket over a chemical rocket is not sufficient to change its field of application relative to other types of power plants. It is not suitable for long range flights inside the earth's atmosphere, but requires instead a trajectory similar to that of the German V-2 missile. Such a flight pattern is suitable only

[redacted]

for a guided missile, since no return flight is possible. Thus, the utility of a long-range nuclear rocket depends upon the development of a satisfactory guidance system.

Preliminary design studies of nuclear rockets have been made by NEPA (1), North American Aviation, Inc. (4), and Johns Hopkins University (5). A summary of typical supersonic missile designs by the last two groups is given in the following table. The results of NEPA were not reduced to a preliminary design.

	North American	Applied Physics Laboratory
Gross Weight, lb	96,790	115,000
Payload, lb	8,000	20,000
Range, miles	10,000	5,000
Reactor Wall Temperature, °F	5,800	4,940
Max. Flight Velocity, ft/sec.	24,900	22,300

In the calculations presented by these groups, no consideration was given to shielding the reactor. An appreciable amount of shielding would probably be required in practice, in order to protect the payload and other components of the missile.

The requirement of a shield makes all of the conclusions of the above groups very optimistic and means that much larger missiles would be necessary in practice for a given payload and range.

From the above discussion it is evident that the nuclear rocket may possibly have prospects of performance somewhat superior to a chemical rocket, but that its size would be very large and its field of application limited to guided missiles. Furthermore, the required reactor wall temperatures would be much higher than those required by other types of power plants.

It is the conclusion of the Lexington Project, therefore, that the advantages to be gained by development of other types of nuclear power plants will be greater and the difficulties encountered less than in the development of a nuclear rocket. Only slight consideration has accordingly been given to the nuclear rocket.

A more detailed review of the work of other agencies is given in Reference 22.

MATERIALS FOR GAS TURBINES AND JET ENGINES

At the present time metals are available for use as blade materials in gas turbines operating with turbine inlet temperatures of about 1500°F

(815°C), for long periods of time (20,000 hours), and for stresses of 10,000 psi and higher. The requirements for aircraft call for higher stresses (20,000 psi) and shorter life (100-300 hours). Various cobalt-chromium alloys are capable of performing quite satisfactorily up to 1500°F (815°C) for over 100 hours at stresses on the order of 20,000 to 30,000 psi. These same cobalt-chromium base alloys have not been utilized to the maximum of their capacity and it is anticipated that the very best alloys of this type may perform at 1600°F (870°C) for 100 hours at stresses from 15,000 to 20,000 psi.

If higher temperatures and/or higher stresses are required, it will be necessary to develop entirely new alloy systems. Research in this direction is under way currently on chromium base alloys. Already there have been developed chromium base alloys which are stronger at 1600°F (870°C) than are the best cobalt-chromium alloys at 1400°F (760°C). This indicates a 200°F advantage which may still be sustained at 1800°F (980°C). Other developments may come in the molybdenum system; however, further refinements must be made for protection against oxidation in molybdenum alloys. The cobalt-chromium and the chromium base alloys will perform satisfactorily for 100 hours in air or helium at temperatures up to 1800°F (890°C) when the molybdenum content is under 6 to 10 percent.

For the sake of engine simplicity, solid, uncooled metallic blades are preferred; however, if the demand for temperatures exceeds 1800°F (980°C) there remain two choices:

(1) Liquid or Air-Cooled Metallic Blades Water-cooled aluminum blades have been operated successfully up to 2000°F at the NACA laboratory in Cleveland. There is no doubt that successful engines employing cooled blades can be built provided the slow-down in the engine building program and the complexity of design are not too great. In the case of a small number of nuclear-powered aircraft neither drawback is too severe.

(2) Ceramic Blades Ceramics with their very high melting points and extreme resistance to oxidation and corrosion are destined to play an important part in high-temperature high-stress applications. Tests have already been made with ceramic-bladed turbine wheels at the NACA laboratory in Cleveland with reasonable success. Considerable additional work is under way with blades made of combinations of metals and ceramics which may lead to success in view of their increased thermal shock resistance.

In the next three years, alloys with good oxidation resistance and high strength will probably be available for blade materials for 1600°F (870°C) performance for 100 hours. The best alloys today are good only for 1500°F (865°C) or less. In the next five to ten years alloys will probably be available for 1800°F (890°C) operation. Turbines utilizing water or aircooled blades could probably be built within five years for operation at 1800 to 2000°F(890 - 1100°C). Ceramic blades are the most doubtful, and another five year period will be required before a reasonable estimate can be made of their future possibilities.

REACTOR HEAT TRANSFER

General Considerations

A primary prerequisite for an aircraft nuclear reactor (or pile) is some provision for controlled dissipation of relatively tremendous amounts of heat per unit volume of reactor. Flying the shield necessary to surround even the smallest reactor requires a great amount of power. Huge volumetric rates of energy generation can indeed be achieved by the use of nuclear fission. The removal of this heat to the working fluid of a heat engine, while maintaining at the same time the integrity of the pile structure is, however, a heat-transfer problem which may often set a minimum for the pile size.

The type of pile which has been most seriously considered by the Lexington Project is a solid matrix perforated by passages which convey the heat-transfer fluid. The heat-evolving fissionable material may be distributed through the solid; in this case the heat must be conducted through the solid matrix and any protective coatings before it can be dissipated into the coolant. In other cases the fissionable material may be located in thin layers on the boundaries of a matrix of moderator so that the primary heat source is separated from the coolant only by whatever protective coating is needed. Again, following the precedent of piles at Hanford, Oak Ridge and Brookhaven, the fissionable material may appear as coated slugs or rods within cooling channels passing through a matrix of moderator; the heat must then flow by conduction through the interior of the rod or slug before reaching the coolant. In both the latter modifications, small amounts of heat are evolved in the surrounding matrix even though the fissionable material is not distributed therein. In piles of this type the solid matrix might conceivably be replaced by a molten metal or other fluid, provided a suitable enclosing structure were devised to separate the fluid from the coolant. In such a case the conduction problem within the matrix is replaced by one of free convection; the primary problem of transferring heat from the walls to the coolant remains the same.

Certain types of piles which might perhaps be advantageous from a heat-transfer point of view have not been considered because they seem impractical for other reasons. Piles in which the coolant is in direct contact with fluid or solid fissionable material, or piles in which the fissionable material is contained in a fluid medium which is itself circulated through an external exchanger, are deemed impracticable because not only the piles but also the external circuits would need heavy shielding.

Conduction in Solid Masses of Moderator or Fissionable Material

Heat generated in the body of a solid must flow by conduction to cooled boundaries if a steady state is contemplated. The interior of the solid must therefore assume a temperature higher than that of the boundaries. Any given solid material loses its structural strength above some temperature; it may even melt or pass through a phase change attended by volume changes. Furthermore, large temperature differences within a

solid may introduce thermal stresses which cause the structure to fail either by rupture or by gross warping. Any proposed design must therefore be analyzed carefully from this point of view. However, since the results are greatly influenced by the details of design, it is unprofitable here to go beyond general considerations.

The techniques of analyzing conduction problems are well understood; they are based on Fourier's equation for heat flow

$$q = -KA \frac{\partial T}{\partial x}$$

where q is the heat flux; K the thermal conductivity; A, the area across which the heat flow occurs; and $\partial T/\partial x$, the temperature gradient. Especially if the desired temperature rise of the cooling medium is large, the temperature gradients in the solid will be expected to have components in all three directions in space. On this account and also because of the usually complex geometry, numerical methods of integration must often be applied in solving the differential equation. Such methods, while tedious, nevertheless almost always provide solutions.

In general, it is found in a given case that the magnitudes of the temperature gradients and of the maximum temperature elevation above the boundaries are directly proportional to the rate of heat generation within the solid per unit volume and inversely proportional to the thermal conductivity. This result would, of course, be expected from a consideration of Fourier's basic equation. It is therefore preferable, if possible, to avoid distributing fissionable material throughout a solid unless the resultant mass has a high thermal conductivity or is endowed with such structural properties that it can withstand severe temperature gradients and high temperatures. Since metals in general have higher thermal conductivities than do non-metals, they are preferable to non-metals for structures in which heat will be generated. Certain non-metals are exceptions from this point of view, for example, graphite. At the rates of heat evolution necessary for aircraft reactors, unsymmetrical cooling of metal structures can readily result in serious warping.

The most critical uncertainties in the conduction problems arise from (a) lack of knowledge of the effect of radiation on the thermal conductivities of materials and (b) the impossibility of securing either definitive or reproducible thermal resistance in fluid films at the joints between adjacent solids. The first of these difficulties is discussed elsewhere; in some known cases a very considerable decrease in thermal conductivity is caused by exposure to radiation. The second enters, for example, when brick-type reactor structures are considered and when unbonded coatings of various kinds are introduced.

Heat Transfer to a Flowing Fluid

At the boundaries of the solid portions of the reactor, the heat is transferred to a moving fluid. In a practical system, fluid-flow

passages will be provided within the pile structure. The mechanism by which the heat generated within the reactor is transferred from the solid boundary to the moving fluid involves both conduction and convection, and in some extreme cases, radiation. It is customary to express the rate of heat transfer by means of Newton's law of cooling:

$$q = hA\ (T_w - T_o)$$

where q denotes the heat flux, h the heat-transfer coefficient, A the area for heat flow, and the quantity $(T_w - T_o)$ denotes the temperature driving force or temperature difference between the solid boundary and the moving fluid.

At steady state, the heat generated by the fission will be removed from the pile regardless of the magnitude of the heat-transfer coefficient and of the area for heat flow. When these terms are small, however, the difference in temperature between the wall and the flowing fluid becomes excessively large, and the wall temperatures may exceed the permissible temperatures within the pile structure. If extremely high temperatures are permitted, radiant heat transfer, which increases as the difference of the fourth powers of the source and sink temperatures, might be exploited. However, refractory materials endowed with the requisite structural and nuclear characteristics for this purpose are not known.

Heat-Transfer Coefficients

In order to keep the wall temperatures down to reasonable values, it is necessary to provide large values of the heat-transfer coefficient and of the heat-flow area. In general, high heat-transfer coefficients may be obtained by the use of (a) liquids having high thermal conductivities, high specific heats and low viscosities; (b) high mass-flow velocities; (c) small passage diameter; (d) boiling within the heat-transfer section.

Liquid metals are almost ideal from a heat-transfer point of view since they exhibit large values of thermal conductivity, and their high density permits the attainment of high mass velocities at relatively low linear velocities. The present empirical equations, which are based on data obtained by heating the common gases and liquids, do not accurately predict the values of the heat transfer coefficients for liquid metals. However, considerable experimental work is planned in this field, notably that of the Knolls Laboratory of the General Electric Company, the Mine Safety Appliance Company, the Oak Ridge National Laboratory, and the Argonne Laboratory. This work, as well as the small amount of research previously done by industry, has largely been confined to sodium and potassium and their mixtures. If the need arises, the techniques and apparatus could be readily adapted for experiments with other liquid metals. Fortunately, however, it is unlikely that the values of the heat-transfer coefficients for liquid metals which have been predicted from theoretical considerations will be grossly in error. Consequently, calculations made at this time can be used with confidence to estimate the performance, from heat-transfer

considerations, of a liquid-metal-cooled reactor. Final design calculations, however, will require experimental corroboration.

Heat-transfer data for gases and ordinary liquids are much more plentiful, and considerable confidence can be placed in the semi-empirical relations already developed. Although at high velocities these fluids exhibit anomalous characteristics due to boundary layer effects, sufficient experimental evidence has been accumulated to permit an adequate preliminary treatment of high-velocity heat transfer. The same type of problem arises when the temperature driving forces are very large, but work now under way at NACA has yielded promising results; it is believed that further work will satisfactorily resolve this problem. Inasmuch as gases have low values of thermal conductivity, large heat-transfer coefficients require the use of high mass velocities. This dictates high linear velocities and high gas densities.

To a good approximation in the turbulent-flow region, the heat-transfer coefficient varies inversely as the 0.2 power of the passage diameter. However, when the passage diameters become so small that the flow is laminar, a decrease in the passage diameter increases the heat-transfer coefficient more strongly, presumably due to the decrease in the length of the radial conduction paths into the fluid. Hence, in all flow regimes, small diameter passages are favorable from a heat-transfer standpoint.

Although the heat-transfer coefficients from a heated wall to a boiling liquid are high, the density fluctuations caused by bubble formation would impose such serious control problems that it is doubtful whether extensive boiling can be permitted within a reactor. Consequently, this promising approach must be held in abeyance unless further work demonstrates that boiling does not render the control of a pile impossible.

The previous discussion of our knowledge of heat-transfer coefficients for fluids is based on the assumption that the fluid will be confined in passages exhibiting regular geometries, i.e., inside cylindrical tubes, in annuli, across banks of tubes, etc. However, an optimum reactor does not necessarily consist of a matrix perforated by uniformly shaped tubes, but may well be composed of oddly shaped passages connected together in a seemingly random fashion. Our knowledge of the effect of such flow geometry on the performance of the heat transfer is fragmentary. Although some work is in progress on the heat-transfer properties of porous materials, notably at Battelle Memorial Institute, it is certain that present experimental work does not encompass all possible reactor configurations and flow geometries. While it is possible that all of the desirable regimes of reactor heat transfer may fall into cases that can be handled by conventional methods, it is felt that a reasonable amount of exploration into regions not yet adequately covered might be worth while. It may be noted further that the heat-transfer coefficient is not uniform over the periphery of irregularly shaped cross sections, but little information is available to permit prediction of its manner of variation. The lack of such information has been troublesome in the development of previous piles.

Reactor Geometry

Large values of heat-transfer area per unit volume of reactor are obtained through the use of flow passages characterized by small values of the hydraulic radius, i.e., small-diameter tubes. Unfortunately, small fluid-flow passages tend to produce a "pancake" reactor configuration which results in unfavorable geometry from a nuclear, shielding and coating standpoint. There is no doubt but that the engineering problems will be greatly decreased in magnitude if relatively large passage diameters are employed. This, however, implies an increase in reactor dimensions. As the passage diameters are decreased, in a situation requiring a fixed temperature rise of the fluid at a fixed mass velocity and a fixed pressure drop, the length of the heated passages decreases. The frontal area required for flow is not changed, however. The total frontal area of the reactor can be decreased by increasing the mass velocity (by increasing the linear velocity or increasing the density,) or by increasing the fraction of the reactor frontal area which is utilized for fluid flow. From a heat-transfer standpoint, it would be desirable to utilize all of the reactor frontal area for flow passages. From a nuclear standpoint this is impossible, and in most cases the physicist would prefer that the entire reactor consist of a solid matrix. Obviously, a compromise must be arrived at. Values of the free-flow ratio lying between 0.1 and 0.7 are probably reasonable; the exact figure depends on a balance of nuclear and heat-transfer requirements, as described further in Chapter VII.

In Figures III-122, III-123, and III-124 are diagrams of reactors which would provide the heat transfer necessary for an open-cycle turbojet of 20,000 pounds thrust operating at 30,000 feet altitude and at 0.9 Mach number with the power plant design conditions as specified. The dimensions for each reactor were estimated on the assumption that the air enters and leaves the headers radially with a mass velocity equal to that in the tubes. In the examples presented, the overall length of the reactor, including headers, is made equal to the diameter; the allowable pressure loss was assumed to be 30% of inlet pressure, and a free-flow ratio of 0.40 was used.

For the reactor in the form of a single-pass square cylinder with a free-flow ratio of 0.4 (Figure III-122) the dimensions are so small (2.54 feet diameter) that it might require a large uranium investment to make the pile critical, particularly if the moderator were graphite. The tube diameter of 0.20 inch, while small, is probably not unreasonable. The size of the bridge, or minimum web, between the tubes, of 0.08 inch, would certainly cause difficulties in construction with materials of construction such as beryllia or graphite. Decreasing the free-flow ratio to 0.20 would raise the diameter and length of the reactor to 3.6 feet, the tube diameter to 0.274 inch, and the bridge between tubes to about 0.21 inch. Thus, a decrease in free-flow ratio, with other things equal, results in a more desirable configuration from a nuclear and structural standpoint, but only at the expense of a larger reactor and a concomitant increase in shield weight.

From considerations of heat transfer alone, a split pile (Figure III-123) with a free-flow ratio of 0.4, which satisfies the same overall requirements, need be only 1.8 feet in diameter with passages 0.066 inch

in diameter. In this particular case, the pile consists of two short right cylinders end-on-end with the coolant entering through the central slot and leaving at the ends. No attempt has been made to optimize the split reactor. In consequence, the apparant nuclear and structural difficulties imposed by the dimensions found in this specific case should not lead one to eliminate split reactors from future consideration. It does, however, appear likely that a gas-cooled split reactor for this service would require a slot width inconveniently large from nuclear considerations.

Figure III-124 indicates how small the reactor might be if the air were heated by passage through plates of a porous matrix material. The dimensions listed can be only rough approximations because of the inadequacy of the present data on heat transfer in porous media. However, the header volume for such a multi-pass reactor appears to become so large a part of the total reactor volume that to render the system critical may be impossible. The high power density would lead to a high rate of radiation damage. Coating the porous material to prevent the escape of fission products and to protect the matrix from oxidation would present an extremely difficult problem. For these reasons, reactor development in this particular direction seems unlikely to be profitable.

Heat Flux Distribution

From a heat-transfer point of view, the ideal power distribution would be such that the wall temperature of the coolant passage was everywhere the maximum allowed by the materials of construction, by thus providing everywhere the highest allowable temperature difference, the cooling surface would be utilized as effectively as possible. In order to achieve this condition, the heat release per unit volume within the reactor must vary exponentially with length. This is probably not a realistic case, because the loading of fissionable material to obtain such a power distribution would be exceedingly difficult. A uniformly-loaded cylindrical reactor would exhibit a symmetrical sinusoidal distribution along the longitudinal axis and a Bessel function distribution radially. The power density would be a maximum at the center and would drop to zero at the edges. The use of reflectors at the boundaries of the pile would serve to smooth out the power distribution, and although it would still exhibit a maximum at the center, it would no longer fall off to zero at the edges. In all probability, the shield will act as a partial reflector. The introduction of control rods into the body of the reactor would perturb the distribution considerably but an exact prediction of this effect would require a knowledge of the location and character of the rods.

Although unfavorable from a heat-transfer point of view, a longitudinal variation in the wall temperature can be tolerated. In the numerical computations made by the Lexington Project to compare different types of power cycles, it was assumed that the longitudinal power distribution within the reactor was a symmetrical sinusoidal function, with the peak power at the center and the power flux at the edges of the pile 20% of that at the center. Other power distributions would affect the heat-transfer calculations, but should not invalidate the general numerical results presented.

A non-uniform radial power distribution is a more serious limitation of the heat-transfer possibilities of the pile. In order to obtain the same temperature rise for all the fluid flowing through the pile, it would be necessary to vary radially the volumetric rate of flow, or the duct distribution, or both. Although this is possible, it does not utilize the pile effectively as a heating system. It would be desirable to flatten the radial power distribution as much as possible, utilizing non-uniform loading of the pile with fissionable materials and, possibly, reflectors. Some radial variation in the flux density can be tolerated, but it should be held to a minimum. In the calculations presented in this report, the radial flux distribution has been taken to be uniform. Although this assumption is optimistic, it will not grossly affect the numerical results.

Friction Forces

The circulation of the cooling fluid through the reactor requires the expenditure of mechanical energy. It is unfortunate that the presence of a heat-transfer boundary results in the generation of shearing wall stresses. These frictional forces unavoidably result in the dissipation of potential work. Indeed, when conduction and convection are the controlling mechanisms of heat transfer, it can be shown that there is a direct relationship between the magnitude of the wall shear stress and the heat-transfer coefficient. This effect is frequently expressed in the form of a so-called analogy between heat and momentum transfer. Since the presence of a bounding wall implies wall shear forces, and hence power losses, the economical use of the passage boundaries indicates that they should also be effective for heat transfer. Consequently, as a general proposition, the use of fuel rods contained within the flow passages is undesirable. In such cases, a significant amount of heat is transferred from only the walls of the fuel rod whereas the shear forces act on both the walls of the fuel rod and on the walls of the outer annulus. Hence, to avoid power wastage, it is extremely desirable to make the entire duct boundary a heat source.

With compressible fluids, in addition to the wall shear forces, there are acceleration effects which result from the density changes associated with the heating process. Acceleration requires the expenditure of mechanical energy, and, in the case of gases, is frequently as important a factor in determining the pumping work as the wall friction. When compressibility effects are important, heat transfer and momentum transfer are interdependent, and the differential equation required to characterize the flow is not solvable by rigorous analytical methods.

If the temperatures in the reactor matrix and the arrangement of the fissionable material were not limited by the properties of materials and by nuclear considerations, the reactor dimensions set by heat-transfer considerations could be absurdly small. This ideal situation does not obtain, and the final design of a successful aircraft nuclear reactor must represent a compromise of the conflicting requirements imposed by nuclear, structural and heat-transfer considerations. Examples of possible designs are presented in Chapter VII.

The detailed methods of analysis used by the Lexington Project to

determine the characteristics of the reactor as a heat-transfer system are presented in Appendix III-E.

COMPARISON OF POWER PLANTS

The different types of power plants discussed in this chapter must in the final analysis be evaluated through the type of integration which is illustrated in Chapter VII.

It is possible here, however, to compare the various types along several lines which bear on the nuclear aircraft as a whole. At the same time the crucial development problems will be outlined. The ramjet and rocket are considered the least feasible and are omitted from the following discussion.

Simplicity Of Power Plant

The open-cycle machines are by far the simplest of the power plants considered. This is a great advantage in itself, especially for aircraft propulsion systems. The helium and bismuth closed-cycle machines are more complex than the open-cycle machines, but they do not seem unreasonably so for aircraft. With the steam or mercury cycles, the additional complication of a ternary fluid (such as liquid bismuth), makes the scheme marginal on the issue of complexity alone.

Weight Of Power Plant (Excluding Shield)

The different types of power plants have similar weight-thrust ratios, ranging in the order of 2 to 4 pounds of weight per pound of thrust at an altitude of 40,000 feet. This figure is low enough so that the weight of machinery is not a controlling factor at altitudes up to 40,000 feet. With increasing altitudes, however, the machinery weight becomes more and more important and ultimately limits the design altitude.

Size Of Power Plant

The most critical dimension of an aircraft power plant as regards installation in the airframe is the required frontal area. The ratios of frontal area to wing planform area for the several power plants are approximately as follows:

Bismuth turbojet:	1%
Helium turbojet:	1-2%
Open-cycle turbojet:	1-2%
Helium gas-turbine compressor-jet:	3%
Vapor cycles:	6-8%

Thus, the maximum frontal area for the attractive closed cycles (sometimes determined by the air compressor and sometimes by the radiator) of reasonable design is usually less than 3% of the wing planform area, or less than about 30% of the wing frontal area projected in the direction of flight.

The latter figure indicates that it will probably be feasible to house the air machinery and radiator of the closed cycles inside the wing of a subsonic aircraft.

Size of Reactor Required for Heat Transfer

The length and diameter of the reactor core, which in the studies of this chapter are determined by heat-transfer considerations and the specification that the reactor shall be a right square cylinder with a single pass of the cooling fluid, is one of the critical characteristics of the power plant because of its great influence on the weight of shield. In the range of thrust from 20,000 to 80,000 pounds at 30,000 feet altitude, the reactor core diameter as determined by heat transfer is of the order of 2 feet both for the bismuth cycle and for the helium cycles at 1000 psia, and with a free-flow ratio of 0.4. Under similar conditions, the open-cycle machines have reactor-core diameters of the order of 2.5 to 5 feet. The corresponding diameters for different free-flow ratios may be found from the simple rule that the diameter is inversely proportional to the square root of the free-flow ratio. This rule applies only to heat transfer considerations. The manner in which the required uranium investment depends on the reactor diameter and free-flow ratio is discussed in Chapter IV.

Size of Airplane

From the preceding comments on reactor size, it is evident that to carry a shield of given thickness and with a given uranium investment will in general require a larger airplane with an open-cycle turbojet than with a closed-cycle machine.

Maximum Design Altitude

Increases in flight altitude, by virtue of the effect on air density, require increases in the size of all the air-swallowing components. In the closed cycle, the size and weight of the air turbine, air compressor and radiator are dependent on altitude. In the open cycle, on the other hand, altitude affects not only the size of the air compressor and air turbine, but, and more important, affects the size of the reactor core as well and ultimately the weight of the shield. The closed-cycle machines are affected by altitude in about the same way as chemical turbojets, and, as will be seen in Chapter VII, have about the same maximum design altitude as chemical turbojets. The open-cycle nuclear turbojet suffers more severely from altitude than does the chemical turbojet, and will have a lower maximum design altitude than the closed-cycle power plants.

Materials

Considering the difficulty of finding materials suitable for the high reactor temperatures which are desirable from a power plant viewpoint, it seems worth estimating in a crude way the minimum reactor temperature for which each type of power plant might lead to reactors of reasonable size with reasonable investments of uranium.

With the open-cycle turbojet it is doubtful whether the temperature can be reduced below a figure lying somewhere in the range 1700-2000°F (927-1093°C).

The closed helium cycles probably require temperatures not lower than a figure lying somewhere in the range 1500-1800°F (815-982°C).

With the closed bismuth cycle the temperature can probably not be less than a figure lying somewhere in the range 1300-1500°F (704-815°C).

Further reductions in reactor wall temperature can be achieved probably only with a steam cycle or mercury cycle, in which case the lower temperature limit might be of the order of 1000-1200°F (538-649°C). This cycle would entail on the other hand considerable extra complication in the power plant, and the frontal area of the radiator required might be excessive for installation in an aircraft.

It is important to note that the temperatures mentioned above should be thought of as the range of probable lower limits set by an exploratory study rather than as figures which seem acceptable on the basis of specific engineering designs. Higher temperatures than these minima would increase materially the performance of the power plant.

Development of Machinery

The open-cycle turbojet resembles chemical turbojet engines, but the compressors require pressure ratios of perhaps 20 or greater whereas the pressure ratios of existing machines are between 3 and 8. To obtain machines with extraordinary pressure ratios will doubtless involve a severe developmental problem, but there appear to be no insurmountable obstacles in this respect.

Turning to the closed-cycle machines, the air compressors and air turbines are very similar to existing chemical turbojets, not only in regard to function, but also in regard to pressure ratio, so that no insurmountable development problems along these lines need be anticipated.

In the helium machines, the helium turbines and compressors are novel, but the conditions of operation are such that the methods of design used with present-day gas turbines can undoubtedly be carried over to the design of the helium gas turbine with success. Perhaps the chief development problems here would be the design of the helium machinery to be light, leak-proof, and reliable with high helium pressures.

In all the machines the construction of light and leak-proof heat exchangers will be a serious but not insurmountable development problem, but the closed-cycle systems require the more complex heat-transfer equipment.

Vulnerability

Because of its relative simplicity and lack of a secondary fluid, the open-cycle air turbojet is the least vulnerable of the different machines

considered. The closed-cycle machines not only have a secondary fluid system aside from the propulsive air, but they have large radiators which are easily damaged. Little armor could be used for the radiators because of their bulk, but on the other hand it might be possible to build them in several sections so that the loss of one or two sections would not be fatal.

The intercooler of the open-cycle turbojet, although fairly large, is less vulnerable than the heat exchangers of the closed cycles.

CONCLUSIONS AND RECOMMENDATIONS

The conclusions of the Lexington Project relating to propulsion systems are as follows:

1. There are several propulsion systems which appear to have some prospect of being suitable for nuclear-powered aircraft at subsonic speeds and at altitudes up to 50,000 feet. These include the open-cycle turbojet, the helium gas-turbine compressor-jet, the bismuth turbojet and the helium turbojet. The open-cycle turboprop has significant advantages over the open-cycle air turbojet only for flight Mach numbers less than about 0.6. The machines employing steam and mercury cycles have many disadvantages as aircraft power plants and should be turned to only as a last resort. The rocket and ramjet appear even more difficult to develop than the types already mentioned, and they should not be actively pursued at this time.

2. Each of the four promising cycles mentioned above involves one or more serious developmental problems.

3. The most pressing demand in the synthesis of a successful nuclear power plant is the perfection of suitable materials for the reactor. In this respect there are five material problems: (a) strength at high temperatures, (b) penetration of fission fragments into coolant, (c) chemical attack, (d) thermal stresses, and (e) radiation damage. The minimum reactor temperatures for which each of the machines has some prospect of feasibility probably lie somewhere in the following ranges:

Open-cycle air turbojet	1700-2000°F (927-1093°C)
Helium closed cycles	1500-1800°F (815-982°C)
Bismuth closed cycle	1300-1500°F (704-815°C)

4. The machinery required for the four promising power plants will require expensive development programs but there is good reason to believe that these programs would be successful.

5. The open-cycle turbojet offers the advantages of simplicity and less vulnerability to attack, but has the disadvantages of very severe material requirements and relatively large shield weights.

6. The closed-cycle machines offer the advantages of relatively moderate material requirements, relatively small shield weights, and somewhat

higher altitude ceilings, but have the disadvantages of complexity and somewhat greater vulnerability to attack.

7. Supersonic nuclear flight with existing designs of supersonic aircraft does not appear to be feasible, but if the lift-to-drag ratio of supersonic aircraft could be made comparable to that of subsonic aircraft, the types of power plants considered here would be suitable for flight Mach numbers up to about 1.5.

The recommendations on power plants are as follows:

1. Analyses of the different possible power plants should be made on a scale much broader than was possible within the time available to the Lexington Project. Such an analytical program should cover a variety of propulsion systems and working fluids, and wide ranges of such variables as altitude, speed, pressure ratio, temperatures, component efficiencies, pressure drops, off-design operation and fuel investment. In addition, preliminary design layouts should be made simultaneously to investigate the structural and installation problems.

2. For the present, the developmental problems of the open-cycle turbojet and the closed-cycle turbojets and compressor-jets should be given primary attention. Vapor cycles should be considered as last resorts and should not be pressed now. Likewise in the interests of economy of time and effort, the development of nuclear rocket and ramjet power plants should await prior development of one of the more promising cycles.

3. The selection of a particular power plant for intensive development and incorporation into a nuclear-powered aircraft should be preceded by a program whose aim is to solve for each engine the crucial materials problems which cast doubt on the feasibility of the engine. The object of the materials program would be to develop suitable reactor structures, including coatings or cannings to protect against the escape of fission fragments and against chemical attack, for reactor temperatures as follows:

Open cycle turbojet	2000 to 2300°F
Helium gas-turbine compressor-jet	1750 to 2000°F
Helium turbojet	1750 to 2000°F
Bismuth turbojet	1500 to 1800°F

Details of a suggested materials program are given in Chapter IV.

4. Assuming that satisfactory progress is made along one or more of the lines outlined immediately above, the choice of power plant may then be made on the basis of all the considerations which enter into the design of a nuclear-powered aircraft. At that stage the intensive development of compressors, turbines, heat exchangers, ducting, etc., should be undertaken.

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(22) Redding, E. M., "The Feasibility of Nuclear-Powered Rockets," Lexington Project Report LP-148.

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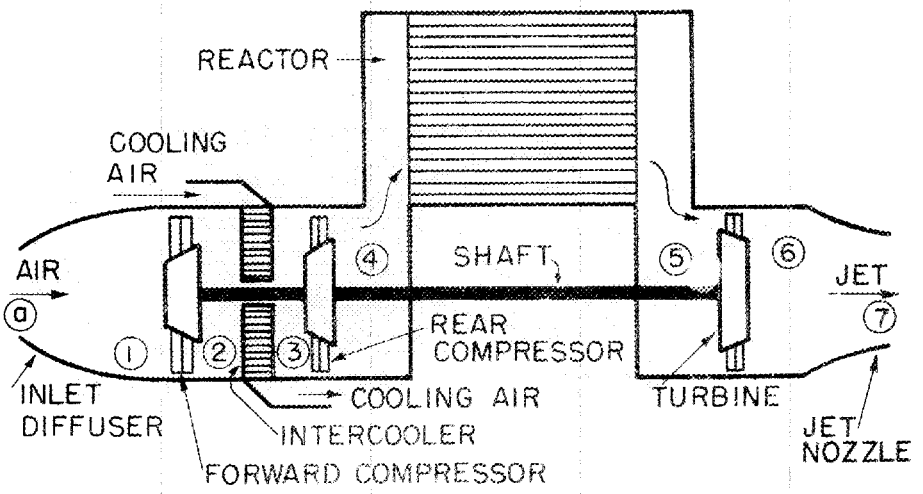

FIGURE III - 1

OPEN CYCLE AIR TURBOJET

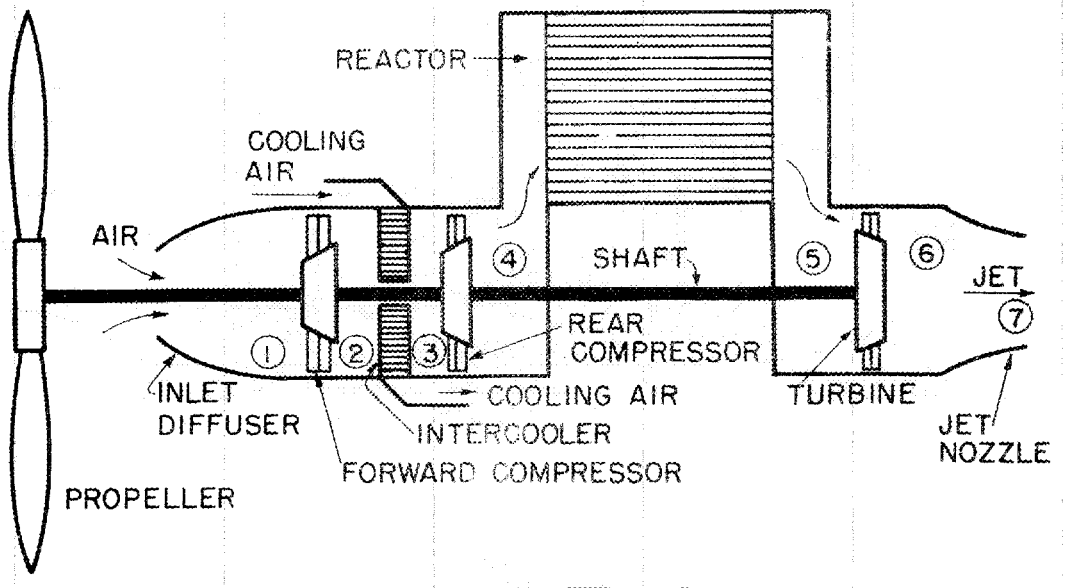

FIGURE III - 2

OPEN CYCLE AIR TURBOPROP

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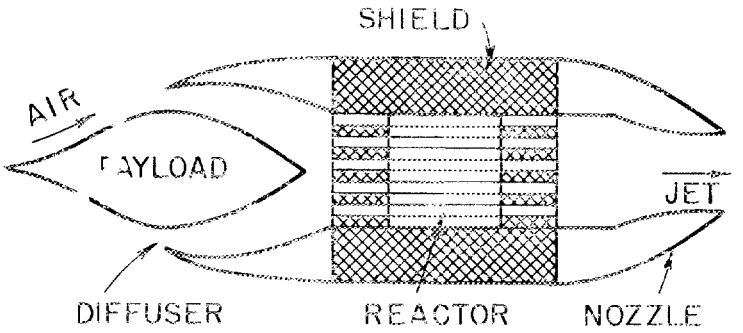

FIGURE III-3

NUCLEAR RAMJET

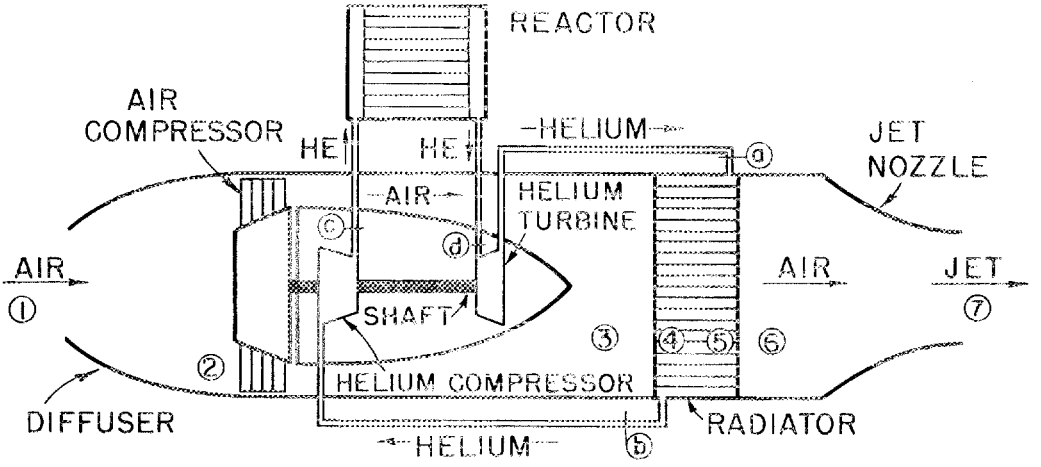

FIGURE III-4

HELIUM GAS-TURBINE COMPRESSOR-JET

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[redacted]

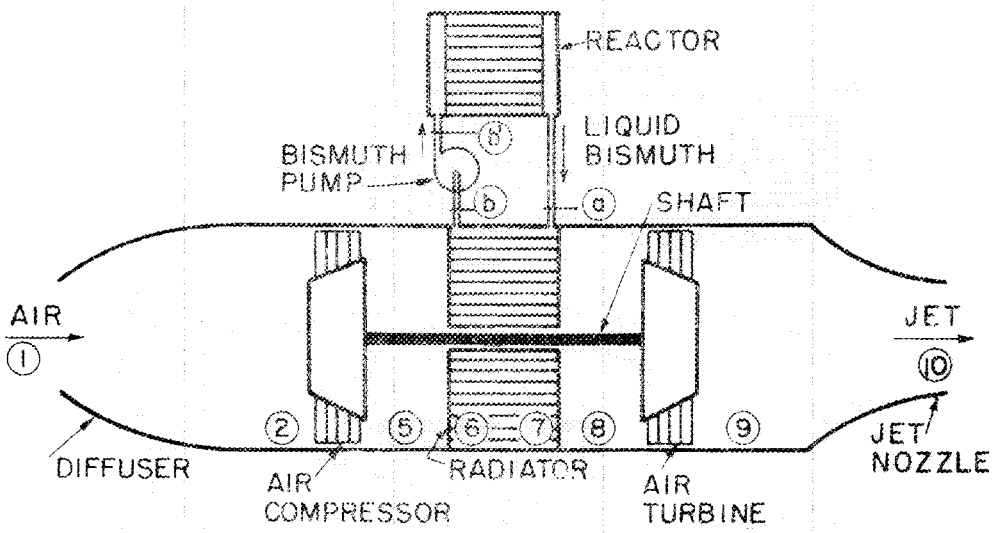


FIGURE III-5

BISMUTH TURBOJET

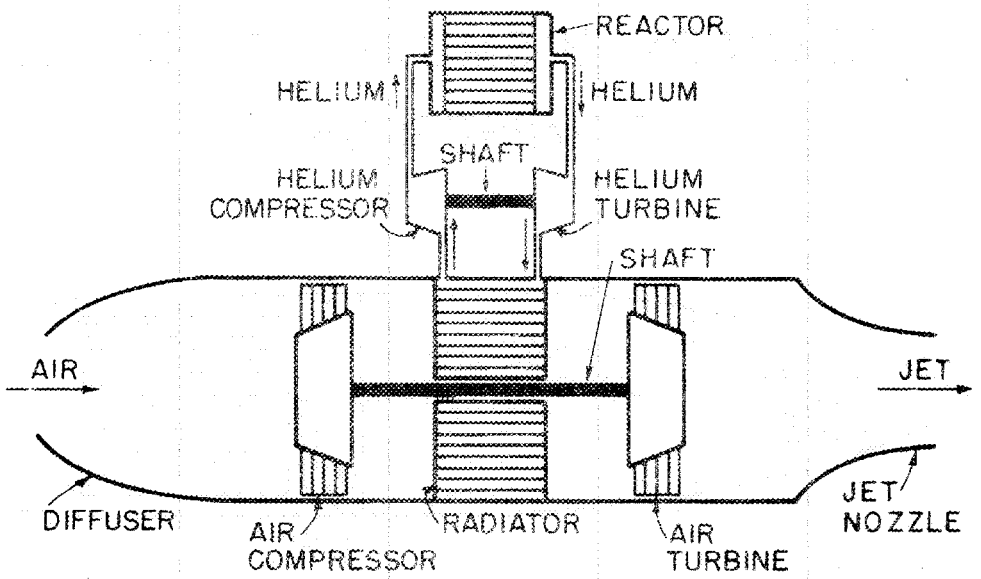


FIGURE III-6

HELIUM TURBOJET

[redacted]

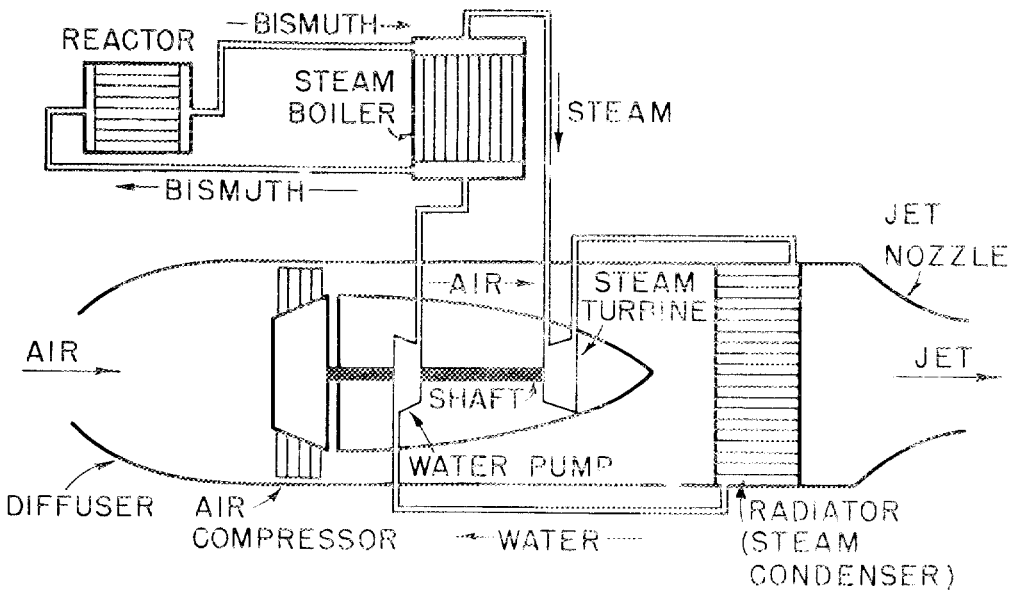


FIGURE III - 7

TERNARY BISMUTH STEAM-CYCLE COMPRESSOR-JET

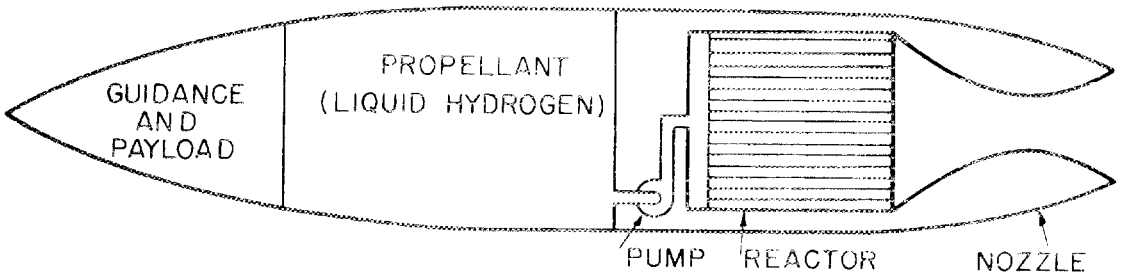


FIGURE III - 8

NUCLEAR ROCKET

[redacted]

FIGURES III-9, III-10, III-11, III-12

OPEN CYCLE POWERPLANT

COMPARISON OF TURBOJET AND TURBOPROP ENGINES

DESIGN CONDITIONS

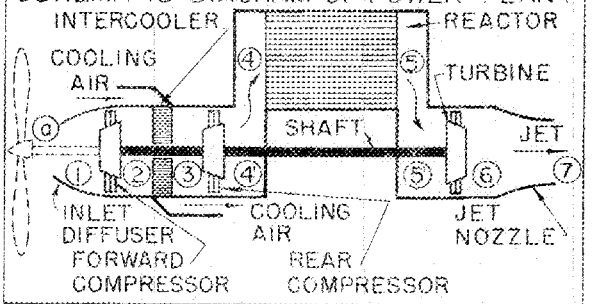
FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40.1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	1800°F

MAX. REACTOR WALL TEMPERATURE: 2500°F

EFFICIENCIES: $\eta_D = .90$; $\eta_C = .82$; $\eta_T = .87$; $\eta_N = .90$; $\eta_G = .95$; $\eta_P = .80$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



POWER PLANT	SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
TURBOJET	43.0 LB-SEC./LB.	6.7 KW/LB.
TURBOPROP	59.5 LB-SEC. LB	4.9 KW/LB.

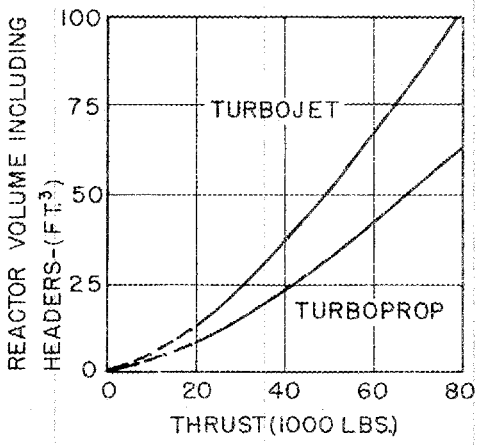


FIGURE III-9

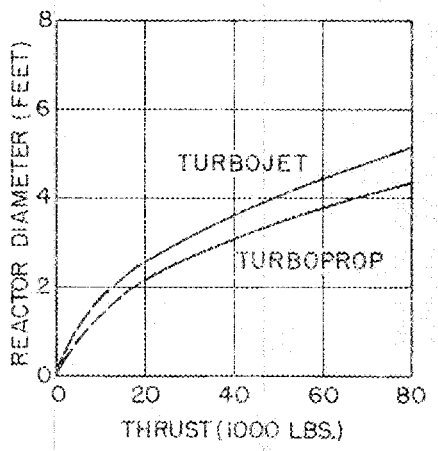


FIGURE III-10

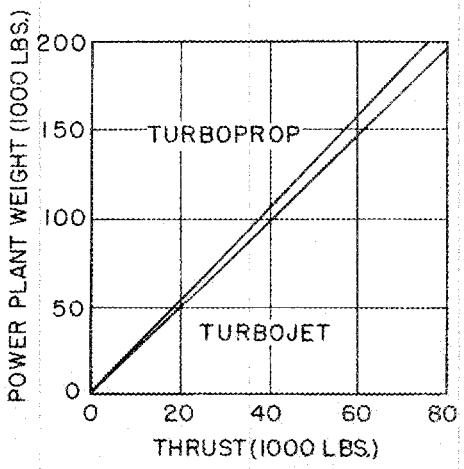


FIGURE III-11

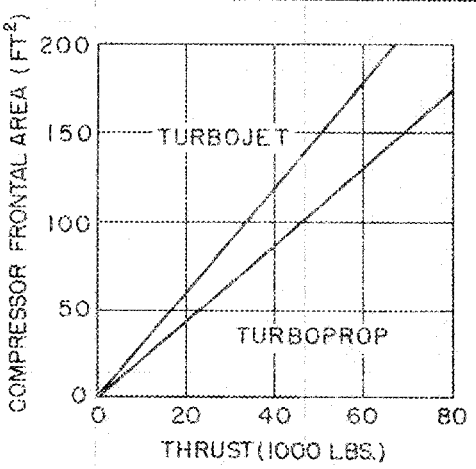


FIGURE III-12

[redacted]

FIGURES III-13, III-14, III-15, III-16

OPEN CYCLE AIR TURBOJET

EFFECT OF FLIGHT ALTITUDE

DESIGN CONDITIONS

FLIGHT MACH NUMBER: 0.9

ALTITUDE: VARIABLE

COMPRESSOR PRESSURE RATIO: 40:1

INTERCOOLER EFFECTIVENESS: 0.8

REACTOR PRESSURE DROP: 30.6

REACTOR FREE-FLOW RATIO: 0.4

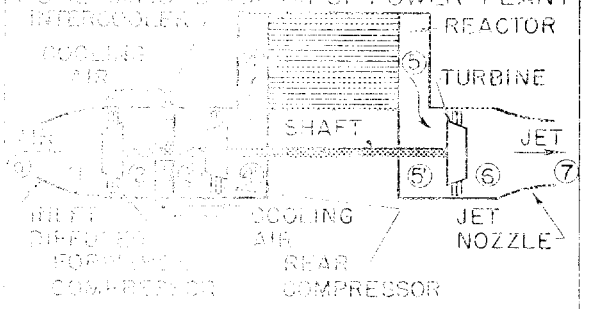
TURBINE INLET TEMPERATURE: 1800°F

MAX. REACTOR WALL TEMPERATURE: 2000°F

EFFICIENCIES: $\eta_D = .90$; $\eta_c = .82$; $\eta_t = .87$; $\eta_n = .95$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



FLIGHT ALTITUDE	SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
30,000 FT	46.0 LB-SEC/LB	6.7 KW/LB.
40,000 FT	47.5 LB-SEC/LB	6.4 KW/LB.
50,000 FT	47.5 LB-SEC/LB	6.4 KW/LB.
60,000 FT	47.5 LB-SEC/LB	6.4 KW/LB.

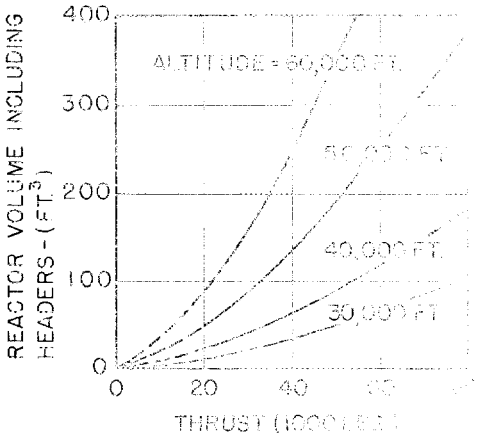


FIGURE III-13

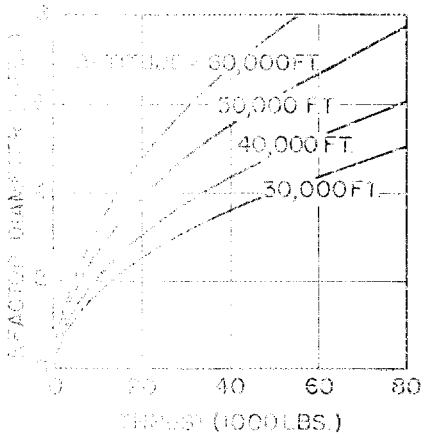


FIGURE III-14

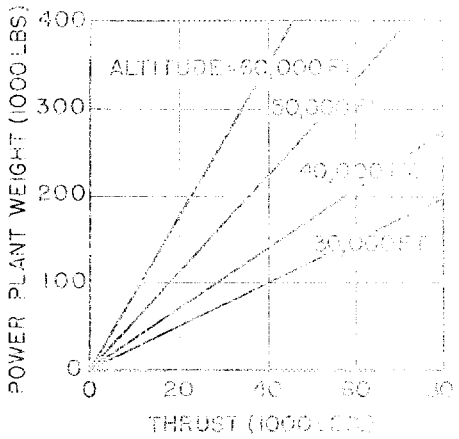


FIGURE III-15

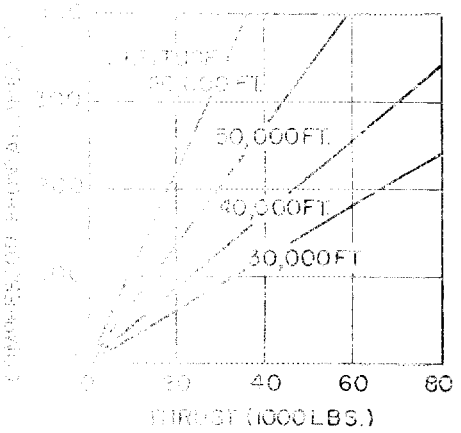


FIGURE III-16

FIGURES III-17, III-18, III-19, III-20

OPEN CYCLE AIR TURBOJET

EFFECT OF AIRPLANE FLIGHT SPEED

DESIGN CONDITIONS

FLIGHT MACH NUMBER: VARIABLE

ALTITUDE: 30,000 FEET

COMPRESSOR PRESSURE RATIO: VARIABLE;

$M=.5, R_c=55; M=.9, R_c=40; M=1.5, R_c=20$

INTERCOOLER EFFECTIVENESS: 0.5

REACTOR PRESSURE DROP: 30 %

REACTOR FREE-FLOW RATIO: 0.40

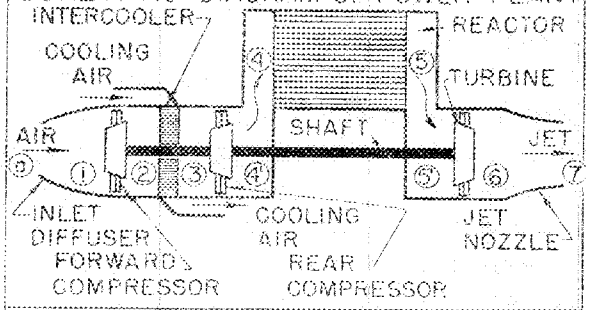
TURBINE INLET TEMPERATURE: 1800°F

MAX REACTOR WALL TEMPERATURE: 2500°F

EFFICIENCIES: $\eta_D=.90; \eta_c=.82, \eta_T=.87; \eta_N=.90$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



FLIGHT MACH NUMBER M	SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
0.5	53.2 LB-SEC/LB.	5.6 KW./LB.
0.9	43.0 LB-SEC./LB.	6.7 KW./LB.
1.5	31.5 LB-SEC./LB.	8.8 KW./LB.

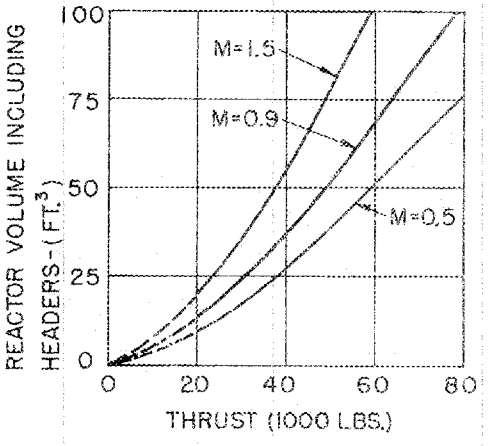


FIGURE III-17

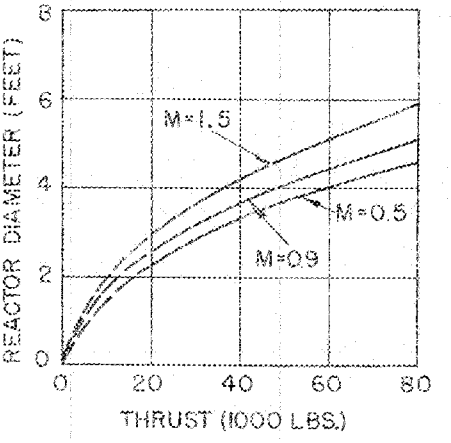


FIGURE III-18

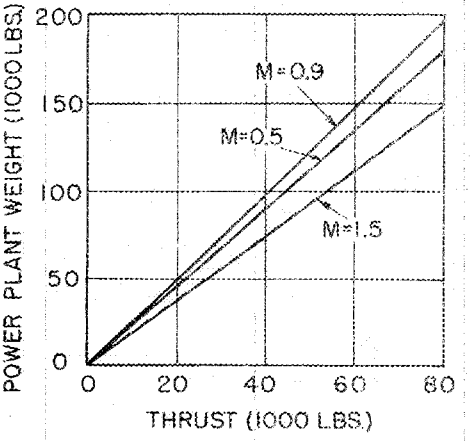


FIGURE III-19

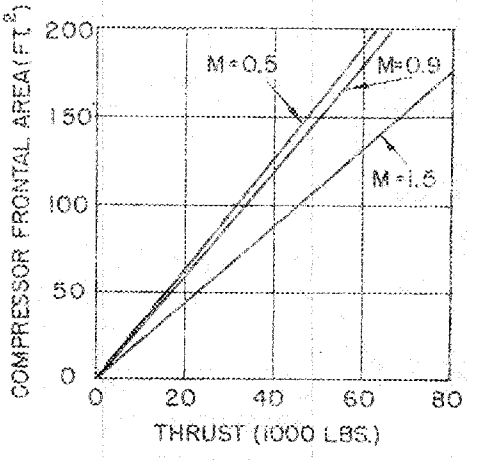


FIGURE III-20

[redacted]

FIGURES III-21, III-22, III-23, III-24

OPEN CYCLE AIR TURBOJET

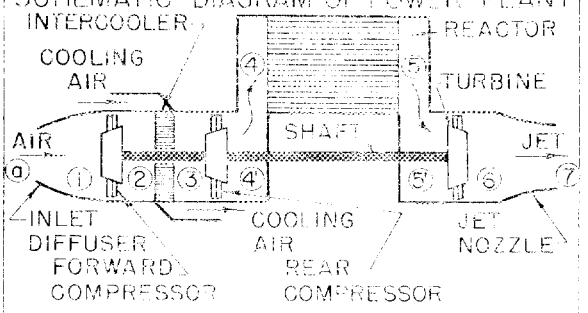
EFFECT OF COMPONENT EFFICIENCIES

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.90
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40:1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	1800°F
MAX. REACTOR WALL TEMPERATURE:	2500°F
EFFICIENCIES:	VARIABLE

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



EFFICIENCIES				SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
η_D	η_C	η_T	η_N		
.90	.78	.83	.90	33.8 LB.-SEC./LB.	8.2 KW./LB.
.90	.82	.87	.90	43.0 LB.-SEC./LB.	6.7 KW./LB.
1.00	1.00	1.00	1.00	75.51 LB.-SEC./LB.	5.5 KW./LB.

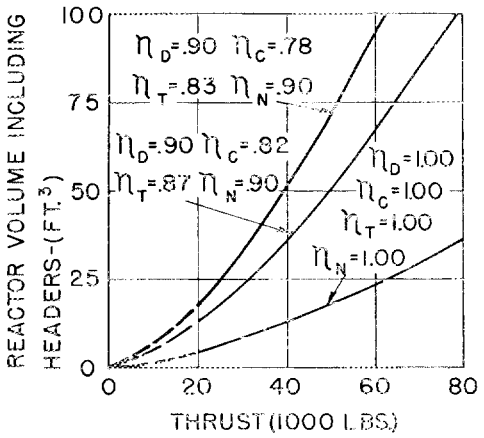


FIGURE III-21

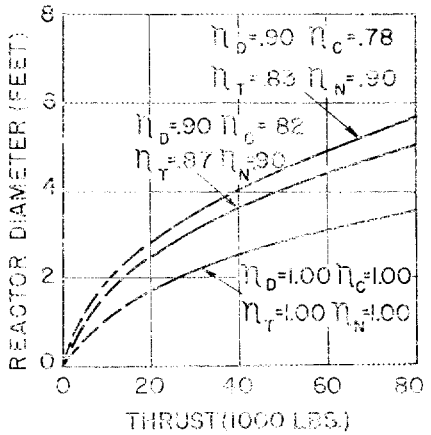


FIGURE III-22

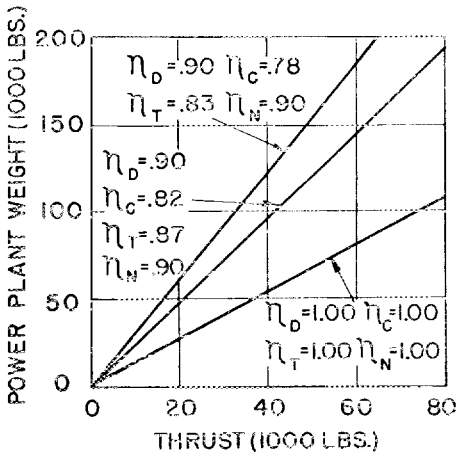


FIGURE III-23

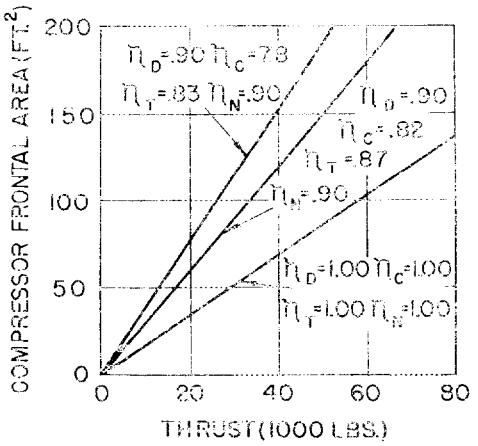


FIGURE III-24

[redacted]

[redacted]

FIGURES III-25, III-26, III-27, III-28

OPEN CYCLE AIR TURBOJET

EFFECT OF TURBINE INLET TEMPERATURE

DESIGN CONDITIONS

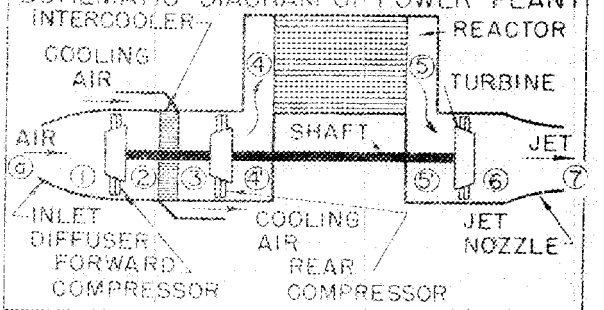
FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40:1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	VARIABLE

MAX. REACTOR WALL TEMPERATURE: 2500°F

EFFICIENCIES: $\eta_D = .90$; $\eta_C = .82$; $\eta_T = .87$; $\eta_W = .90$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



TURBINE INLET TEMP T_{05}	SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
1500° F	22.6 LB.-SEC./LB.	9.0 KW./LB.
1800° F	43.0 LB.-SEC./LB.	6.7 KW./LB.
2100° F	58.6 LB.-SEC./LB.	6.4 KW. LB.

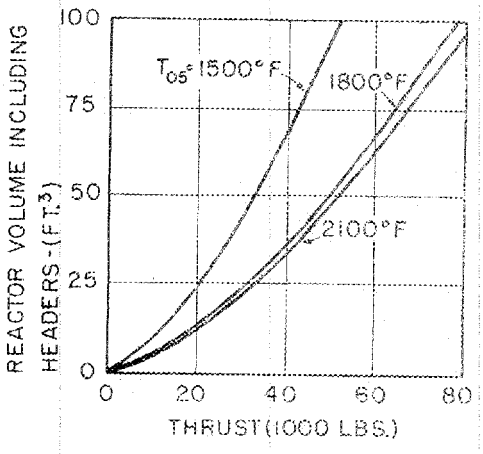


FIGURE III-25

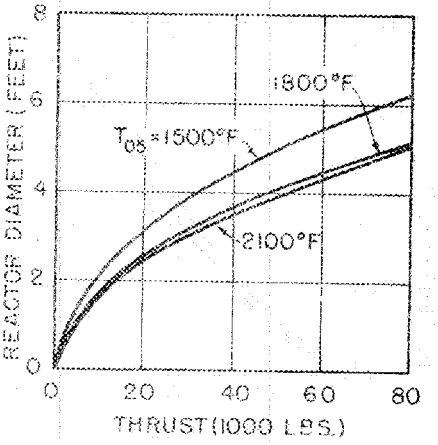


FIGURE III-26

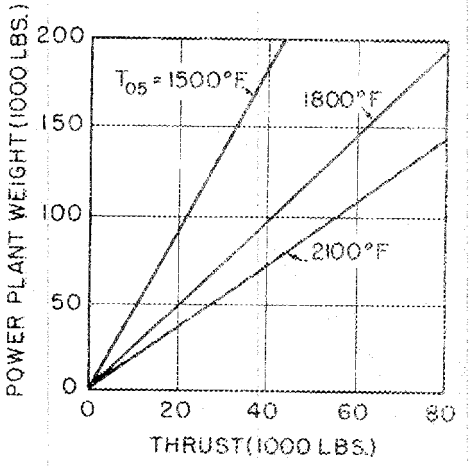


FIGURE III-27

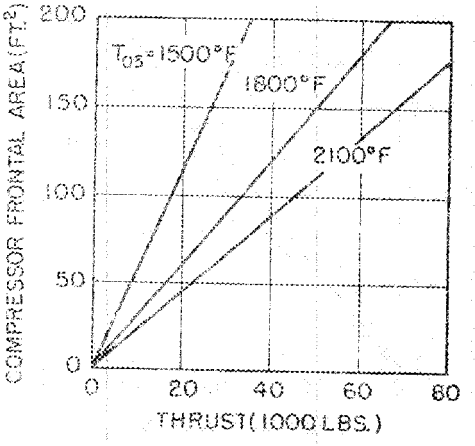


FIGURE III-28

[redacted]

[redacted]

FIGURES III-29, III-30, III-31, III-32

OPEN CYCLE AIR TURBOJET

EFFECT OF TURBINE INLET TEMPERATURE

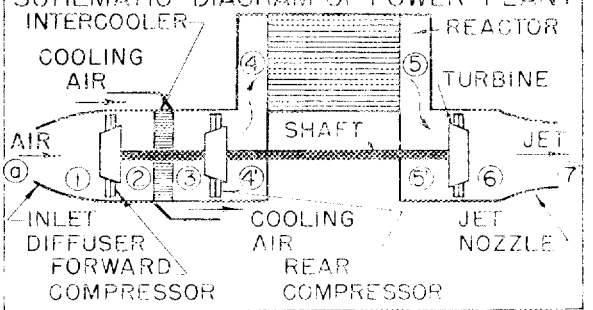
DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40:1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	VARIABLE
MAX. REACTOR WALL TEMPERATURE:	1832°F

EFFICIENCIES: $\eta_D = .90$; $\eta_C = .82$; $\eta_T = .87$; $\eta_N = .90$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



TURBINE INLET TEMP. T_{05}	SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
1500°F	22.6 LB-SEC/LB	9.0 KW/LB.
1650°F	33.8 LB-SEC/LB.	7.3 KW/LB.
1750°F	40.2 LB-SEC/LB.	6.9 KW/LB.

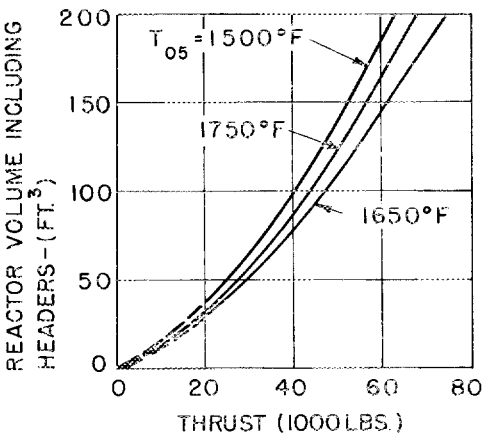


FIGURE III-29

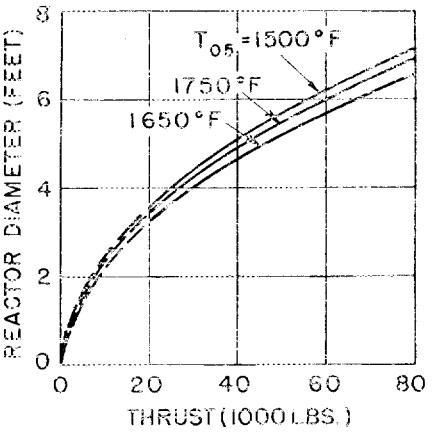


FIGURE III-30

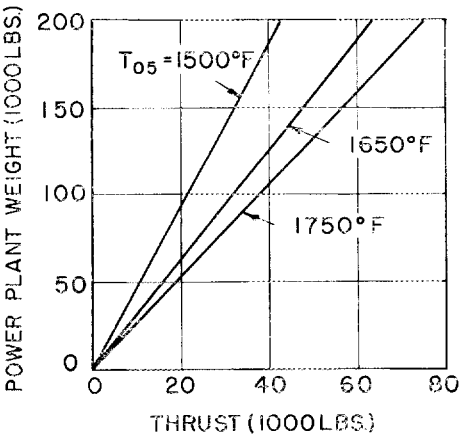


FIGURE III-31

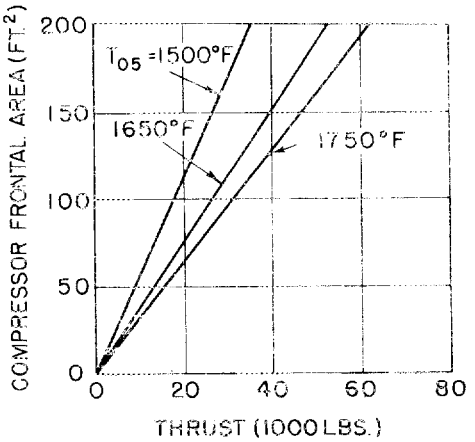


FIGURE III-32

[redacted]

[redacted]

FIGURES III-33, III-34, III-35, III-36

OPEN CYCLE AIR TURBOJET

EFFECT OF MAXIMUM REACTOR WALL TEMPERATURE

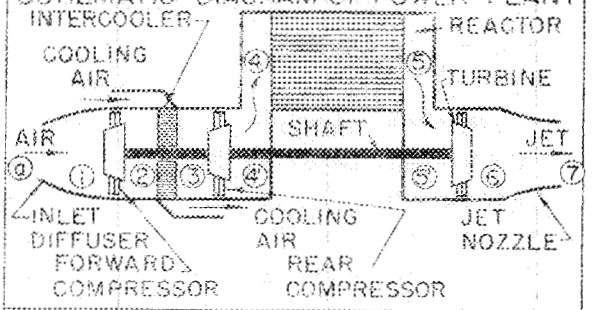
DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40:1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	1800°F
MAX. REACTOR WALL TEMPERATURE:	VARIABLE

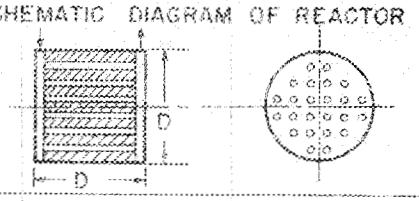
EFFICIENCIES: $\eta_c = .82$; $\eta_t = .87$; $\eta_N = .90$; $\eta_D = .90$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



SCHEMATIC DIAGRAM OF REACTOR



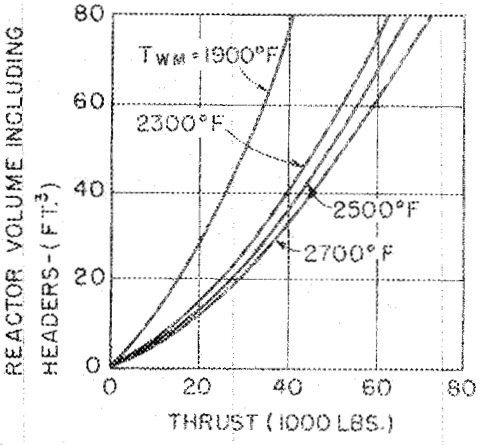


FIGURE III-33

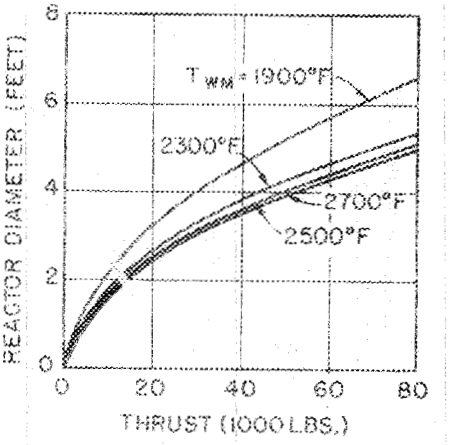


FIGURE III-34

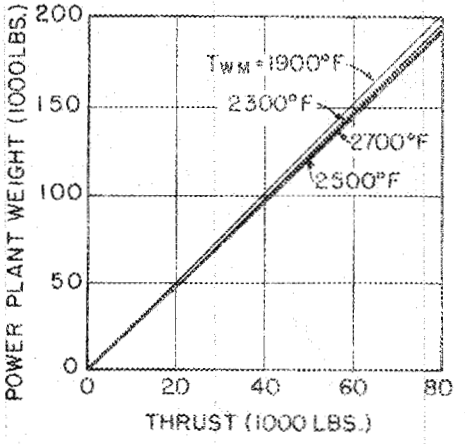


FIGURE III-35

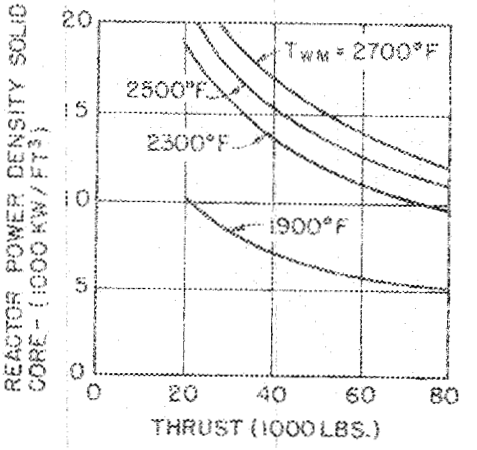


FIGURE III-36

[redacted]

[redacted]

FIGURES III-37, III-38, III-39, III-40
OPEN CYCLE AIR TURBOJET
EFFECT OF COMPRESSOR PRESSURE RATIO

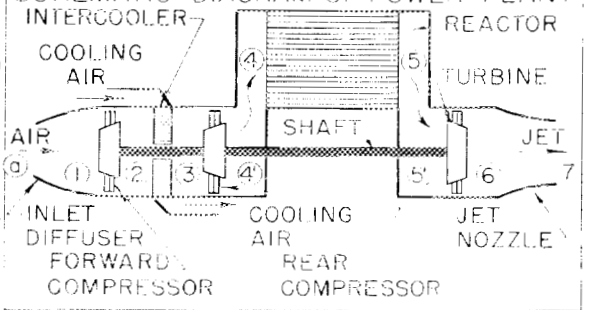
DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	VARIABLE
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	1800°F
MAX REACTOR WALL TEMPERATURE:	2500°F

EFFICIENCIES: $\eta_D = .90; \eta_C = .82; \eta_T = .87; \eta_N = .90$

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



COMP. PRESS RATIO R_C	SPECIFIC IMPULSE I	REACTOR POWER PER UNIT THRUST
20	52.5 LB-SEC./LB.	6.6 KW./LB.
40	43.0 LB-SEC./LB.	6.7 KW./LB.
60	35.1 LB-SEC./LB.	7.2 KW./LB.
80	26.4 LB-SEC./LB.	8.3 KW./LB.

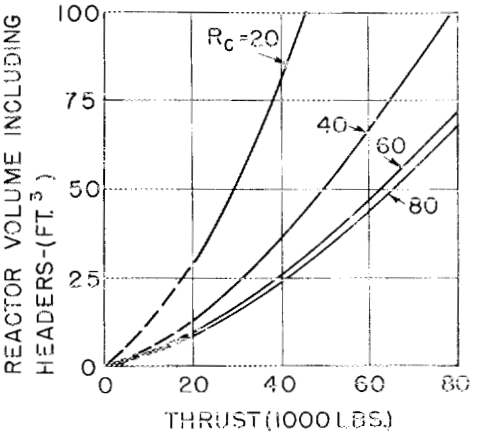


FIGURE III-37

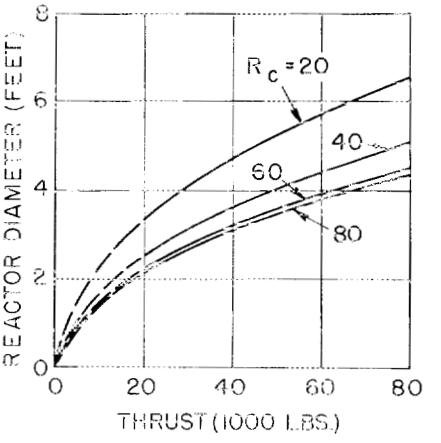


FIGURE III-38

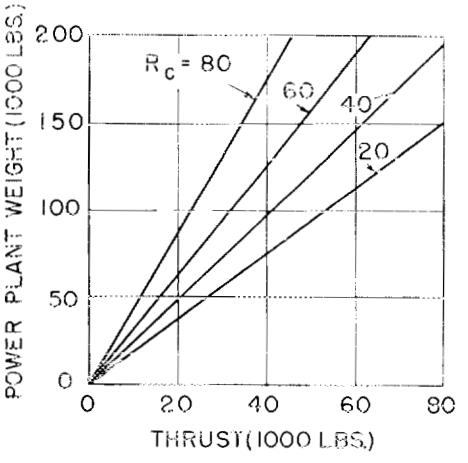


FIGURE III-39

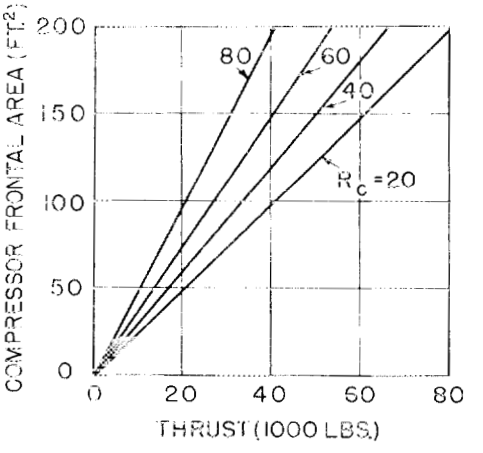


FIGURE III-40

[redacted]

FIGURES III-41, III-42, III-43, III-44

OPEN CYCLE AIR TURBOJET

EFFECT OF REACTOR PRESSURE DROP

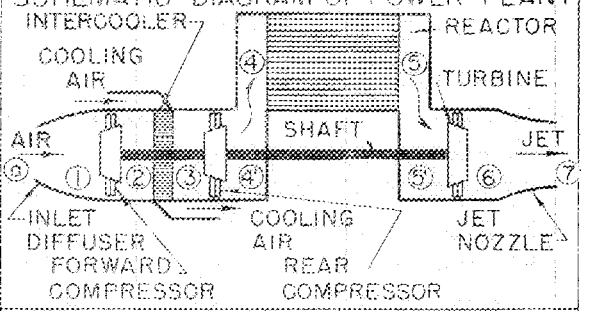
DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40:1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	VARIABLE
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	1800°F
MAX. REACTOR WALL TEMPERATURE:	2500°F

EFFICIENCIES: η_D=.90; η_C=.82; η_T=.87; η_N=.90

SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



REACTOR PRESSURE DROP, ΔP	SPECIFIC IMPULSE, I	REACTOR POWER PER UNIT THRUST
20%	45.5 LB-SEC/LB.	6.4 KW/LB.
30%	43.0 LB-SEC/LB.	6.7 KW/LB.
40%	40.0 LB-SEC/LB.	7.2 KW/LB.

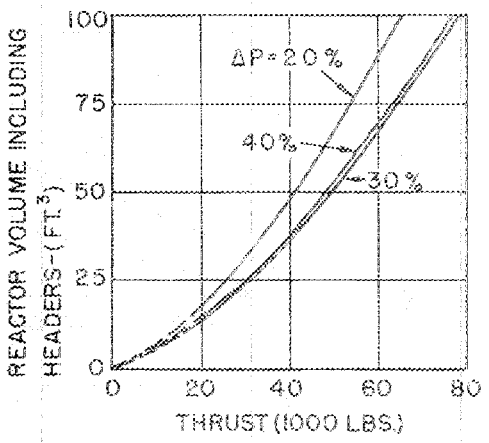


FIGURE III-41

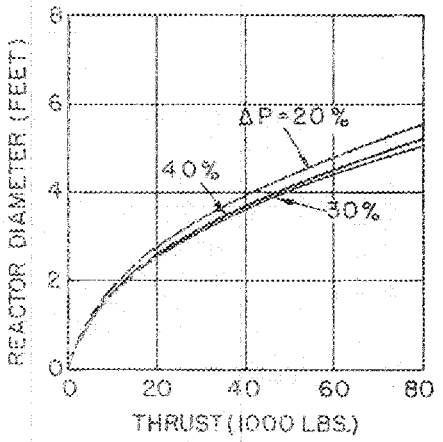


FIGURE III-42

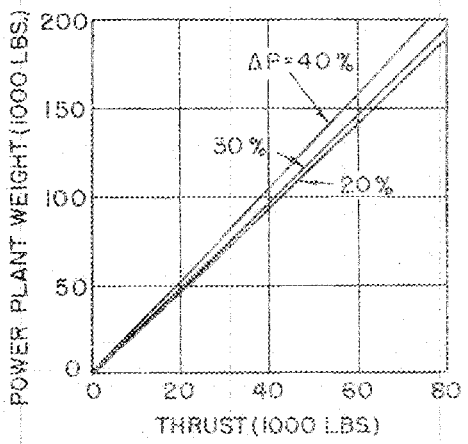


FIGURE III-43

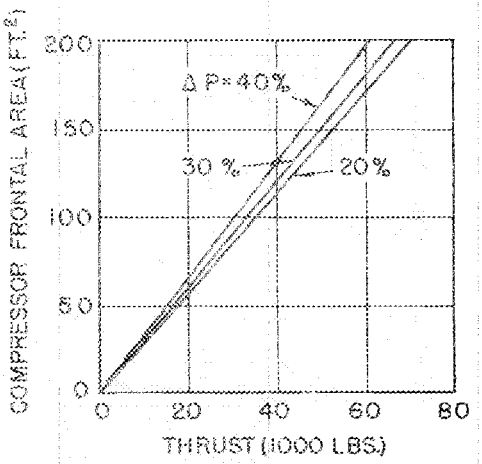


FIGURE III-44

FIGURES III-45, III-46, III-47, III-48

OPEN CYCLE AIR TURBOJET

EFFECT OF REACTOR GEOMETRY

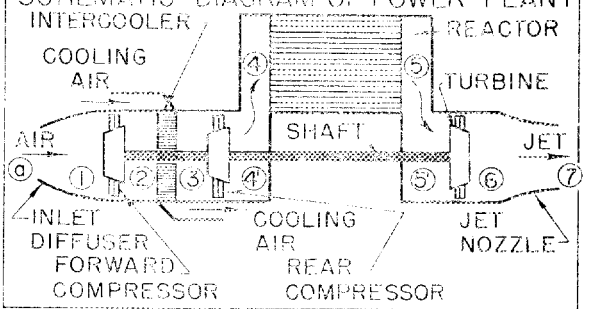
DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FEET
COMPRESSOR PRESSURE RATIO:	40:1
INTERCOOLER EFFECTIVENESS:	0.50
REACTOR PRESSURE DROP:	30%
REACTOR FREE-FLOW RATIO:	0.40
TURBINE INLET TEMPERATURE:	1800°F
MAX. REACTOR WALL TEMPERATURE:	2500°F

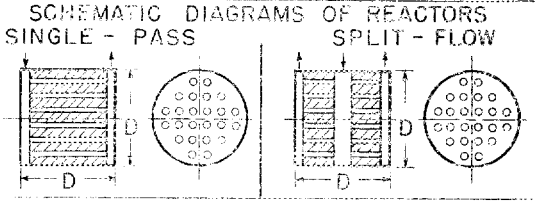
EFFICIENCIES: $\eta_D = .90$; $\eta_C = .82$; $\eta_T = .87$; $\eta_N = .90$

SINGLE PASS AND SPLIT FLOW REACTORS WITH HEADERS, SIZE BASED ON HEAT TRANSFER ONLY

SCHEMATIC DIAGRAM OF POWER PLANT



SCHEMATIC DIAGRAMS OF REACTORS



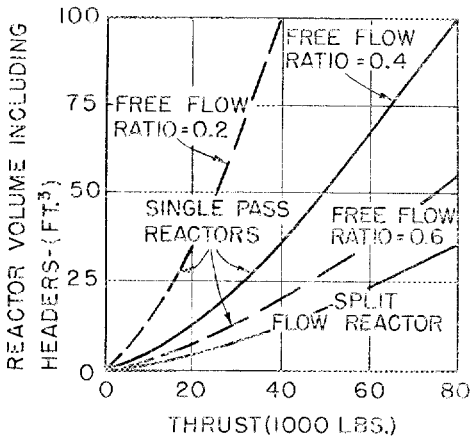


FIGURE III-45 SQUARE CYLINDER REACTOR WITH HEADERS

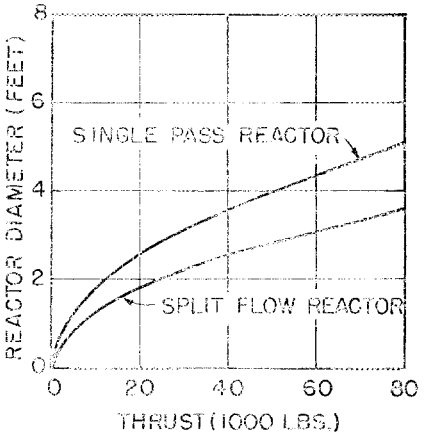


FIGURE III-46 SQUARE CYLINDER REACTOR WITH HEADERS

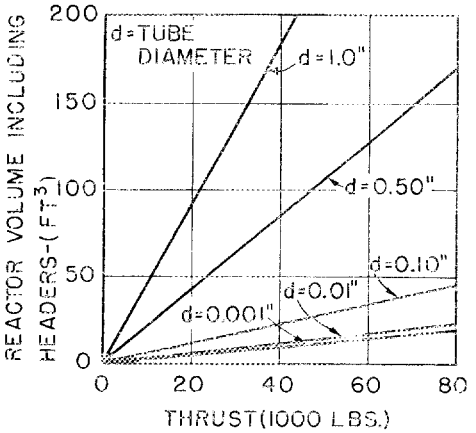


FIGURE III-47 CYLINDRICAL REACTOR WITH HEADERS

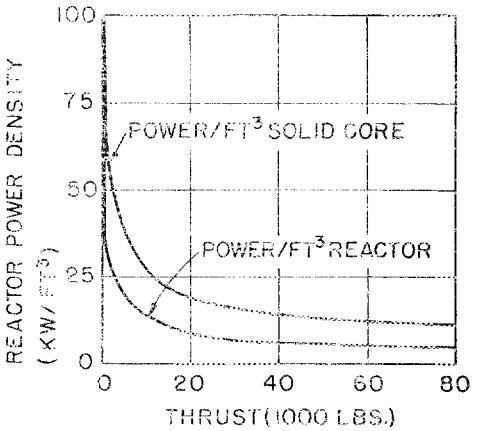


FIGURE III-48 SINGLE-PASS SQUARE CYLINDER REACTORS WITH HEADERS

[redacted]

FIGURES III-49, III-50, III-51

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF HELIUM CYCLE (BASED ON WORKING ASSUMPTIONS)

DESIGN CONDITIONS

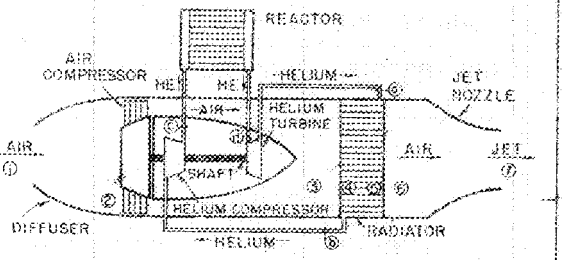
COMPRESSOR EFFICIENCY:	85%
TURBINE EFFICIENCY:	88%
TURBINE PRESSURE RATIO / COMPRESSOR PRESSURE RATIO:	0.9

NOMENCLATURE

T_d = TURBINE INLET TEMPERATURE

T_b = COMPRESSOR INLET TEMPERATURE

ROMAN NUMERALS REFER TO SPECIFIC HELIUM CYCLES USED IN COMPLETE PROPULSION SYSTEM (SEE TEXT)



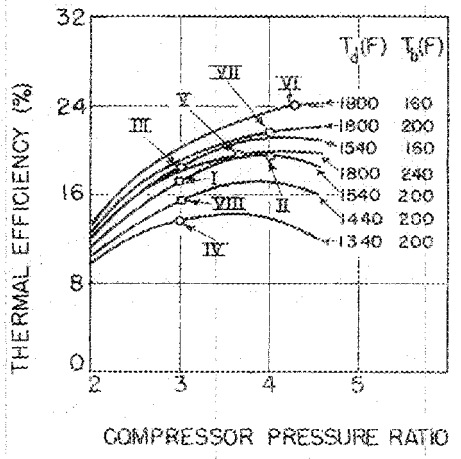


FIGURE III-49

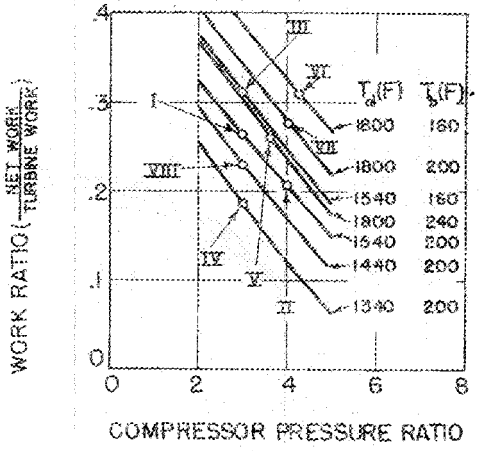


FIGURE III-50

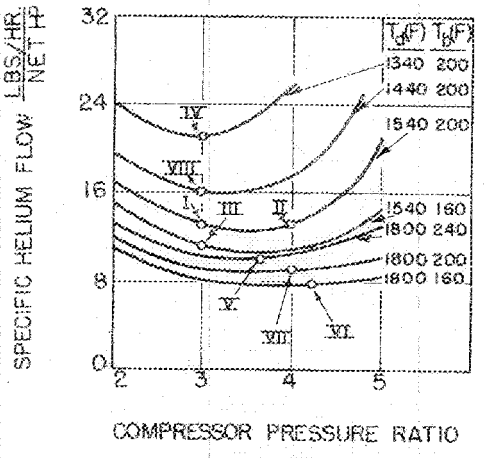


FIGURE III-51

[redacted]

[redacted]

FIGURES III-52, III-53, III-54

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF HELIUM CYCLE (BASED ON OPTIMISTIC ASSUMPTIONS)

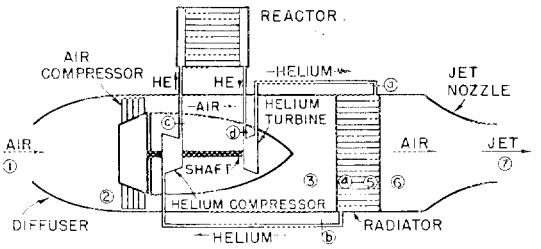
DESIGN CONDITIONS

COMPRESSOR EFFICIENCY:	90 %
TURBINE EFFICIENCY:	92 %
$\frac{\text{TURBINE PRESSURE RATIO}}{\text{COMPRESSOR PRESSURE RATIO}}$:	0.95

NOMENCLATURE

T_d = TURBINE INLET TEMPERATURE

T_b = COMPRESSOR INLET TEMPERATURE



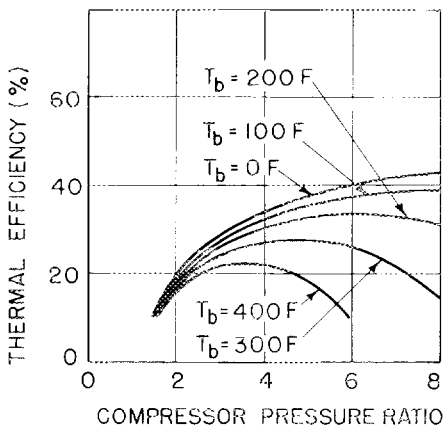


FIGURE III-52

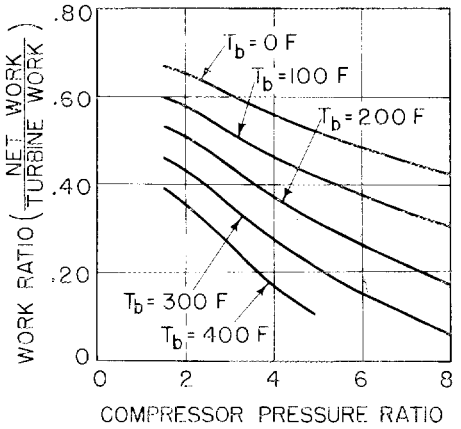


FIGURE III-53

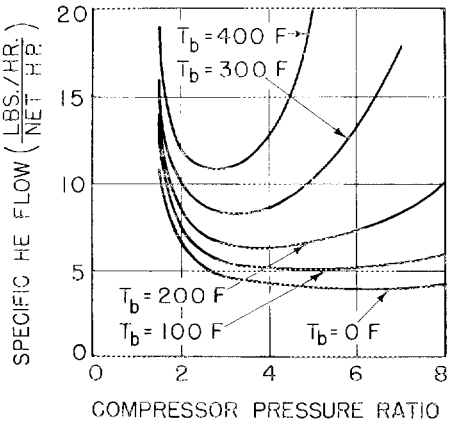


FIGURE III-54

[redacted]

[redacted]

FIGURES III-55, III-56, III-57

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF HELIUM CYCLE (BASED ON PESSIMISTIC ASSUMPTIONS)

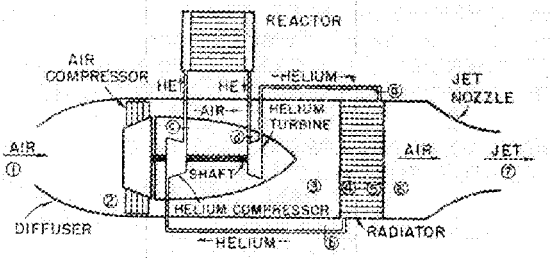
DESIGN CONDITIONS

COMPRESSOR EFFICIENCY :	80 %
TURBINE EFFICIENCY :	82 %
TURBINE PRESSURE RATIO / COMPRESSOR PRESSURE RATIO :	.80

NOMENCLATURE

T_d = TURBINE INLET TEMPERATURE

T_b = COMPRESSOR INLET TEMPERATURE



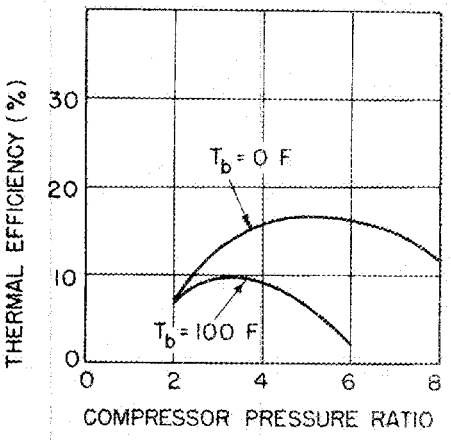


FIGURE III-55

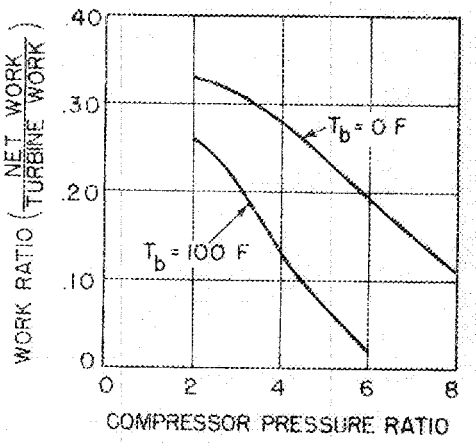


FIGURE III-56

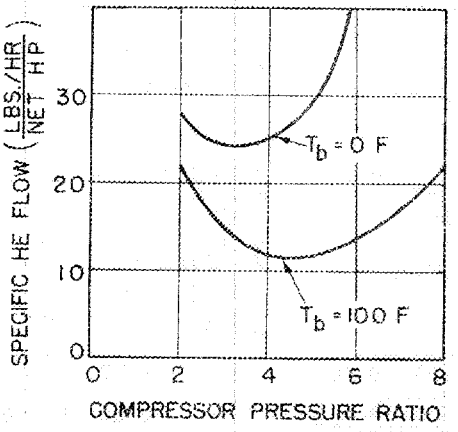


FIGURE III-57

[redacted]

[redacted]

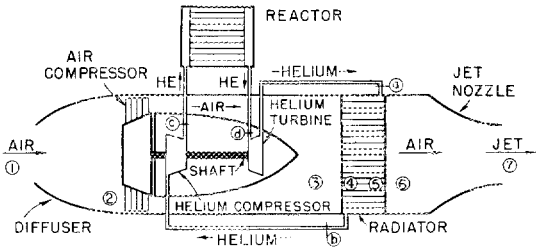
FIGURES III-58, III-59, III-60, III-61

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE I)

DESIGN CONDITIONS

FLIGHT MACH NO.: 0.9
ALTITUDE: 30,000 FEET
HE COMPRESSOR PRESSURE RATIO: 3.0
HE TURBINE PRESSURE RATIO: 2.7
HE COMPRESSOR INLET TEMPERATURE: 200 F.
HE TURBINE INLET TEMPERATURE: 1540 F.
EFFICIENCIES: (HE CYCLE) $\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE) $\eta_D = 90\%$; $\eta_C = 85\%$
$\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET: 0.5
RADIATOR TUBES: DIAMETER 1/8 IN.
WALL THICKNESS 0.004 IN.
DENSITY 490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE
W_A = AIR FLOW RATE
P_R = REACTOR INLET PRESSURE

HELIUM CYCLE I

THERMAL EFFICIENCY: 0.17
WORK RATIO: 0.265
SPECIFIC HELIUM FLOW: 13.2 LBS/HP-HR.

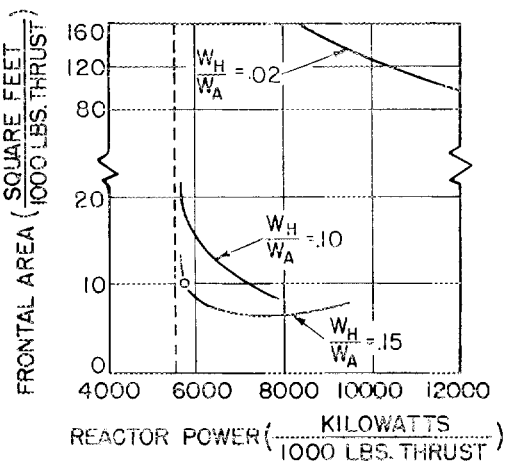


FIGURE III-58

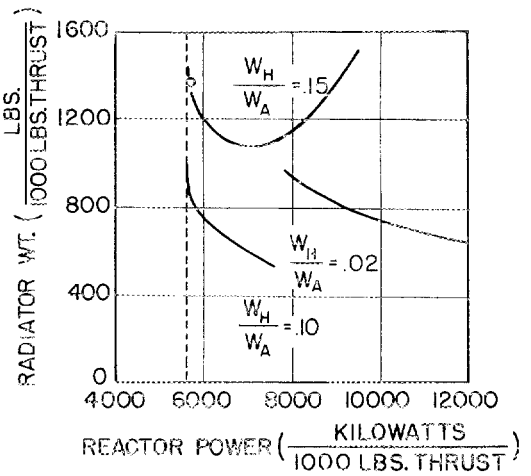


FIGURE III-59

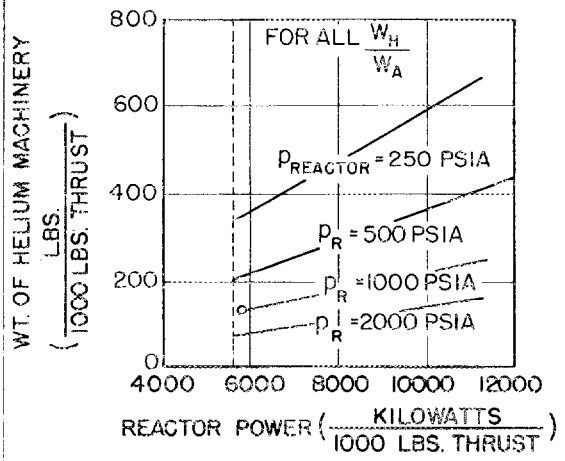


FIGURE III-60

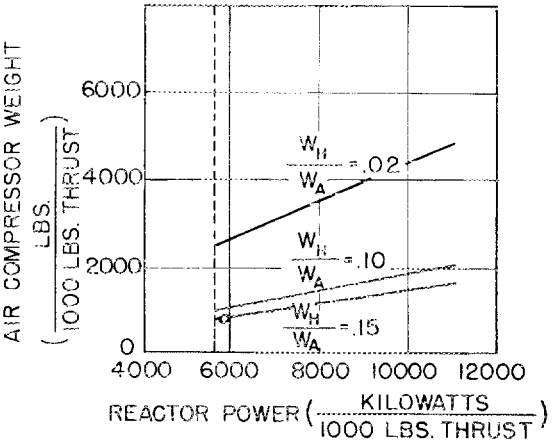


FIGURE III-61

[redacted]

[redacted]

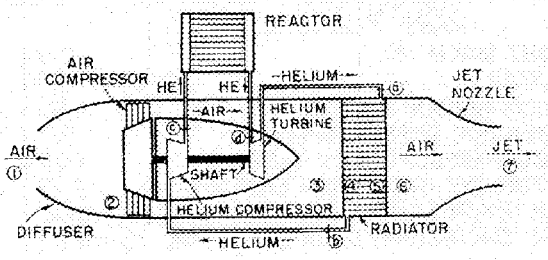
FIGURES III-62, III-63, III-64, III-65

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE II)

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO:	4
HE TURBINE PRESSURE RATIO:	3.6
HE COMPRESSOR INLET TEMPERATURE:	200F
HE TURBINE INLET TEMPERATURE:	1540 F
EFFICIENCIES: (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE
W_A = AIR FLOW RATE
P_R = REACTOR INLET PRESSURE

HELIUM CYCLE

THERMAL EFFICIENCY:	0.20
WORK RATIO:	0.21
SPECIFIC HELIUM FLOW:	13.4 LBS/HP-HR.

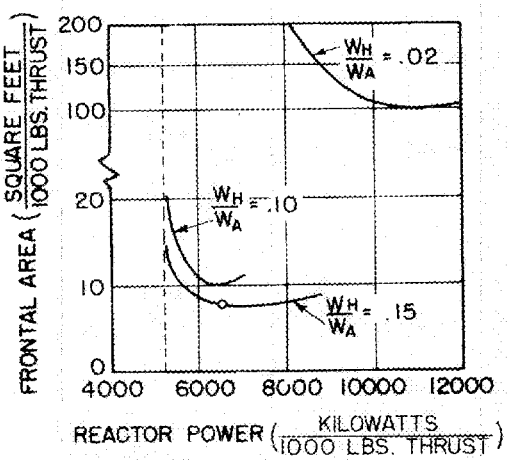


FIGURE III-62

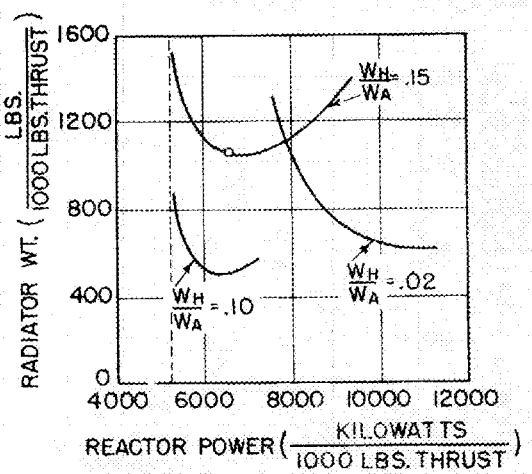


FIGURE III-63

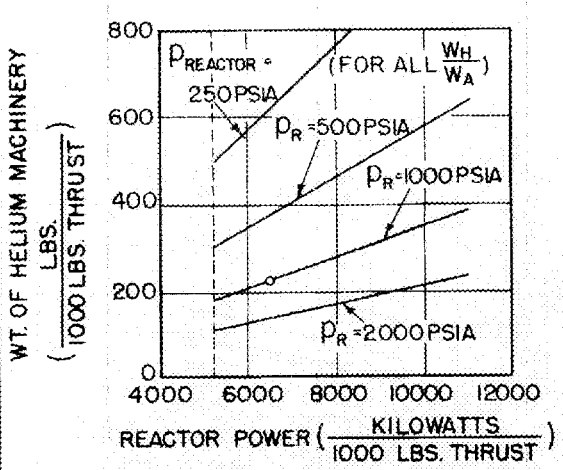


FIGURE III-64

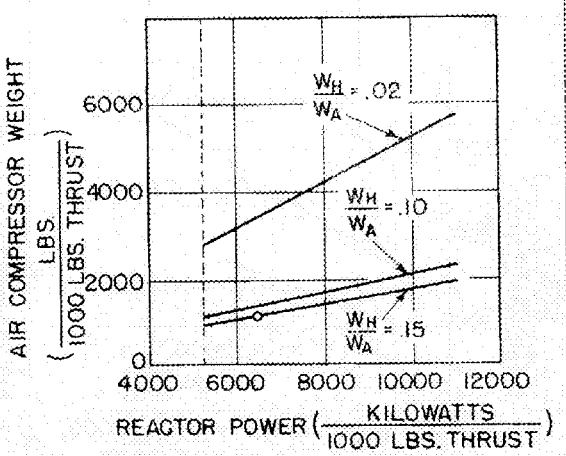


FIGURE III-65

[redacted]

[redacted]

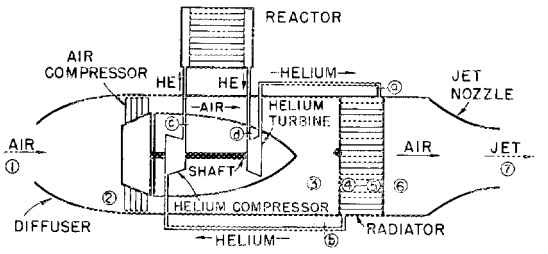
FIGURES III-66, III-67, III-68, III-69

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE III)

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO:	3.0
HE TURBINE PRESSURE RATIO:	2.7
HE COMPRESSOR INLET TEMP.:	160F.
HE TURBINE INLET TEMPERATURE:	1540F.
EFFICIENCIES: (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$
	$\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE
W_A = AIR FLOW RATE
p_R = REACTOR INLET PRESSURE

HELIUM CYCLE III

THERMAL EFFICIENCY:	0.185
WORK RATIO:	0.31
SPECIFIC HELIUM FLOW:	11.3 LBS/HP-HR.

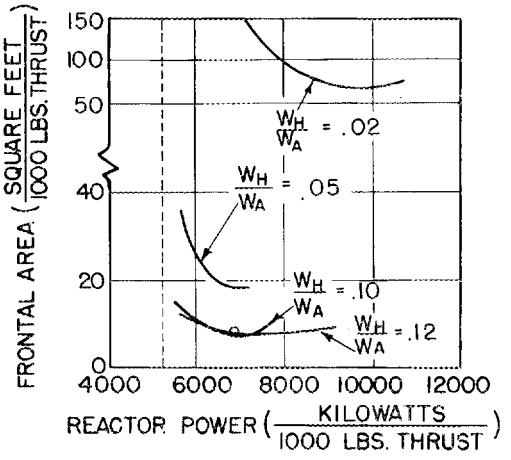


FIGURE III-66

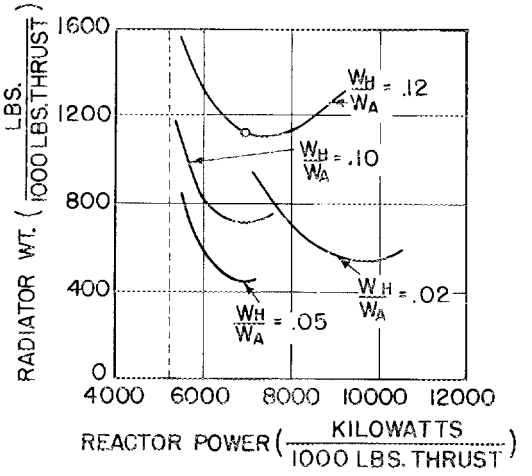


FIGURE III-67

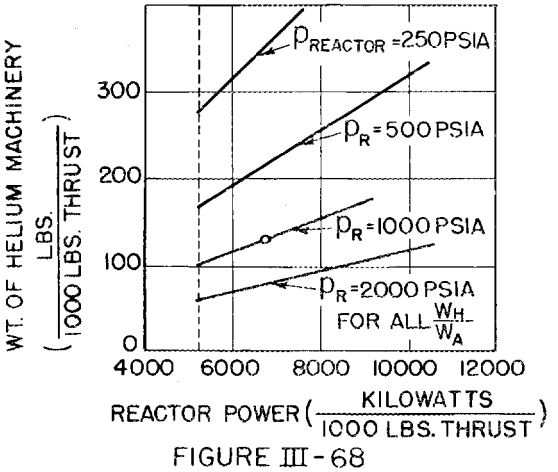


FIGURE III-68

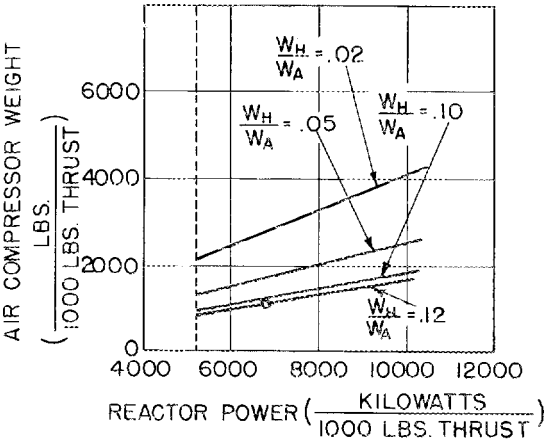


FIGURE III-69

[redacted]

[redacted]

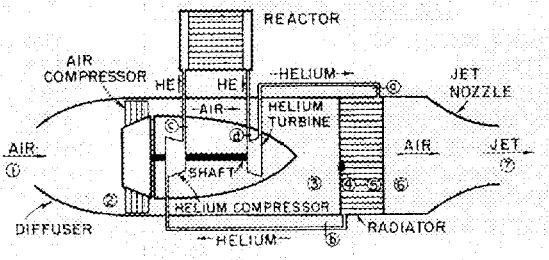
FIGURES III-70, III-71, III-72, III-73

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE IV)

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO:	3.0
HE TURBINE PRESSURE RATIO:	2.7
HE COMPRESSOR INLET TEMP.:	200 F.
HE TURBINE INLET TEMPERATURE:	1340 F.
EFFICIENCIES (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE
W_A = AIR FLOW RATE
p_R = REACTOR INLET PRESSURE

HELIUM CYCLE IV

THERMAL EFFICIENCY:	0.135
WORK RATIO:	0.183
SPECIFIC HELIUM FLOW:	21.2 LBS/HP-HR.

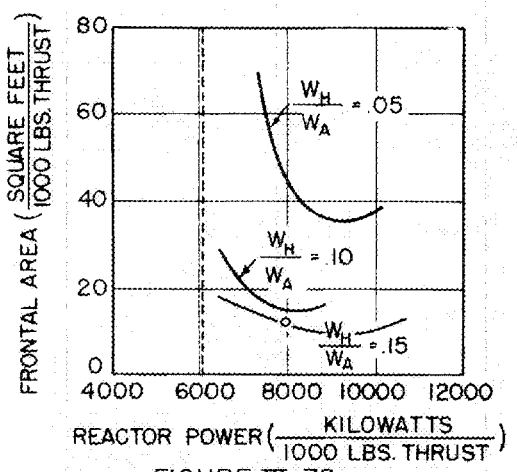


FIGURE III-70

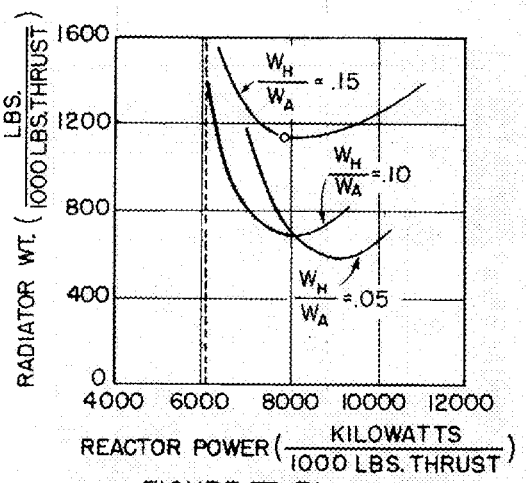


FIGURE III-71

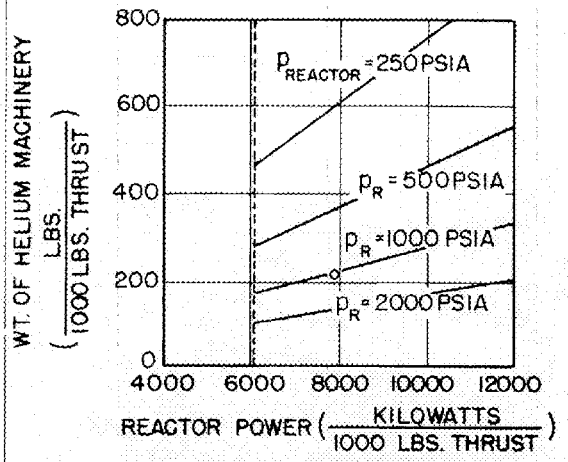


FIGURE III-72

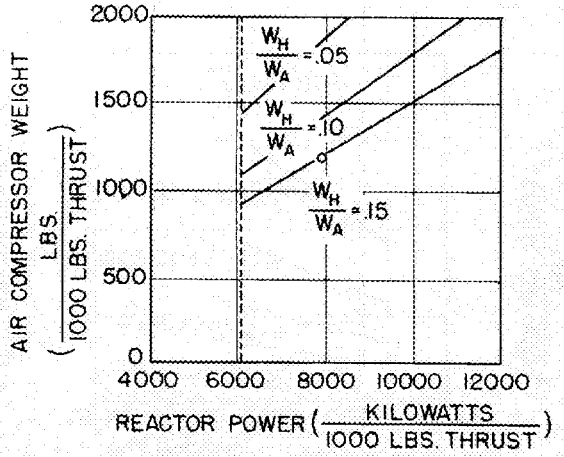


FIGURE III-73

[redacted]

[redacted]

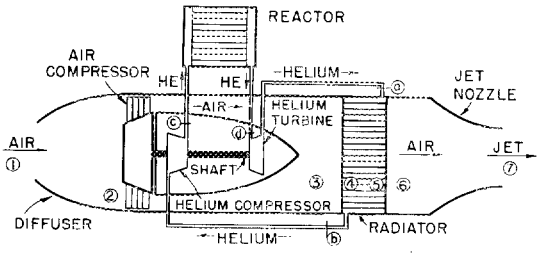
FIGURES III-74, III-75, III-76, III-77

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE V)

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO:	3.65
HE TURBINE PRESSURE RATIO:	3.28
HE COMPRESSOR INLET TEMP.:	240 F.
HE TURBINE INLET TEMPERATURE:	1800 F.
EFFICIENCIES: (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE

W_A = AIR FLOW RATE

P_R = REACTOR INLET PRESSURE

HELIUM CYCLE V

THERMAL EFFICIENCY:	0.196
WORK RATIO:	0.257
SPECIFIC HELIUM FLOW:	10.4 LBS/HP-HR.

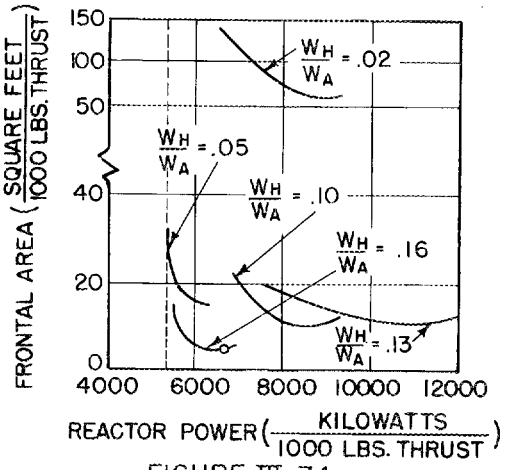


FIGURE III-74

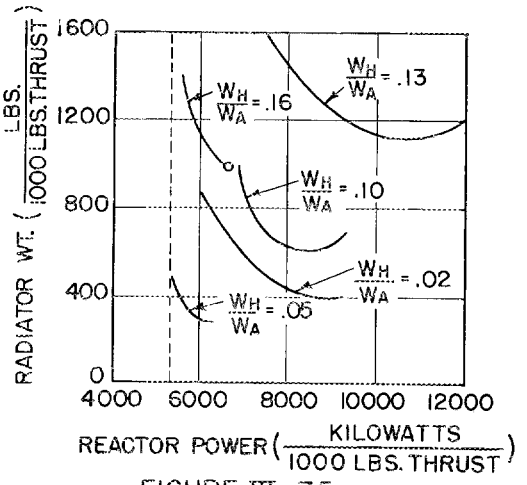


FIGURE III-75

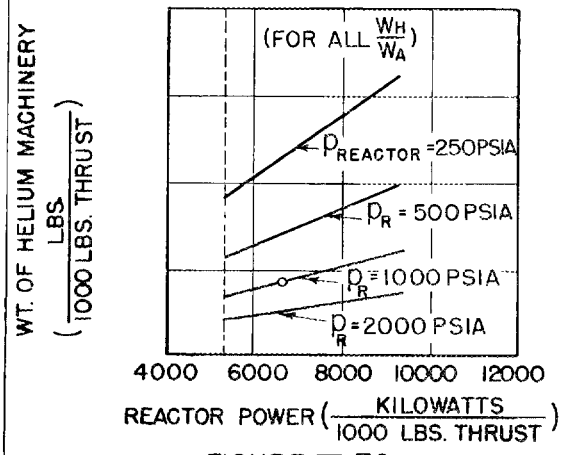


FIGURE III-76

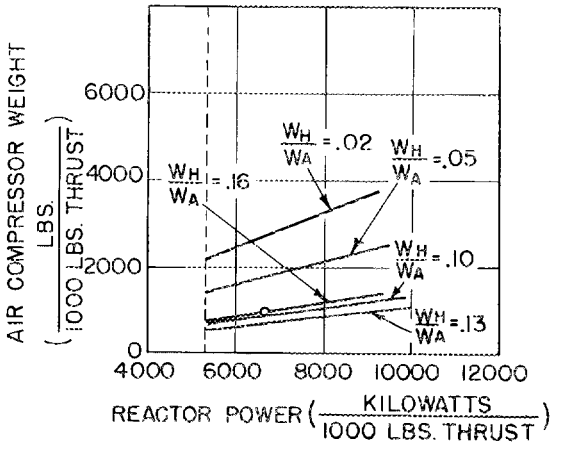


FIGURE III-77

[redacted]

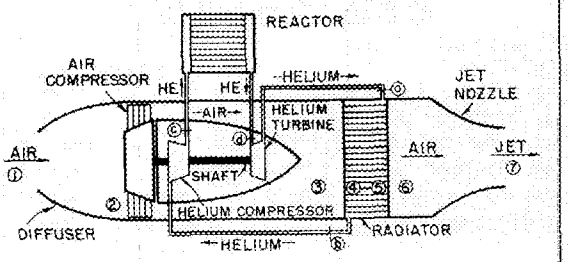
FIGURES III-78, III-79, III-80, III-81

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE VI)

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO	4.25
HE TURBINE PRESSURE RATIO:	3.82
HE COMPRESSOR INLET TEMP.:	160°F.
HE TURBINE INLET TEMPERATURE:	1800°F.
EFFICIENCIES: (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE
W_A = AIR FLOW RATE
P_R = REACTOR INLET PRESSURE

HELIUM CYCLE VI

THERMAL EFFICIENCY:	0.237
WORK RATIO:	0.308
SPECIFIC HELIUM FLOW:	8.0 LBS./HP-HR.

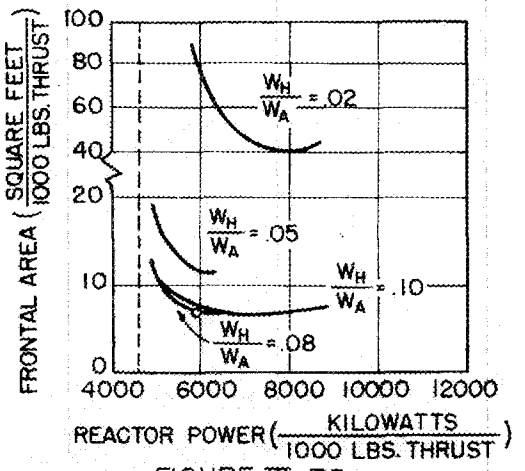


FIGURE III-78

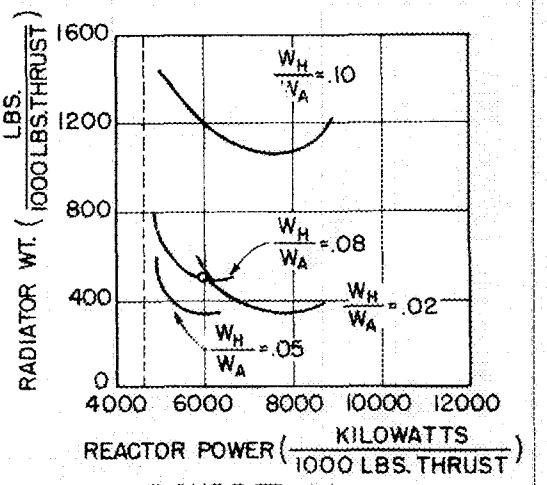


FIGURE III-79

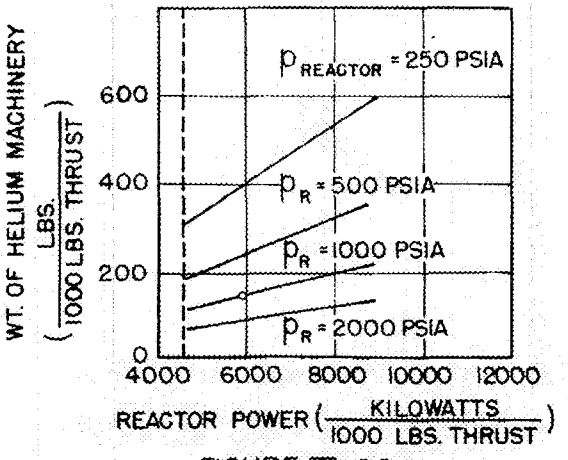


FIGURE III-80

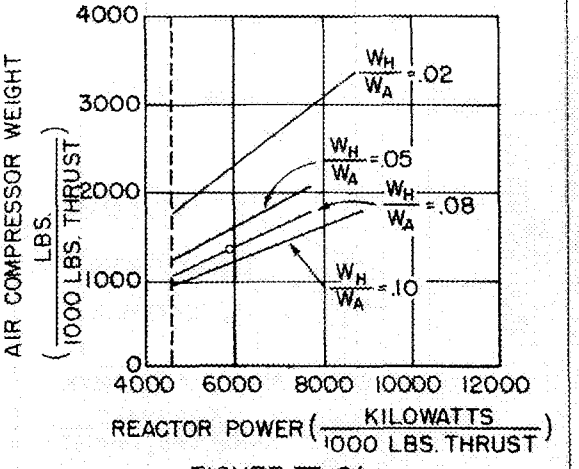


FIGURE III-81

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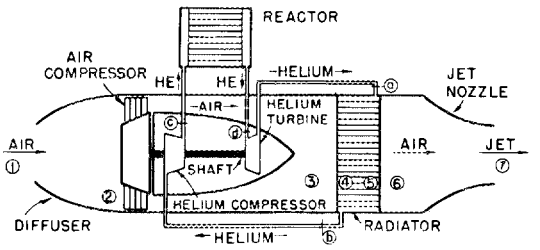
FIGURES III-82, III-83, III-84, III-85

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE VII)

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO:	4.0
HE TURBINE PRESSURE RATIO:	3.6
HE COMPRESSOR INLET TEMP.:	200 °F.
HE TURBINE INLET TEMPERATURE:	1800 °F.
EFFICIENCIES: (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE

W_A = AIR FLOW RATE

P_R = REACTOR INLET PRESSURE

HELIUM CYCLE

THERMAL EFFICIENCY: 0.216

WORK RATIO: 0.277

SPECIFIC HELIUM FLOW: 9.2 LBS./HP-HR.

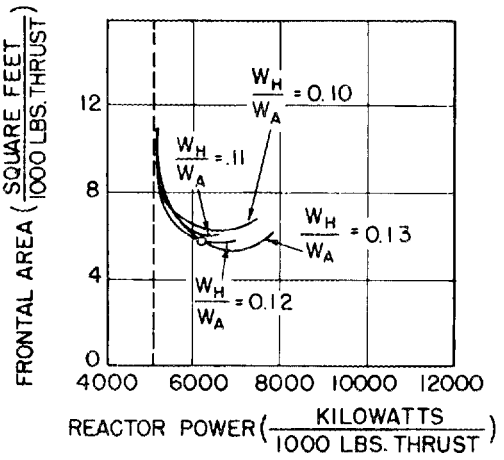


FIGURE III-82

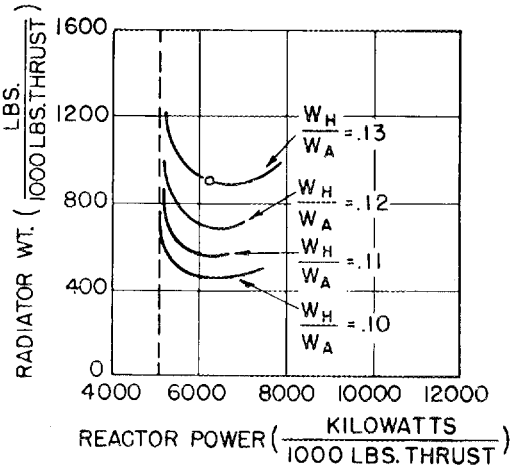


FIGURE III-83

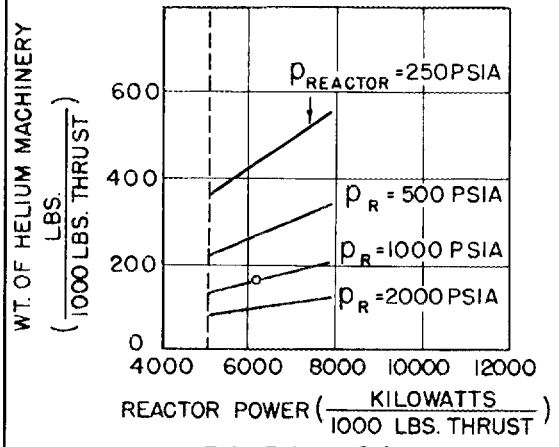


FIGURE III-84

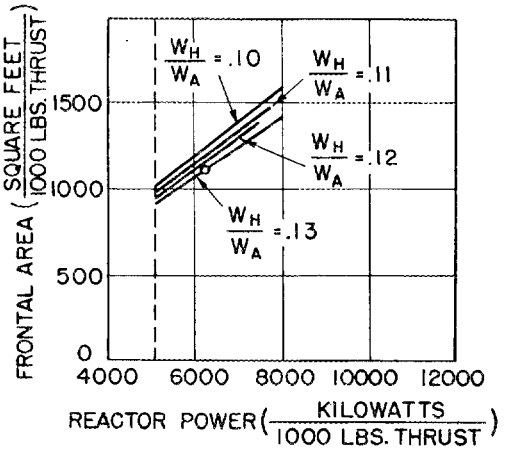


FIGURE III-85

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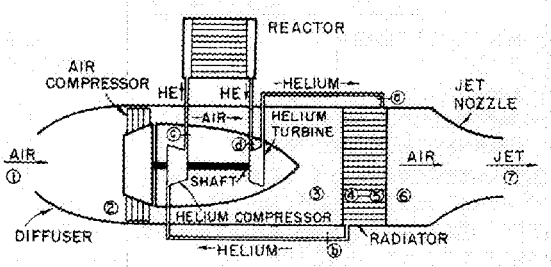
FIGURES III-86, III-87, III-88, III-89

HELIUM GAS-TURBINE COMPRESSOR-JET

CHARACTERISTICS OF PROPULSIVE DUCT (FOR HELIUM CYCLE VIII

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FEET
HE COMPRESSOR PRESSURE RATIO:	3.0
HE TUBINE PRESSURE RATIO:	2.7
HE COMPRESSOR INLET TEMP.:	200 F.
HE TURBINE INLET TEMPERATURE:	1440 F.
EFFICIENCIES. (HE CYCLE)	$\eta_C = 85\%$; $\eta_T = 88\%$
EFFICIENCIES: (AIR CYCLE)	$\eta_D = 90\%$; $\eta_C = 85\%$
	$\eta_N = 92\%$
MACH NO. AT AIR COMPRESSOR INLET:	0.5
RADIATOR TUBES: DIAMETER	1/8 IN.
WALL THICKNESS	0.004 IN.
DENSITY	490 LBS/FT.^{3}



NOMENCLATURE

W_H = HELIUM FLOW RATE
W_A = AIR FLOW RATE
P_R = REACTOR INLET PRESSURE

HELIUM CYCLE VIII

THERMAL EFFICIENCY:	15.6
WORK RATIO:	0.228
SPECIFIC HELIUM FLOW:	16.2 LBS./HP-HR.

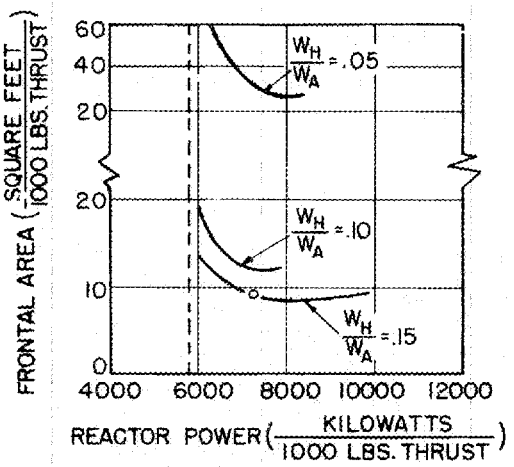


FIGURE III-86

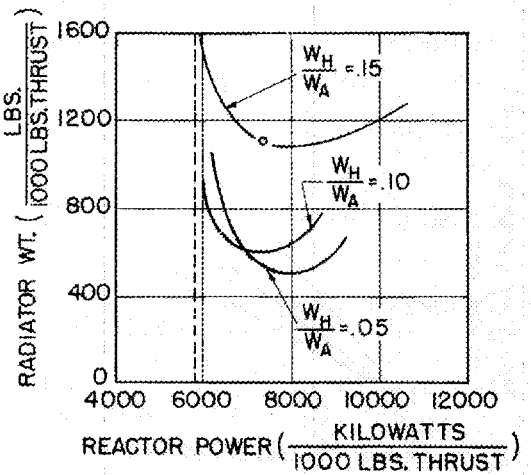


FIGURE III-87

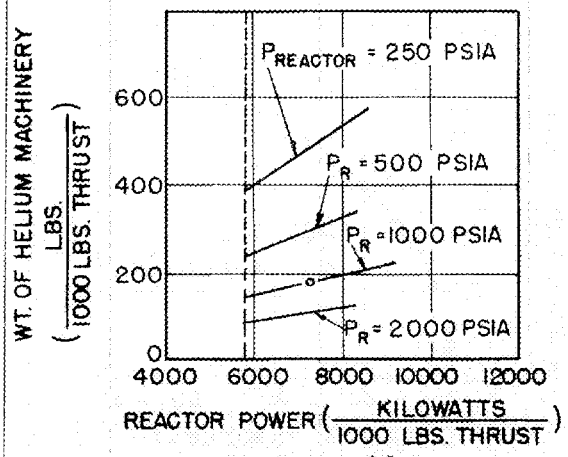


FIGURE III-88

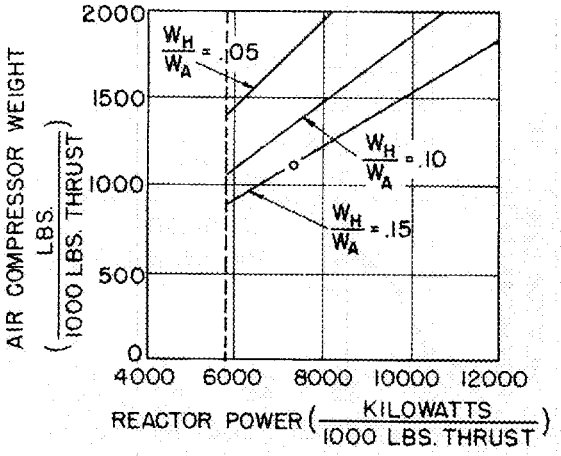


FIGURE III-89

[redacted]

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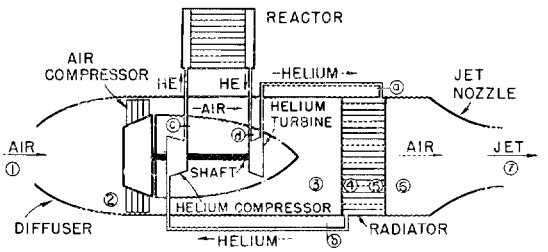
FIGURES III-90, III-91, III-92, III-93

HELIUM GAS-TURBINE COMPRESSOR-JET

EFFECT OF ALTITUDE ON COMPLETE PROPULSION SYSTEM

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FT.
HELIUM CYCLE:	I
COMPRESSOR PRESSURE RATIO:	3.0
TURBINE PRESSURE RATIO:	2.7
COMPRESSOR INLET TEMPERATURE:	200°F
TURBINE INLET TEMPERATURE:	1540°F
EFFICIENCIES: $\eta_C = 85\%$; $\eta_T = 88\%$	
PRESSURE AT REACTOR INLET:	1000 PSI
RATIO OF HELIUM TO AIR FLOW:	0.15
AIR COMPRESSOR PRESSURE RATIO:	1.99
AIR EFFICIENCIES: $\eta_D = 90\%$, $\eta_C = 85\%$, $\eta_N = 92\%$	
REACTOR FREE FLOW RATIO (FFR):	0.4
SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS.	
MAXIMUM REACTOR WALL TEMP.:	1832 °F
PRESSURE AT REACTOR INLET:	1000 PSIA



ALTITUDE (FEET)	REACTOR POWER THRUST (KW/1000 LBS.)	AIR SPECIFIC IMPULSE (LBS.-SEC./LB.)
30,000	6850	26.3
40,000	6220	29.0
50,000	6220	29.0

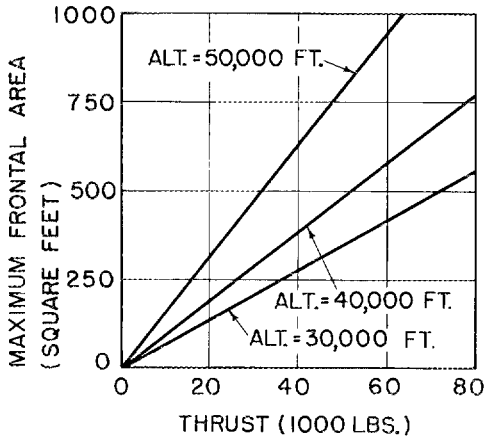


FIGURE III-90

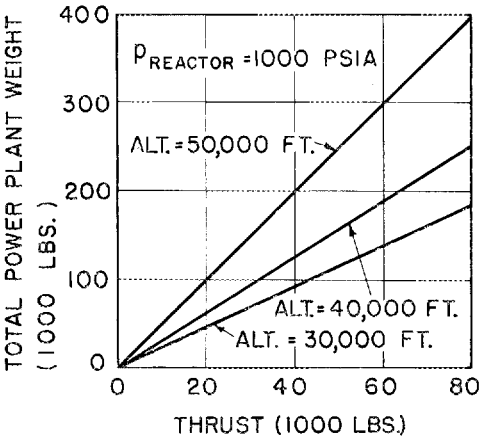


FIGURE III-91

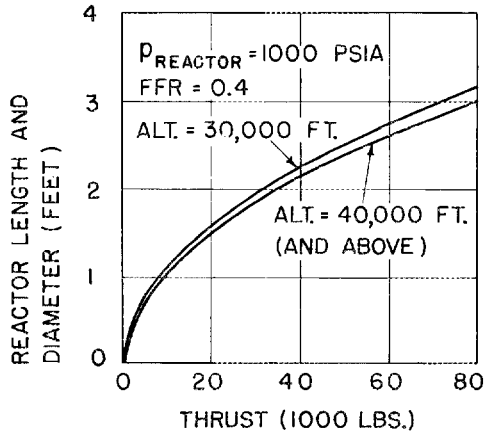


FIGURE III-92

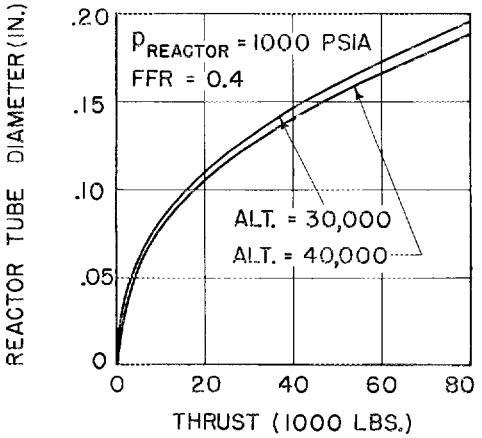


FIGURE III-93

[redacted]

FIGURES III-94, III-95, III-96, III-97

HELIUM GAS-TURBINE COMPRESSOR-JET

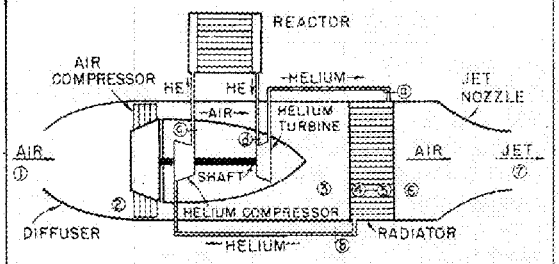
EFFECT OF TURBINE INLET TEMPERATURE ON COMPLETE PROPULSION SYSTEM

DESIGN CONDITIONS

FLIGHT MACH NO.: 0.9
ALTITUDE: 30,000 FT

HELIUM CYCLE:	I	VIII	IV	VII
HE COMP. PRESS. RATIO	3.0	3.0	3.0	4.0
HE TURBINE PRESS. RATIO	2.7	2.7	2.7	3.6
HE COMP INLET TEMP (F)	200	200	200	200
HE TURBINE INLET TEMP (F)	1540	1440	1340	1800
AIR COMP. PRESS RATIO	1.99	1.76	1.55	2.27
RATIO OF HE. TO AIR FLOW	0.15	0.15	0.15	0.13

MAX. REACTOR WALL TEMP. 1832°F
HELIUM EFFICIENCIES: $\eta_C = 85\%$; $\eta_T = 88\%$
AIR EFFICIENCIES: $\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$
REACTOR FREE-FLOW RATIO (FFR): 0.4
PRESSURE AT REACTOR INLET 1000 PSIA
SINGLE-PASS SQUARE CYLINDER REACTOR WITH HEADERS



HELIUM CYCLE	REACTOR POWER THRUST (KW / 1000 LBS.)	AIR SPECIFIC IMPULSE (LBS.-SEC./LBS.)
I	6850	26.3
VIII	7310	21.9
IV	7920	17.7
VII	6200	28.2

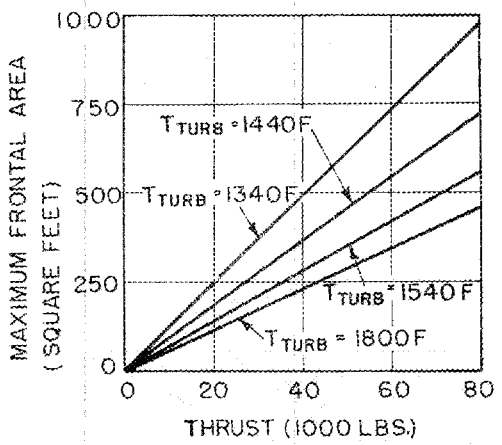


FIGURE III-94

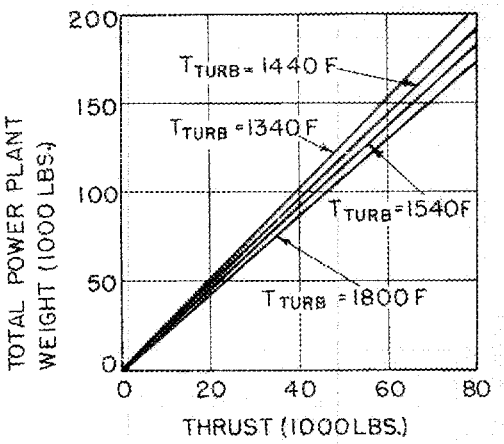


FIGURE III-95

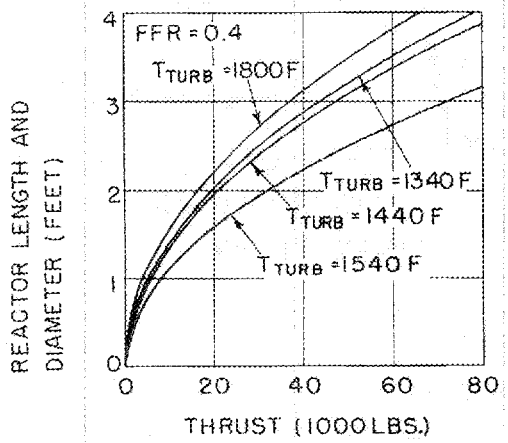


FIGURE III 96

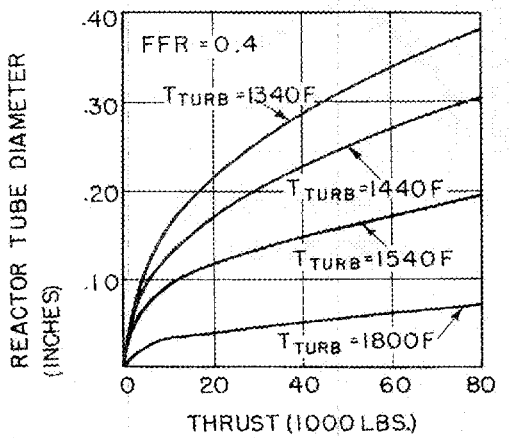


FIGURE III-97

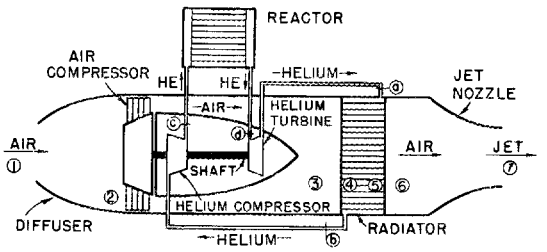
FIGURES III-98, III-99

HELIUM GAS-TURBINE COMPRESSOR-JET

EFFECT OF MAXIMUM REACTOR WALL TEMPERATURE ON COMPLETE PROPULSION SYSTEM

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FT.
HELIUM CYCLE:	I
COMPRESSOR PRESSURE RATIO:	3.0
TURBINE PRESSURE RATIO:	2.7
COMPRESSOR INLET TEMPERATURE:	200°F
TURBINE INLET TEMPERATURE:	1540°F
EFFICIENCIES: $\eta_C = 85\%$; $\eta_T = 88\%$	
PRESSURE AT REACTOR INLET:	1000 PSI
RATIO OF HELIUM TO AIR FLOW:	0.15
AIR COMPRESSOR PRESSURE RATIO:	1.99
AIR EFFICIENCIES: $\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_N = 92\%$	
REACTOR FREE FLOW RATIO (FFR):	0.4
SINGLE PASS SQUARE CYLINDER REACTOR WITH HEADERS.	
MAXIMUM REACTOR WALL TEMP. ($T_{W_{MAX.}}$):	VARIABLE
$\frac{\text{REACTOR POWER}}{\text{THRUST}}$:	6850 KW/1000 LBS
AIR SPECIFIC IMPULSE:	26.3 LBS. SEC./LB



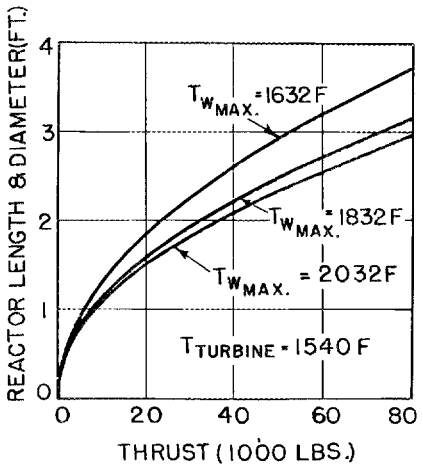


FIGURE III-98

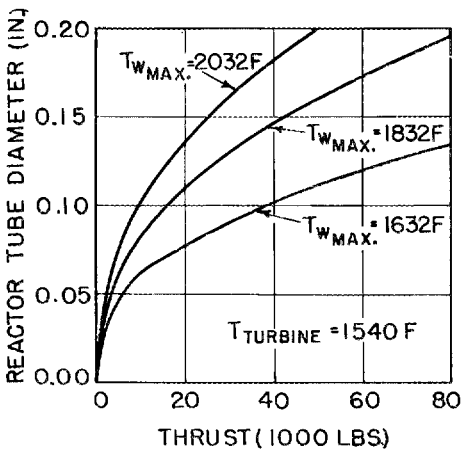


FIGURE III-99

[redacted]

FIGURES III-100, III-101

BISMUTH TURBOJET

CHARACTERISTICS OF BISMUTH COOLED REACTOR

DESIGN CONDITIONS

SINGLE-PASS SQUARE CYLINDER REACTOR, BASED ON HEAT TRANSFER ONLY

SINUSOIDAL HEAT FLUX DISTRIBUTION WITH 20% OF MAXIMUM POWER AT ENDS

NOMENCLATURE

T_d = BISMUTH TEMPERATURE LEAVING REACTOR

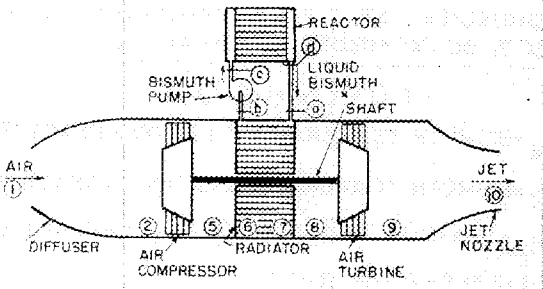
T_c = BISMUTH TEMPERATURE ENTERING REACTOR

$T_{W_{MAX.}}$ = MAXIMUM REACTOR WALL TEMPERATURE

FFR = FREE-FLOW RATIO

d = TUBE DIAMETER

$$\gamma_{WM} = \frac{T_{W_{MAX.}} - T_c}{T_d - T_c}$$



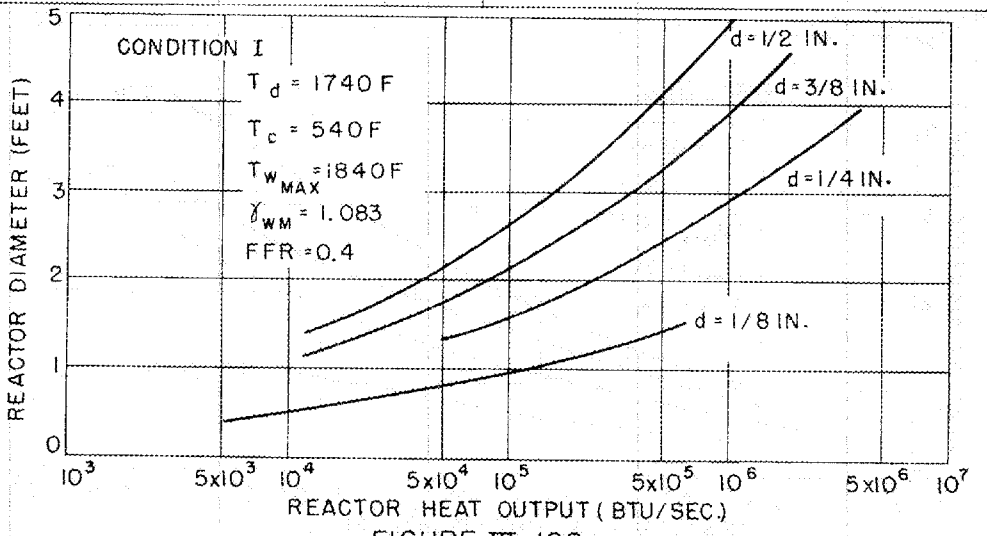


FIGURE III-100

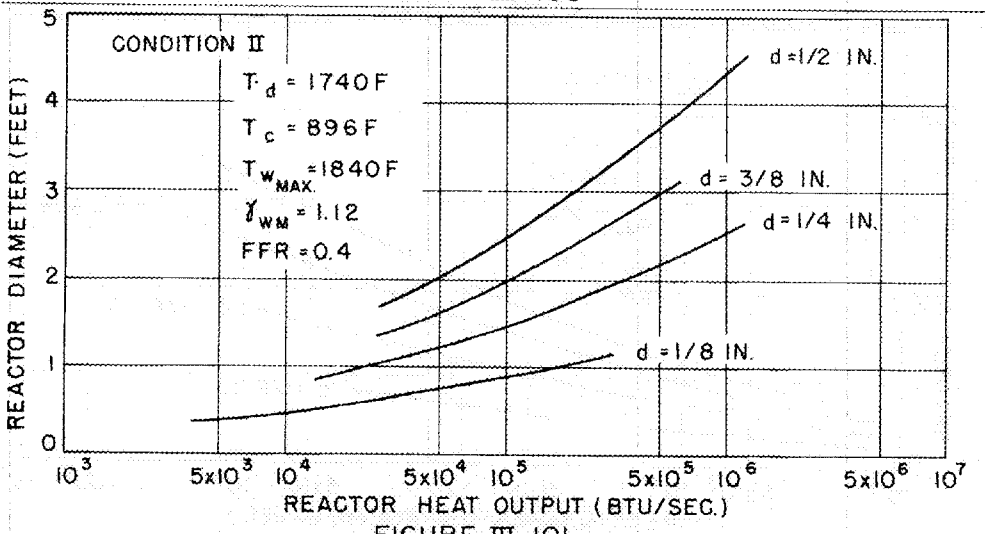


FIGURE III-101

[redacted]

[redacted]

FIGURES III-102, III-103

BISMUTH TURBOJET

CHARACTERISTICS OF BISMUTH-COOLED REACTOR

DESIGN CONDITIONS

SINGLE-PASS SQUARE CYLINDER REACTOR, BASED ON HEAT TRANSFER ONLY

SINUSOIDAL HEAT FLUX DISTRIBUTION WITH 20% OF MAXIMUM POWER AT ENDS

NOMENCLATURE

T_d = BISMUTH TEMPERATURE LEAVING REACTOR

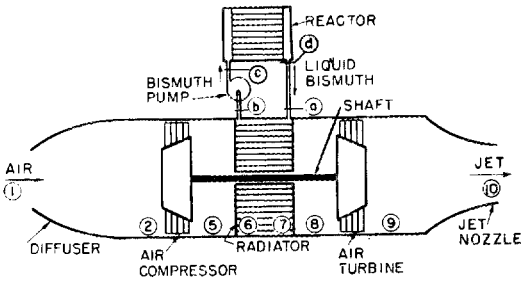
T_c = BISMUTH TEMPERATURE ENTERING REACTOR

$T_{W_{MAX.}}$ = MAXIMUM REACTOR WALL TEMPERATURE

FFR = FREE-FLOW RATIO

d = TUBE DIAMETER

$$\gamma_{WM} = \frac{T_{W_{MAX.}} - T_c}{T_d - T_c}$$



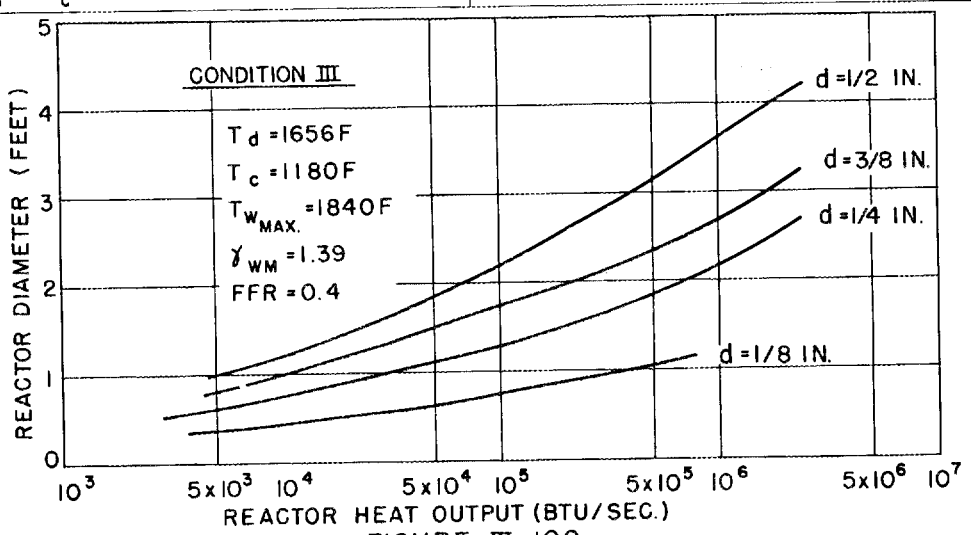


FIGURE III-102

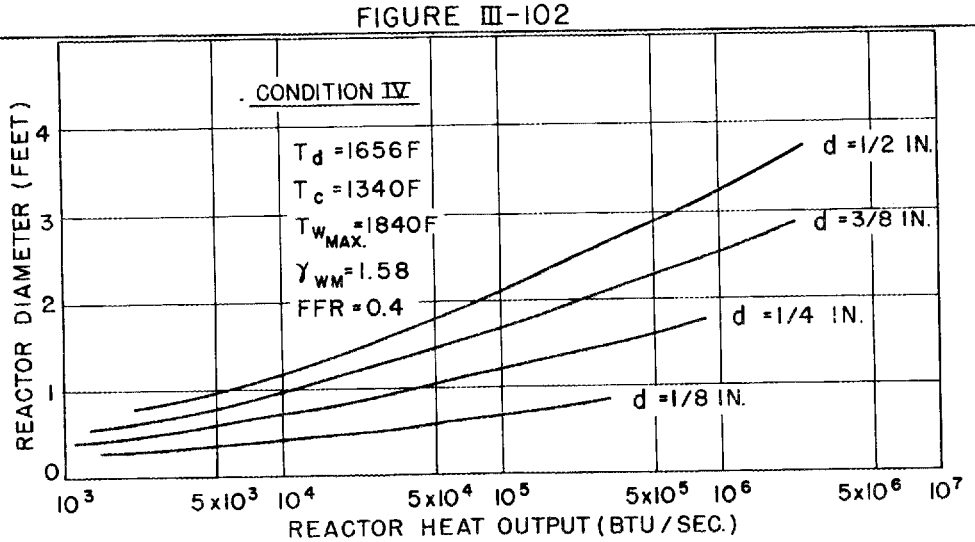


FIGURE III-103

[redacted]

[redacted]

FIGURE III-104

BISMUTH TURBOJET

CHARACTERISTICS OF BISMUTH-COOLED REACTOR

DESIGN CONDITIONS

SINGLE-PASS SQUARE CYLINDER REACTOR, BASED ON HEAT TRANSFER ONLY

SINUSOIDAL HEAT FLUX DISTRIBUTION WITH 20 % OF MAXIMUM POWER AT ENDS

NOMENCLATURE

T_d = BISMUTH TEMPERATURE LEAVING REACTOR

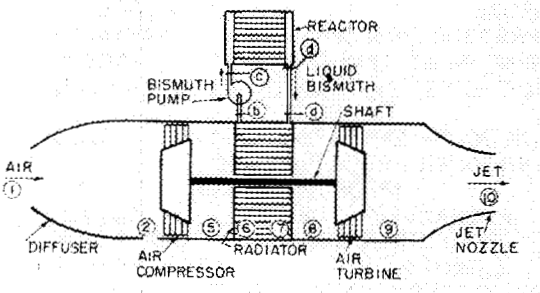
T_c = BISMUTH TEMPERATURE ENTERING REACTOR

$T_{w_{MAX.}}$ = MAXIMUM REACTOR WALL TEMPERATURE

FFR = FREE-FLOW RATIO

d = TUBE DIAMETER

$$\gamma_{WM} = \frac{T_{w_{MAX.}} - T_c}{T_d - T_c}$$



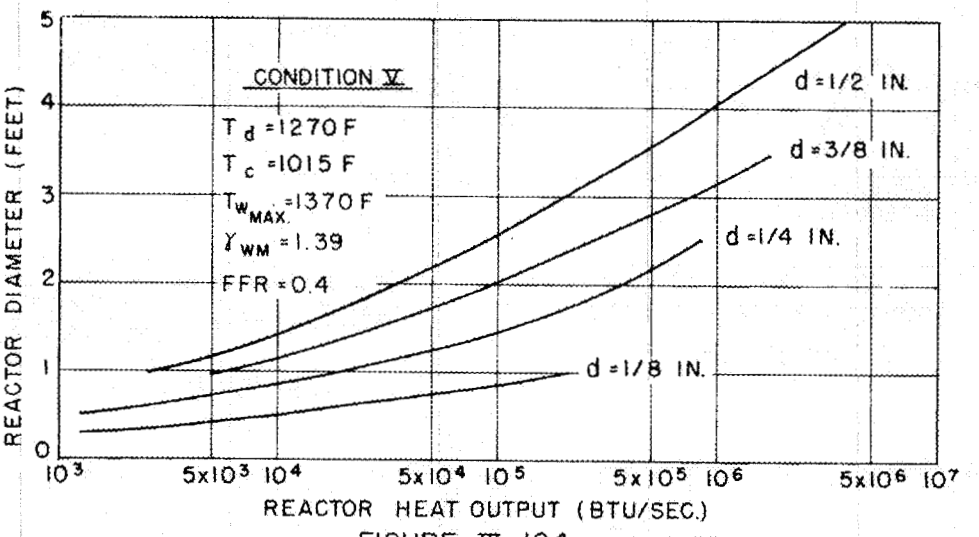


FIGURE III-104

[redacted]

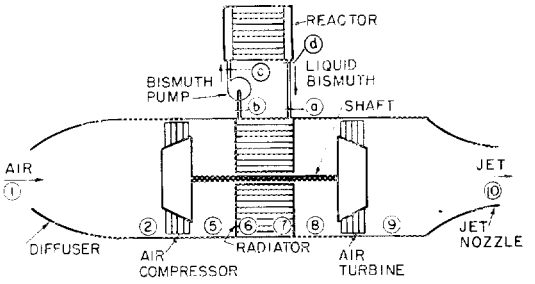
FIGURES III-105, III-106, III-107, III-108

BISMUTH TURBOJET

EFFECT OF COMPRESSOR PRESSURE RATIO ON AIR CYCLE

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	30,000 FT.
COMPRESSOR PRESSURE RATIO:	VARIABLE
TURBINE INLET TEMPERATURE:	1500°F
EFFICIENCIES:	$\eta_D = 90\%$; $\eta_C = 83.5\%$; $\eta_T = 87\%$; $\eta_N = 90\%$
INTERCOOLER EFFECTIVENESS:	0.50
INTERCOOLER PRESSURE DROP (PROPULSIVE AIR):	5%
HEAT EXCHANGER: TUBE DIA.	1/8 INCH
WALL THICKNESS	0.004 INCH
DENSITY	490 LBS./FT.^{3}
BISMUTH TEMP. ENTERING HEAT EXCHANGER:	1656°F
BISMUTH TEMP. LEAVING HEAT EXCHANGER:	1180°F



NOMENCLATURE

CPR = COMPRESSOR PRESSURE RATIO

* = INDICATES THAT INTERCOOLER IS USED

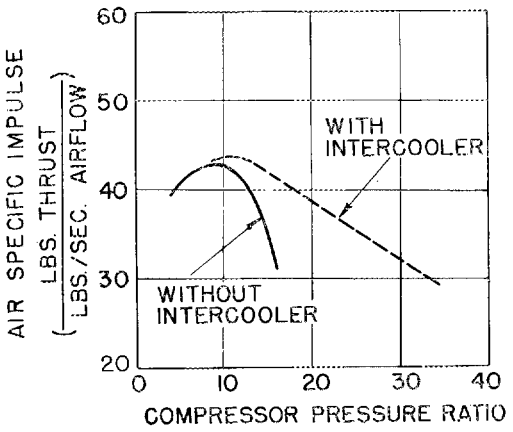


FIGURE III-105

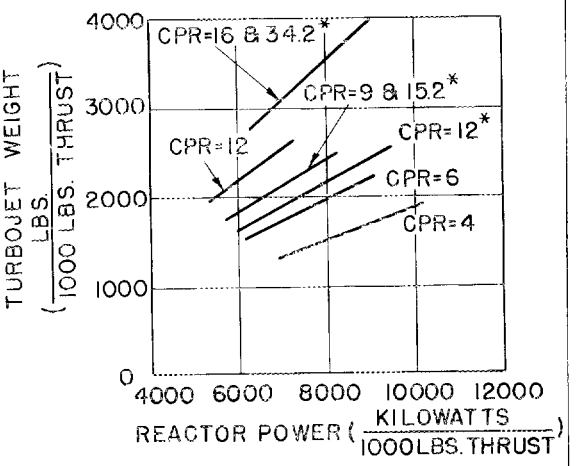


FIGURE III-106

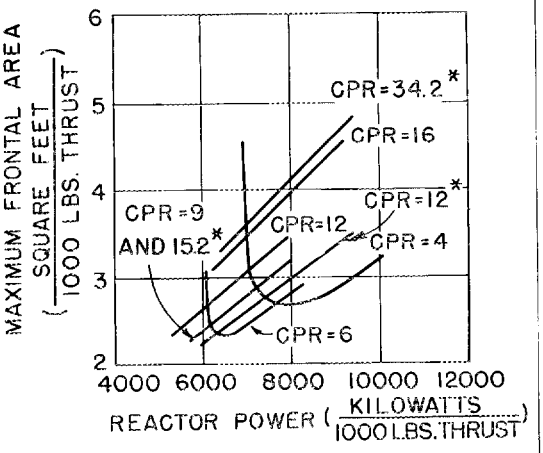


FIGURE III-107

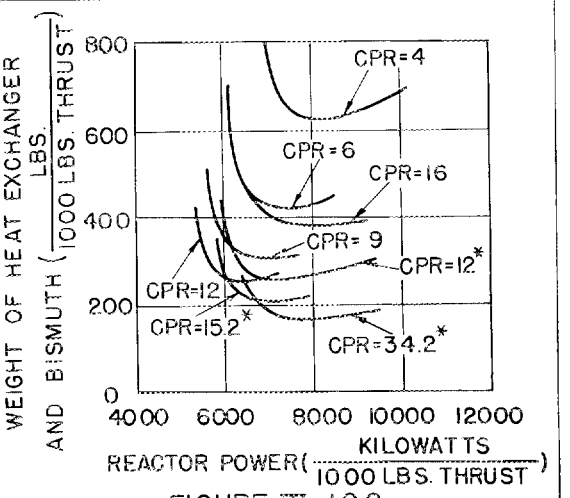


FIGURE III-108

[redacted]

FIGURES III-109, III-110

BISMUTH TURBOJET

EFFECT OF TURBINE INLET TEMPERATURE ON AIR CYCLE

DESIGN CONDITIONS

FLIGHT MACH NO.: 0.9

ALTITUDE: 30,000 FT.

COMPRESSOR PRESSURE RATIO: 6.0

TURBINE INLET TEMPERATURE: VARIABLE

EFFICIENCIES: η_D=90%; η_C=83.5%; η_T=87%; η_N=90%

NO INTERCOOLER

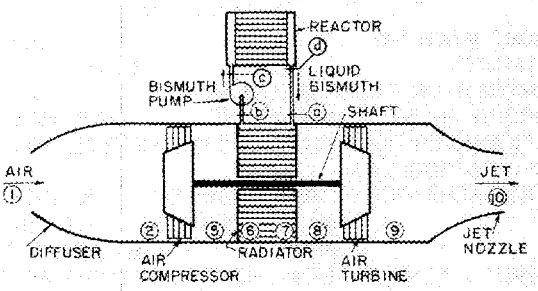
HEAT EXCHANGER: TUBE DIAMETER 1/8 IN.

WALL THICKNESS 0.004 IN.

DENSITY 490 LBS./FT.^{3}

BISMUTH TEMP. ENTERING HEAT EXCHANGER: 1656°F.

BISMUTH TEMP. LEAVING HEAT EXCHANGER: 1180°F.



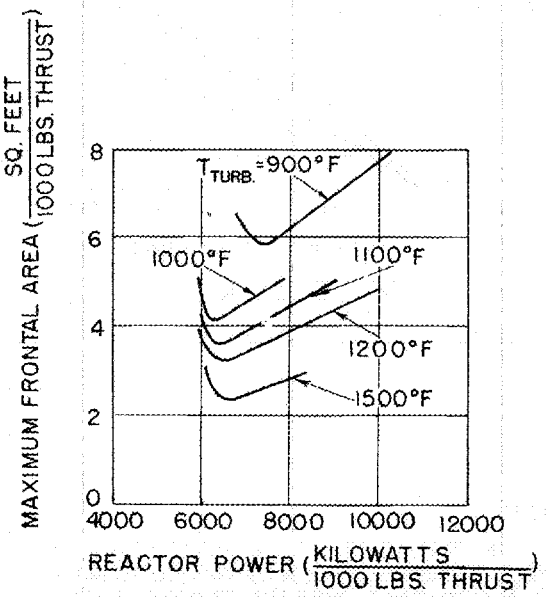


FIGURE III-109

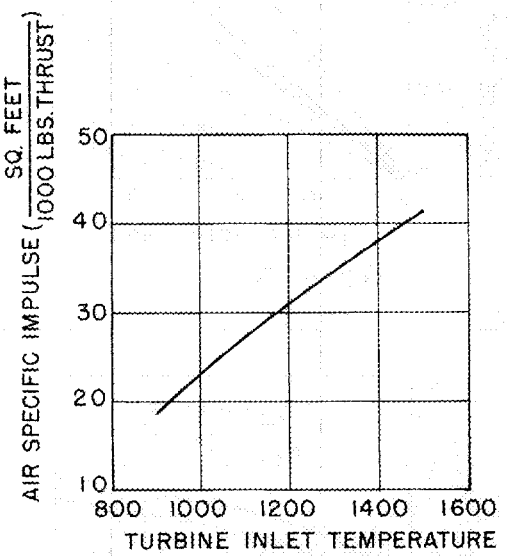


FIGURE III-110

(CORRESPONDING TO MINIMUM POINTS ON FIGURE III-109)

[redacted]

[redacted]

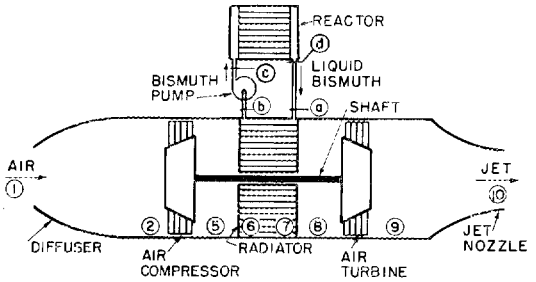
FIGURES III-111, III-112, III-113

BISMUTH TURBOJET

EFFECT OF TURBINE INLET TEMPERATURE ON COMPLETE POWER PLANT

DESIGN CONDITIONS

FLIGHT MACH NO.:	0.9
ALTITUDE:	30,000 FT.
COMPRESSOR PRESSURE RATIO:	6.0
TURBINE INLET TEMPERATURE:	VARIABLE
EFFICIENCIES $\eta_D = 90\%$; $\eta_C = 83.5\%$ $\eta_T = 87\%$; $\eta_N = 90\%$	
NO INTERCOOLER:	
HEAT EXCHANGER: TUBE DIAMETER	1/8 INCH
WALL THICKNESS	.004 IN.
DENSITY	490 LBS/FT^3
BISMUTH TEMP. ENTERING HEAT EXCHANGER:	1656 F
BISMUTH TEMP. LEAVING HEAT EXCHANGER:	1180 F
REACTOR FREE-FLOW RATIO:	0.4
REACTOR TUBE DIAMETER:	3/8 INCH



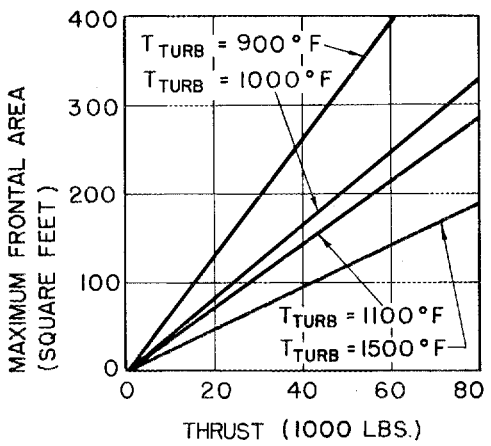


FIGURE III-111

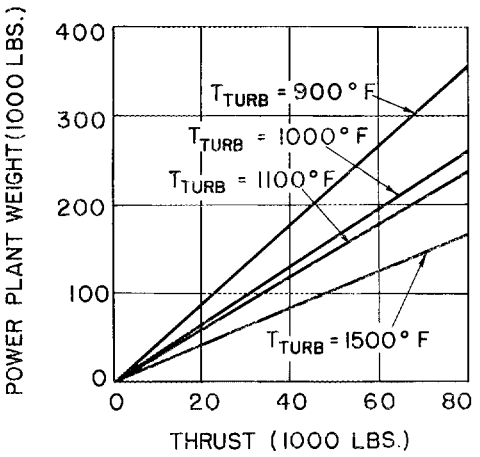


FIGURE III-112

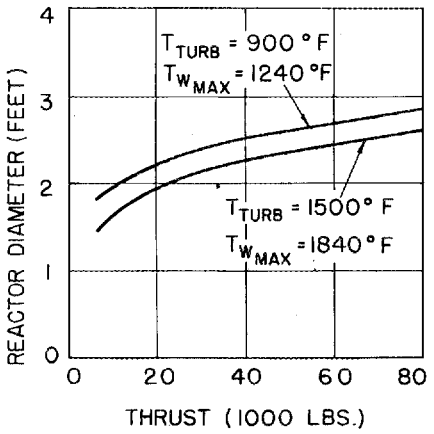


FIGURE III-113

NOTE:

IN FIGURE III-113, MAXIMUM REACTOR WALL TEMPERATURE WAS VARIED TO MAINTAIN CONSTANT AVERAGE TEMPERATURE DIFFERENCE BETWEEN WALL AND COOLANT. THUS, THE DIFFERENCE BETWEEN THE TWO CURVES IS DUE ENTIRELY TO CHANGE OF CYCLE EFFICIENCY WITH TURBINE INLET TEMPERATURE.

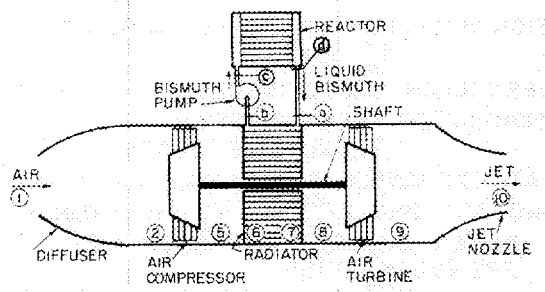
[redacted]

[redacted]

FIGURES III-114, III-115, III-116
BISMUTH TURBOJET
EFFECT OF ALTITUDE ON COMPLETE POWER PLANT

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
ALTITUDE:	VARIABLE
COMPRESSOR PRESSURE RATIO:	6.0
TURBINE INLET TEMPERATURE:	1500°F
EFFICIENCIES: $\eta_D = 90\%$; $\eta_C = 83.5\%$; $\eta_T = 87\%$; $\eta_N = 90\%$	
NO INTERCOOLER	
HEAT EXCHANGER: TUBE DIAMETER	1/8 INCH
WALL THICKNESS	0.004 INCH
DENSITY	490 LBS/FT^3
BISMUTH TEMPERATURE ENTERING HEAT EXCHANGER:	1656°F
BISMUTH TEMPERATURE LEAVING HEAT EXCHANGER:	1180°F
REACTOR FREE-FLOW RATIO:	0.4
REACTOR TUBE DIAMETER:	3/8 INCH
REACTOR MAXIMUM WALL TEMP.:	1840°F



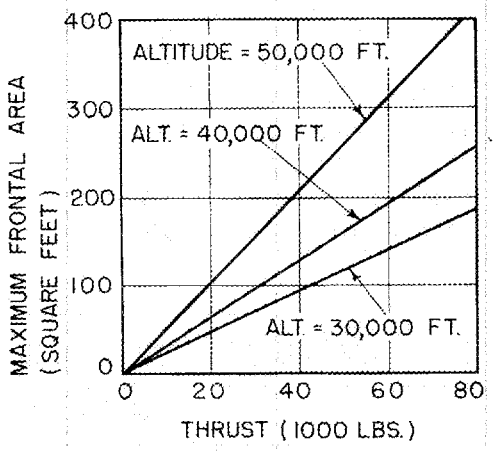


FIGURE III-114

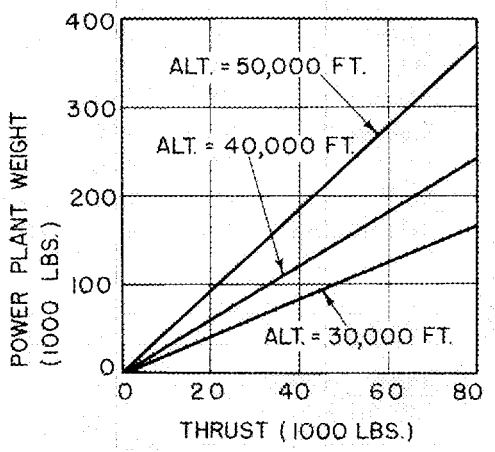


FIGURE III-115

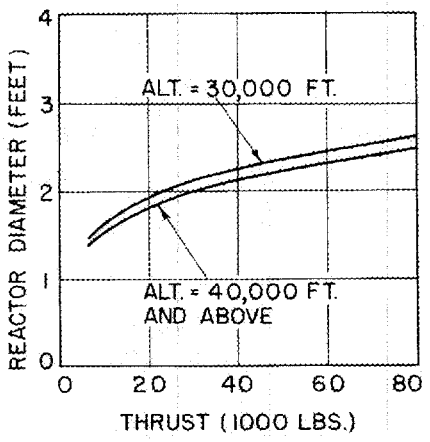


FIGURE III-116

[redacted]

[redacted]

FIGURES III-117, III-118

BISMUTH TURBOJET

EFFECT OF BISMUTH PUMPING POWER ON REACTOR DIAMETER

DESIGN CONDITIONS

FLIGHT MACH NUMBER: 0.9

ALTITUDE: 30,000 FT.

EFFICIENCIES: $\eta_D = 90\%$, $\eta_C = 83.5\%$, $\eta_T = 87\%$, $\eta_N = 90\%$

INTERCOOLER EFFECTIVENESS: 0.50

INTERCOOLER PRESSURE DROP: (PROPULSIVE DUCT) 5%

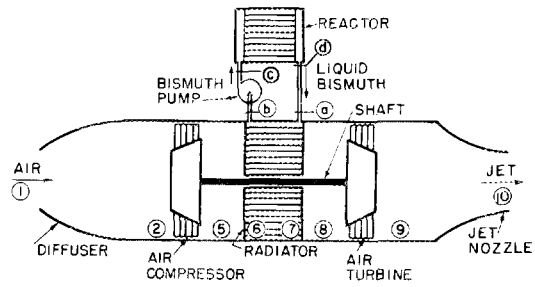
HEAT EXCHANGER TUBE DIAMETER: 1/8 INCH

REACTOR: SINGLE-PASS SQUARE CYLINDER, BASED ON HEAT TRANSFER AND PRESSURE DROP

SINUSOIDAL HEAT FLUX DISTRIBUTION WITH 20% OF MAXIMUM POWER AT ENDS

REACTOR FREE-FLOW RATIO (FFR): 0.4

REACTOR TUBE DIAMETER: VARIABLE



NOMENCLATURE

T_d = BISMUTH TEMP. LEAVING REACTOR

T_c = BISMUTH TEMP. ENTERING REACTOR

$T_{W_{MAX.}}$ = MAXIMUM REACTOR WALL TEMP.

$T_{TURBINE}$ = TURBINE INLET TEMP.

CPR = COMPRESSOR PRESSURE RATIO

CASE	A	B	C	D	E	F	G	H
CPR	6	6	6	6	6	6	6	16
$T_{W_{MAX.}}$, °F	1840	1840	1840	1840	1370	1370	1370	1840
$T_{TURBINE}$, °F	1500	1500	1500	1500	1200	1100	1000	1500
T_d, °F	1656	1740	1740	1656	1270	1270	1270	1656
T_c, °F	1180	896	540	1340	1015	1015	1015	1180
INTERCOOLER	NO	NO	NO	NO	NO	NO	NO	YES

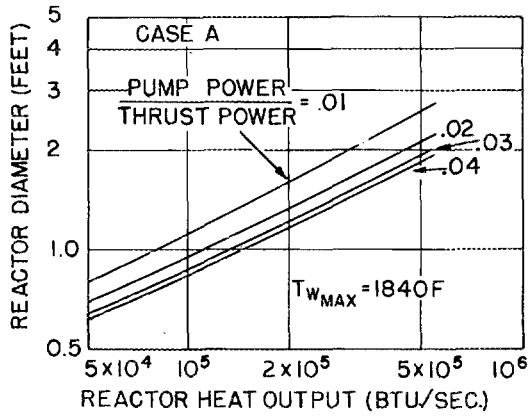


FIGURE III-117

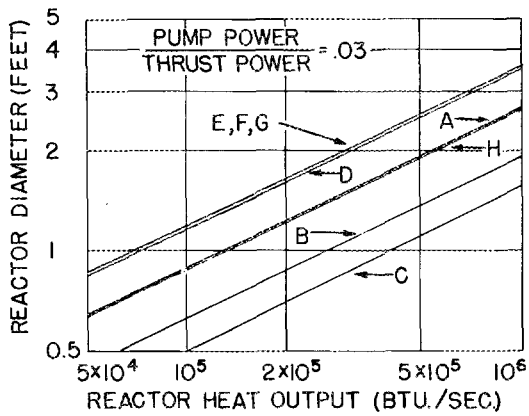


FIGURE III-118

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[redacted]

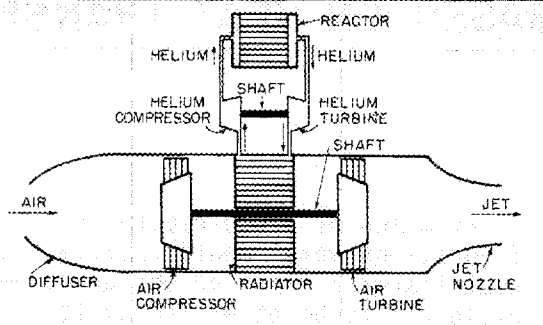
FIGURE III-119, III-120, III-121

HELIUM TURBOJET

EFFECT OF ALTITUDE ON COMPLETE POWER PLANT

DESIGN CONDITIONS

FLIGHT MACH NO:	0.9
ALTITUDE:	VARIABLE
HELIUM COMPRESSOR PRESSURE RATIO:	1.3
HELIUM TURBINE PRESSURE RATIO:	1.17
TURBINE INLET TEMPERATURE:	1800°F
HELIUM EFFICIENCIES: $\eta_C = 85\%$; $\eta_T = 88\%$	
AIR EFFICIENCIES: $\eta_D = 90\%$; $\eta_C = 85\%$; $\eta_T = 88\%$; $\eta_N = 90\%$	
AIR COMPRESSOR RATIO:	6.0
AIR TURBINE INLET TEMPERATURE:	1448°F
HEAT EXCHANGER: TUBE DIAMETER	1/8 IN.
WALL THICKNESS:	0.004 IN.
HELIUM TEMPERATURE ENTERING HEAT EXCHANGER:	1680°F
HELIUM TEMPERATURE LEAVING HEAT EXCHANGER:	470°F
HELIUM TEMPERATURE ENTERING RATIO:	590°F
REACTOR FREE FLOW RATIO:	0.4
REACTOR INLET PRESSURE:	1000 PSI
REACTOR MAXIMUM WALL TEMP:	2000°F



ALTITUDE (FEET)	HELIUM FLOW RATE ($\frac{\text{LBS. SEC.}}{\text{1000 LBS.}}$)	AIR FLOW RATE ($\frac{\text{LBS. SEC.}}{\text{1000 LBS.}}$)
30,000	3.88	23.2
40,000	3.52	21.1
50,000	3.52	21.1

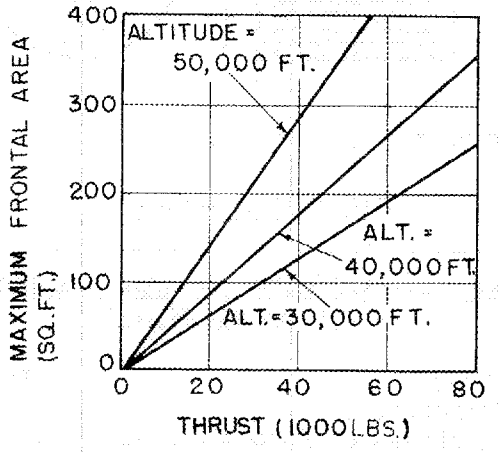


FIGURE III-119

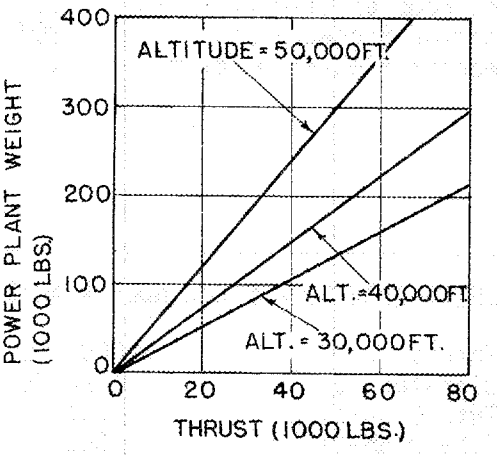


FIGURE III-120

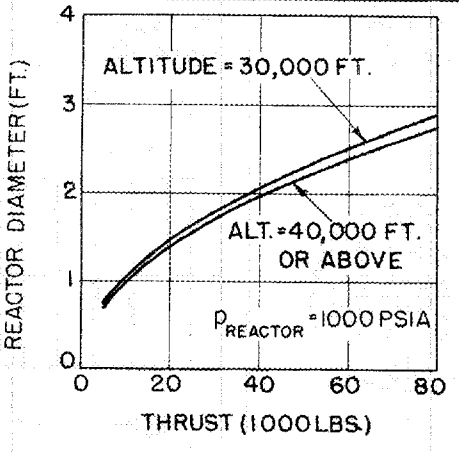


FIGURE III-121

[redacted]

[redacted]

FIGURE III-122, III-123, -III-124
EXAMPLES OF POSSIBLE REACTOR CONFIGURATIONS BASED ON HEAT TRANSFER CONSIDERATIONS ONLY

Examples of three reactor configurations which would provide the heat transfer required for an open-cycle air turbojet at the following design conditions:

Flight Mach Number	0.9
Altitude	30,000 ft.
Turbine Inlet Temperature	1800°F
Max. Reactor Wall Temp.	2500°F
Efficiencies: η_D =.90; η_C =.82; η_T =.87; η_N =.90	
Compressor Pressure Ratio	40:1
Intercooler Effectiveness	0.50
Reactor Pressure Drop	30%
Reactor Free-Flow Ratio	0.40

SINGLE-PASS SQUARE CYLINDER REACTOR

Thrust, F	20000 lb.
Diameter, D	2.54 ft.
Tube Length, L	2.03 ft.
Tube Diameter, d	0.20 in.
End Header Width, h	3 in.
Minimum Wall Thickness	0.08 in.
Number of Tubes	9300

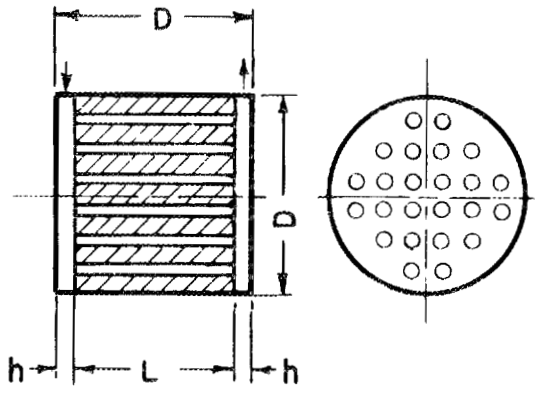


FIGURE III-122

SPLIT-FLOW SQUARE CYLINDER REACTOR

Thrust, F	20000 lb.
Diameter, D	1.80 ft.
Tube Length, L	6.5 in.
Tube Diameter, d	0.066 in.
End Header Width, h	2.15 in.
Center Gap Width, S	4.3 in.
Minimum Wall Thickness	0.026 in.
Number of Tubes	43,000

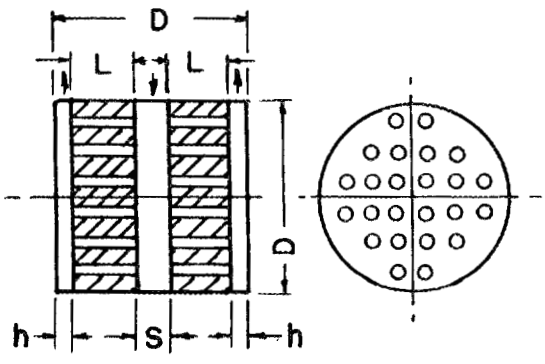


FIGURE III-123

MULTI-PASS SQUARE CYLINDER REACTOR (FIVE POROUS PLATES)

Thrust, F	20000 lb.
Diameter, D	1.14 ft.
Thickness of Porous Plates	0.02 in.
Equivalent Diameter of Pores	0.001 in.
End Header Width, h	1.4 in.
Plate Spacing, S	2.7 in.

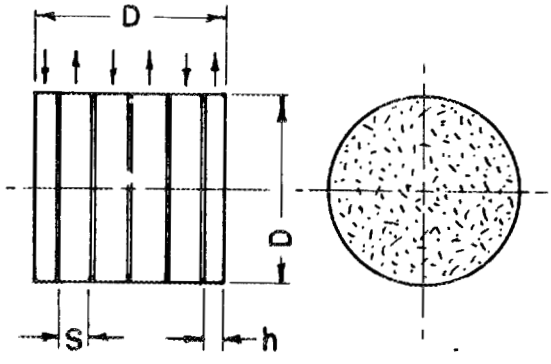


FIGURE III-124

[redacted]

[redacted]

Chapter IV

PART A - THE REACTOR

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Chapter IV

THE REACTOR AND THE SHIELD

PART A -- THE REACTOR

INTRODUCTION

A nuclear reactor for aircraft service must deliver large amounts of heat at a high temperature from a small volume. For example, some of the aircraft described in Chapter VII require that several hundred thousand kilowatts of heat be delivered from a reactor having a volume of around 500 liters, with the cooling fluid leaving the reactor at a temperature not lower than perhaps 820°C (1500°F).

A small reactor is essential so that the weight of shield around it will not have to be so great as to prohibit its use in an airplane. High temperature is needed so that sufficient mechanical power can be developed to drive the plane. In addition, the amount of fissionable material which the reactor requires must be kept within moderate limits because of the scarcity of supplies. The degree to which these requirements can be met by the pile designer may well prove to be the crucial factor which will determine the feasibility of nuclear-powered flight.

This section on the reactor takes up the types of reactors based on the energy at which the neutrons are utilized, including some general consideration of coolants and moderators and fuel needs, the requirements for aircraft piles, control of the reactor, mechanical design problems and a discussion of reactor materials.

REACTORS IN GENERAL

One way of classifying reactors is based on the energy of the neutrons which cause fission. "Fast" piles utilize neutrons at high energy. "Thermal" piles, which contain moderators, rely upon fission by neutrons which have been slowed down to thermal energies. "Intermediate" piles are, as the name implies, between these two extremes.

Most of the existing reactors are thermal moderated piles: Hanford, Clinton, Argonne, Chalk River, and the Los Alamos water boiler. The only operating example of a high-energy neutron reactor is the Los Alamos fast pile.

Table IVA-1 presents data on (1) existing piles, (2) some piles which have been rather thoroughly designed, and (3) other piles which are in a very preliminary stage. This table is a general reference for much of the following discussion.

Fast Piles

Very small sizes can be obtained with fast reactors, the lower limit

being set by the bomb itself. The existing Los Alamos and the proposed Zinn fast piles are, in effect, somewhat diluted and slightly cooled bombs, with cores about 10 cm. in radius. The Zinn pile, which is considerably the more powerful of the two, is designed for a heat output of 1000 kilowatts; this is something like 1% of what would be needed in an aircraft reactor.

Some larger and more dilute fast piles with thinner solid structure have been suggested for high-power outputs. Thus Menke has considered some reactors comprised simply of iron plates, inside which is the fuel, and between which flow bismuth cooling streams. Some further estimates on such systems are presented in Figure IVA-1; to a first approximation the results are independent of the relative volumes of iron and bismuth used (1). The pile in Figure IVA-1 is assumed to be spherical. From the upper curves one can see what fraction of the reactor volume is available for the coolant and structural materials. Other high-powered fast reactors have been considered by Soodak, General Electric, and Soodak and Dancoff (2).

As may be seen in Figure IVA-1, the amount of fissionable material needed for a fast pile increases rapidly with the size of the pile. Fast reactors which are big enough to permit the removal of such large amounts of heat as are needed for an airplane will probably require an undue amount of fuel as compared to a thermal reactor for the same service. Even with a large fuel investment, the power densities (kilowatts of heat per liter) which would be needed in airplane fast reactors would probably be quite high in comparison with the densities envisaged at present. Furthermore, the control of a fast reactor is rather more hazardous in some important respects than the control of a thermal reactor, and less experience is available.

The fast pile, however, has a notable advantage for aircraft consideration. It permits a wider choice of materials to be used in the pile, because it does not require that they possess good moderating properties. Hence if materials problems turn up to block the development of moderated reactors, fast piles may offer another possible line of attack.

Finally, although fast piles tend to be <u>too</u> small from the standpoint of heat density, the small size does minimize the weight of the shield.

Thermal Piles

At the other extreme from fast piles are thermal piles. These have the potentialities of very low requirements of fissionable material -- for example, the Los Alamos water boiler will run on less than 1 kilogram of uranium 235. Furthermore, there is much more experience available to the designer in the case of thermal piles, and methods of solving the control problem are further developed.

While thermal piles need less fuel than fast piles, they generally require larger volume. The size depends rather markedly on the moderator which is used. Water-moderated piles are inherently smaller than those which are moderated with beryllium, and these latter in turn are smaller

than graphite-moderated piles. Moderators are discussed in a later section of this chapter.

Intermediate Piles

Between the extremes of fast and thermal reactors there can be a whole sequence of reactors with intermediate neutron energy spectra. Intermediate piles tend to be intermediate also in size, but are not always intermediate in fuel requirement. It is quite possible that the desire to get a small reactor for aircraft use may result in a design which was initially predicated on a thermal pile but which became an intermediate pile as a result of decreasing the size of the reactor. Examples of intermediate reactors are found in work by General Electric (3)(4), Friedman and Goldberger (5), and Weinberg (6).

Coolants

The selection of coolant for use in a reactor has a considerable effect on design; in particular, it affects the size and fuel requirements of the pile.

Liquids are better scatterers than gases and so, provided their neutron absorption is not too high, tend to permit smaller piles from the reactivity point of view. Figure IVA-2 illustrates, in engineering units, the difference in this respect between gas and liquid coolants. The curves are for bare uniform unpoisoned reactors in the shape of right circular cylinders with height about equal to diameter. The gas-cooled graphite curves are taken from a NEPA report (7). The remaining curves were crudely estimated by a method described elsewhere (4).

Moderators

Hydrogen would be an excellent moderator for an airplane reactor from the standpoint of permitting a very small pile and operating on a small amount of fissionable material. Unfortunately, it is hard to hold hydrogen at high density and high temperature. Low-temperature water piles have been designed to run on a few kilograms of fuel and to give tens of thoudands of kilowatts of heat from a core with a radius of only about one foot. Current work on nuclear reactors for submarines is seriously considering the use of water at very high pressures so that temperatures which are of interest for power can be realized. It should be noted that in the submarine a water-cooled condenser can be used.

Another possible way of using a hydrogen moderator might be by employing a metal hydride such as zirconium hydride (8).

Heavy water need not be seriously considered for aircraft reactors. It has the temperature limitations of ordinary water and it requires larger piles.

Beryllium, its oxide and its carbide, are approximately the same from the nuclear standpoint and come next to high-density hydrogen as a moderator. Beryllium-moderated reactors are somewhat larger than those

which would be necessary with hydrogen, but they have higher temperature potentialities and may prove to be the most attractive types for aircraft. Figure IVA-2 shows estimates of pile sizes and fissionable material requirements for an idealized unpoisoned reactor moderated by beryllium or by its oxide or its carbide. In this figure the reactor diameter in feet is plotted against the free-flow ratio in the reactor. Free-flow ratio is the fraction of the reactor volume which is occupied by coolant. Table IVA-1 gives some examples of beryllium-moderated reactors, and a number of the hypothetical aircraft cases which appear in Chapter VII are predicated on reactors which are moderated with beryllium.

Graphite is the other moderating material which warrants consideration. From the nuclear standpoint, graphite is poorer than beryllium, and its use therefore entails still larger reactor dimensions for the same fuel inventory. On the other hand, it has good high-temperature characteristics in the absence of oxidizing conditions and it is relatively strong under thermal stress.

Figure IVA-2 shows that graphite-moderated reactors are inherently rather large. The integrated designs which are presented in Chapter VII indicate that reactor diameters greater than four feet are hardly tolerable in an aircraft under conditions which are otherwise favorable. It is, of course, possible to hold a graphite-moderated reactor down in size by increasing the amount of fissionable material in it. It should be emphasized, however, that the amount of fissionable material shown in Figure IVA-2 is calculated on an unpoisoned reactor and that the actual requirement will be greater, depending largely upon the amount of poisons which have to be introduced in the form of coatings and the like. These results are predicated on a graphite having a specific gravity of 1.7; increased graphite density would permit smaller reactors.

Mixed Piles

In view of the fundamental nature of the temperature problem in nuclear-powered aircraft, it may be worth while to consider composite reactors. Grebe, during a visit to the Lexington Project, suggested using a medium-temperature central region in the pile to shrink the total volume and provide mechanical power for the compressor of a jet heated by a higher-temperature outer part of the pile. Similar suggestions for "mixed piles" have been made by other people (9)(10)(11).

Whether or not one splits the power plant, the notion of a mixed pile with different temperature zones and materials may be worth looking into. Thus a medium-temperature hydride or beryllium core might be used to keep the pile dimensions down, while a graphite outer shell provides the peak temperatures needed.

A few related notions might be mentioned. If it were possible to maintain temperature differences in a finer scale structure, one might try using separate cooling streams and run the fuel elements hot while keeping the moderator cool. Lawrence seems to have considered a pile with water moderator to get small size, and liquid-metal-cooled fuel

elements to get power temperature (12).

Fuel Requirements

Uranium 235 is the most likely fuel for airplane reactors. It is more abundant, less toxic, better known, and probably more expendable. Fast pile fuel requirements would be lower with plutonium 239 (see Figure IVA-1) or with the (at present unavailable) uranium 233 isotope. Plutonium would also give some reduction in the fuel needed for thermal piles. Besides nuclear properties, differences in material properties may also enter in choosing between uranium and plutonium.

The majority of the pile examples which have been mentioned above are rather idealized and incomplete. In actual cases the amount of fuel needed will depend considerably upon the details of the design followed, and upon the poisoning that has to be introduced for jackets and control provisions and the like. It will also depend upon the reflector-shield employed; upon the coolant distribution system and ducts through the reflector; on such fission product poisons as xenon 135; upon the temperature, power and depletion ranges envisaged in operation; and upon the temperature, power and coolant distribution in the reactor (13)(14).

Presumably one would move in the direction of heavy reflectors which are themselves part of the shield (15). There would then be but thin layers of moderating reflector between the core and the heavy reflector; this possibility is illustrated to some extent in certain of the fast and intermediate pile examples in Table IVA-1. Involved are combined considerations of reactivity and shielding and power distribution, and the relative costs of fuel and gross weight.

Besides the fuel actually in the reactor, the amount held up in pile reprocessing plants may also be of concern. Thus, in the Clinton high-flux design the expected processing hold-up was some 10 to 20 times that in the pile proper. While this can probably be reduced materially, work will be needed to develop the technology (16).

AIRCRAFT REACTORS

The demands made of a reactor for airplane application were mentioned in the introduction to this chapter, and are further discussed in Chapters III and VII. This section will briefly illustrate these requirements.

Temperature

Table IVA-2, taken from Chapter III, indicates the probable ranges of temperatures that will need to be exceeded in the reactor for various aircraft power plants.

[redacted]

TABLE IVA-2

TEMPERATURE REQUIREMENTS

Type of Power Plant	Required Reactor Temp.
Open-cycle turbojet	930 - 1090°C
Helium compressor-jet or turbojet	820 - 980°C
Bismuth turbojet	700 - 820°C
Vapor cycle	540 - 650°C

Size and Power

Figures IVA-3 and IVA-4 illustrate the reactor size and heat output for manned and tug planes respectively. It was here supposed that a reasonable estimate for shield weight in the case of a manned plane is obtained by using a shield of specific gravity 6 and thickness 120 cm. all around a spherical hole which contains the reactor and its immediate adjuncts. Ducts, shadow shielding, change in shield thickness with reactor size and composition and heat output, external system shielding, etc. were ignored. For a tug plane the shield thickness was reduced to 60 cm., the other assumptions being unchanged; this is thought to be on the conservative side (see section IVB on shielding). Typical values* were assumed for the aircraft and power plant parameters (see Chapters III, V, VII). With these assumptions the gross weight of the manned planes comes out to be twice the shield weight, while in the tug planes it turns out to be 9300 kg. plus twice the shield weight.

From the point of view of applying nuclear power, one of the greatest advantages of the tug-tow arrangement is that more room is made available for the reactor. Table IVA-3 shows, for several shield weights, the dimensions made available for the reactor and the average heat densities required.

TABLE IVA-3

REACTOR SIZES AND HEAT DENSITIES

	Hole Radius in cm.		Kilowatts per Liter	
Shield Wt. in kg.	Manned	Tug	Manned	Tug
18,000		30		510
24,000		40		250
30,000		50		150
44,000	0	66		80
55,000	10	80	20,300	54
75,000	24	97	2,100	40
100,000	40		580	
200,000	85		120	

*Speed of plane = 0.8 speed of sound; aircraft lift-drag ratio = 15; machinery weight = 1 kg. per thrust kw.; structural weight = 0.28 gross weight; miscellaneous weight = 0.07 gross weight; thrust work = 0.2 heat release; drag of towed plane and cable = 2000 kg.

[redacted]

Pile Examples

None of the piles which have been built shows a combination of size and heat output anywhere near that needed for aircraft application.

As an example from Table IVA-3, a tug plane with a shield hole radius of 50 cm. requires in this hole a mean power density of 150 kw. per liter. The maximum heat density in the uranium slugs of a Hanford pile is about 50 kw. per liter (the average heat density in such a pile as a whole is less than 1 kw. per liter). The planned power density in the core of the proposed Clinton high-flux pile was over 300 kw. per liter. The proposed General Electric intermediate pile and the Zinn fast pile are expected to give 250 and 125 kw. per liter respectively, with an exit coolant temperature of about 350°C (660°F).

In considering the applicability of a pile to airplane use, one should note such characteristics as volume of pile and reflector (radius of shield hole), amount of fuel, heat output, coolant used, and maximum temperature of coolant or pile. As recopied from Figures IVA-3 and IVA-4, the required airplane heat output is plotted as a function of shield hole radius in Figure IVA-5. Also shown are points for some of the reactors of Table IVA-1 with the fuel mass indicated at each point.

In studying Figure IVA-5, one should bear in mind that the reactors presented were not designed for aircraft application but for stationary (and, in some cases, submarine) applications. Thus, in many instances the reflectors used are larger than one would consider in the aircraft case.

Some idea of the interplay of critical mass, moderator and radius of shield hole is given by the critical masses for bare solid piles as plotted on Figure IVA-6. Also plotted are some piles from Table IVA-1, with the power outputs written next to the points.

CONTROL

It is not thought that control difficulties will determine the feasibility of nuclear-powered airplanes, but they will doubtless present substantial problems and require considerable progress in the field of pile control as a whole (17)(18).

Work on automatic control systems for reactors has started in recent years (cf. studies on the Clinton high-flux pile, work at NEPA, and the partially constructed Brookhaven pile), and continued effort should produce substantial progress. However, little attention has yet been given to the composite system of pile plus external equipment (19). In the aircraft case this composite aspect is present to an unusual degree, since an airplane reactor is but one component of a complex power system. This system has to be manageable over a considerable range of operating conditions, by a small crew having limited access to the equipment, and yet attain a high degree of reliability. Successful

integration of the characteristics of the various components into a single satisfactory system will require a carefully planned program throughout the entire development. This includes analytical studies of the dynamics of the complete system under its variety of operating conditions, to go hand-in-hand with the component development.

One would like, for reasons of pile size or fuel economy, to reduce the amount of excess reactivity that has to be provided. Similarly, because of the aggravated mechanical problems discussed in the next section, one would like to simplify the control mechanism as much as possible. Some help in this direction may be obtained by reducing the reactivity allowance for xenon. Shortly after a pile is shut-down, highly absorbing xenon 135 builds up in the course of fission product decay and strongly poisons the reactor. Presumably, an airplane reactor will not be expected to override this shut-down poisoning to any great extent, and thus the situation will be eased considerably in this respect as compared to the high-flux and Navy piles (20).

It has been suggested that xenon poisoning instability might limit the flux at which thermal piles can be controlled, but more recent considerations tend to be reassuring on this point (21). The position of the xenon absorption line in the neutron energy spectrum is not yet known, but experiments aimed at determining this have been under way at Oak Ridge and Hanford for a year or so.

Additional simplification of the system will result from keeping some of the control mechanisms and operations on the ground. The start-up operation is rather a delicate one and would presumably be done on the ground with auxiliary equipment (preheaters, externally powered cooling, supplementary control drives, measuring instruments, etc.) not borne aloft with the airplane (22). The plane would, however, carry mechanism for shutting off the reaction. It is also likely that the control of thrust will not rely solely upon fission rate, but may involve other portions of the cycle; such additional controls would be viewed with more favor if arranged to lower the pile temperature at reduced thrust. In any case it is necessary to have some degree of control over temperatures as well as heat rates.

In open-cycle air-cooled reactors, changes in moisture content of the air through which the plane is moving may produce fluctuations in pile reactivity. Offhand it does not appear that this would normally be a serious problem at high altitude, but the considerations are inconclusive in regard to extreme gradient conditions at low altitude (23).

MECHANICAL DESIGN PROBLEMS

General Remarks

Detailed study of the mechanical problems of aircraft reactors has not been possible in the absence of a definite design, and would in any event be premature at this stage. However, several mechanical problems which are new or are intensified in this application have been explored

to outline potential solutions. The results of these studies have been generally encouraging, as discussed in the following paragraphs. It should be emphasized that many of the mechanical design problems of reactors are difficult and require long study even when the principles of the solution are clear. In the present section only these principles have been considered, and much detailed work therefore remains to be done.

Structurally, reactors may be broadly divided into two types, metallic and non-metallic (or ceramic). The metallic reactors, with the exception of beryllium, are considered to involve simpler design problems in view of the more highly developed technology and the superior structural properties of metals. Unfortunately, those metals which retain their structural properties at high temperatures can be used liberally only in fast reactors. Since the fuel requirements of such reactors are considered to be too high, the aircraft reactor is urged toward a thermal or near-thermal pile employing moderating material as the major component. The moderating materials potentially available at these temperatures are graphite, beryllium oxide and beryllium metal. Graphite and beryllium oxide are non-metallic and result in typical ceramic construction. Beryllium metal cannot at present be handled by conventional metal techniques, but may yield to a design treatment that considers it as a ceramic.

Thermal Stresses

One of the more troublesome mechanical problems is the possibility that the moderator will disintegrate under thermal stresses. The resistance of available moderators to thermal cracking is evaluated in the discussion of materials. Of the three non-metallic materials suitable as moderators, graphite is best, beryllium oxide next best and beryllium carbide poorest from this point of view. Because of the good nuclear properties of beryllium oxide it has been given particular study in this connection.

Thermal stresses in beryllium oxide have been estimated for two representative cases, the first with the fuel intermixed with the oxide, and the second with the fuel separate from the oxide (24). In the former case, the high heat production and severe radiation damage to thermal conductivity result in thermal stresses in excess of the strength at all but the ends of the pile where power production is below the average. Thus plastic relief of the stress must be accomplished even at the temperatures existing near the cool end of the pile (1100°F for the "minimum" open-cycle case of Table IVA-2, and even less for the closed cycles). The possibility of achieving this while retaining strength in the high-temperature parts of the pile seems remote. When the fuel is separate from the oxide, on the other hand, the lower heat production and the reduced radiation damage in the moderator reduce the stresses sufficiently that they are well below the available strength even at the 1800°F temperature of the "minimum" open-cycle case. However, moderate increases in power combined with increases in temperature would increase the stress and decrease the strength. Since the physical properties are not well established above about 2000°F, this points to the need for investigating these properties up to 2500-3000°F, and of introducing some

plasticity into beryllium oxide at temperatures above about 1800°F.

Reactor Construction Problems

In order to illustrate the nature of some of the reactor construction problems, a general description of a possible assembly is desirable. It is not intended as a proposed design since far more detailed consideration would be necessary before such a proposal would begin to emerge. No mention of the type of coolant is made, since the fundamentals here described are applicable to either gas or liquid metal coolants, though detailed designs would vary considerably with the coolant selected.

One general design of a beryllium oxide pile might be as follows. The oxide pieces might take the form of "bricks" roughly three inches laterally by one to six inches in length, depending upon the thickness of wall as determined by the number, size and disposition of the coolant passages. These small elements are to be assembled in the shape of a cylinder two to four feet in diameter and length, providing continuous coolant passages throughout the length, into which thin-walled fuel-bearing tubes are inserted.* With coolants that are not inert these tubes must form tight joints with headers at either end of the pile in order to prevent the coolant from attacking the oxide. With inert coolants only one header, presumably at the cool end, will be required to hold the tubes against the coolant thrust and to take the thrust of coolant pressure against the solid portions of the structure.

In a graphite pile, the construction might follow the same general pattern, though the greater possible size of the graphite bricks will reduce the complexity of the structure to some extent. It may be that the uranium need not be confined within tubes when graphite is employed, but the use of tubes for fission-product retention may prove to be the most promising approach. If the fuel is mixed with the graphite, headers must be employed at both ends for fission product retention; if the fuel is in the tubes, the requirements for headers are the same as those of the beryllium oxide pile.

The design problem in the ceramic pile structure centers around the necessity for holding the bricks in the desired configuration in spite of thermal expansion and externally introduced acceleration loads. Cumulative expansions of 1/4 to 1/2 inch will be encountered in the transverse direction, and up to roughly one-half of this in the axial direction. The retention means, in addition to confining the assembly of bricks, must withstand loads caused by vertical accelerations up to

*It should be pointed out that this concentration of the fuel will introduce problems of its own. The most obvious of these is the reduced thermal capacity of the mass containing the fuel, leading to greater temperature excursions with deviations from normal power. Since metallic tubes might well prove necessary, because of greater promise of successful fission product retention, the temperature margin will probably be less than with ceramic materials. In addition, the introduction of the metal will increase the uranium investment required to overcome the added absorption.

2.5 x gravity and longitudinal accelerations of as much as 1 x gravity. Although transverse accelerations are greater and would appear to involve a worse retention problem than the longitudinal retention system, the thermal isolation of the transverse supporting structure is easier than for the longitudinal structure and the transverse support is not required to avoid obstructing cooling passages. These facts make the transverse support the simpler problem.

Because of the changing pile temperature pattern during operation the most promising approach to the problem appears to be the incorporation of elasticity in the restraint. The elasticity presumably need be introduced on only three of the six "sides" of the reactor; the remaining sides would then be rigid thrust members which in the transverse directions could be incorporated in the confining shield. Stressed members with low elastic constant or hydraulic pressure within extensible tubes may be used for the elastic members. Of these two, the former appears the simpler and more reliable. The hydraulic system must rely upon an external source of pressure, which introduces additional equipment whose failure can cause failure of the entire unit.

A small but important item in the transverse retention problem is the cross-sectional shape of the "brick." In the power piles considered to date attention has been devoted chiefly to a hexagonal shape, in order to take advantage of its interlocking properties. Because of this interlocking property, however, it is possible for the retaining forces to be diverted around individual elements, thus leaving them free to move around (within limits) under the influence of accelerations. While this may not be injurious, the use of square or rectangular shapes whose sides lie parallel to the direction of the retaining forces gives promise of eliminating such looseness.

In the longitudinal direction the thrust member should be relegated to the hot end of the pile in order to take advantage of the lower temperature region for the elastic members. At the hot end the expansion problem is magnified both by the higher average temperature and by the variations in the temperature. To eliminate differential expansion of dissimilar materials, various designs of mechanically interlocked bricks of the same material as the pile have been proposed. However, when the design is based upon a brick which is simple enough in shape to permit fabrication, the bending stiffness of the structure must be produced by a relatively high compressive load at the perimeter (25). Since the means for providing this load must accommodate variations in expansion, the design problem involved is of considerable magnitude.

It is felt that simple beams of high-temperature material spanning the pile and laterally located by the bricks themselves show more promise. Expansion along the length of the beam would introduce no obstruction to the flow of coolant, and the lateral dependence on the pile expansion would prevent overlapping in that direction.

Control Mechanisms

The control mechanism for the entire aircraft power plant cannot be

intelligently considered until the selection of the type of cycle has been made. The portion of the control associated with the reactor, however, presents some mechanical problems that are common to all types of reactor. The more important of these may be summarized as follows:

(a) Alignment of control elements with reactor in spite of high temperatures, accelerations and thermal distortion.

(b) Simplicity and compactness of the reactor control system. (This obviously applies to the overall system).

(c) Heat removal from the control elements.

(d) Achieving the necessary reliability in the control mechanism, i.e., drives, position indicators, etc.

The alignment problem clearly implies the use of a flexible linkage between the control element in the pile and the external parts of the drive. Though such flexible mechanisms are available, an accuracy of location of the control element may be required which will not permit lost motion in the driving system. Should this be the case, the more common flexible elements may be ruled out, and, indeed, development work may be necessary to obtain sufficiently refined elements.

The desirability of simplicity and compactness in the control is obvious. Not so obvious, however, are the means for achieving the objective. The possibility of including in the air-borne system only that equipment which is essential to flight operations has been mentioned and is a step in the right direction. Further gains can be made by careful attention to the design of the control system. Minimum mass of the moving elements and proper support to prevent unnecessary friction will be rewarded by reduction in size of the entire drive system. Since the controls must be able to operate even while experiencing externally applied accelerations, excess mass in control elements is reflected several-fold in the power of the driving means, with consequent increase in size and weight.

It is fortunate that, at least in the case of absorption control of thermal piles, the cooling requirement points also in the direction of minimum mass. The heat generation in a control rod is strongly influenced by gamma ray absorption, and since this is proportional to the mass of the material, the cooling is reduced by a reduction in mass of the element.

Reliability of the control system will result only from a well-executed developmental program. This is true not only for individual components but for the entire system, since an assembly of tested components requires further development in order to achieve the proper interrelation of the various elements.

The need for an experimental program associated with the mechanical design can hardly be overemphasized. Preparation of construction

drawings, far from indicating that the mechanical problems are solved, signals instead the start of a program which will require many times the amount of money, manpower and facilities that were expended to produce the drawings.

MATERIALS FOR REACTORS

Introduction

As stated previously, a thermal reactor for aircraft should be a comparatively small, compact unit consisting of a considerable proportion of moderating material containing passages for the flow of the coolant and a relatively small volume of fuel, either incorporated in the moderator or in separate tubes or rods. In some cases it is necessary to protect the moderator, or the fuel, by a protective layer from the corrosive influence of the coolant. Further, the coating or matrix must prevent fission fragments from entering the coolant stream. In addition to the main parts of the reactor enumerated above, it is necessary to use some control material with a large absorption cross section and perhaps a reflecting envelope enclosing the whole reactor.

The fast reactor differs from the thermal reactor in that the moderator is eliminated. It will be seen that the requirement of containing the fission fragments within the fuel body and protection against interaction between the fuel body and coolant are substantially the same as in the previous case. In the fast reactor the materials problem is somewhat simplified in that a much wider selection of materials is possible, since the neutron absorption is much the same for all materials.

It is well to review briefly the conditions existing in the reactor. In the first place, the temperature will be as high as the materials of construction will stand. All parts of the reactor will be exposed to a neutron flux, and in the immediate vicinity of the fuel to the damaging effect of fission fragments. A constant temperature cannot be depended upon, either throughout the pile, or in relation to time, so that differential expansion and thermal stresses must be allowed for in the design. In addition, the structure must stand the shocks and accelerations due to operation in the aircraft. It should be remembered that the aircraft reactor may have a relatively short life, perhaps 100 hours.

Materials for the Moderator

In order to produce a thermal reactor, it is necessary to incorporate a considerable amount of moderating material with the fuel to slow down the fast neutrons to thermal velocities. Such material, to be efficient, must have a high atomic density, a low capture cross section, a high scattering cross section and high energy exchange per collision.

In addition to the favorable nuclear properties of the moderating material, it must also be stable at the desired operating temperatures when in contact with the coolant or protecting layer. It must be resistant to temperature stresses, be stable under high intensity radiation, as well as being free from impurities with large capture cross sections.

In addition, it must be capable of being formed accurately into shapes of maximum density. All of these factors make the selection of the moderating material one of the most difficult problems to solve in the design of the reactor. Some of the pertinent properties of these materials are shown in Table IVA-4.

TABLE IVA-4

PROPERTIES OF MODERATING MATERIALS (29)

Materials	Nuclear Density $N \times 10^{24}$/cc	Melting Pt. °C	Boiling Pt. °C	Specific Gravity	Tensile Strength psi at 1000°C	Chemical Stability in Air
Water	0.10	0°C	100	1	-	Stable
Heavy Water	0.10	1.8	100	1	-	Stable
Beryllium	0.12	1280	2500	1.85	2500*	Up to 600°C
Beryllium Oxide	0.14	2530	3900	3.02	10,000	Stable up to 1200°C (in air containing water vapor)
Beryllium Carbide	0.15	>2100 (decomposes)	-	2.44	-	Oxidizes
Graphite	0.081	3650 (sublimes)	-	1.9	3000	Stable up to 350°C
Zirconium Hydride	0.11	(decomposes above 800°F)		5.6	-	----------

*Long-time creep strength only 25 psi

The resistance to temperature stresses in the moderating material is such a critical problem in reactor design that it is necessary to take some space to discuss it. In a high-power reactor all of the heat (perhaps 1 kilowatt per cubic centimeter) is liberated in the moderator if the structure is homogeneous, with the fuel dispersed completely through the moderator (24)(26). In a heterogenous reactor, however, where the fuel is concentrated, the heat generated in the moderator is of the order of only 7 per cent of the total heat, so that under these conditions the moderator is working under much more favorable conditions. When the reactor is producing power, the heat is evolving continuously and is carried

off by the coolant flowing through passages; so that a temperature difference is set up between the surface of the cooling passage and the interior of the moderator. This temperature difference sets up stresses in the moderating material of such magnitude that in many cases they seem to be a limiting factor in the power that can be produced in a given design.

Considerable study has been carried out on the temperature stresses in brittle materials. It has been shown that the stress set up is closely proportional to the temperature differences in the material, which are proportional directly to volumetric rate of heat generation and inversely to the thermal conductivity (24)(26). For a given material it has been shown that the stress or tendency to crack is inversely proportional to the thermal conductivity and the structural flexibility, and proportional to the coefficient of expansion, for a given temperature difference (27).

There are, at present, only two solids which have been seriously proposed for moderators at high temperatures; that is, graphite and beryllium oxide. In Table IVA-5, below, are listed their properties and the computed tendency to crack under temperature stresses.

TABLE IVA-5

PROPERTIES AT 1000°C (1830°F)

Material	Coef. of Expansion (°C)	Coef. of Therm. Cond. c.g.s. units	Tensile Strength psi	Mod. of Elast. psi	Relative Flexibility	Relative Tendency to Crack
Graphite	1.1×10^{-6}	.2	3000	$.7 \times 10^{6}$	.0013	1
Beryllium Oxide	8.9×10^{-6}	.05	10,000	28×10^{6}	0.5	380

It will be seen that graphite has three to four hundred times greater resistance to cracking than beryllia. Beryllium metal, although not included in the table, shows an excellent resistance to temperature cracking because of its hot ductility, but it may not have sufficiently high creep strength to permit its use in pure forms in high-temperature reactors. However, recent work indicates that some alloys of beryllium give high creep strength. Beryllium carbide has been suggested as a moderator but little is known of its physical properties.

It is well known that refractory materials, even with a high purity, will flow plastically under stresses at elevated temperatures. While there are few data available for beryllia, it can be assumed to be very similar to pure alumina, which flows readily above 1150°C (32). It is felt, therefore, if the reactor is brought up to temperature and cooled slowly, that the temperature stresses in a moderator of beryllia will be greatly minimized by plastic flow. Moreover, by adding a small proportion of flux to the beryllia the plastic range can readily be extended to lower temperatures.

It is also possible, by proper arrangement of the material in the moderator, to minimize the thermal stress -- such as by the introduction of slots in the proper locations. In commercial and laboratory furnaces, conditions are often found where temperature differences in a thin wall of fairly pure refractory are as high as 1500°C (2730°F) without cracking in long service (27). This resistance to cracking is probably due to a slight plastic flow which relieves the stress.

The moderator must have a relatively high fusion point to permit elevated temperatures and it must be stable when in contact with the other materials in the reactor. For example, graphite as a moderator could not be used with air cooling because of the rapid oxidation, and beryllium oxide could not be used above 1200°C with moist air because of the volatilization of the beryllia. On the other hand, all of the moderator materials could be used with helium as a coolant without deterioration up to the point where their vapor pressures become appreciable. When the coolant is used in a closed cycle, losses by volatilization may be less than in an open cycle because the coolant will become saturated with the vapor at the temperature of the coolest part of the cycle (which may itself be quite high); on the other hand, trouble may result from distillation of the moderator from the hotter to the cooler regions.

The moderating material must also be stable against the damaging effect of the high-intensity radiation in the reactor. These radiations are of four types; gamma rays, beta particles, high-velocity neutrons and fission fragments. The first two, apparently, have very little damaging effect on the types of materials being considered here, but the neutrons, which permeate the whole pile structure, cause physical changes in many materials (26). The fission fragments have a very small range of action and will damage material closely adjacent to the fuel itself. Unfortunately, there are very little data now available on the effect of radiation damage on reactor materials, especially at the high-flux densities proposed in the aircraft reactor. Some conclusions can be drawn from the data that are now available, but they are scarcely adequate for the designer.

At the present time there is no evidence to indicate that the physical properties of metals are appreciably altered by fast-neutron bombardment. This has, however, been actually confirmed only with aluminum and beryllium at room temperature. There is no doubt that many of the atoms of the metal are moved around during an extended exposure but the atoms seem to be able to return to their lattice sites so that there are no permanent changes in properties. For operation at elevated temperature there is even less likelihood of changes in physical properties, with the possible exception of creep, because of annealing during bombardment.

Because of the greater damaging effect of fission fragments it might be expected that a metal exposed to them would be seriously affected. There are no experimental investigations on this point where the fissionable atoms are dispersed on an atomic scale in the matrix, with the exception of natural uranium. In this case it is known that the metal swells and warps after several hundred megawatt days' exposure in the Hanford pile.

The only other information has been obtained on an alloy of aluminum and uranium where the uranium occurs as the compound UAl_3 which is dispersed as discrete particles in a matrix of almost pure aluminum. In this case after one-half per cent of the uranium has fissioned (original content was 17%), there are only minor changes in electrical resistivity and hardness and no appreciable change in dimensions. It is quite probable that most of the fission fragments remain within the islands of the compound and hence that the aluminum received relatively little damage (26).

The idea of having the uranium 235 dispersed in the form of discrete particles whose size is greater than the range of the fission fragments is now believed to be a sound principle. In this way the bulk of the material receives no fission fragments and hence is less damaged. In addition to bombardment, the fission fragments will act as impurity atoms (some of them gaseous) in the lattice containing them. Again it seems desirable to retain these products in small islands in order to spare the matrix.

Because of the extensive mobility of metal atoms during bombardment, it is believed that there may be extensive stress relief or creep in case the material carries a load. This phenomenon has not been extensively studied, but it may become one of the serious problems with metals in a high-temperature reactor. It is possible also that bombardment may give accelerated diffusion rates and this may enable fission products to diffuse through solid layers of metal.

It has been shown by a large body of experimental data that graphite is profoundly altered by fast-neutron bombardment at room temperature. The strength and elastic modulus increase rapidly two or threefold and then level off while the thermal conductivity is gradually reduced to 1/40 of its original value (26). The total energy stored in the lattice increases continuously with exposure. There is also a change in dimensions which presents serious construction problems.

All of these changes in graphite can be removed by heating, but each property reverts to its original value over a different temperature range. For example, much of the stored energy is released at 200°C (390°F) but the elastic modulus recovers only after exposure to 800-1000°C (1470-1830°F) while electrical resistivity and probably thermal conductivity are completely restored only after regraphitizing. Probably misplaced atoms return to vacant lattice sites at temperatures up to 800°C (1470°F) while holes diffuse out of the lattice only at the highest temperatures.

The differences in annealing behavior indicate that different properties of graphite will be altered by widely different amounts if bombarded at elevated temperatures. For example, the elastic modulus may not be altered at all at 800°C while thermal conductivity still will show large changes. The only observations at elevated temperatures have been made on thermal conductivity. In this work the samples contained uranium 235 to give bombardment by fission fragments since neutron damage alone would produce effects much more slowly. With uranium 235 particles one micron or less in size, a sample exposed at 800°C (1480°F) showed a thermal conductivity decrease to one-fortieth and leveled off at a value which

is equal to that of the value of ungraphitized carbon at the same temperature. On the other hand, coarse uranium 235 particles of 100-mesh size caused the conductivity to decrease only to one-third. On the basis of this latter result it is believed that graphite containing coarse uranium particles will not show serious damage during a 100-hour life in a 10^{14} neutron flux.

This leads one to the hope that materials in the reactor at high temperatures might escape extensive damage if the displaced atoms are sufficiently mobile to return shortly to their starting point and thus restore the crystal lattice to its original condition. This is important since the loss of thermal conductivity in the reactor materials is particularly harmful, as it increases the temperature gradient in the moderator sections and augments the temperature stresses.

Some work has been done on beryllia after exposure to a neutron flux and it is shown again that there is significant damage at room temperature, particularly a decrease in the thermal conductivity to about one-half. Annealing at moderate temperatures appears to restore the original conductivity value, so that on the whole this material is less affected by neutron bombardment than graphite. On the other hand, if beryllia is subjected to fission recoils, the damage is more extensive and the thermal conductivity is decreased about five times. What little information is available would indicate that the mechanical strength is not particularly influenced by these effects.

It should be emphasized that before a high-intensity reactor can be designed, it is necessary to have extensive knowledge of radiation damage effects on moderator materials under the conditions of high flux and high temperature that would be expected in the aircraft reactor. This should be one of the immediate problems and it will probably be necessary to design a special reactor for the purpose, as discussed in Chapter I.

From the preceding discussion it will be clear that the moderating materials must have very special properties; that is, the efficient moderating materials must be stable under high-neutron fluxes and fission recoils, and have chemical and physical properties that will give them sufficient strength and stability against temperature cracking under the operating conditions. In addition to this, they must be of extreme purity in order to eliminate any atoms with large capture cross section; and furthermore, they must be of a nature that will permit them to be formed into intricate shapes of maximum density necessary for the reactor. Fabrication of moderating material to maximum density and intricate shapes is a difficult problem and should receive a major amount of attention.

Fuels

The fuel in a reactor can be uranium 235, uranium 233 or plutonium. The amount required in a reactor has been discussed earlier in this chapter, and will run from perhaps 20 pounds to 200 pounds in proposed designs. The fuel can be dispersed uniformly through the material in a homogeneous

type of construction or it may be concentrated in rods or tubes in passages in the moderator. The latter construction seems to be necessary if beryllia is used as a moderator to prevent excessive temperature stresses.

Metallic uranium 235 may be used as the fuel under certain types of construction, but because of its low melting point the temperature of operation is limited. Higher temperature limits may be attained by using alloys of uranium with other metals of small capture cross section. An alloy of uranium and beryllium with a melting point of 1900°C seems of interest. On the other hand, uranium oxide (UO_2) may be used as a fuel with high permissible operating temperatures if air is excluded. Uranium carbide is a possible fuel material, especially when in contact with carbon as a moderator, but must also be protected from air.

Little information is at hand concerning the alloys or compounds of plutonium. Recent work indicates that the oxide or carbide might be satisfactory for fuel material, but much more work needs to be done with these before they can be seriously considered in reactor design.

Consideration must be given to radiation damage in the fuel itself if it is to be depended upon to have any appreciable amount of mechanical stability in the reactor design. It is known that uranium undergoes rather extensive structural changes under the influence of fission recoil in some of the existing reactors.

Altogether the material problems associated with fuel assemblies are very serious ones when dealing with elevated temperatures, and much more data must be made available to the designer on fundamental properties of the fuel and contacting materials.

Coolants

The coolant fluid which is required to take the heat from the reactor and bring it outside for the production of useful power needs very careful selection. In the first place, this fluid must not have nuclear properties which are harmful to the operation of the reactor, such as a large capture cross section, and it would be highly desirable if it had other useful nuclear properties such as a strong moderating effect for the thermal reactor. The fluid must not attack the passages through the reactor nor the material in the heat interchanger; it must have a high heat-transfer efficiency, which means high thermal conductivity, high specific heat and perhaps good wetting ability. To conserve pumping power the viscosity should be low at the working temperature and the materials should have good thermal, chemical and radiation stability under the conditions of use. In addition, it is very desirable that the fluid have a low induced radioactivity so that it will be unnecessary to shield the heat interchanger (31).

The general types of coolants are gases, low-temperature liquids, and high-temperature liquids. As discussed in Chapter III the gas coolants, unless highly compressed, tend to require a larger volume of cooling passages through the reactor than do liquids. Because of this, and because of lack

of neutron-scattering power, gases tend to increase both the size of the reactor and the amount of fuel needed. On the other hand, air cooling has peculiar interest for an aircraft reactor, and helium also merits consideration.

The gases which have been seriously proposed for coolants include hydrogen, helium, and air. Air is an oxidizing agent and cannot well be used in contact with such moderators as graphite or most metals at high temperatures. Hydrogen is a strong reducing agent and seriously affects the properties of some refractories at high temperature. Helium, on the other hand, is completely inert and can be used in contact with any of the materials in the reactor.

Water, the principal liquid so far used as a coolant, has the advantage of being an excellent moderator; has excellent heat-transfer properties, but is limited in the maximum temperature to which it can be used, even when working to critical pressures. Moreover, if water is used to form steam in the reactor, serious control problems arise. Heavy water is also a good moderator, but is difficult to produce in quantity.

Perhaps the most interesting coolants for airplane reactors are the elements lead and bismuth and their alloys. These elements and alloys have good nuclear properties, are efficient heat-transfer media, and have negligible induced radioactivity. However, practical problems must be solved in melting the metals for starting operations and to prevent expansion effects during the melting or freezing. In Tables IVA-6 and IVA-7 are listed other metals and alloys which may be of interest as coolants.

There is a possibility that some of the lower-fusing salt mixtures might be useful as coolants, but little is known of their stability or corrosiveness. Salts containing bismuth, lead, or sodium, together with phosphorous, silicon, oxygen, sulphur or fluorine would have sufficiently good nuclear properties, and a number of low melting point compounds or mixtures can be formed in these systems.

To sum up the coolants, it may be said that a number of gases and liquid metals are quite practical in reactor construction, although more data are needed on the reactions between various possible coolants and moderators and containers.

TABLE IVA-7

LOW-MELTING BINARY AND TERNARY ALLOYS FOR COOLANTS (31)

Alloy Per Cent by Weight		Specific Gravity	Melting Point °C	Average Neutron-Capture Cross Section in Barns
45 Al	55 Ge		423	.36
32 Al	68 Mg		436	.11
97 Bi	3 Na		218	.11
56.5 Bi	43.5 Pb	10.3	124	.18
58 Bi	42 Sn	8.59	139	.34
76.5 Bi	23.5 Ti	10.2	198	.69
97.5 Pb	2.5 Mg		250	.33
38.1 Pb	61.9 Sn	8.48	183	.47
53 K	47 Na		7	1.13
18.5 K	81.5 Rb		32.8	1.05
56 Rb	44 S		149	.43
~80 Na	~20 Pb		~200	.48
8 Na	92 Rb		-4.5	.49
31 Na	69 S		232	.42
~90 Na	~10 Sn		~200	.48
~90 Na	~10 Te		~210	.58
92 Sn	8 Sb	7.25	246	.84
~72 Sn	~28 Ga		~160	1.0
46.5 Mg	53.5 Zn		340	.42
52 Bi	31 Pb 17 Sn		96	.25
~67 Mg	~32 Al ~1 Be		~430	.11

Protecting Materials

Protecting materials are used in the reactor, first, to prevent fission fragments from escaping into the coolant and thus circulating outside of the reactor where they would make necessary additional shielding, and, second, to prevent interaction between the coolant and the moderator, such as the reaction between air and graphite.

The protecting materials may be rolled or drawn thin metal layers, such as tubes lining the coolant passages in the moderator, or thin sheets surrounding the fuel. For the more moderate temperature use (1500-1800°F) (815-980°C), stainless-type steels would seem to be the most practical. At more elevated temperatures molybdenum has been proposed, but in an oxidizing or carburizing atmosphere it has to have additional protection. The embrittlement following recrystallization is a difficulty that must be overcome. Recently it has been possible to produce pure ductile zirconium and because of its nuclear and mechanical properties, this material would seem promising under certain conditions, but here again it would need protection from air.

The protection may also be a coating deposited on the surface of a moderator or other parts of the reactor. For example, molybdenum has been protected from air by a silicide coating which probably fuses to a self-healing silica glass, thus protecting the surface. Certain high-temperature enamels have been proposed consisting of a glassy material which is melted down on the surface of the body to be protected. Other protective coatings have been suggested, such as bonded silicon carbide and alumina. However, all of these coatings are subject to a very serious limitation, that is, the difficulty of coating the tremendous area of the reactor without a single pin hole or any porosity which would permit local action.

It has been stated that the loss of a small portion of the reactor by corrosion will not seriously affect the operation, but even a little corrosion may release enough fission fragments to contaminate the plane or even affect the towed plane. It should be remembered that only one-millionth of the fission fragments produced can be permitted to pass into the coolant stream in a manned airplane. In fact, this difficulty seems at present so great that it is felt that no deposited coatings for this purpose are practical, at least in the present stage of development.

It should be realized that protecting coatings absorb neutrons, so the amount of fuel must be increased. The last column of Table IVA-8 gives some idea of the order of fuel increase necessary.

TABLE IVA-8

INFLUENCE OF COATINGS ON FUEL ECONOMY (30)

(Example: Reactor 2.5 feet right cylinder with 0.4 free-flow ratio and 0.25 inch cooling passages)

Coating Material	Capture Cross Section	Wt. of coating per mil thickness in kg.	Ratio of coating atoms per mil of thickness to atoms in kg. of U235	Ratio of absorption per mil of coating to absorption by 1 kg. of U 235	Number of kg. of U235 equivalent in absorption to 20 mils of coating
Zirconium	0.4	14.6	37	0.027	0.5
Molybdenum	2.8	22.8	56	0.28	5.6
Iron	2.5	17.6	74	0.34	6.8
Beryllium	0.01	4.2	108	0.002	0.04

Control Materials

It is, of course, necessary to control the output of a thermal reactor by introducing some material with a high absorption cross section in the form of a control rod. This control rod may run at the temperature of the reactor or it may be separately cooled below the prevailing temperature.

Materials suitable for controls must have a high neutron-capture cross section. The material most suitable for this purpose, either as an element, or as an alloy, is boron. It is sufficient to add only a few per cent of boron to steel or other alloys to attain sufficient control, together with good mechanical properties of the rod itself. It is not anticipated that any serious problems will arise in these control materials. However, in order to permit rapid movement of the controls, it may be desirable to use very light materials. Since the control of fast reactors is carried out by reflector movement, the material problem is tied into the reflector.

Reflecting Materials

Sometimes, but especially in fast reactors, the reactor is surrounded by a reflector which prevents the escape of the neutrons. This reflector must consist of a material having a large scattering cross section and a low absorption cross section. As the temperature in the reflector may be lower than in the reactor itself, there would seem to be no particular difficulty in adapting materials to this purpose. Beryllium, iron, bismuth, tungsten, uranium and lead are possible materials.

Summary on Materials

While this section covers the available materials for reactor construction, it is shown that in most cases very meager information is available on their properties under the working conditions of high-temperature and high-intensity radiation. Before the engineer can intelligently design a high-temperature reactor, he must have far more data on materials than he has at present, as it is impossible to predict properties under conditions differing greatly from those of the test.

In the particular, he must know the amount of change in physical properties undergone by the materials under high intensity of radiation. A proposed program for this study, discussed in Chapter I, entails the construction of a reactor especially for this work.

Also, the high-temperature properties of both metals and refractories must be known. The high-temperature values of thermal conductivity are at present relatively unreliable and new means must be developed for this extremely difficult measurement. Moreover, the reactivity of the various materials with the coolants must be known over a wide temperature range as well as the reaction rate of one solid in contact with another.

Especially important is a thorough study of thermal stresses in the moderator at the working temperature. Efforts should be made to develop the very desirable beryllia moderator body so that it will permit release of high-temperature stresses.

The rather extensive research on protective coatings now under way must be continued and extended to see whether or not absolutely dependable protection of such materials as graphite and molybdenum can be obtained.

At the same time protection by thin metal tubes or "cans" should be further studied for high-temperature operations.

To indicate more specifically the possibilities of various materials in the airplane reactor a series of hypothetical combinations is shown in Table IVA-9. The temperature limitations, reactor size, amount of fuel and the uncertain factors are all indicated. The absolute values will need correction when specific material properties are available. These examples clearly show relative trends and deserve careful study by anyone dealing with the materials problem.

RECOMMENDATIONS

Further investigation on types of reactors for aircraft might include studies of:

(a) Thermal beryllium or beryllium oxide piles with metal fuel tubes. One would try to keep the fuel inventory modest, and would attempt to get the liquid metal or gas coolant up to gas-turbine temperatures.

(b) Reactors using high-temperature metal fuel assemblies with liquid-metal cooling, and moderated with water or hydride assemblies which run at a lower temperature. The two types of assemblies might be separated by a heat insulator such as lampblack. Such a reactor would be mechanically complicated, but avoids the shortcomings of beryllium.

(c) Piles divided into different temperature zones with appropriate different moderators, e.g., reactors partially moderated with graphite.

(d) All-graphite-moderated piles, if the estimates of allowable reactor size are increased by shielding developments.

(e) Fast piles.

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Table IVA-1

SUMMARY OF PILE DESIGN DATA 1. Existing Piles

	1.1 Clinton	1.2 Argonne P-9	1.3 Hanford 105	1.4 LA Water Boiler	1.5 Chalk River-NRX	1.6 LA Fast
Core						
Fuel	U235	U235	U235	U235	U235	Pu
Enrichment,%	0.71	0.71	0.71	14.3	0.71	--
Fuel Mass,kg	320	20	1600	0.58	58	13
Diluent	Graphite	D_2O	Graphite	H_2O	D_2O	Fe
Diluent Mass,1000 kg	150	5	890	0.015	10	0.014
Coolant	Air	D_2O	H_2O	H_2O	H_2O	Hg
Volume Fraction Coolant,%	3	~100	0.52			33
Shape*	R	C	C	S	C	C
Dimensions,cm	457x457x457	92x183	500x670	15	132x223	7.6x15.2
Volume,liters	96,000	4,800	530,000	15	12,100	3
Radius of equi-volume sphere,cm	284	104	500	15	142	9
Reflector						
Material	Graphite	Graphite	Graphite	BeO	Graphite	U
Thickness,cm	92	61	61	46	84	20
Heat Output						
Total,1000 kw.	3.4	0.3	275	0.005	30	0.010
Specific,core,kw./liter	0.035	0.063	0.52	0.33	2.5	3.3
Specific,fuel,kw./kg	11	15	170	8.7	520	0.7
Specific,fuel element,kw/liter	1.5	2.0	23	0.33	70	10
Heat Removal						
Coolant Velocity,meters/sec	41	--	5.5	--	~4	
Inlet Temp., °C	25	25	30			
Outlet Temp.,°C	90	40	75	30		150
Max.pile or wall Temp., °C	245		280			
Max.heat transfer,cal/cm^2,sec.	0.45	0.70	12.8		25	0.23
References	M-3445 M-3554	M-3445 M-3554	M-3445 M-3554	M-3445 M-3554	M-3445 M-3554	M-3554

Notes: * R = rectangular parallelopiped
S = sphere
C = right circular cyclinder, the two dimensions given in this case are respectively radius and height

Enrichment = 0.71 indicates natural uranium piles

Table IVA-1

SUMMARY OF PILE DESIGN DATA 2. Pile Designs

	2.1 Clinton Hi-Flux	2.2 Daniels Power	2.3 Clinton-Navy	2.4 Zinn Fast	2.5 G.E. Power	2.6 Brook-haven
Core						
Fuel	U235	U235	U235	U235	U235	U235
Enrichment,%	100	30	100	100	100	0.71
Fuel Mass,kg	3	5.1	8.5	40	40	~500
Diluent	Al	BeO	Be	Fe	Be,Fe	Graphite
Diluent Mass,1000 kg	.09	2.8	0.16	~0.005	0.03	220
Coolant	H_2O	10 atm He	124 atm H_2O	Na-K	Na	Air
Volume Fraction Coolant,%	66	28	50	~45	40	3
Shape*	R	C	C	C	C	R
Dimensions,cm	40x40x60	62x114	30.5x61	9.9x19.8	23x51	~520x520x520
Volume,liters	96	1380	178	6.1	85	140,000
Radius of equi-volume sphere,cm	28.4	69.0	34.9	11.3	27.3	325
Reflector						
Material	radial Be ends H_2O	BeO	H_2O	U	Be;Be/U=1	Graphite
Thickness,cm	50	14.6	20	11.7	5;30	~60
Heat Output						
Total,1000 kw	30	12	66	1.0	20	30
Specific,core,kw/liter	312	8.7	370	125	240	~0.2
Specific,fuel,kw/kg	10,000	2350	7800	25	500	~60
Specific,fuel element,kw/liter	910	11	740	330	400	~8
Heat Removal						
Coolant Velocity,meters/sec	9.2		6.1	0.83		
Inlet Temp., °C	38	260	226	257		25
Outlet Temp., °C	55	760	248	340	350	200
Max.pile or wall Temp., °C	85	1080	270	450	400	
Max.heat transfer,cal/cm^2,sec	38	38		23	35	~2.5
References	M-3554 Mon T 433 Mon P 272	M-3554 Mon N 383	ORNL-22	M-3554 ANL-WHZ-IL2 LP-33, 33b	M-3554 LP-82	M-3554

Table IVA-1

SUMMARY OF PILE DESIGN DATA 3. Pile Suggestions

	3.1 Young	3.2 Young	3.3 Young	3.4 Moore-Navy	3.5 Clinton Power	3.6 Clinton Power
Core						
Fuel	U235	U235	U235	U235	U235	U235
Enrichment,%	100	100	100	30	30	30
Fuel Mass,Kg.	17.3	45.7	18.4	3.5	6.2	10.2
Diluent	Be	Be	Be	Be	BeO	BeO
Diluent Mass,1000 Kg	5.8	4.6	1.8	1.0	5.6	3.15
Coolant	50 atm He	50 atm He	50 atm He	60 atm He	40 atm He	40 atm He
Volume Fraction Coolant,%	50	60	37.5	20	46	53
Shape*	C	C	C	C	C	C
Dimensions,cm	103x190	103x190	65x121	49x92	86x156	75x134
Volume,liters	6340	6340	1600	700	3620	2360
Radius of equ-volume sphere,cm	115	115	72.5	54.8	95.3	82.6
Reflector						
Material	None	None	None	Be	BeO	BeO
Thickness,cm	0	0	0	25	43	37
Heat Output						
Total,1000 kw	617	743	164	57.5	250	250
Specific,core,kw/liter	97	117	102	83	69	106
Specific,fuel,kw/kg	35,600	16,300	8,900	16,400	40,300	24,500
Specific,fuel element,kw/liter	196	297	166	103	130	225
Heat Removal						
Coolant Velocity,meters/sec	145	145	138			
Inlet Temp., °C	300	300	300	232	260	260
Outlet Temp., °C	818	818	761	704	760	760
Max.pile or wall temp., °C	880	880	880	815	860	870
Max.heat transfer,cal/cm^2,sec	25	25	36			
Reference	Mon P-190 CF-3290	Mon P-190 CF-3290	Mon P-190 CF-3290	ORNL-CF-47-10-628	Mon N-383	Mon N-383

Table IVA-1

SUMMARY OF PILE DESIGN DATA 3. Pile Suggestions, Continued

	3.7 Clinton Power	3.8 Clinton Power	3.9 Clinton Power	3.10 Clinton Power	3.11 Clinton Power	3.12 Ethering-ton
Core						
Fuel	U235	U235	U235	U235	U235	U235
Enrichment,%	30	30	30	30	50	100
Fuel Mass,Kg.	6	8.7	15.8	3.5	15	8.0
Diluent	Be	Be	Graphite	BeO	BeO	Be
Diluent Mass,1000 Kg.	2.88	1.72	6.6		9.2	2.1
Coolant	40 atm He	40 atm He	40 atm He	40 atm He	10 atm He	10 atm He
Volume Fraction Coolant,%	40	50	40		25.8	43
Shape*	C	C	C	C	C	C
Dimensions,cm	77x140	69x125	105x189	58x104	91.5x168	70.1x130
Volume,liters	2610	1870	6560	1100	4420	2000
Radius of equi-volume sphere,cm	85.3	76.3	116	63.9	102	78.2
Reflector						
Material	Be	Be	Graphite	BeO	BeO,Graph.	Be
Thickness,cm	38	34	52	29	46	56
Heat Output						
Total,1000 kw	250	250	250	60	40	40
Specific,core,kw/liter	96	134	38	55	9.1	20
Specific,fuel,kw/kg	41,700	28,800	15,800	17,100	2,700	5,000
Specific,fuel element,kw/liter	160	268	65		12	35
Heat Removal						
Coolant Velocity,meters/sec					92	
Inlet Temp., °C	232	232	260	260	260	260
Outlet Temp., °C	594	594	760	760	760	760
Max.pile or wall temp., °C	810	810	790	830	950	920
Max.heat transfer,cal/cm^2,sec					8.3	8.1
References	Mon N-383	Mon N-383	Mon N-383	Mon N-383	Mon N-188 M-3554	M-4157

Table IVA-1

SUMMARY OF PILE DESIGN DATA 3. Pile Suggestions, Continued

	3.13 Etherington	3.14 Young	3.15 Young	3.16 Young	3.17 Soodak Fast	3.18 Menke Fast
Core						
Fuel	U235	U235	U235	U233	Pu	U233
Enrichment,%	100	100	100	100	--	100
Fuel Mass,Kg	4.7	8.4	40	18	125	46
Diluent	BeO	Be	Be	Be	U238	Fe
Diluent Mass,1000 Kg.	9.6	0.84	1.0	2.1	0.5	0.088
Coolant	10 atm He	Na	Na	Bi	Na	Bi
Volume Fraction Coolant,%	25.8	9.1	33.3	50	45	50
Shape*	C	C	C	C	S	S
Dimensions,cm	91x172	44x82	52x96	73.6x136	25.2	18.7
Volume,liters	4460	499	815	2320	67.0	27.4
Radius of equi-volume sphere,cm	102	49.2	57.9	82.2	25.2	18.7
Reflector						
Material	BeO	None	None	None	U	Be;U
Thickness,cm	45	0	0	0	30	3;27
Heat Output						
Total,1000 kw	40	100	500	255	125	34
Specific,core,kw/liter	9.0	200	613	110	1860	1240
Specific,fuel,kw/kg	8,500	12,000	12,500	14,200	1,000	740
Specific,fuel element,kw/liter	12	220	920	225	3,400	2,480
Heat Removal						
Coolant Velocity,meters/sec		10	10	2	10	3
Inlet Temp., °C	260	$\Delta T=333$	$\Delta T=333$	$\Delta T=232$	100	500
Outlet Temp., °C	760				300	715
Max.pile or wall temp., °C					570	
Max.heat transfer,cal/cm^2,sec	14	96	160	38		
References	M-4157	Mon P-190 Muc-GY-39	Mon P-190	Mon P-264	CF-3107	NDA-JRM-2, ORNL-71

[redacted]

Table IVA-1

SUMMARY OF PILE DESIGN DATA 3. Pile Suggestions, Continued

	3.19 Menke Fast	3.20 Menke Fast	3.21 G.E. Navy Thermal	3.22 Weinberg Breeder	3.23 Clinton-Navy	3.24 Clinton-Navy
Core						
Fuel	U233	U233	U235	U233	U235	U235
Enrichment,%	50	33.3	100	100	100	100
Fuel Mass,Kg.	97	157	15	11	8	14.8
Diluent	Fe	Fe	Be	Al,Th	Be	Be
Diluent Mass,1000 Kg.	0.325	0.751	1.3	0.44Th	0.17	0.17
Coolant	Bi	Bi	Na	80 atm H_2O	307 atm H_2O	307 atm H_2O
Volume Fraction Coolant,%	50	50	20	80	50	50
Shape*	S	S	C	C	C	C
Dimensions,cm	29.4	38.8	53x102	35x68	30.5x61	30.5x61
Volume,liters	106	245	900	262	179	179
Radius of equi-volume sphere,cm	29.4	38.8	59.9	39.7	35.0	35.0
Reflector						
Material	Be;U	Be;U	Be	Th	H_2O	radial Be; ends H_2O
Thickness, cm	3;27	3;27	30.5		20	20
Heat Output						
Total,1000 kw	107	208	45	98	55	55
Specific,core,kw/liter	1010	850	50	374	307	307
Specific,fuel,kw/kg	1,100	1,300	3,000	9,000	6,900	3,700
Specific,fuel element,kw/liter	2,020	1,700	250	1,870	614	614
Heat Removal						
Coolant Velocity,meters/sec	3	3		10		
Inlet Temp., °C	500	500	455	300	370	370
Outlet Temp., °C	715	715	540	308	400	400
Max.pile or wall temp., °C			594			
Max.heat transfer,cal/cm^2,sec			30	70		
References	NDA-JRM-2	NDA-JRM-2	M-3554 M-3557 A-4208	Mon P-93	ORNL-66	ORNL-66

Table IVA-1

SUMMARY OF PILE DESIGN DATA 3. Pile Suggestions, Concluded

	3.25 Clinton-Navy	3.26 Weinberg	3.27 NEPA Superponic
Core			
Fuel	U235	Pu	U235
Enrichment,%	100	--	90
Fuel Mass,Kg.	18.1	85	45
Diluent	Zr	Al	Graphite
Diluent Mass,1000 Kg.	0.57	0.92	2.1
Coolant	307 atm H_2O	CF_2	Air
Volume Fraction Coolant,%	50	64	30
Shape*	C	C	C
Dimensions,cm	30.5x61	45x167	66x133
Volume,liters	179	1060	1820
Radius of equi-volume sphere,cm	35.0	63.3	76
Reflector			
Material	radial Be ends H_2O	None	Graphite
Thickness,cm	20	0	end 50 radial 34
Heat Output			
Total,1000 kw	55	1,600	132
Specific,core,kw/liter	307	1510	73
Specific,fuel,kw/kg	3,000	19,000	2,900
Specific,fuel element,kw/liter	614	4,200	104
Heat Removal			
Coolant Velocity,meters/sec		10	
Inlet Temp., °C	370	50	480
Outlet Temp., °C	400	120	990
Max.pile or wall temp., °C		190	1430
Max.heat transfer,cal/cm^2,sec			19
References	ORNL-66	CF-2848	LP-78

Table IVA-6

PROPERTIES OF PURE METALS

USABLE AS COOLANTS

[redacted]

Element	Atomic Weight	Melting Point °C	Boiling Point °C	Specific Gravity*	Viscosity in Centipoises*	Vapor Pressure in Atmospheres*	Specific Heat in Cal/gm*	Thermal Conductivity in c.g.s. units*
Aluminum	26.97	660.2	2060	2.4		.0001 at 1110°	.25	
Bismuth	209.0	271.3	1420	10.04	.998 at 600°	.001 at 1008°	.03	.0418 at M.P.
Gallium	69.72	30	2070			.0001 at 1178°	.08	
Lead	207.21	327.4	1725	10.69	1.38 at 600°	.00002 at 1023°	037	.038 at 400°
Magnesium	24.32	650	1107	1.57		.001 at 608°	.28	
Phosphorus	30.98	17.0	280		1.21 at 80°			
Rubidium	85.48	39	705			.00055 at 353°	.091	
Sodium	22.997	97.7	892	.93	.71	.01 at 534°	.32	0.32
Sulfur	32.06	119.0	444.6	1.81	18600 at 220°		.22	
Tin	118.70	231.9	2270	6.96	1.20 at 500°	0.1 at 1932°	.076	

*Pertains to liquid element

[redacted]

Volumetric Expansion %	% Shrinkage on Solidification	Heat of Fusion Cal/gm.	Electrical Conductivity micro-ohm/cm.*	Thermal Neutron-Capture Cross Section in barns	Neutron-Scattering Cross Section in barns	Induced Radioactivity
	6.6	94.6	20.1 at 660°C	.23	1.5	Worse than magnesium
12.5×10^{-6}		12.6	139.9 at 500°	.016	9.2	Negligible radiation; suitable for both manned and tug-tow planes
		19.2		2.3	18	
	4	11.9	107.2 at 600°	.4	13	Negligible radiation; suitable for both manned and tug-tow
	4	89		.06	3.6	Not usable in manned aircraft. Marginal in tug-tow
		5.0		.30	10.0	Excellent; no gammas emitted in thermal capture
			19.6 at 140°	.5	11.0	
71×10^{-6}		27.5	10.3 at 140°	.48	3.5	Not usable in manned aircraft or thermal reactor in tug-tow. Marginal for fast reactor in tug-tow
		13.2		.43	1.5	
	2.7	14.0	61.2	.54	5.0	Not usable in manned aircraft. Suitable in tug-tow

TABLE IVA-9

SOME HYPOTHETICAL REACTOR COMBINATIONS

FROM THE STANDPOINT OF MATERIALS

CODE

(1) Recent work has shown that Be mixed with BeO has good strength and ductibility at 1150°C.

(2) In piles of this size graphite is an ineffectual moderator and acts mostly as a structural material.

(3) It is hoped that enough Be_2C can be added to give some moderation without losing the good structural properties of graphite. The uranium investment would be lower than for pure graphite.

(4) This alloy is 35-15 carburizing box grade. It is both oxidation and carburization resistant.

(5) The molybdenum would be coated with silicide for oxidation resistance in the open cycle.

(6) Pu oxide is supposed to be stable in air at high temp. Nothing is known of dilute Pu alloys.

(7) Fuel investment is lower with compressed He than with air because smaller free-flow ratio is required.

(8) Extensive alloying between Be and Bi does not occur. Zr forms liquid with Be at about 1000°C and hence cannot be used.

(9) A layer of pure Be at surface of cooling tubes to keep fission fragments out of the gas or liquid metal.

(10) Bi dissolves C to a small extent.

(11) Pure graphite pitch impregnated at the cooling surfaces might prevent diffusion of fission fragments into gas.

(12) In a fast reactor Pu can be used in which event less of it will be required than U^{235}.

LIST OF QUESTIONABLE FEATURES

A - Can moderator be shaped so that heat can be removed without melting Be in interior?

B - Evaporation of Be.

C - Extent of solubility of Be in Bi.

D - Radiation damage to fuel tube.

E - Cracking of BeO due to thermal stresses.

F - Diffusion of fission fragments thru graphite.

G - Radiation damage to the Be.

H - Diffusion of fission fragments thru fuel tube walls.

I - Possible alloying between Be and fuel tube.

J - Carburization of the coolant tubes.

K - Extent of solubility of tube alloy in Bi.

L - Effect of rate of radiation damage on the graphite.

		I	II	III	IV	V	VI	VII	VIII	IX	X	XI	XII	XIII	XIV	XV	XVI	XVII
Neutron Energy	Thermal		x	x	x					x					x	x	x	
	Intermediate					x	x	x	x		x	x	x	x	x	x		
	Fast																	x
Moderator Nos. give dia. and length of core in ft.	Be or Be-BeO (1) or Be-Be_2C		3	3	3										2	2	2	1 foot - No moderator
	BeO					3	3	3	3	3								
	Graphite (2)	4																
	Graphite-Be_2C (3)										3	3	3	3				
Coolant	Air	x	x			x			x		x							
	High press. He(7)			x			x	x				x		x	x	x		
	Bi or Bi-Pb Alloy				x					x			x				x	x
Distribution of fuel	In moderator	x		x	x						x	x	x	x	x		x	
	In sandwich tubes or sheets		x			x	x	x	x	x						x		x
Material for Coatings or Fuel Tubes	Fe, Cr, Ni, Si Alloy (4)	x		None (9)	None (8) (9)						x	x	(10) x	None (11)	None		None (8) (9)	
	Stainless Steel		x			x	x			x						x		x
	Molybdenum (5)							x	x									
Fuel Material	UO_2 or UC_2	x									x	x	x	x				
	U Alloy		x	x	x	x	x	x	x	x					x	x	x	Pu(12)
Maximum Wall Temp.	Centigrade	1000	1000	1000	1000	1000	1000	1400	1400	1000	1000	1000	1000	1000	1000	1000	1000	1000
Fuel Invest. in Kg.	U^{235}	150-300	30-80	20-50	10-20	30-80	25-60	30-70	40-100	15-30	100-200	70-140	30-60	50-100	80-150	90-170	40-80	Pu 60-70
Questionable Features		J	D	A	A	D	D	D	D	D	J	J	J	F	A	D	C	D
		L	G	B	C	E	E	E	E	E	L	L	K	L	G	H	G	H
			I	G	G	H	H	H	H	H			L			I		K

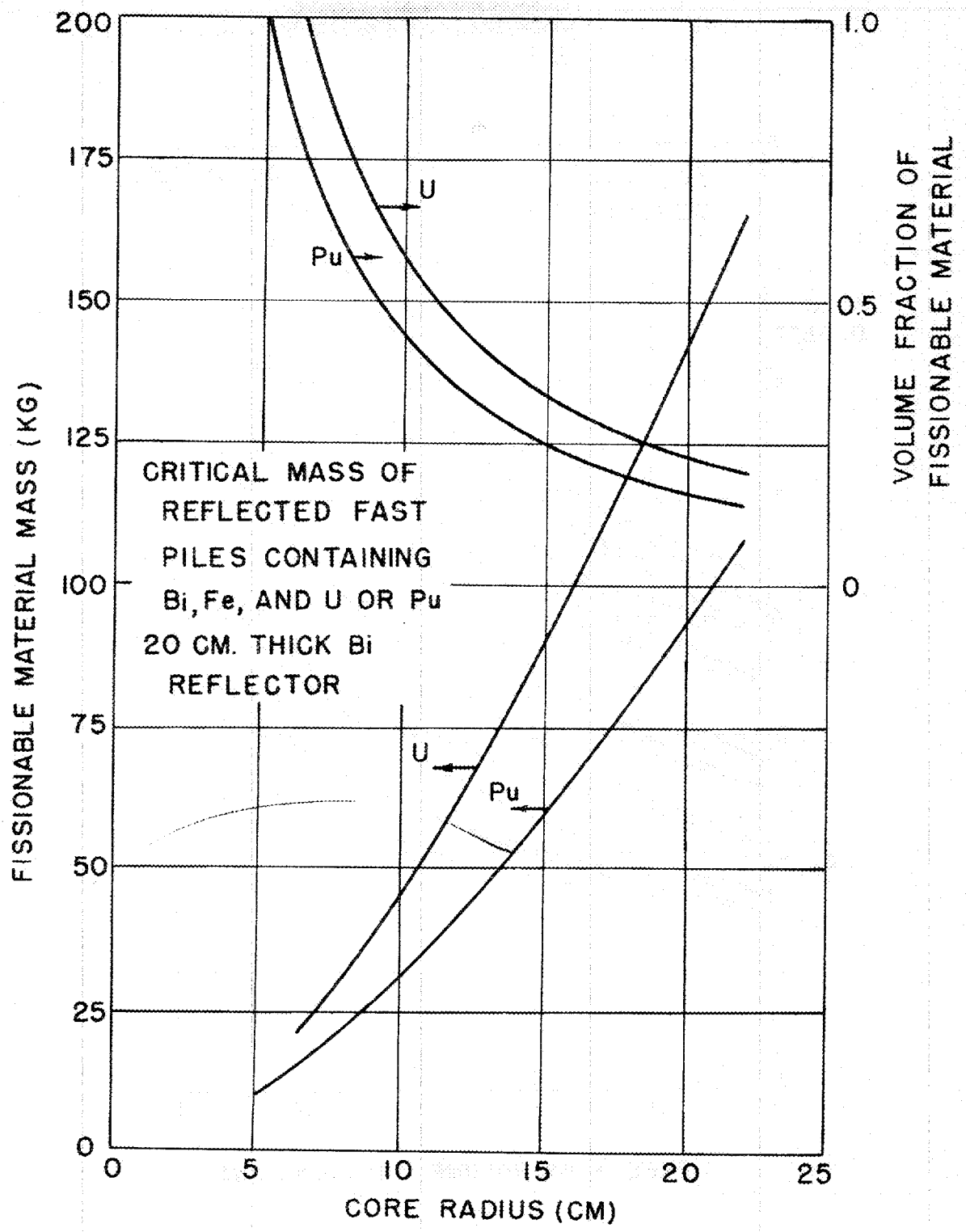

200
175
150
125
100
75
50
25
0
1.0
0.5
0
0
5
10
15
20
25
FISSIONABLE MATERIAL MASS (KG)
VOLUME FRACTION OF FISSIONABLE MATERIAL
CORE RADIUS (CM)
U
Pu
U
Pu
CRITICAL MASS OF REFLECTED FAST PILES CONTAINING Bi, Fe, AND U OR Pu
20 CM. THICK Bi REFLECTOR

FIGURE
IV-A-1

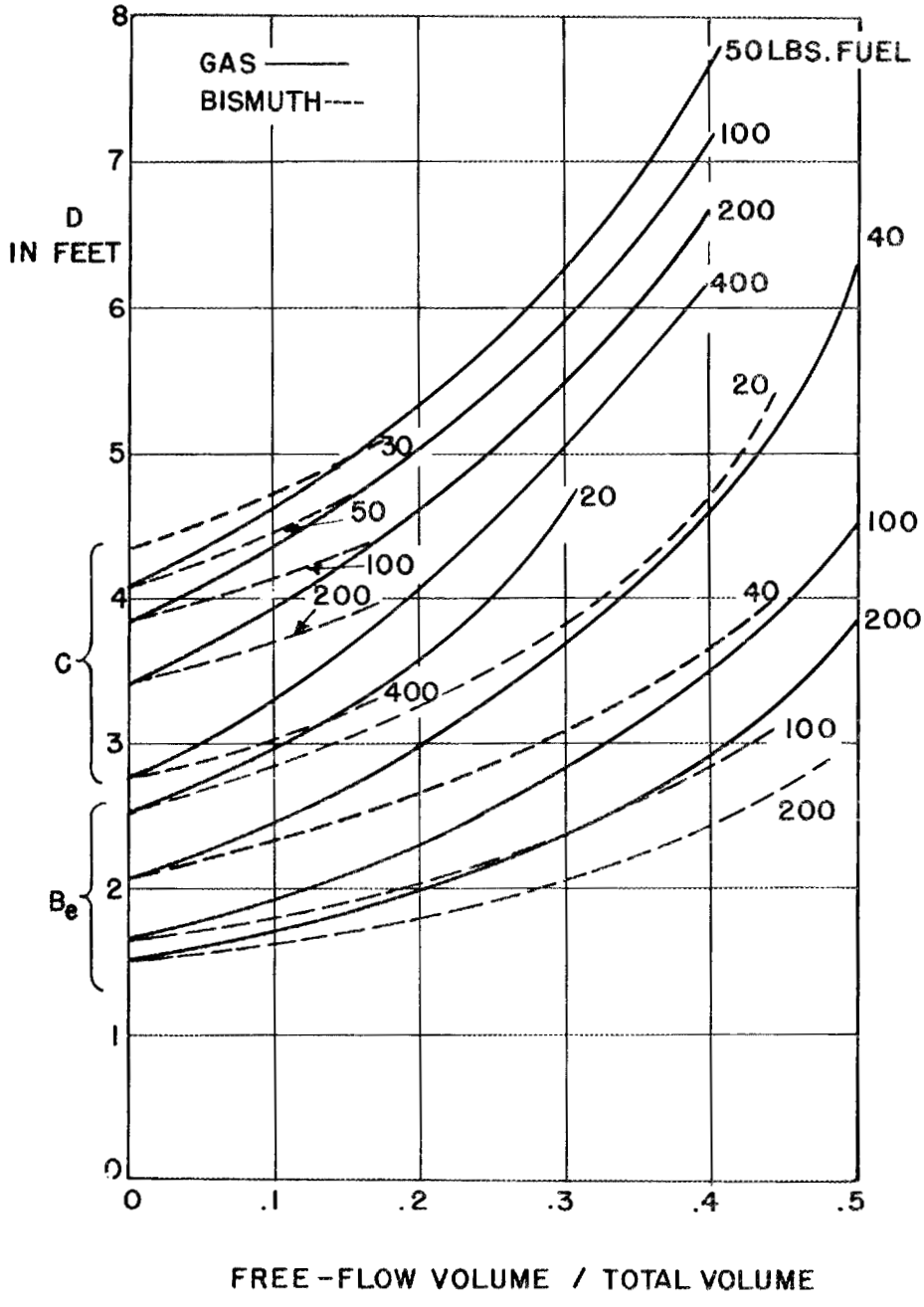

GAS
BISMUTH
D
IN FEET
C
Be
50 LBS. FUEL
100
200
400
40
20
30
50
100
200
20
40
400
100
200
100
200
0
1
2
3
4
5
6
7
8
0
.1
.2
.3
.4
.5
FREE-FLOW VOLUME / TOTAL VOLUME

FIGURE
IV-A-2

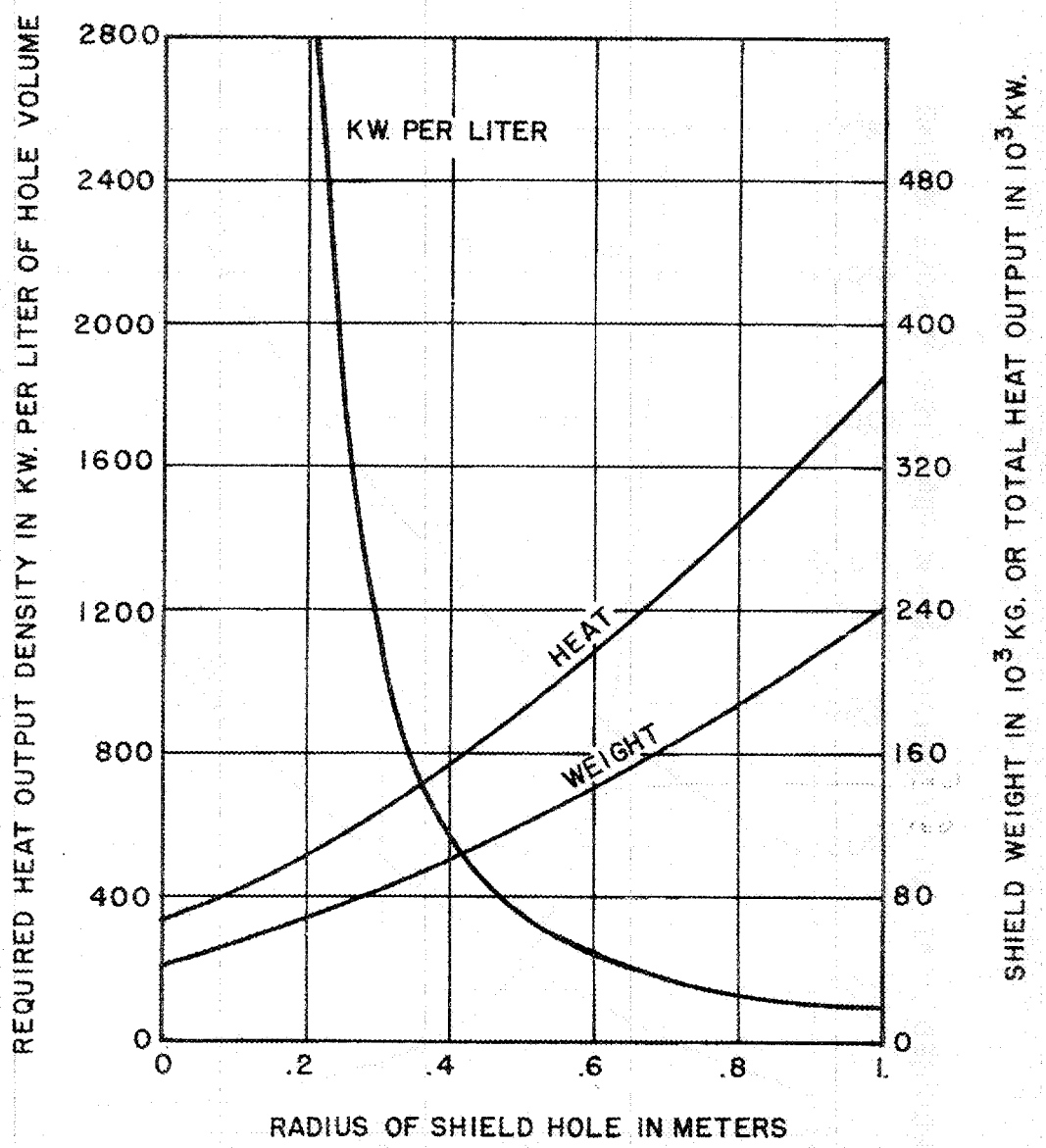


FIGURE IV-A-3

MANNED PLANE

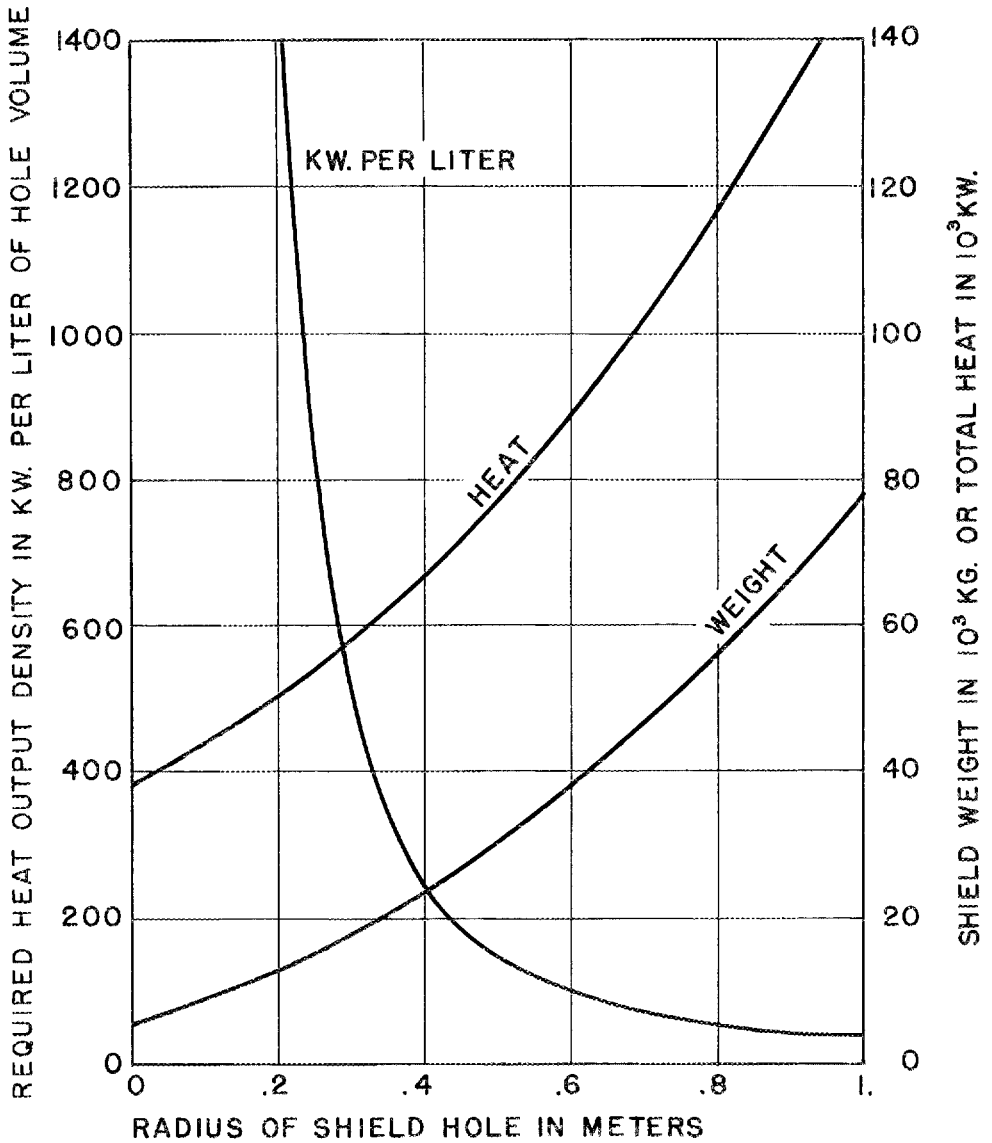
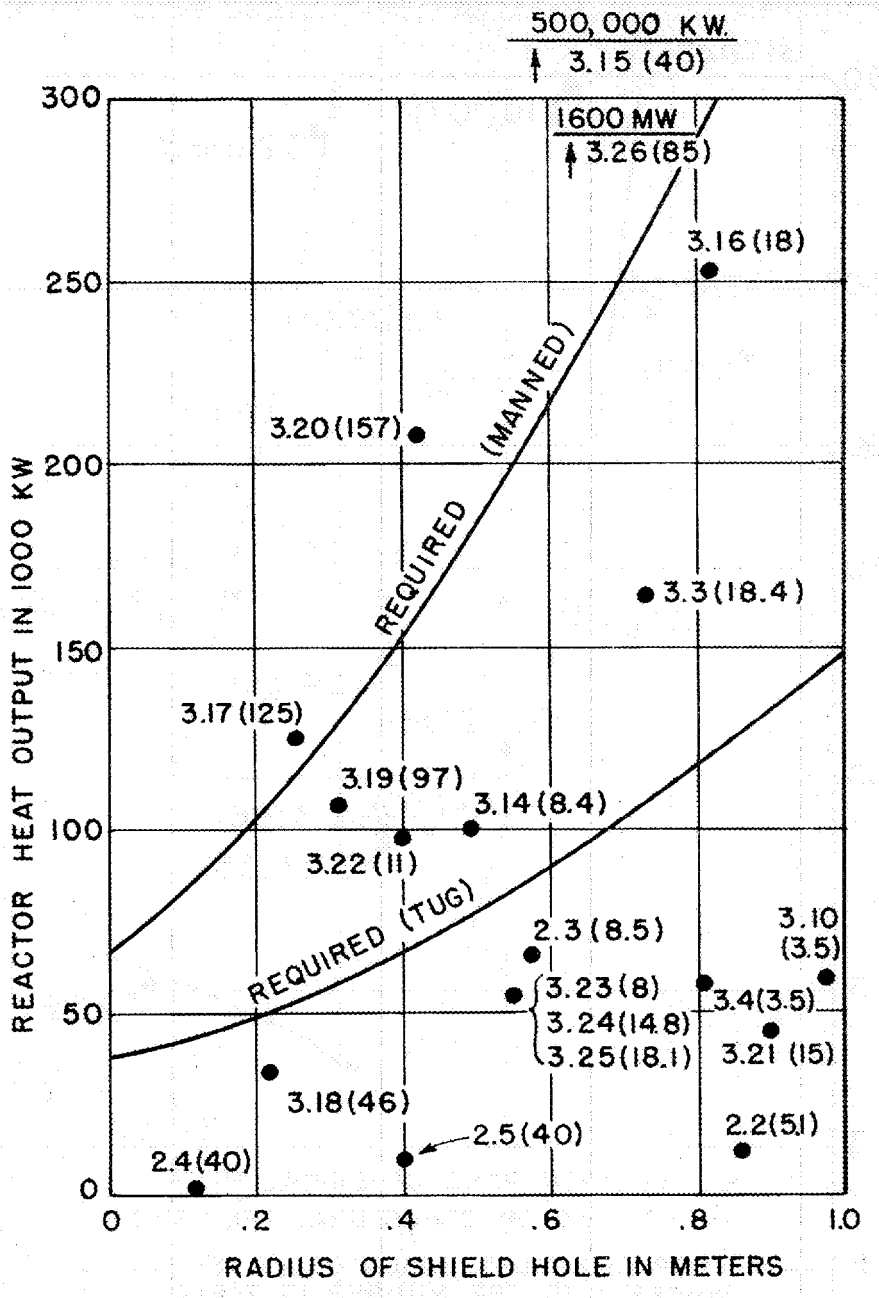


FIGURE IV-A-4

TUG PLANE

[redacted]



LEGEND:– EACH PILE IS INDICATED BY ● A (B) WHERE A IS NUMBER OF PILE IN TABLE I; B IS FISSIONABLE MATERIAL MASS (KG.)

FIGURE IV-A-5 ESTIMATED HEAT OUTPUT OF VARIOUS REACTORS

[redacted]

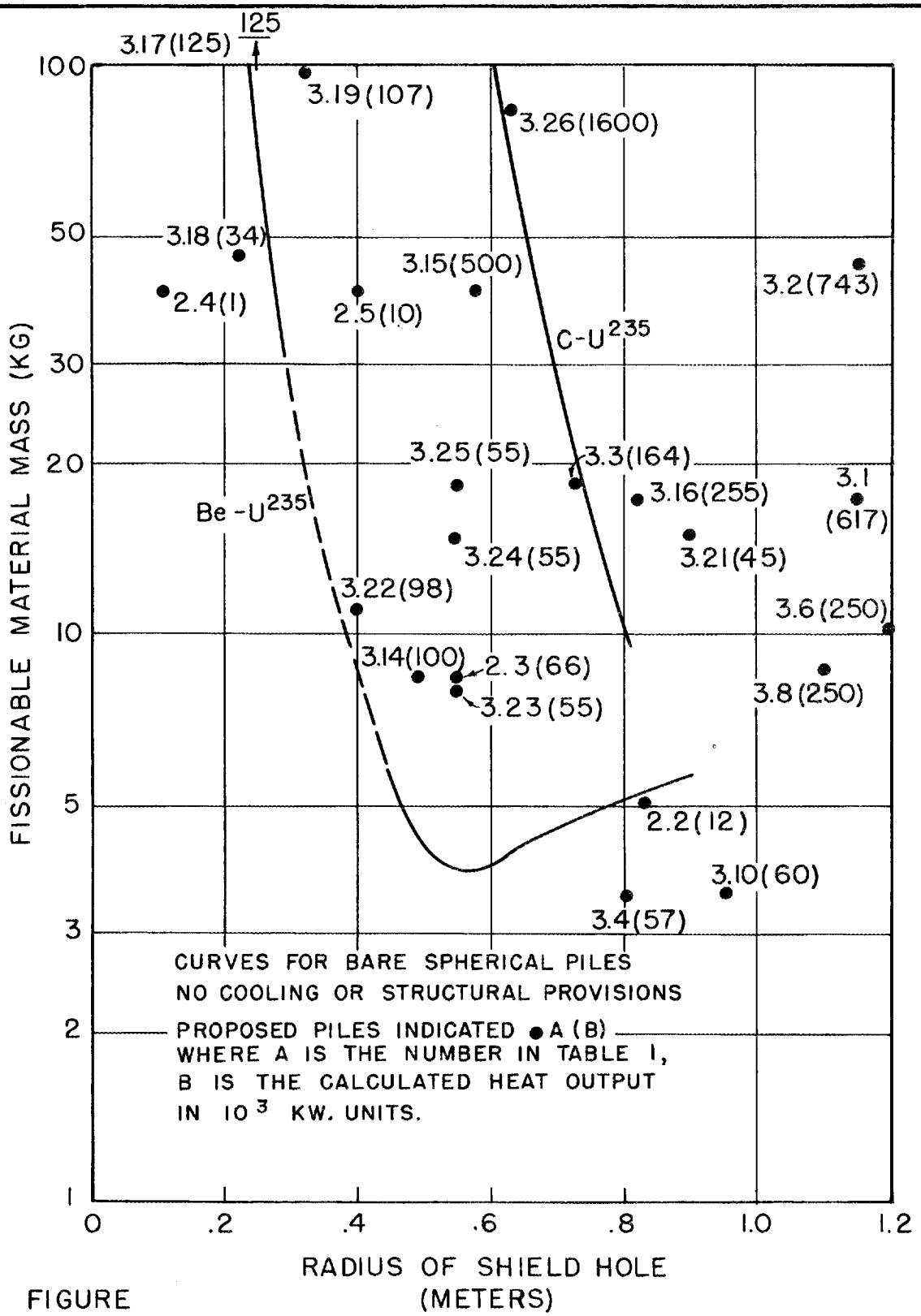

3.17(125)
125
3.19(107)
3.26(1600)
3.18(34)
2.4(1)
2.5(10)
3.15(500)
3.2(743)
C-U235
3.25(55)
3.3(164)
3.16(255)
3.1
(617)
Be-U235
3.24(55)
3.21(45)
3.22(98)
3.6(250)
3.14(100)
2.3(66)
3.23(55)
3.8(250)
2.2(12)
3.10(60)
3.4(57)
CURVES FOR BARE SPHERICAL PILES
NO COOLING OR STRUCTURAL PROVISIONS
PROPOSED PILES INDICATED ● A (B)
WHERE A IS THE NUMBER IN TABLE I,
B IS THE CALCULATED HEAT OUTPUT
IN 10³ KW. UNITS.
FISSIONABLE MATERIAL MASS (KG)
100
50
30
20
10
5
3
2
1
0
.2
.4
.6
.8
1.0
1.2
RADIUS OF SHIELD HOLE
(METERS)

FIGURE
IV-A-6

[redacted]

Chapter IV

PART B - THE SHIELD

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[redacted]

[redacted]

Chapter IV

THE REACTOR AND THE SHIELD

PART B - THE SHIELD

INTRODUCTION

The shield problem will be broken into four parts: (a) the production of neutrons and of gamma rays (γ-rays), (b) the tolerance of various components for neutrons and gamma rays, (c) the attenuation of neutron and gamma ray fluxes, and (d) the design of the shield. The first two parts combine to give the shield requirements.

General references on the shield problem (1)(2) may be consulted to fill in the reasons for selecting only neutrons and gamma rays in this consideration. (These two general references plus the best bibliography (3) and the most recent analytic work (4) give a reasonably complete picture of present knowledge in the shielding art.)

The importance of the gamma rays of 2 to 5 Mev. energy is emphasized in Reference (2), p. 9, and the importance of neutrons about 1 Mev. and above is considered there (p. 12). The gamma rays in the 2 to 5 Mev. region will be called "bad" gamma rays in this chapter; the neutrons of about 1 Mev. and above are called fast neutrons.

For mobile power units and especially for aircraft, a high premium is placed on light weight for the shield. This consideration does not appear in the general shielding analysis until near the end, but it strongly influences the last part of the shielding section of this report.

REQUIREMENTS

The Production of Neutrons and of Gamma Rays: Fluxes

The production of neutrons is almost entirely confined to the reactor itself. A small neutron production by (n, 2n) or (γ,n) reactions outside the reactor may take place in exceptional circumstances, and a more significant production may take place in a reflecting shield (tamper) containing fissile material -- uranium or uranium hydride for example. If used at all, such a shield would almost certainly be used close around the reactor, and the neutron and power generation may be lumped with that of the reactor as a first calculation.

The production of gamma rays is likely to be spread over a much larger region. We shall separate the gamma ray production somewhat arbitrarily into primary and secondary, that taking place in the reactor and that taking place elsewhere. This separation is desirable for the subsequent estimation of shielding requirements.

[redacted]

The primary gamma rays, originating in the reactor, come from four sources:

(a) "directly" in the fission process ("prompt gammas"),

(b) from the radioactive fission products,

(c) from the inelastic scattering of neutrons and (n, γ) reactions in the reactor, and

(d) from radioactive atoms created by neutron capture in the reactor.

The secondary gamma rays, originating outside the reactor, come from two sources:

(a) the inelastic scattering of neutrons and (n, γ) reactions and

(b) radioactive atoms created by neutron capture.

During operation of the reactor the activated atoms make only a small contribution to the radiation. After operation ceases, they and the fission products are the only sources of radiation remaining.

Using the fundamental knowledge of the fission process and some estimates of (n, γ) reactions, Table IVB-1 showing the production of neutrons and gamma rays in the reactor is constructed. This analysis applies crudely to any reactor. The power is a multiplicative factor, and the unit of power used here is 100,000 kilowatts.

TABLE IVB-1

PRODUCTION OF NEUTRONS AND BAD GAMMA RAYS IN THE REACTOR

(10^5 KW.)

Radiation	Number per Second
Fast Neutrons - emitted at fission	1×10^{19}
Primary Gamma Rays	
Prompt - (total = 1.5×10^{19} per sec.) " - bad (1/3 of total)	0.5×10^{19}
From fission products (total = 3×10^{19}) " " " - bad (1/3 of total)	$\sim 1 \times 10^{19}$
From (n, γ) reactions - bad	$\sim 1 \times 10^{19}$

The total amount of the primary bad gammas listed above may, for practical purposes, be regarded as 1×10^{19}.

The result may be summarized as 10^{19} neutrons and 10^{19} bad gamma rays emitted per second in the reactor at a power level of 100,000 kilowatts. The accuracy is probably quite adequate since greater uncertainties are introduced in the final calculation of the shielding requirements by the lack of precise information on tolerances of men and equipment to radiation and by the uncertainties in estimating the attenuation factors which will be obtained with shields of definite weights or thicknesses.

In applying these results to a shielding problem, the specific characteristics of the reactor must be considered. The spatial distribution of the neutron and gamma ray sources within the reactor and the composition of the reactor itself make a difference in the fractions of the neutrons and of the gamma rays which escape in the "bad" energy ranges. The reactor characteristics are of even greater significance in finding the numbers of neutrons and gamma rays which escape in other energy ranges.

For example, about half the neutrons produced will be absorbed in the reactor; and in order for this to be true in a thermal or epithermal (slightly above thermal) reactor much more than half of the total neutron production must be slowed down to energy ranges which are not dangerous from the shielding point of view. The quantity 10^{19} neutrons per second represents the neutron production and not the number of neutrons which get out of the reactor (the fission neutron outflux). The shielding calculation must proceed either by computing the attenuation from the sources taken at their real positions or by computing the fast neutron outflux from the reactor, using the reactor characteristics, and then computing their attenuation through the shield. No single ratio of fast outflux to fission neutron production can be given, but a typical number for a small reactor -- still not a "fast" pile -- might be

$$\frac{\text{fast neutron outflux}}{\text{fission neutron production}} = \frac{1}{10}$$

Similarly, the ratio of bad gamma outflux to bad gamma production in the reactor depends quite largely not only on the size but also on the density of the pile. Again no typical number can be given, but the ratio between outflux and production for a pile of density 2 grams/c.c. is likely to be somewhat greater for the gammas than for the fission neutrons; perhaps the ratio may be taken as 1/3. (See also (5).)

Using Table IVB-1 and the ratios indicated in the last paragraph, one can calculate the fluxes of fast neutrons and of bad gamma rays at various distances from the center of an unshielded reactor. For example, a production of 10^{19} neutrons per second from a 100,000 kw. reactor gives a 10^{18} outflux of fast neutrons, which at a distance of 1 meter is distributed over about 10^5 sq. cm. to give a fast neutron flux of 10^{13} per sq. cm. per sec. Table IVB-2 gives fluxes at various distances. It includes a column

of energy outflow in kilowatts, calculated by estimating fractions of the total power, using the fundamental data on neutrons and gamma rays. (See also (6) and (7).)

TABLE IVB-2

FLUXES OF FAST NEUTRONS AND BAD GAMMA RAYS

FROM A BARE REACTOR (10^5 KW.)*

Distance in meters	1	10	100	1000*	Energy Outflow (in Kw.)
Fast neutrons, no./sq.cm.-sec.	10^{13}	10^{11}	10^9	2.5×10^7	400
Bad gamma rays no./sq.cm.-sec.	3×10^{13}	3×10^{11}	3×10^9	1.5×10^7	1200

*For a discussion of the departures from the $1/r^2$ law see reference (7). The altitude is assumed to be 30,000 ft. and the bad gamma rays mostly about 2 Mev.

For shielded reactors these numbers are the essential ones because of the high penetration of the neutrons and gamma rays listed. For an unshielded reactor, fluxes in other energy ranges should also be considered. It is hoped that these estimates may help in general orientation. Specific calculations will be given for definite reactors and shields in a later section.

It should perhaps be noted here that the apparent attenuations which are measured by detectors placed in shields are likely to be greater than the attenuations obtained by the type of calculation here outlined. The reason is that many neutrons and gamma rays come out of the reactor in energy regions which are not significant for the thick-shield attenuation calculation. These neutrons and gamma rays, on the other hand, contribute to the measured intensity nearer the inside face of the shield.

The secondary gamma ray production problem is tied up in detail with the exact design of the shield. It cannot be treated here and will be considered later with the specific shield designs.

Neutron and Gamma Ray Tolerances

In considering both manned and unmanned nuclear-powered aircraft to carry a payload, there are quite a number of sensitive components that should be evaluated. Of these man is probably the most sensitive component of large size.

The picture presented here of the radiation tolerances of man is the result of discussion with several radiologists and biologists. Details of the discussion are found in references (8-11).

The gamma ray tolerances of man are usually determined in Roentgen units, abbreviated here as r. One r corresponds to approximately 10^9 gamma rays/sq.cm. in the energy range of the so-called bad gamma rays. The effect of fast neutrons on man can be compared with gamma ray effects by calculating the volume density of ionization and correcting for the effects of a higher linear ionization density along the tracks of the recoil protons created in the man's body. Biological experiments show that the effectiveness of fast neutrons in causing biological damage is about 5 to 10 times higher than that of gamma rays when both are measured in terms of the volume density of ionization. To give an example, the net flux of fast neutrons of 2 Mev. which is assumed to be biologically equivalent to an r of gamma radiation is about 5×10^7 neutrons/sq.cm., and the ratio of biological effectiveness in terms of flux is approximately 20. (The number of 2 Mev. gamma rays which produce a given physiological effect divided by the number of 2 Mev. neutrons producing the same effect equals twenty.)

The equivalences assumed are given in Table IVB-3.

TABLE IVB-3

ROENTGEN EQUIVALENTS FOR MAN

Radiation		No./sq.cm. equivalent to 1 r
Bad gamma rays (about 2 Mev.)		10^9
Neutrons	2 Mev.	5×10^7
	1 Mev.	6.5×10^7
	0.5 Mev.	9×10^7
	thermal	4.3×10^9

These numbers are crude but will serve for present purposes. They are to be used in conjunction with the tolerances of man in terms of r.

An outline of the tolerances of man in terms of r is given in Table IVB-4. Since it is useless to carry a man who will not be functional, the first two numbers indicate an extreme upper limit for tolerable radiation; the table goes on through numbers to indicate lethal effects, then probable nausea and a prolonged period of weakness, and finally, some numbers which may indicate freedom from all short-run effects. Further discussion is given below the summarizing table.

TABLE IVB-4

TOLERANCES FOR MAN

(Full Body Radiation)

Characteristics	Time of Exposure	Dose in r
Unable to complete mission	10 hr	>2,000
Able to complete mission	10 hr	1,000
Certain death within a few weeks after mission	any	> 800
50% mortality within a few weeks	"long" "short"	550 450
No deaths within a few weeks	any	300
Probable nausea during mission; weakness for many weeks; 25% incapacitated for several months	20 hr.	> 200
Nausea during or after flight unlikely except from psychological origin	any	100
No noticeable permanent effects; possible temporary psychological ones, and possible slight temporary lymphopenia. Effects on life expectation arguable.	any	25

Genetic effects have been omitted from Table IVB-4. This omission is the result of several considerations: (a) it is assumed that very few people are subjected to the high-radiation doses involved here; (b) 25 r is only approximately three times the dose received from cosmic rays and other background radiation in the normal genetic lifetime (8)(9 - p. 13, ff.)(10)(12), and cosmic rays apparently are too little to account for the normal mutation rate (9); (c) between 50 and 100 r, the spontaneous mutation rate is doubled (8), and (d) experts disagree on the seriousness of the effects to be expected. Further evaluation of genetic effects may be desirable. In this regard, as in general, reports of the board of medical consultants for NEPA are expected to prove of value.

Effects on life expectation in the range around 25 r are undemonstrated. This fact alone shows that they can not be great, and should probably be disregarded in any real emergency unless new information to the contrary becomes available.

data on tolerances to continued radiation is omitted as irrelevant. It is available in many places.

The large factor between the highest and the lowest total dose listed, 2000 and 25 r, does not imply a similar range in shield thickness requirements. The ratio between 1000 r (the highest dose which it is estimated might still permit a successful suicide mission by an enemy) and 25 r (which it will be argued is likely to be in the range of normal consideration) is 40. The estimates of attenuations required for a manned nuclear aircraft lie in the range of 10^8. Consequently by subjecting the aircraft crew to the greatest possible radiation dose the needed attenuation would fall only to the range of 10^6. Since shield thickness is believed to be closely proportional to the exponent -- not to the attenuation factor -- this implies that the thickness is not reduced by more than 25% between "normal" and extreme radiation doses. Calculation then shows that the shield weight around the reactor may be halved by adopting this lethal measure.

The same reasoning can be applied to other cases. Thus the saving in thickness in going from "normal" (25 r) to barely sublethal doses (300 r) is expected to be about 13½% and the saving in weight not more than 35%. Taking advantage of such a saving would appear extremely dangerous since an accidental increase in the dose rate or an additional exposure would quickly place one in the range in which 50% of the population (aircraft crew) would die.

Again we note that the difference between giving the crew the "normal" 25 r and what may be a really safe civilian dose of about 5 r costs only 8% in shield thickness and less than 20% in additional shield weight for a manned aircraft. (These estimates in the preceding three paragraphs may have to be modified because of refinements like shadow shielding. They are not expected to undergo gross modification.)

It may be that changes in shield weight comparable to those mentioned in the foregoing paragraphs will turn out to be important for the feasibility of manned aircraft, but they do not change the order of magnitude as a first glance at Table IVB-4 might seem to indicate.

It should be clear from the character of the estimates -- and their uncertainties multiply as they are carried further -- that no design built on present data should be trusted radiation-wise to give the predicted dose within a factor of 4. Therefore it is impossible at the present time to recommend design dosages above 50 r, even though the most thorough job of eliminating unexpected radiation sources and leaks be done. When the possibility of accidents and the uncertainty in biological knowledge at high sublethal doses are considered, even 25 r may be accepted with reluctance. Under the most optimistic assumptions, 25 r per flight would leave room for no more than 8 flights -- test, training mission -- and no accidents in a crew man's life (10). 25 r is used here as the normal tolerance of man for the military aircraft crew, but it is used with the realization that it may have to be revised as

knowledge increases, emergencies threaten, or as mastery of the problem makes greater safety either available or requisite. 25 r is merely used as the basis for the present discussion.

In considering man as a radiation-susceptible component, the question of possible selection or treatment for greater radiation resistance arises. Past attempts at evaluating this possibility are inconclusive because any effects of physiological selection or pretreatment for the radiations important here have apparently been obscured by psychological effects. (These radiations are penetrating and skin effects are not the question.) Although this problem is apparently complex and unpromising, it has but recently become important; further advice and work are needed (8)(10)(11).

In addition to man, there are other components sensitive to radiation. Table IVB-5 lists estimates of the radiation tolerances of several components. Some of the tolerances in Table IVB-5 are quoted in terms of the total dose, some in terms of rates, depending upon which is more fundamental. The fundamental rates may be converted to total doses or the total doses into rates where the length of the flight is known. Throughout the shielding section of this report, 25 hours is assumed to be the time of a mission. On this assumption, a factor of 1.1×10^{-5} will convert tolerable total doses into tolerable rates.

The status of the 25-hour value for man has been discussed above. Many points illustrated in the case of man apply similarly to other components. There are abundant uncertainties, but on the other hand, the shield thickness is sensitive to the order of magnitude (exponent) of the required attenuation rather than to the attenuation factor itself. Further remarks on the other components follow:

1. Aerial photographic film as listed here refers to film currently used. The gamma ray tolerance in this case is quite accurately determined but the neutron tolerance estimated may be in error even as to order of magnitude (13). The possibility of special film for use in nuclear aircraft should be investigated.

Despite the high sensitivity to gamma rays, the relatively small size of film as compared to men or reactors indicates that specific shielding for the film should not prove too difficult; perhaps a few tons may suffice (12)(13).

2. The tolerance for the payload is based on the present-day atomic bomb. The uncertainty in the estimate may be of the order of a factor of 4, and future developments which might change the tolerance to fast neutrons should be considered (14)(15).

Payloads other than the atomic bomb may be significant; "bombs" for radioactive or biological warfare may serve as examples.

3. The tolerance of guidance devices is based on that of star-tracking systems. Figures are very uncertain. Nuclear power

TABLE IVB-5

RADIATION TOLERANCES OF VARIOUS COMPONENTS

Component	r		Fast Neutrons/cm^2 (2 Mev.)	Bad Gamma Rays/cm^2 (2 Mev.)	Energy Outflow (kw.)
Man	25		1.3×10^9	2.5×10^{10}	
Aerial Photographic Film	0.1	for gamma rays	1.3×10^9	5×10^7	
	25	for neutrons (type of r used for man)*			
Payload			10^8 per sec.	10^{11} per sec.	
Guidance Devices				10^{10} per sec.	2
Radar Signals	See text and especially (16)				
Radar Transmitter					2
Electronic Equipment			10^{11} per sec.	10^{11} per sec.	
Lubricants: Oils,	10^7	ordinary**	2.7×10^{15}	10^{16}	
Gasolines, etc.	10^8	selected **(aromatic and fluoro-compounds)	2.7×10^{16}	10^{17}	
High Explosives	10^7	on some explosives	2.7×10^{15}	10^{16}	

* 25 r of the type used for man is equivalent to 5 r measured by volume ionization.

** Gamma rays and neutrons, measured in terms of volume ionization (not type of r used for man) probably give the same order of magnitude here.

certainly makes guidance more difficult. The limiting factors are the response of the photosensitive surface to gamma radiation, and stray light from the cloud of ionization around the plane (16)(17). Except for an experiment showing the failure of a photocell under high fluxes (18), no experimental work is available.

4. Propagating a radar signal seems feasible at high frequencies (16)(17).

5. For radar transmission the limiting factor lies in the high-voltage breakdown due to ionization; the figure is very uncertain (16)(17). Experimental work is lacking and seems called for (16).

6. Some electronic components which are generally used cannot stand the quoted intensities. However, it is believed that substitutes can be found that will stand up under the stated tolerances (16)(17)(18).

Experimental work ((17) and especially (18)) has been used (16)(17) in making the estimates quoted; a further experimental and theoretical program embracing guidance, radar and radio, and general electronics is suggested.

7. The lubricant estimates have some experimental basis (19)(20)(21) but are largely theoretical (21)(22). They may be revised upward (by not more than a factor of ten) or downwards when more realistic experiments (at appropriate temperature and pressure, for example) are performed. Further experimental planning appears needed (21).

8. The explosives situation seems similar to the one for lubricants (21)(23).

The results listed in Table IVB-5 imply that for a manned aircraft, man and aerial photographic film determine the necessary attentuation factor; for a tug-tow, the critical items are man, film, and intercommunication; and for a pilotless craft (carrying no film), probably guidance devices and payload determine the shielding.

Estimate of Needed Attenuation Factors

In order to make the previous figures on production and tolerance more meaningful, a crude estimate is given here of the attenuation of primary neutrons and primary gamma rays needed to reduce the radiation from a 400,000 kw. reactor to a tolerance of 1 r per hour, a value in the range for man. The 400,000 kw. rating appears at present to be reasonable for a reactor shielded to carry a man -- assuming that any such vehicle can be built and operated. 1 r per hour corresponds to about 25 r for a mission as conceived at present. It is also assumed that the man can be 10 m removed from the center of the reactor.

From Table IVB-2, the primary bad gamma rays passing at 10 meters distance is 1.2×10^{12}. Using the tolerance value given in Table IVB-5

($2.5 \times 10^{10} \times 1.1 \times 10^{-5}$), the needed reduction factor would be estimated as 4.3×10^{6}. Care should be taken in the use of this attenuation factor. It accounts only for the gamma rays in the specific group considered. A primary gamma ray reduction of this amount would not be sufficient. In addition to the primaries penetrating the shield, a certain number of gamma rays degraded in energy will emerge from the shield. The problem of allowing for these degraded gamma rays can be handled in various ways (see section on Primary Neutron Attenuations). If the attenuation factor is specified in such a way that the degraded gamma rays will not need independent consideration, we must multiply the above result by a factor of the order of 3, and perhaps even 10. In these terms the needed attenuation factor is 1.3×10^{7} or 4.3×10^{7}. Since the build-up of degraded gamma rays will be allowed for later in this treatment, the number 4.3×10^{6} is used as the attenuation requirement.

Using the same procedure, an estimate of the needed attenuation for fast neutrons can be made. Since the fast neutrons are about 20 times more dangerous biologically, one for one, than the bad gamma rays, and since the ratio of outfluxes to production, as stated earlier, is likely to be about 1/10 for the neutrons instead of the 1/3 used for the gamma rays, the comparable attenuation for the fast neutrons is 3×10^{7}. Any degraded neutrons are excluded from the estimate. (See section on Neutron Attenuation.)

An estimate similar to the one above, and excluding the degraded radiations, shows that for the tug-tow craft the attenuations needed are: 2000 for fast neutrons and 60 for bad gamma rays. This estimate is based on a reactor power of 100,000 kilowatts and a distance of 1000 meters between tug and tow.

At the attenuations indicated for the tug-tow, lubricants and electronic equipment will be carried on the tug according to the tolerance estimates in Table IVB-5. At 10 meters the more sensitive component (the electronic equipment) exhibits a sufficient margin of safety (a factor of 20) so that it may be thought that it will be regarded as safe against the radiations even when other energy regions are considered.

In terms of shielding thicknesses, the numbers quoted for the manned plane are relatively insensitive to factors of the order of 3 which might arise from differences in the amount of absorption of the radiations inside the reactor itself or from errors in the biological tolerances. The situation is reversed, however, for the tug-tow craft. Since the overall attenuations required are low, the shield thickness and shield weight will be extremely sensitive to such factors. It is therefore desirable to give some reasonably sure overestimates of the required attenuation in the case of the tug-tow; 6000 for fast neutrons and 300 for bad gamma rays might constitute such overestimates (7). It may be added, however, that the small amounts of shielding needed for the tug-tow may lead to the use of a large reactor (see Chapter IVA). Consequently, the internal absorptions are likely to be larger than those used and the original estimates more nearly safe ones.

Since the needed attenuation factors for the tug-tow craft are so low, the type of estimate given here should eventually be supplemented by a more thorough study which includes primary radiations at energies other than those included here. Such a study may not prove too difficult for the very reason that makes it necessary: that is, because the attenuation factors are not particularly large. (See section on Shield Types.)

Estimates of the attenuation factors needed for a pilotless aircraft taken here as the last example, are probably less certain than those of either the manned or the tug-tow aircraft. At 100,000 kw. reactor power, the following attenuation factors are obtained by use of Tables IVB-2 and IVB-5.

1. Needed for the payload at 3 meters:

   (a) 10^4 attenuation of fast neutrons and

   (b) 30 attenuation of bad gamma rays.

2. Needed for star-tracking guidance system at 2 meters or more from the reactor (all figures uncertain):

   (a) 2×10^2 for fast neutrons and

   (b) 10^3 for bad gamma rays.

The remarks on the sensitivity of predicted shield weight and shield thickness to inaccuracies in the attenuation estimates for the tug-tow apply equally to the case of the pilotless aircraft.

For both the tug-tow and the pilotless aircraft some additional considerations enter the picture. Around a bare reactor, considerable illumination will be produced. If it is desired to reduce light output to the point where the aircraft is essentially invisible, it may be necessary to introduce enough shielding to produce an attenuation of the order of 1000 on the gamma rays and perhaps somewhat less on the fast neutrons. These estimates are quite uncertain because of extreme uncertainties in the quantum yields (16).

Any airplane which is only partially shielded for a man nearby creates a handling problem when it is on the ground after operation. In the case of insufficient neutron shielding, induced activities will be produced in the shielding material and in the body of the plane. In any case, the accumulated fission products will continue to emit gamma rays after shutdown. For example, about an hour after shutdown the gamma radiation is expected to be approximately $\frac{1}{2}$% of that which was present during operation. Further decay of the gamma radiation from the fission products proceeds approximately at $t^{-1.2}$%, where t is in days after operation. The decay of the induced activities depends on the details of the composition of reactor, shield and aircraft (24). It is clear that considerable precautions will be required in approaching either the tug-tow or the pilotless plane and that these precautions will have to be more elaborate the more the shielding is reduced (6).

[redacted]

In general, needed attenuations in shields for components other than man may be estimated by comparing the component tolerance to that of man and evaluating the appropriate distance factor.

It should be noted that the mechanisms of shielding so far considered include shielding internally by the reactor itself and shielding by air (or distance) between the reactor and the sensitive component. The needed attenuation factors estimated here must therefore be obtained by the use of heavy shields either around the sensitive components or around the reactor. Preferably, the shield will be placed around the smaller volume, in order to save weight. For example, extra shielding for aerial photographic film goes around the film and not around the reactor.

The estimates of the required attenuation factors have been computed on the assumption that the specific type of radiation being discussed was the only one that needed to be considered. For example, in the calculations for the manned craft, the fast neutrons are attenuated in such a way as to give a dose of 25 r, and the bad gamma rays are also attenuated in such a way as to give the 25 r dose. The effects of the various radiations are, however, additive and the reduction should be correspondingly greater in order that the total dose shall not exceed the tolerance. Nevertheless, it is more useful to leave the attenuation factors temporarily in their present form. It may prove to be the case that, in an actual shield, certain radiations are attenuated more than is required and only the least attenuated component may need to be considered. Also, it is possible that other sources of radiation will have to be added to those discussed already. For example, secondary gamma rays must be evaluated and also the effects of induced activities or fission products escaping into the cooling streams of the reactor.

The results of this section are summarized in Table IVB-6, which lists the probable attenuation factor needed for fast neutrons and for bad gamma rays in order to reduce each individually to the required tolerance level.

TABLE IVB-6

REQUIRED ATTENUATION FACTORS

	Manned (4×10^5 kw.)	Buggy (10^2 kw.)	Pilotless (10^2 kw.)
Fast neutrons	10^8	2×10^3	10^4
Bad gamma rays (primary)	4.3×10^6	60	10^3

The probable attenuation factors quoted in Table IVB-6, it may be noted, are in agreement with other estimates (6). (In making comparisons, however, care must be taken to check the assumed power, distance factors, and the amount of internal absorption assumed for the radiations.)

[redacted]

It is important to note that the secondary gamma ray problem has been completely omitted from this section. This omission does not imply that the secondary gamma ray problem is unimportant. It will be discussed at some length in the later sections.

THE ATTENUATIONS

The requirements which the shield must meet in terms of the attenuation of fast neutrons and of primary bad gamma rays have been laid down. How to meet these requirements is the theme of the following sections. The main points under discussion are outlined here in advance.

It is instructive to try to evaluate the product of density times thickness (ρt in grams per square centimeter) needed to produce a specified attenuation of the dangerous radiations. If the thickness were not important, the shield weight required would vary directly with the required ρt. In fact the thickness is important because of the geometric problem of wrapping the shield around the reactor. A small increase in ρt may be compensated by a decrease in thickness. The play-off of ρt against thickness is illustrated in the third part of the shielding discussion -- the design part. Here the discussion is largely in terms of ρt alone.

By considering only those gamma rays which are completely untouched in going through the shield -- the undeviated gamma rays -- a definite underestimate is made of the ρt needed as a shield around the reactor. For the manned plane the underestimate shows that shield density times thickness, ρt, must be greater than 370 gm./sq.cm. For the tug-tow, 98 gm./sq.cm. is needed if the requirements are correctly determined; and on the same basis, but with less certainty because of the dubious nature of the requirements, 160 gm./sq.cm. is needed for the pilotless plane.

Better estimates could be given if the build-up of degraded gamma rays were completely known. Although an accurate estimate cannot be given, the present knowledge of the build-up leads to an overestimate of the change required by this effect. For shields made of heavy elements (lead and heavier) an overestimate of the primary gamma ray requirements results. The estimates are given in Table IVB-7.

For the thin shields of pilotless and tug-tow airplanes the overestimates of ρt are not too reliable. They may not be valid because of the possibility that some gamma rays not considered in the bad group are contributing to the radiation.

For shields which contain light elements the overestimates are not valid and the underestimates are misleading because they are too low to be of great interest. Nevertheless, considering the primary gamma rays alone, it is possible to get interesting estimates for shields containing light elements. For example, the manned airplane may easily require between

500 and 600 gm./sq.cm. when light elements are present in the shield. Again believing the estimated requirements, some reasonable range is indicated in Table IVB-7 for each aircraft.

Table IVB-7

ESTIMATES OF ρt REQUIRED FOR BAD PRIMARY GAMMA RAYS (gm./cm^2)

	Underestimate	Heavy Element Overestimate	Light Element Reasonable Range
Manned	370	550	500 - 600
Pilotless	160	280	220 - 300
Tug-Tow	98	170	130 - 190

The presence of light elements in the shield seems necessary to provide neutron attenuation. The requisite proportion of light elements is unknown. Present data on neutron attenuation merely indicate the needed value of ρt for a few specific shields. The two most important examples are given in Table IVB-8. The composition of these shields is given in the section on neutron attenuation. Some estimates for water are also included. (In the table the hydrogen content increases to the right.)

Table IVB-8

ρt REQUIRED FOR FAST NEUTRONS (gm./cm^2)

	MO	Canadian	Water
Manned	590	430	~120
Pilotless	315	230	50
Tug-Tow	260	190	40

Neutrons in the shield produce secondary gamma rays. When the neutron behavior in the shield is known the outflux of secondary gamma rays may be predicted. Usually shielding for the secondary gamma rays appears to require about the same or greater ρt than that required for the primary gamma rays, but there may be important exceptions to this crude rule.

Because it is necessary to obtain a simultaneous solution of the problems of bad primary gamma rays, of fast neutrons, and of secondary gamma rays, only the MO shield, one of the two appearing in Table IVB-8, seems to meet the present requirements for a manned plane. No doubt better shields can be designed, but no other adequate tested shield can be said to exist.

For the tug-tow or pilotless planes a thinner MO shield could be used. For these planes, however, the MO appears to be far from an ideal design. A comparison of Tables IVB-7 and IVB-8 would indicate, for example, that the Canadian shield is more appropriate for the tug-tow. (The increased thickness indicated later in Table IVB-12 shows that it is not quite so good as Tables IVB-7 and IVB-8 indicate.)

The design problem is dealt with briefly for manned, tug-tow, and pilotless planes after the sections on attenuations.

Primary Gamma Ray Attenuation

An underestimate of required weight per unit area for a shield can be obtained from a consideration of the undeviated gamma rays. In heavy shields the undeviated primary beam of gamma rays is reduced by a factor of ten for every 58.5 gm./sq.cm. of any material which consists of elements with atomic number greater than or equal to that of lead (25). The meaning of this statement is not that the emerging gamma radiation is less than the incident radiation by this factor but that the gamma radiation which remains at exactly the same energy and travels in the same direction as it entered the shield is so reduced. This statement should also be qualified in two directions. First, the incident radiation must have an energy somewhere around 3 Mev. and for any other incident energy a smaller weight of shielding material will suffice. Second, this rule is accurate to about 6%.

Since the gamma radiation emerging from the reactor contains a reasonable number of gamma rays in the range of energy around 3 Mev., the number 58.5 gm./sq.cm. of material for a tenfold attenuation should apply rather accurately to the gamma radiation from the reactor. A reference to the discussion of the attenuation factors needed now shows that for heavy shields a weight of 390 gm./sq.cm. is required to reduce the undeviated primary bad gamma rays to the tolerance level in the case of the manned airplanes (4.3×10^6 attenuation). To allow for the 6% uncertainty in the number 58.5 gm./sq.cm., the 390 is reduced to 370 gm./sq.cm.

Since some gamma rays which have been removed from the pure primary beam, for example by Compton scattering, are still present in the emerging gamma ray flux, this estimate constitues an absolute lower limit for the value of ρt for a heavy shield.

In passing to shields made of elements of atomic number lower than that of lead, the attenuation of 3 Mev. gamma rays stays approximately the same for the same ρt, but the attenuation of higher energy gamma rays is not so great. For example, the undeviated primary attenuation of 5 Mev. gamma rays in iron requires about 71 gm./sq.cm. for each factor of 10; 76 gm./sq.cm. is required in water and about 85 gm./sq.cm. in carbon. Consequently the ρt required for shields in which light elements play a part is greater than that required for shields made with heavy elements.

Under certain assumptions an upper limit to the required ρt for a gamma ray shield can be obtained. The presence of the gamma rays which

have been deviated from their original course and degraded in energy increases the shielding requirement. An estimate has been made to take account of the build-up of these degraded gamma rays as a function of shield thickness.

The effect of the build-up of degraded gamma rays can be represented by a factor which gives the ratio of emergent gamma ray flux to undeviated primary gamma ray outflux from the shield. An overestimate of this factor is given in Table IVB-9 (7).

Table IVB-9

BUILD-UP FACTORS FOR DEGRADED GAMMA-RAYS

Attenuation (power of 10)	10^0	$10^{0.5}$	10^1	10^2	10^4	10^6	10^8
Build-up Factor	1	2.1	2.8	4.2	7.1	10.5	13.9

The numbers in Table IVB-9 are computed for the case of pure Compton effect. Iron has a sufficiently low atomic weight so that in the neighborhood of 3 Mev. its true absorption coefficient for gamma rays is very largely due to Compton scattering. Furthermore, heavier elements are better than iron, gram for gram, in eliminating gamma radiation. The heavy elements substitute pair production (a real absorption process which does not produce further bad gamma rays of any consequence) for Compton scattering. The value of ρt required in iron should therefore constitute an upper limit to the values of ρt required in heavier elements to attenuate bad gamma radiation from the reactor.

In the case of the manned aircraft the needed attenuation for undeviated primary gamma rays is 4.3×10^6. In order to take care of the degraded gamma ray build-up, a greater attenuation of the primaries is required -- an attenuation great enough to include the build-up factor. Table IVB-9 gives a build-up factor of about 12. The required attenuation of the undeviated primaries is thus about 4.7×10^7 and the corresponding ρt is 550 gm./sq.cm. The corresponding estimates for tug-tow and pilotless aircraft appear in Table IVB-7.

These values should constitute overestimates of ρt needed if the shield could be designed solely from the point of view of primary gamma radiation. In practice, however, consideration must be given to the problem of eliminating the neutrons, and this problem may require the use of relatively light nuclei.

For shields containing light elements, an estimate of weight required per unit area may be made. For this purpose the more relevant values of ρt(for an attenuation factor of 10 at 5 Mev.) are: in water, 76; in carbon or aluminum, 85; and in iron, 71. Using these values and the attenuation factors required for undeviated primaries and primaries with degraded

gamma ray build-up, it becomes clear that a ρt of 500 to 600 gm./sq.cm. for the manned aircraft should not be at all unexpected. The value for the MO shield, for example, should be between 520 and 600 gm./sq.cm. from the primary gamma ray point of view.

Ranges of ρt determined in this way are given in Table IVB-7 for the various aircraft.

Some data are available on the attenuation of gamma rays through various materials (26). The data extend only over attenuations of the order of 100 to 1000. The analysis of these data agrees with the picture presented here. The true absorption cross section can be used for an exponential attenuation and a reasonable build-up factor then accounts for the actual observations. The build-up factors seem to be smaller than those of Table IVB-9, as is expected, but the data are too scanty to place much reliance on this fact (26).

Neutron Attenuations

Although there is a considerable amount of data on neutron attenuations through thick shields, rather little of it is either good enough or relevant enough to be of much use in evaluating fast neutron attenuation for aircraft reactors. Most of the data have been taken on a single shield type, a combination of iron and hydrogenous material. Other data apply to ordinary concrete, a material which is unsuitable for the present problem. Finally there are some data on water (27)(28). Unfortunately, for many of the shields measured the observed neutron attenuations are not reproducible. Also the relaxation distances typically increase in traversing the shield from inside to outside. This departure from exponential behavior is of course to be expected.

The most extensive collection of data comes from work of E. P. Blizard and collaborators at Oak Ridge National Laboratory (29). The example chosen for the most thorough analysis is the so-called MO shield, and the experimental data are taken from the work of the Oak Ridge group.

Result of the analysis of the MO shield may be expressed as follows (4):

$$C(r) = \frac{A\mu^2}{2\pi} \frac{e^{-\mu r}}{\mu r}$$

where $A = 2.3$ and the reciprocal relaxation distance $\mu = .176$ per cm. $C(r)$ is the number of neutrons absorbed at a distance r (cm.) from a unit fission source at $r = 0$ in MO shielding material. The density of this material is 5.8 gm./c.c. and the composition is given in Table IVB-10.

The above analysis is so performed that the function C (r) shall be applicable to the computation of thick shields (shields with attenuations

of the order of 10^6 to 10^9). The constant A may be interpreted as being proportional to the fraction of the fast neutrons getting into the penetrating exponential. μ is related to the cross-sections which scatter neutrons out of the fast beam.

Table IVB-10

COMPOSITION OF MC SHIELD

Element	Atoms/cm^3 x 10^{-24}
Fe	.0517
H	.0232
Mg	.0077
O	.0184
Cl	.0018

Perhaps the other shield in which the relaxation distance has been accurately measured and is most meaningful is the bottom shield of the Canadian heavy water reactor (30). It is suggested that the data on this shield and several others should be analyzed in the near future. Tentatively the crude relaxation distance extracted from the data will be used in place of an adequate analysis (31).

The relaxation distance measured in the Canadian shield is 6.7 cm. at large distances from the reactor. The composition of the shield is essentially described by saying that it is built of 10-inch units, 4 inches of which are iron, 6 inches of which are water. The average density is 3.72 gm./c.c.

If the data for water (28) are fitted with the above equation one obtains different values of A and of μ depending on where the fit is made. The results are summarized on Table IVB-11.

Table IVB-11

A AND μ FOR WATER

Distance from Source to Point Where Data are Fitted (cm.)	A	μ
30	0.83	0.17
70	0.18	0.128

As compared with 2.3 for the MO shield data, the value of A is low. The low values of A indicate that the fraction of the neutrons from the source which contribute to the penetrating exponential at large distances is smaller for water than it is for MO. The rapid fall in the hydrogen cross section may account for the low A's by sorting out the very high-energy portion of the fission spectrum and robbing the rest of the fast neutrons of their usual importance.

In addition to the data summarized above, there are some measurements of the neutron flux coming through the top shield of the Hanford reactor (32). These data do not suffice to determine the relaxation distance inside the laminated portion of the Hanford shield. The total attenuation factor between the edge of the reactor core and the outside may be determined. Unfortunately, in order to determine the attenuation ascribable to the laminated shield, it is necessary to know the behavior of the reflector (2 feet of graphite) and of the "thermal shield" (10 inches of iron).

Further analysis of the Hanford shield is extremely desirable. It is possible to make measurements through the shield from inside to outside, thus avoiding the necessity for knowing the behavior of reflector and thermal shield, and in future Hanford reactors it may be possible to improve the situation still further by placing uranium slugs next to the thermal shield so that the reflector effect can be eliminated.

In the analysis used here the attenuation of a plane source of neutrons by a shield is given by

$$\frac{2.5e^{\mu t}}{A}$$

where μ and A are the same as before. For the MO shield then, the required fast neutron attenuation determined earlier implies that 17.2 relaxation distances are necessary for the neutron shielding of the manned plane. This number gives a ρt for the MO shield of 590 gm./sq.cm. Comparable ρt values for other shields and other aircraft are given in Table IVB-8. (It is necessary to know the value of A, and A has been determined only for water and MO; however, for most shields it is probably safe to use $A \approx 2.5$. This assumption is made in calculating the Canadian shield.)

For the neutrons, the build-up of degraded flux doubtless takes place as it does in the gamma ray case. It can have no large effect on the present consideration, since the available neutron attenuations (relaxation distances) are empirically determined at appropriate positions in real shields. The effect of the degraded neutron build-up is, therefore, already included in the analysis.

Estimates of the flux of neutrons which are more than slightly degraded and estimates of the energy which may be transported by them also show that no correction to the calculation is necessary (33); the experimental data for MO have also led to the same conclusion (4).

No adequate purely theoretical approach to the neutron-attenuation problem is known at present. There is, however, an apparent correlation between the measured relaxation distances and the total neutron cross section per unit volume at neutron energies of about 2 Mev. In making the correlation, allowance is necessary for the forward nature of the scattering by hydrogen nuclei (34).

The inelastic scattering cross section may also play a role in correlating the relaxation data. In a shield containing water, for example, the hydrogen cross section falls off at high energies. On the other hand, the inelastic cross sections of heavy elements are likely to increase with energy, and the energy loss of neutrons by inelastic scattering may bring the neutrons to a low enough energy so that they are easily eliminated by subsequent collisions with hydrogen (34)(35).

Present data can be fitted by a wide range of possible formulas and do not allow the conclusion that any particular prescription or any prescription at all is correct. However, the fact that mixtures of iron and water in various proportions give shield thicknesses (cm.) which are approximately the same may be an example of the interplay of inelastic cross section, total cross section and of the peculiar properties of hydrogen. (See Table IVB-12 and compare with Table IVB-8). Sufficient data are just becoming available to warrant a detailed study of the methods for computing relaxation distances.

Table IVB-12

NEUTRON SHIELD THICKNESS (cm.)

	MO	Canadian	Water
Manned	102	115	~ 120
Pilotless	53	62	50
Tug-Tow	45	51	40

It is necessary at present to rest the discussion of neutron attenuations almost completely upon the analysis of the MO shield.

Secondary Gamma Ray Attenuations

When the neutrons in the shield are inelastically scattered, they produce gamma rays. Gamma rays are also produced when the neutrons are captured. These gamma rays from inelastic scattering and capture of neutrons in the shield are the secondary gamma rays in the classification used here. Even if the primary gamma rays are adequately shielded or even if there were no primary gamma rays, the secondary gamma rays present a new problem.

Experimental data on gamma rays in actual shields might shed some light on the secondary gamma ray problem. Unfortunately the penetration of gamma radiation in the shields discussed in the last section is not well known. In the case of the Canadian shield no measurements are available. In the case of the MO shield there are significant discrepancies between various sets of experiments, and in addition the most recent experiments give results which are hard to understand. On an experimental basis, therefore, it cannot be stated that these shields are adequate for both neutrons and gamma rays simultaneously. They are employed so far merely for the determination of the neutron properties.

The theory of the interaction of gamma rays and matter is excellent. Once the fast-neutron attenuation is given and the secondary gamma ray source is thus known, the outflux of secondary gamma rays can be computed quite accurately. Although an adequate experimental verification of these specific results is lacking, the results are probably correct.

As was pointed out in the last section, although the primary-neutron attenuation may not be really understood, nevertheless there is at least one example in which the primary-neutron attenuation is known, that of the MO shield. Furthermore, in any case a nearly exponential relaxation of neutron flux is expected. It seems worthwhile, then, to investigate the secondary gamma ray problem reasonably closely. The investigation will refer especially to the MO shield and will cast some light on the general problem of the secondary gamma rays.

A detailed treatment of the secondary gamma ray problem of the MO shield in spherical geometry surrounding a spherical reactor of the same density has been carried through (4)(36). The result of the secondary gamma ray analysis for the MO shield indicates that the shield is adequate for secondary gamma rays as well as for neutrons. At ρt = 590 gm./sq.cm. the emergent gamma ray flux from the secondary gamma rays is about 8.5 times the emergent neutron flux. When account is taken of the different tolerances for neutrons and gamma rays, the biological effect is seen to be about the same (i.e., neutrons and gamma rays are each separately reduced to the 25 r tolerance). The estimate of the primary gamma ray effect made above shows that the primary gamma rays are of less importance in the overall shielding than are the secondaries, or possibly of the same importance.

The result that the secondary gamma rays are of greater consequence in the shielding problem than are the primary gamma rays is not unexpected. The contribution from secondary gamma rays may be crudely estimated by comparing the neutron outflux from the reactor with the primary gamma ray outflux, multiplying by the number of penetrating gamma rays produced per neutron in the shield, and taking account of the fact that since they are made near the outside the secondaries are less attenuated than the primary gamma rays.

The flux of fast neutrons entering the shield is approximately one-third the flux of bad gamma rays, to use the example which has been employed throughout the shielding section of the report. As a reasonably

pessimistic estimate of the number of penetrating secondary gamma rays per neutron absorbed in the shield, the number 2 is adopted. (In the MO analysis the conversion coefficient for turning neutrons into bad gamma rays was taken to be 1.) In order to compensate for the fact that the source of secondary gamma rays is nearer the radiation-sensitive components than the source of primary gamma rays, a factor of 3 is introduced. (This factor is probably too low. Its origin is discussed more formally a little further on.) In the simple calculation here, the ratio of secondary to primary gamma ray outflux is $(1/3) \times 2 \times 3 = 2$, and the result of the estimate is that the secondary gamma rays are comparable with primary gamma rays or of slightly greater importance.

The argument given above, which shows that secondary gamma ray effects may be comparable with or slightly worse than primary gamma ray effects, is not completely general. There are two ways in which the secondary gamma effects may turn out to be considerably different from the crude prediction.

First, it may be possible to suppress the production of secondary gamma rays in the penetrating energy ranges by employing elements with little or no inelastic scattering and by implanting in the shield elements of high-capture cross section which do no produce bad gamma rays upon neutron capture. Boron is the type element in the latter category. Some of the light elements probably get into the former category, but no heavy elements seem to be peculiarly advantageous, although lead has a low inelastic cross section if the neutron energy is not too high. (For lead the inelastic scattering cross section is negligible for 1 Mev. and probably somewhat less than 1 barn for 2 Mev.) (37). It should be recalled, of course, that a successful suppression of the secondary gamma rays still leaves a comparable primary gamma ray problem.

The second modification necessary in the above result arises for light shields if the energy of secondary gamma rays is high. Certain work, both experimental and theoretical (38)(39) on gamma ray energies in (n,γ) reactions shows that secondary gamma rays of the order of 6 Mev. and even higher energy may be produced in light shields. (5 Mev. was taken as the secondary gamma ray energy in calculating the MO shield.) Such penetrating gamma rays may decay with a relaxation distance which is greater than the relaxation distance for neutrons, or for the so-called bad primary gamma rays. Consequently, the secondary gamma rays effects will be worse than appears either from the description above or from the formalism in the simple cases which are described below.

A recent measurement (32) of the ratio of gamma rays to neutrons emerging from a shield has been made by the Physics Group at Hanford. It seems to be an example of the effect of too long a relaxation distance for gamma rays. The relaxation distance for hard secondary gamma rays in the Hanford shield is probably about 7.3 cm. The neutron relaxation distance has been measured in preliminary tests as 6.7 cm. and this value fits in the sequence in Table IVB-12 rather too well. Using a slight modification of the formalism presented below, a modification which allows for the difference in relaxation lengths, the ratio of secondary gamma rays

to neutrons emerging from the shield should be about 30. The experimentally measured value is somewhere between 20 and 70 with a most probable value of about 34. The agreement is probably fortuitous.

Various theories have been worked out to give the secondary gamma ray outflux. In order to keep the formal discussion of the secondary gamma ray problem within bounds, the special case of equal relaxation distances for fast neutrons and bad secondary gamma rays will be the only case considered. In this case, the ratio, R, of secondary gamma ray outflux to fast-neutron outflux is variously given for thick shields as

$$R_1 = \tfrac{1}{2}\mu t\, C_{n\gamma}, \text{ and}$$

$$R_2 \simeq \tfrac{1}{2}[-.42 + \log_e (\mu t)]\, C_{n\gamma}$$

depending on what the distribution of gamma rays around a point source is assumed to be (2)(40). R_1 applies to a distribution of gamma ray absorption around the source of the form $\frac{\mu^2 e^{-\mu r}}{4\pi\mu r}$, and R_2 applies to a distribution of the form $\frac{\mu e^{-\mu r}}{4\pi r^2}$. C_n is the conversion coefficient of neutrons to gamma rays.

R_1 may be used when it is desired to include an approximation to the build-up factor in a simple way. R_2 does not include the build-up but it can be modified to include the build-up factor (when it is known) to any desired accuracy. For $\mu t = 17.2$ and $C_{n\gamma} = 1$,

$$R_1 = 8.6 \text{ and } R_2 = 1.2$$

It should be noted that R_1 and R_2 are both derived for use with a neutron distribution of the form $Ae^{-\mu x}$, where x = distance from reactor shield interface in a slab shield.

R is the factor needed to compensate for the fact that the source of secondary gamma rays is closer to the outside of the shield than is the source of primary gamma rays. R = 3, as used in the crude argument several paragraphs above, is between R_1 and R_2. R_2 is probably too low by a factor of about 7 because of the build-up of degraded secondary gamma rays. (7 is a rough average build-up factor for the secondary gamma rays -- see Table IVB-9.)

Since the important production of secondary gamma rays can be shown to take place, relatively speaking, near the outside of the shield, the difference between spherical geometry and slab geometry should not play much of a role in these calculations. The slab geometry formulae have the advantage that they are simpler to use in their present form. The spherical geometry formalism presents no real difficulties, however, when more exact calculations are desired. It has been applied to the MO shield (36)(4).

A certain amount of data which are relevant to the gamma radiation problem in general has been collected from the experiments on thick shields. No thorough analysis of it has been attempted and a crude agreement obtained between a selected set of data and approximate theoretical predictions may

be illusory. The experiments are not yet in such a form that it is feasible to separate primary and secondary gamma ray effects at all unambiguously, but the program of the Oak Ridge National Laboratory group includes some proposed experiments which should make the effects separable. Much further work will be required, and the shield analysis will remain in doubt until theoretical and experimental results can really be checked against each other at many more points.

The MO Shield

In the MO shield on the manned aircraft, it appears that $\rho t = 590$ gm./sq.cm. will allow only 25 r of neutrons in a mission, only about 25 r of secondary gamma rays and probably less than 25 r of primary gamma rays. The total dose should be under 75 r. Since the relaxation distances are all about equal, the addition of a little over one relaxation distance will bring the total dose given the crew down to tolerance. The ρt value is now 625 gm./sq.cm.

The conclusion that 625 gm./sq.cm. is the ρt for a usable shield for the manned plane depends on many assumptions.

Six may be mentioned here as somewhat doubtful:

(a) The reactor power is 4×10^5 kw.

(b) The biological tolerances are correct.

(c) The bad gamma ray leakage is one-third the bad gamma ray production in the reactor.

(d) The bad gamma ray production in the reactor is 10^{19} per sec. at 10^5 kw. reactor power

(e) The fast-neutron leakage from the reactor is one-tenth the fast neutron production.

(f) The conversion coefficient of neutrons to bad secondary gamma rays is one.

If there is no change in (a) and (b), very pessimistic assumptions about (c), (d), (e) and (f) would not increase the needed ρt above 700 gm./sq.cm.

Effect of Curvature

As was mentioned above, the analysis of required ρt can be carried through in the case of spherical geometry. All of the ρt values so far discussed correspond to an infinite inner radius of the shield. As the radius of the shield decreases in order to fit it onto smaller and smaller piles the required ρt rises. In the range of ρt values of 625 to 750 the increase in ρt required in going from a slab shield to a shield around a point source of the same strength turns out to be about 10% (36)(4).

DESIGN

In the first part of the discussion on shielding the requirements for the manned airplane, the tug-tow combination and the pilotless aircraft were investigated. In the second part, the attenuations which might be produced by unperforated shields were studied. The requirements were then rewritten in terms of the number of gm./sq. cm. (ρ t) of needed shield material. The remaining problems are those of designing an actual shield and of estimating its weight.

Effect of Ducts

The real shield cannot be unperforated. Ducts must be provided to let the working fluid in and out of the nuclear reactor. The ducts will usually decrease the attenuation given by the shield and thus lead to a greater shield thickness for adequate radiation protection.

For the case of the open cycle, the duct problem is an extremely serious one. In addition to reducing the density of the shield, the channels, which are essentially vacuous, allow long free rides to be taken by the neutrons and gamma rays. Those neutrons and gamma rays which take their free rides in the direction of the outside of the shield will cause an additional radiation dosage. As a result not only is the average density decreased but the thickness is so increased that the needed value of ρt increases.

An increase in thickness at constant ρ t has a very serious effect in increasing shield weight. Material which has been removed for ducting near the reactor face must be added back on at the outside of the shield. Since it is necessary to replace the same number of grams per square centimeter, the total weight rises because the surface area is greater on the outside. (See section on shield weights.)

The experimental work on air ducts is scanty and inconclusive. An experiment comparing the radiation attenuation produced by an iron shield and by a shield made of iron punchings indicates that there is a higher radiation dose penetrating the iron punchings, although the ρ t value is the same for punchings and for solid iron.

Another recent experiment, one on a spiral duct, is not easy to interpret. Considerable extra radiation came through the shield in the vicinity of the spiral. Both of these experiments were performed by the Oak Ridge National Laboratory group (41) (42).

In the present state of ignorance there is little to be said about the ducting problem for the open cycle. In order to make some estimates of the minimum effect of ducts, the average density change at a constant value of ρ t will be adopted despite the fact that the average density change is an inadequate description.

For a closed cycle which uses a high-pressure gas, the ducting area is lower than that needed for the open cycle, primarily because of the

[redacted]

higher pressure. Consequently the increase in shield weight due to ducting will not be so great as in the case of the open cycle.

If the reactor coolant in the closed cycle is reasonably dense the duct spaces, instead of being gaps, may contain material with a density which is comparable to that of the shield. In certain cases even the nuclear properties important in shielding may be substantially unchanged by the presence of the working fluid. For example, liquid bismuth and lead are not too dissimilar.

The case of liquid bismuth illustrates another interesting possibility. The density of bismuth is high, 10 grams per cubic centimeter, and the presence of the coolant may even increase the shield density in certain cases rather than decrease it. Pending an adequate investigation of the duct problem it will be assumed that there is no effect on the shield if the coolant introduced is at the same or higher density than the shield, and that there is a density effect only if the coolant is of lower density than the shield.

Calculations which neglect ducting are likely to be very far from the true answer in the case of open-cycle systems combined with thin shields such as might be employed for the tug-tow. Large solid angles will be subtended by the open spaces of the ducts and neutrons and gamma rays will leak more freely because of the wide apertures.

Shadow Shielding

The possibility of reducing the weight of a shield by reducing the shield thickness on the side of the reactor away from radiation-sensitive components has come in for a little investigation by Project RAND (43). It is as yet too early to give any accurate results. It is possible that the attenuation required at the far side of the reactor is less than that required at the side toward the radiation-sensitive components by approximately one order of magnitude for neutrons and somewhat more for gamma rays.

Since advantage of shadow shielding can be taken only over part of the surface of the reactor, and since one order of magnitude in attenuation is only approximately 15% in shield thickness for the manned aircraft, the possibility of shadow shielding is not likely to change the weight requirements for the manned aircraft radically. Even in this case, however, the extent to which shadow shielding can be employed depends on the precise relative position of reactor and man.

The energy of the radiations which have been turned around in going from the far side of the reactor to the position of the crew is certain to be somewhat lower on the average than the energy of radiation in the direct beam from reactor to man. The difference in energy opens up the possibility that for gamma rays at least it may be advantageous to shift some of the shielding from the far side of the reactor to a position near the crew quarters. Such details of design as these may eventually play a role in the determination of the weights.

[redacted]

In the case of the tug-tow and of the pilotless aircraft, shadow shielding is likely to be of far greater significance. For thin shields, an order of magnitude less in required attenuation introduces a big reduction in the necessary shield weights. The dubious nature of the shadow-shielding estimates is an additional reason for believing that the shielding requirements of tug-tow and pilotless aircraft are especially uncertain and for recommending a careful evaluation of the shielding problem in these cases.

Radioactivity of the Reactor Coolant

Gamma rays will be emitted by the coolant when it is outside the nuclear reactor. The wake will be radioactive in open-cycle systems, and in closed-cycle systems the reactor coolant will carry sources of gamma rays into the heat exchanger and pump. The gamma rays, emitted in the external system, create an additional shielding problem.

The gamma ray activity in the coolant may be produced by:

(a) neutron-induced activity in the coolant;

(b) neutron-induced activity in impurities present in the coolant;

(c) radioactive contamination of the coolant by corrosion and erosion of activated atoms from the walls of the tubes through which the coolant flows; and

(d) radioactive contamination of the coolant with fission products leaking through tube walls.

For closed-cycle systems the coolant activity has been investigated in some detail (44). The conclusions are summarized in the following paragraphs:

The computations are based on a thermal or fast reactor of 100,000 kilowatts containing a coolant volume of 1 cubic foot and operating at neutron fluxes of 2.4×10^{14} and 8×10^{15} per square centimeter per second, respectively. It has further been assumed that only 10% of the coolant is in the reactor at any time, the rest being in the heat exchanger and associated power plant equipment.

If helium or hydrogen gas is used as the working fluid the induced radioactivity of the coolant proper is negligible. In the case of helium at 1000 psi pressure and a temperature of 1000°C, an argon impurity as great as 0.01% will result in a total gamma ray dosage over one day of only 2 r for a man 10 meters away. Thus, these gases present no serious problem in the manned aircraft and a negligible one in the tug-tow.

Among the liquid metals considered for coolants, pure lead and bismuth give insignificant radiation doses. Tin, magnesium and sodium lead to total gamma ray doses between 3×10^4 and 6×10^5 r over one day in the case of the manned plane. Because the shielding of the heat exchanger would introduce

intolerably large weights, these three liquid metals must be excluded from consideration for manned planes.

For the tug-tow, on the other hand, tin and magnesium turn out to be marginal (less than 10 r per day). Sodium becomes too radioactive in the thermal reactor, but may be tolerable in a fast reactor.

In liquid metal coolants the impurities present a serious problem. However, it turns out that bismuth of the purity used at Hanford would probably be suitable even in the manned aircraft.

The tolerances on the amount of tube wall material which can be allowed to get into the closed-cycle coolant stream by corrosion and erosion do not appear to be impossibly stringent if steel is considered for the coolant tubes. In the case of a thermal reactor of the above-mentioned properties for a manned aircraft, assuming tube walls of 18-8 stainless steel, as much as 1.7 kg. of steel can be permitted to get into the coolant before a man (at 10 meters) receives a total radiation dose of 25 r in one day.

For the open-cycle system, the radioactivity induced in the air is negligible both for the manned and the tug-tow aircraft.

The problem of fission product contamination of the coolant has been considered (7). The results are summarized in Table IVB-13, in which the maximum allowable fission product contaminations (those which would cause a total of 25 r over one day) are given on two different assumptions as to how the fission products leak into the coolant.

The first, and the more pessimistic, assumption is that a certain fraction of the fission products get into the coolant immediately after they are formed. This assumption would be appropriate if, for instance, part or all of the uranium were exposed to the coolant, so that some of the fission recoil nuclei emerged directly into the coolant. The second, and the more optimistic, assumption is that fission products diffuse into the coolant through a tube wall at a rate which is proportional to the total number of fission products in the fuel elements. On this assumption many of the fission products will decay before they get into the coolant. For the same total fission product contamination of the coolant, the resulting radiation will be less in this case than in the first one. The numbers given in the table under the optimistic "diffusion" assumption correspond to the reciprocal of the mean time elapsing between the birth of the fission product and its entrance into the coolant. This is equivalent to the fraction of fission product nuclei entering the coolant per unit time.

In computing the numbers given in Table IVB-13 the towed aircraft is assumed to be at a distance of 1000 meters from the tug and at the reasonable distance of 150 meters below the wake. It should be noted that, for the open cycle, only a few minutes in the wake probably results in a dosage equivalent to a few hours at 150 meters. There is no wake radioactivity for the closed cycle.

Table IVB-13

ALLOWABLE FISSION PRODUCT ESCAPE INTO COOLANT

For a dosage of 25 r in one day -- 100,000 kw. reactor

	Pessimistic (Instantaneous Escape)		Optimistic (Diffusion Rate)	
	Fraction of all the fission products which are in coolant		Reciprocal mean fission product hold-up time, per day	
	manned	tug-tow	manned	tug-tow
closed-cycle	10^{-6}	10^{-2}	10^{-4}	1
open-cycle	10^{-3}	10^{-3}	10	10

For the manned plane with a closed-cycle, even minute instantaneous fission product escape (10^{-6}) is intolerable. For the other possibilities listed, however, the requirements do not seem to be too stringent. For example, when measured in terms of fission recoils escaping from uranium fuel elements the figures indicate that the tolerances might even be satisfied without coatings at reasonable surface-to-volume ratios for the fuel elements.

A comparison of the numbers in Table IVB-13 brings out the striking conclusion that, for the open cycle, the manned and tug-tow aircraft have the same tolerances. This is radically different from the situation with the closed-cycle aircraft, where the tug-tow is more favorable by a factor of 10^4.

The figure of 10 per day for the open cycle, arrived at on the optimistic diffusion assumption, indicates that for a mean hold-up time of 0.1 day, all of the fission products could be allowed to escape into the air. For instantaneous emergence only one-thousandth can be allowed out.

In comparing the open and closed cycle for the tug-tow the two assumptions lead to opposite conclusions as to which is more favorable, by a factor of 10 in each case.

Shield Types

In considering the design of a shield for a manned airplane it is reasonable to start with the MO shield. It is not unlikely that the required ρt value for such a shield may be reduced somewhat. It does not seem likely that it will be reduced very greatly. Apparently the primary gamma rays, fast neutrons and secondary gamma rays are all contributing about the same biological effect according to the calculation made before. (The primary gamma and even the secondary gamma rays may be a bit behind the neutrons.) This indicates that the shield constituents are mixed in nearly optimal proportions. More hydrogen, for the purpose of slowing down neutrons, would increase the required ρt based on the lower limit for

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primary gamma rays, which is already 520 gm./sq. cm. Therefore, any reshuffling of the proportions of the present constituents cannot be expected to decrease the ρt value beneath 0.8 of the present value of 725.

From the absolute lower limit on primary gamma rays it follows that a ρt less than 0.6 times the present one is out of the question. Since there must be some degraded gamma ray build-up and some small deviation from the ideal gamma ray properties in order to take care of fast neutrons, the limit of ρt values is likely to be in the range above 450 gm./sq. cm. for any shield (i.e., above 0.7 times the present ρt needed for MO). The gap between MO shields and ideal ones is probably not too great.

In order to reduce shield weights it is desirable either to increase the average density or to find a method of moving high-density components in towards the interface between reactor and shield without causing too great an increase in the overall ρt required.

Probably the large attenuation factors needed for the manned airplane do not allow very much advantage to be taken of the differences in the attenuation required for fast neutrons and for bad gamma rays. In these circumstances shifting high-density components to the inside is unlikely to create a large saving. The main hope for reducing the shield weight of the manned aircraft is to increase the shield density without changing the nuclear characteristics which make it a shield.

A change in density of the type required may be produced by finding higher-density compounds of the identical shielding materials or by finding higher-density materials with the same nuclear characteristics. Perhaps the latter is the best hope. The search for high-density materials with the appropriate nuclear characteristics is a combined materials and nuclear physics problem. It is one of the guiding themes of the next section, on shield materials.

Although the main information on the possible densities is contained in the shield materials section, it may be useful to get an idea of the possible increase in density over the density of 5.8 gm./c.c. of the MO shield. A shield of comparable characteristics might be made by the use of a heavy metal hydride. Uranium hydride would be a relevant example if it were not for the neutron-multiplying characteristics of ordinary uranium. Because of the neutron multiplication, borated uranium hydride has received a small amount of consideration (45). Practically, the densities achieved are likely to lie in the range of 8 to 10 gm./c.c. (46).

In addition to making heavy metal hydrides, it is possible to conceive of a shield made of alternate layers of heavy metals and water. (This suggestion may seem a little impractical from the structural point of view, but its equivalent can probably be worked out as in the Canadian shield.) Since heavy metals are known with densities as high as 20 gm./c.c. it may be possible to attain shield densities in the range between 12 and 16. In this case, however, the problems of limited supply in the very heavy metals may be acute.

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As was pointed out in the section on neutron attenuations most of the experimental data applies to a particular type of shield, of which the MO shield and the Canadian shield are examples. Other types of shields have been conceived in the past. Although there will be no extended analysis of any other type of shield, at least two other possibilities should be mentioned here. They may help in visualizing how lower ρt values and more attractive density distribution may be achieved. They will be called the "high energy" shield and the "low density" shield.

The idea behind the high-energy shield is that a certain amount of shield thickness can be saved by capturing the neutrons at high energies. This saving takes the form of the elimination of the slowing down distance from high energies to thermal or slightly epithermal energies. In the conventional type of tested shield this slowing-down distance is not expected to be very long and the importance of the slowed-down neutrons in contributing radiation doses seems to be negligible (see the section on neutron attenuation). Nevertheless, the high-energy shield may have some advantages. By utilizing heavy material of high inelastic scattering cross section (this is part of the usual prescription for any shield) combined, for example, with boron in quite high concentrations, it may be possible to cut the neutron relaxation distance somewhat further than it would be cut by merely increasing the density of the present type of shield. The high-energy shield may be viewed as one of the more natural outcomes of the present direction of development of shields which are already in use.

Despite the fact that shielding tricks do not look too promising for the manned reactor, one rather radical idea is worth outlining. This is the possibility of the low-density shield. The low-density shield goes in the opposite direction as far as modification of physical properties is concerned. It is known that neutron attenuation in water is quite good, approximately the same as the neutron attenuation in present shields (see Tables IVB-12). The problem in water is to take care of the gamma rays, both secondary and primary. By borating the water to suppress secondary gamma radiation from (n, γ) reactions (note that no great contribution of secondary gamma radiation is expected to result in the water from the inelastic scattering of neutrons) it may be possible to decrease the secondary gamma ray problem to one which can reasonably be handled. In order to take care of the primary gamma ray problem, the density might be increased by dissolving lead and by using a gamma ray shield of heavy material at the reactor side of the shield (perhaps 30 cm of lead). The final result is to place a comparatively thin heavy layer near the reactor, where it will contribute the least weight, and only a moderately heavy layer extending to a fairly large distance beyond it.

Such a low density shield as is described in the last paragraph becomes a real possibility when the consideration shifts from the manned airplane to the tug-tow combination. In this case the primary gamma ray shielding required is extremely low and the shield should clearly be designed primarily as a light-weight neutron shield. Suppression of the secondary gamma rays rather than an attempt to handle them along with a large amount of primary gamma radiation is clearly called for.

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Low-density shields have been insufficiently investigated. The only known work bearing on the determination of their value is scheduled by the Oak Ridge shielding group, partially at the suggestion of the Lexington Project (47). The Oak Ridge investigation will start with some water-attenuation experiments and continue with boron, iron and lead solutions; possibly it may be extended to more complicated mixtures.

The discussion of shield weights for the tug-tow and for the pilotless aircraft (in fact for any system requiring only a thin shield) should really await better calculations than can be provided by the methods used in this report. It has already been pointed out that because the shield is thin the importance of gamma rays and of neutrons in energy regions which have not been considered here is increased. On the other hand, the fact that the attenuations required are in the order of 10^2 and 10^3 suggests that the whole shielding problem may be amenable to a solution by the Monte Carlo method if a suitable computing machine is used (48). Such an investigation of thin shields in general and particularly of low-density shields is strongly recommended.

Shield Materials

The preceeding section points out that the presence of heavy elements is desirable in order to attenuate gamma radiation. Also high densities are desirable in order to reduce shield thickness. Fortunately, heavy elements have high densities.

The following elements, shown with their density and melting point values, are of interest in this respect:

Element	Specific Gravity	Melting Point	Element	Specific Gravity	Melting Point
Osmium	22.5	2720° C	Tantalum	16.6	2850° C
Iridium	22.4	2450° C	Mercury	13.55	-39° C
Platinum	21.45	1773° C	Rhodium	12.44	1965° C
Rhenium	20.	3000° C	Ruthenium	12.2	2450° C
Gold	19.32	1063° C	Palladium	12.0	1554° C
Tungsten	19.3	3380° C	Thallium	11.85	303° C
Uranium	18.7	1130° C	Thorium	11.5	1845° C
			Lead	11.34	326° C

There are objections to many of these on the grounds of scarcity. Osmium, iridium, platinum, rhenium, rhodium, ruthenium, and palladium are in extreme demand. Nevertheless, the total production of these elements (combined) in the world is only about 20 tons per year, of which the United States production is about one-tenth. Since total shield weight may run as high as 100 tons, these elements would not permit many shields to be constructed. The elements might find limited use as materials for special layers.

Although tungsten is more plentiful than the elements just considered, the demand for it by industry in the United States is so great that its use

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as a shield material should be considered very carefully. Our reserves of tungsten are poor, and even at the peak production in 1943 only 50% of our needs was met by U. S. minerals.

The use of gold as a shielding material is largely complicated by its position as a monetary standard. At 35 dollars per ounce the value of 100,000 pounds, for use in a manned aircraft, would be about 50 million dollars. One-tenth of this amount, as in a tug-tow, seems reasonable.

Tantalum at 16.6 density is also very scarce and in demand for other uses where substitutes cannot be found for it. Peak U. S. production of tantalum metal was only about 60,000 pounds in 1943.

Uranium is one of the few high-density metals which is reasonably abundant. (Depleted uranium is all right for shielding.) The problem introduced by thermal neutron multiplication in uranium 235 and by fast neutron multiplication in uranium 238 has barely been looked at (45). It may not be too serious, and the thermal neutron reproduction can probably be suppressed.

The three most abundant heavy elements are mercury, thorium and lead. Of these, mercury is liquid and must be contained, and lead has a low melting point and its density is only 11.3.

On the basis of abundance in the earth's crust, high melting point, and relatively high density, thorium appears to be a potentially important shield material. It has an inelastic cross section which is considerably higher than that of lead, its density is slightly higher, and its atomic weight is considerably greater. Fission caused by fast neutrons leads to a neutron reproduction problem similar to that met in natural uranium, but probably less serious.

The use of thorium or uranium as a shield material in nuclear aircraft is complicated by the fact that both materials are valuable in other uses, in a possible breeding program, for example.

Elements of lower specific gravity than lead are relatively abundant. Molybdenum, nickel, and iron are examples. Since iron is the most abundant, it would of course be used in preference to the more expensive elements, nickel and molybdenum. These three elements have the advantage that they will permit a strong shield construction.

In the shield, next to the reactor core, temperatures may be high. Accordingly, either high melting point materials will be required or cooling must be provided. In general metals are good thermal conductors and it is, therefore, easy to cool them. On the other hand, oxides of some of these elements may serve as insulators to prevent heat loss from the reactor.

It may be desirable for the shield to contain an effective fast neutron absorber (see last section). Since boron and lithium are the only known elements which produce no hard gamma radiation on capturing a neutron and since they have large fast-neutron-capture cross sections they are probably

the best choice for this role. Boron is probably the better because its capture cross section is higher than that of lithium. The large cross sections of these elements are due to the isotopes, B^{10} and Li^{6}, which are present in natural material in amounts of 18.4% and 7.5% respectively. Because the supplies of boron and lithium are relatively large and the atomic weights are small it is not at all preposterous to consider the use of separated B^{10} or Li^{6} for shields.

Various borides can be produced, as well as alloys of a given boron content. Tungsten boride for example could be made to accomplish the dual role of stopping gamma radiation and absorbing fast neutrons. The melting temperature of such an alloy is very high; but while the temperature stability is excellent, the radiation damage is unknown. Probably in the shield, radiation damage is unimportant, but it may be wise to evaluate it prior to application.

To slow down fast neutrons, hydrogen is desirable in a shield. While various hydrides and hydrogen-bearing materials have been suggested, few have been tried under sufficiently exacting conditions. The hydrogen-bearing materials must resist radiation damage and temperature instability. Preferably they should be solids which would fit into the construction scheme.

Water is a satisfactory hydrogen source in some respects but must be kept cool.

Some metallic hydrides which have possibilities as shield materials are those of zirconium (ZrH_2), thorium (ThH_2), uranium (UH_3), titanium (TiH_2), cerium (CeH_3), and of the other rare earths. These hydrides are relatively stable up to 700° C, except for UH_3, (46), although some slow decomposition takes place. Cerium hydride is relatively stable up to 1000° C. Thorium and uranium hydride are the most dense.

The hydrides of the heavy metals tungsten and tantalum, and of molybdenum and chromium, are unstable. Because chromium hydride (CrH_3) has the highest hydrogen content per unit volume of any metallic hydride, it is worth considering, even though it may have to be contained under pressure.

One possible method of reducing the hydrogen vapor pressure over an hydride would be to put in a small amount of the pure metal with the hydride, since 1% of the unreacted metal can reduce the vapor pressure to about one-fifth its normal pressure. Also the use of heavy-walled cans should make it possible to contain hydrides despite thermal decomposition by maintaining a high hydrogen gas pressure in the can.

Although hydrogen diffuses readily through most metals, a 1/2 inch section of steel might contain a positive hydrogen pressure for sufficiently long to permit use of the less stable hydrides. The real problem, however, may be shifted to the creep of the container.

The use of a container may also make it possible to introduce other hydrogen-bearing materials (water, paraffin hydrocarbons) into the shield, and would remove the problems of thermal instability or instability under

irradiation by fast neutrons.

The introduction of both boron and hydrogen may be possible by the use of borohydrides with other elements. Sodium borohydride ($NaBH_4$), for example, has much better stability than either sodium hydride or boron hydride.

Table IVB-14 lists some data on possible shield materials:

Table IVB-14

PROPERTIES OF SHIELD MATERIALS

	Melting Point °C	Tensile Strength psi at 20° C	Tensile Strength psi at 1000° C	Specific Gravity
Boron	2300	----	----	2.3
Iron (Steel)	1535	80,000 - 200,000	2000 - 8000	7.9
Lead	327	2500	----	11.34
Thorium	1845	40,000	----	11.5
Tungsten	3380	200,000	10,000	19.3
Uranium	1130	60,000	----	18.7
Cerium hydride (CeH_3)	decomposes	----	----	3.4
Chromium hydride (CrH_2)	decomposes	----	----	6.76
Zirconium hydride (ZrH_2)	decomposes			5.67
Thorium hydride (ThH_2)				8.
Uranium hydride (UH_3)				10.92

In addition, a series of borides of chromium (CrB and Cr_3B_2), of cobalt (Co_2B, CoB_2, and CoB), of molybdenum (Mo_3B_4) and of tungsten (WB_2) are available and can be produced to large quantities. The physical data on these are lacking.

It is quite evident that an expanded research program is desirable to obtain additional data on various shield materials. Typical subjects for investigation are:

(a) Metal hydrides - manufacturing processes, time-temperature stability, decomposition temperature, vapor pressure, etc. must be investigated.

(b) Metal borohydrides - similar investigations.

(c) Radiation damage to hydrides, borides, etc.

(d) Canning problems - diffusion of hydrogen through various metals of increasing thickness.

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Shield Weights

The concept of an unperforated shield of uniform density and thickness surrounding a spherical pile is useful for orientation purposes, such as showing how the weight of shield will be affected by reactor dimensions, by the magnitude of the attenuations required or by the shield density. The estimation of shield weight in this manner involves (1) determining, for a given shield composition, the ρt which is needed to effect the required attenuation and (2) determining the density attainable with the given composition.

Table IVB-15 shows weights thus obtained for shields of MO and Canadian composition. Certain effects have been disregarded in these estimates. No advantage was taken of the possibilities of shadow shielding or of using shields whose density decreases outward, and no allowance was made for the extra weight which might be entailed by ducts, departure of the hole in the shield from sphericity or increases in ρt with curvature of the system. For an aircraft reactor, the shields for closed-cycle systems which are cooled by liquid metals may be approximated by a calculation which ignores ducts. Gas-cooled systems will probably have heavier shields than are figured by this technique.

The table includes a few cases which are based upon different densities than the actual densities of the MO and Canadian shields. The heavier densities would be realized if heavier material of equal ρt effectiveness could be substituted. The lighter densities might result from the necessity of providing gas ducts in the shield.

The whole foregoing analysis on shielding is an attempt to present a complex problem on which there is both inadequate theory and inadequate experiment. There are two reasons for the importance of a strong and early program. First, the shield weight almost completely controls the remainder of the basic engineering decisions underlying a nuclear-powered aircraft development -- to such an extent that the choices of open *versus* closed cycle, tug-tow *versus* manned aircraft, and gas cooling *versus* liquid-metal cooling cannot be finally made without firm shield weight estimates. Second, some of the simplifying assumptions in the present analysis -- and this represents a summary of the present state of knowledge in this country -- are sufficiently dubious as to require further clarifications before it would be safe to base any final engineering decisions on them. The present analysis, which indicates the tremendous uncertainties in every phase of this subject, might serve as a point of departure for a shielding program.

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Table IVB-15

CALCULATED SHIELD WEIGHTS

(Spherical Shells)

Shield Type	Aircraft Type	ρt Required	Density ρ	Shield Hole Dia.-cm.	Weight 1000 kg.
MO	Manned	625	5.8	60	63
				90	85
				120	110
	Pilotless	340		120	36
	Tug-Tow	285		60	11
				120	26
				180	47
	Manned	625	7.45	60	45
				120	86
			4.6	60	87
				120	140
Canadian	Manned	590	3.72	60	104
				120	160
	Pilotless	265		120	32
	Tug-Tow	215		60	10
				120	22
				180	39
	Tug-Tow	215	5.8	60	6
				120	17
				180	32
				240	52
			3.0	60	13
				120	26
				180	44

[redacted]

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Chapter V

AIRCRAFT DESIGN STUDY

Scope of the Chapter

Chapters III and IV have discussed the nuclear power plant, including the reactor and shield. This chapter discusses the other elements of the complete aircraft -- primarily the airframe. A synthesis of all the elements into the complete aircraft will be given in Chapter VII. In connection with the discussion of the airframe, certain matters relating the flight behavior of the aircraft will naturally be taken up.

It would be impracticable in a study of this sort to attempt to arrive at a detailed design of the aircraft. By the time such basic components as the reactor, power plant, and control system are available, the art of aircraft design will have progressed to a point where present concepts will have been greatly altered. A number of aircraft configurations have been studied, however, to determine whether or not it will be possible to meet the specifications demanded by the nuclear power plant, and as a result of these studies, a tentative standard aerodynamic efficiency (lift-drag ratio), and standard component weights have been developed, for use in the study of the complete aircraft in Chapter VII.

Performance Criteria

The various strategic considerations which influence the choice of performance goals are given in Chapter II. In selecting design performance figures, it is necessary to effect a reasonable compromise between tactical and design requirements. This has led to a selection of a design speed of 0.9 Mach number and design altitudes of 30,000 to 50,000 feet for the studies of this chapter, and for most of those of other chapters. Lower speeds and altitudes increase tremendously the probability of interception by the enemy. Higher speeds bring the aircraft into the supersonic region, where the aircraft efficiency, expressed as ratio of lift to drag, falls off greatly. Higher altitudes bring serious reductions in thrust available from a power plant of given weight and frontal area, as well as a decrease in airframe structural efficiency, since the wing must be larger and heavier to support the same weight in the rarefied upper atmosphere.

Peculiarities of Nuclear-Powered Aircraft

Nuclear-powered aircraft possess several fundamental characteristics entirely different from those of conventionally powered aircraft. The weight of nuclear fuel consumed in a flight is negligible. However, the weight of the power plant including the reactor and shielding is many times that of the conventionally fueled power plant. Because of the negligible consumption of fuel the nuclear-powered aircraft, if it will fly at all at a given altitude and speed, will fly many thousands of miles at that altitude and speed.

Since with a nuclear-powered aircraft there is no fuel weight loss during flight, the landing weight equals the take-off weight, minus the bomb load, which is only a small percentage of the gross weight. Consequently the landing weight is approximately double that of a similar chemically fueled aircraft, of which approximately half the take-off weight is fuel. The corresponding increase in landing speed (proportional to the square root of the ratio of landing weights) will be approximately 40 per cent. For this reason an aircraft whose wing area has been selected for optimum performance at lower altitudes may have to have its wing area increased to keep the landing speed within reason. Because of the characteristics of the nuclear power plant, jet assist will be required at take-off to a greater extent than is the case for normal jet aircraft.

Preliminary studies show that because of the shield weight a manned aircraft with nuclear power will necessarily be very large. At best, the speed and altitude performance of these very large machines cannot be improved without further very large weight increases.

THE TUG-TOW AIRCRAFT

It is apparent that the greatest reduction in total weight of the aircraft is to be obtained by reducing the thickness of shielding. Reasons have been given in previous chapters for regarding the pilotless aircraft as of little value. In general, radiation effects diminish as the inverse square of the distance from the reactor. It is therefore a logical step to place the crew and other sensitive equipment at a sufficient distance from the reactor to allow substantial reductions in shield thickness and weight. This leads to a study of the tug-tow concept. In this arrangement an unmanned remotely controlled nuclear-powered tug aircraft tows at a distance of about 3000 feet a smaller chemically fueled aircraft containing the crew, the bomb, and the communicating and control equipment necessary for flying the tug and accomplishing the mission. Such a combination reduces the required shield thickness by at least one half and the shield weight by at least two thirds.

Flight Plan of the Tug-Tow Aircraft

Using techniques which are presently in use or in process of development, the flight plan of the tug-tow combination might be as follows. The tug would take off under ground-directed radio control, unreeling the cable after take-off. The towed plane is taken off under its own power from a separate base. After a rendezvous has been accomplished, the towed aircraft assumes guidance of the tug, attaches itself to the cable, and then shuts off its power. During the run-in to and run-out from the heavily defended target area, the towed airplane might turn on its power plant to augment the thrust and so permit an increase in altitude or speed in the danger area. Upon return to base the towed airplane casts off and returns to its base under its own power while the tug would be guided in to a landing by ground-directed radio control. In this study it is assumed that the towed plane will have fuel sufficient for one and a half hours of flight -- fifteen minutes each for take-off and landing and one hour at the target area. Actually the tug-tow arrangement would permit repeated

missions without the necessity of landing the nuclear-powered tug between missions. The tug could be cruised under remote control until a new towed plane with a fresh crew was attached. The number of missions accomplished without landing the tug would be limited only by the reliability of the mechanical components of the aircraft.

Operational Advantages of the Tug-Tow Aircraft

From an operational viewpoint the tug-tow combination offers several advantages. The crew is not subjected to the hazardous take-off and landing operations of a nuclear-propelled aircraft. In the event of an accident during normal flight which might increase the radiation appreciably, the crew would have an opportunity to cast off and proceed under independent power to an emergency landing or ditching with the possibility of rescue.

While it may be extremely difficult to design the tug for very high altitudes and supersonic speeds, the problem will be greatly eased for the smaller chemically fueled towed aircraft. If such a high-performance towed airplane were used it would be possible for the towed plane to cast off outside the heavily defended target area, set the tug on a predetermined course, proceed to the target at extreme speed and altitude with reduced possibility of interception and then rejoin the tug using radio homing devices.

Other variations may suggest themselves. For example, fuel for the towed plane might be stored on the tug, and fed to the towed plane as required through a tube in the cable.

Design Advantages of the Tug-Tow Aircraft

The great advantage of the tug-tow combination is the much smaller size of aircraft that it makes possible -- in the range of present-day airplanes. Some weight increase can be tolerated without going to excessive sizes of airplane, and this increase in size can be used in many ways to ease the problems of design which afford the greatest difficulty. Thus the tug-tow could be designed if necessary for lower reactor wall temperatures, or lower turbine blade temperatures, or a shield giving better protection, than could the manned airplane, without giving up the advantages of a piloted aircraft. Or the altitude performance could be improved by the use of additional weight. For these reasons the development of a tug-tow nuclear-powered airplane can be expected to be completed in a considerably shorter time than that of a manned airplane.

THE CONFIGURATION OF THE STANDARD AIRPLANES

The procedure adopted in fixing on the standard component weights and lift-drag ratio was as follows:

For the manned airplane and for the tug of the tug-tow combination, a configuration was first selected. This was the same in the two cases, except that conventional landing gear was assumed in the tug, whereas for the

manned airplane, with its large size, a skid landing was assumed, 1½% of gross weight being allowed for strengthening the under parts of the airplane.

Next, information was obtained on an actual preliminary design of a conventionally fueled airplane of this configuration, weighing about 100,000 pounds gross. The detailed carefully computed weights of component parts were available.

For other gross weights, appropriate weight-estimating formulas were used to predict component weights (except power plant). Thus curves were built up showing, for each gross weight, the weight of the aircraft components except power plant (which includes reactor, shield, etc.). A similar procedure was followed for altitudes other than the design altitude for the basic 100,000-pound airplane.

The towed plane was assumed to be a conventional one of fixed size, regardless of the size of the tug.

The Selection of the Airplane Configuration

It will be the object of the designer to produce an airplane of given gross weight which will allow for the maximum possible weight of power plant per pound of thrust, since this will ease most effectively the problem of developing the power plant. In doing this, the designer has to balance two main factors. (a) The L/D or lift-over-drag ratio should be as high as possible so that the thrust required for a given weight of aircraft will be low. (b) The structural weight of the aircraft must be kept to a minimum in order to allow the highest possible value to the power plant weight. These two factors are to some extent incompatible. Maximum L/D is increased with larger wing span but at the same time structural weight is also increased so that there is an optimum value of wing span and L/D beyond which there is no benefit to be gained.

For the particular case of the nuclear-powered airplane, the design selected was a "delta" wing aircraft of an aspect ratio of 4.0, a wing thickness of 10%, with a relatively short body and no horizontal tail. This airplane has a low structural weight for the following reasons:

(a) The wing weight is reduced by the low ratio of wing span over wing root thickness, which reduces the wing bending moments and increases the moment of inertia at the wing root where the power plant weight is concentrated.

(b) The fuselage weight is reduced by reduction in length and the large part of the fuselage area enclosed within the wing root.

(c) The weight of horizontal tail surfaces is completely eliminated.

The low parasite drag largely overcomes the effect of the low aspect

ratio. As a result, the design has the satisfactory L/D of 15 at a Mach number of 0.9. In addition, the type is well adapted to a concentrated load, and the thick sections are particularly favorable for the introduction of the bulky reactor, shield, engines, and heat exchangers with little parasite drag. Control of this type of airplane is satisfactory.

This discussion is not meant to imply that the design selected is optimum; no extensive study of all possible designs could be carried out, and it is recognized that among designs with higher aerodynamic efficiencies will be some of satisfactory characteristics. It is believed, however, that none will be found to be more desirable to an important degree.

DESIGN DATA - WEIGHTS

Basic Design Information - Standard Manned Airplane and Tug

Basic design information for the manned airplane and the tug plane (identical for both) are given in Table V-1.

TABLE V-1

BASIC DESIGN INFORMATION

Gross Weight	101,000 lb.
Aspect Ratio ($Span^2$/Area)	4.00
Wing Area	2,800 sq. ft.
Tail Area (Vertical)	200 sq. ft.
Fuselage Length	77.40 ft.
Fuselage Diameter	8.33 ft.
Wing Span	105.80 ft.
Wing Root Chord (C_c)	47.67 ft.
Wing Tip Chord (C_t)	5.30 ft.
Taper Ratio (C_c/C_t)	9.00
Wing Thickness Ratio (t/c)	0.10
Wing Sweepback ($\Lambda c/4$)	36.00 °
Wing Loading (Gross Wt./Area)	36.10 lb./sq. ft.
Max. L/D for Cruise	15.0
Lift Coeff. (C_L) for Max. L/D for Cruise	0.25
Optimum Cruising Altitude	49,000 ft.

Basic Design Information - Towed Airplane

Since the towed plane is expected to be a conventional one, no elaborate calculations have been made. The L/D ratio is assumed to be 10, this rather low figure being adopted because the parasitic drag is expected to be rather high on a relatively small airplane carrying a bulky bomb. The assumptions are shown in Table V-2.

[redacted]

TABLE V-2

DESIGN DATA FOR TOWED AIRPLANE

Gross Weight	40,000 lb.
Crew	3
Guns	None
Bomb Load	10,000 lb.
Cruise Speed	Mach No. 0.9
Endurance at Max. Continuous Power	1.5 hours
L/D at Cruise Speed with No Power	10.0
Drag at Mach No. 0.9	4,000 lb.

Basic Weight Information - Standard Airplanes

The breakdown of the manned airplane weight into component weights is given in Table V-3, with dimensions derived from Table V-1.

TABLE V-3

GROUP WEIGHT STATEMENT

101,000 LB. MANNED AIRCRAFT

ITEM		WEIGHT (LB.)	% GROSS WT.
* Wing Group		13,155	13.03
* Tail Group (Vertical)		942	0.93
* Body Group		4,158	4.11
Fixed Equipment Group		5,959	5.90
Instruments	250		
Surface Controls	820		
Hydraulic System	277		
Electrical System	869		
Communicating	716		
Armament Provisions	1875		
Furnishings	1152		
Landing Provisions		1,515	1.50
Useful Load		11,144	11.03
Crew (3)	750		
Bomb	10,000		
Equipment	394		
Gross Weight Less Power Plant & Inst'n		36,873	36.50
** Total Power Plant Plus Installation		64,127	63.50
Installation of Power Plant Only		6,413	6.35
*** Total Power Plant		57,714	57.15

* Refer to Table V-1 for Basic Design Information

** Power Plant Installation Only is placed here to facilitate computations and would normally have been included in structural weight. Total structural weight, with this item included, equals 24.42% of gross weight.

*** Includes reactor, shield, machinery, heat exchangers, and ducts.

[redacted]

The tug aircraft has the same overall configuration and gross weight as the manned aircraft, only the detail weights being rearranged. The group weight statement for the 101,000-pound tug aircraft is given in Table V-4.

TABLE V-4

GROUP WEIGHT STATEMENT

101,000 LB. TUG AIRCRAFT

ITEM		WEIGHT (LB.)	% GROSS WT.
Wing Group		13,155	13.03
Tail Group		942	0.93
* Body Group		3,800	3.76
Fixed Equipment Group		4,968	4.92
Instruments	250		
Surface Controls	820		
Hydraulic System	277		
Electrical System	869		
Guidance System	1,752		
Communicating System	1,000		
Landing Gear Group		7,070	7.00
Gross Weight Less Power Plant & Inst'n.		29,935	29.64
Total Power Plant Plus Installation		71,065	70.36
** Installation of Power Plant Only		7,106	7.04
*** Total Power Plant		63,959	63.32

* Body Group is less than for the manned aircraft due to removal of crew and bomb provisions.

** Power Plant Installation Only is placed here to facilitate computations and would normally have been included in structural weight. Total structural weight with this item included equals 24.76% of gross weight.

*** Includes reactor, shield, machinery, heat exchangers and ducts.

Weight Estimates - Airplanes of Other Sizes

The next step is to estimate the component weights for other airplane gross weights, and also for other altitudes.

The weight statements in Tables V-3 and V-4 apply only to aircraft designed to fly most efficiently at approximately 50,000 feet where the optimum wing loading at a Mach number of 0.9 equals 36.1 pounds per square foot. At an altitude of 40,000 feet the air density is greater by a factor of approximately 1.6, so the optimum wing loading is increased proportionately to 58 pounds per square foot, which results in a smaller wing area. The wing area has not been reduced at altitudes below 40,000 feet for the

reason that 58 pounds per square foot for a delta wing is considered the maximum loading which will provide landing speeds not in excess of 180 miles per hour. Therefore, at 30,000 and 40,000 feet the aircraft size and gross weight are identical.

In making weight estimates for other wing loadings and gross weights, the weights of wing, body, and tail groups are extrapolated using commonly accepted weight-estimating formulas. The hydraulic and electrical systems are assumed to vary directly with wing area. The other weights, exclusive of the power plant and its installation, are held at constant values. In the manned aircraft 1-1/2 per cent of the gross weight has been allocated to skid landing provisions, while in the tug aircraft 7 per cent is allowed for conventional landing gear. In the tug-tow combination, the bomb and bomb sight are located in the towed aircraft.

Figures V-1 and V-2 give the ratio of group weights to gross weight plotted against gross weight for the manned aircraft at 30,000 to 50,000 feet. In Figures V-3 and V-4 similar data are given for the case of the tug aircraft.

Figure V-5 gives the ratio of allowable power plant weight to gross weight plotted against gross weight for all altitudes selected and for both the manned and tug cases. This is the end result of subtracting the summation of the group weights given in Figures V-1 through V-4 from the gross weight. This graph gives the final information to determine the maximum power plant weight which can be allowed and still have the airplane perform with its ceiling at the altitudes specified. All that remains in determining the primary factors in the design of the power plant is to determine the thrust required.

DESIGN DATA - THRUST

Determination of Thrust Requirements

The thrust required for the manned aircraft will be equal simply to the gross weight divided by the L/D ratio of 15. Dividing the allowable power plant weight by the thrust gives the maximum allowable specific weight of the power plant. Considering the tug aircraft, the required thrust will be equal to the sum of the drags of the tug, the tow cable, and the towed aircraft. The drag of the towed airplane is seen from Table V-2 to be 40,000 pounds divided by the L/D ratio of 10, or 4000 pounds. The drag on the tow cable is discussed next.

Aerodynamic Loads and Resultant Forces on the Tow Cable

A towed aircraft can be located with considerable latitude either above or below the tug as long as its location is clear of the aerodynamic and exhaust wake of the tug. However, it is considered more efficient to have the tow cable trailing down in order that the resultant air loads will contribute to rather than detract from the lift of the combination. For this reason the towed aircraft is located at a point behind and down on a 10-degree angle from the tug.

If a free cable unweighted at the end is suspended in a free air stream, it will seek an angle where the lift, drag, weight and cable tension forces are in complete balance. The diagram and equations for the solution of these forces are shown in Figure V-6, where the lift and drag forces of the cable are derived from the corresponding coefficients shown in Figure V-7. When the forces on the cable are in balance the cable will trail aft in a straight line because the aerodynamic and weight loadings are constant along its length and thus there are no forces present to induce bending. Now if the towed aircraft is attached to the end of the free trailing cable, the resultant of its lift, drag and weight forces must lie in a direction along the cable and thus this resultant drag is simply added as an increment to the tension due to the cable drag alone.

Figure V-8 presents a graphical trial and error solution to find the normal trailing angles and drags of a 3/8 inch diameter 1 x 19 cable, 1000 feet long. The cable is assumed covered with a smooth plastic coating. At any trail angle assumed, the required weight for this angle and the drag has been calculated from the equations in Figure V-6. This has been calculated for several angles and plotted against trail angles. The point where the actual cable weight of 318 pounds intersects the required weight, determines the natural trail angle of the cable. The results indicate trail angles of approximately 5, $7\frac{1}{2}$, and 13 degrees for altitudes of 30,000, 40,000, and 50,000 feet respectively. With the towed aircraft at a 10-degree position behind the tug, the cable will be straight at an altitude of approximately 45,000 feet. At lower altitudes the cable will be bowed slightly upwards while at higher altitudes it will sag slightly. The drag of the towed aircraft will tend to straighten the cable so that it is felt justified to approximate the cable drag on the assumption of a straight cable at ten degrees. This drag for a 3000 foot cable will be 1575, 980, and 610 pounds for altitudes of 30,000, 40,000, and 50,000 feet respectively. An average drag of 1000 pounds has been used for the cable since the variations therefrom have little effect on the end computations for required thrust. If a thicker cable is desired the cable drag will increase in proportion to the diameter.

Adding the 4000-pound drag for the towed plane to the 1000-pound drag for the cable gives a total of 5000 pounds tension in the cable at its attachment to the tug. Since a 3/8 inch steel cable has a breaking strength of 17,800 pounds it is considered to have ample margin of strength. Since the L/D of the tug has been estimated at 15, the thrust required of the tug is

$$\frac{\text{Gross Weight of Tug}}{15} + 5000 \text{ pounds}$$

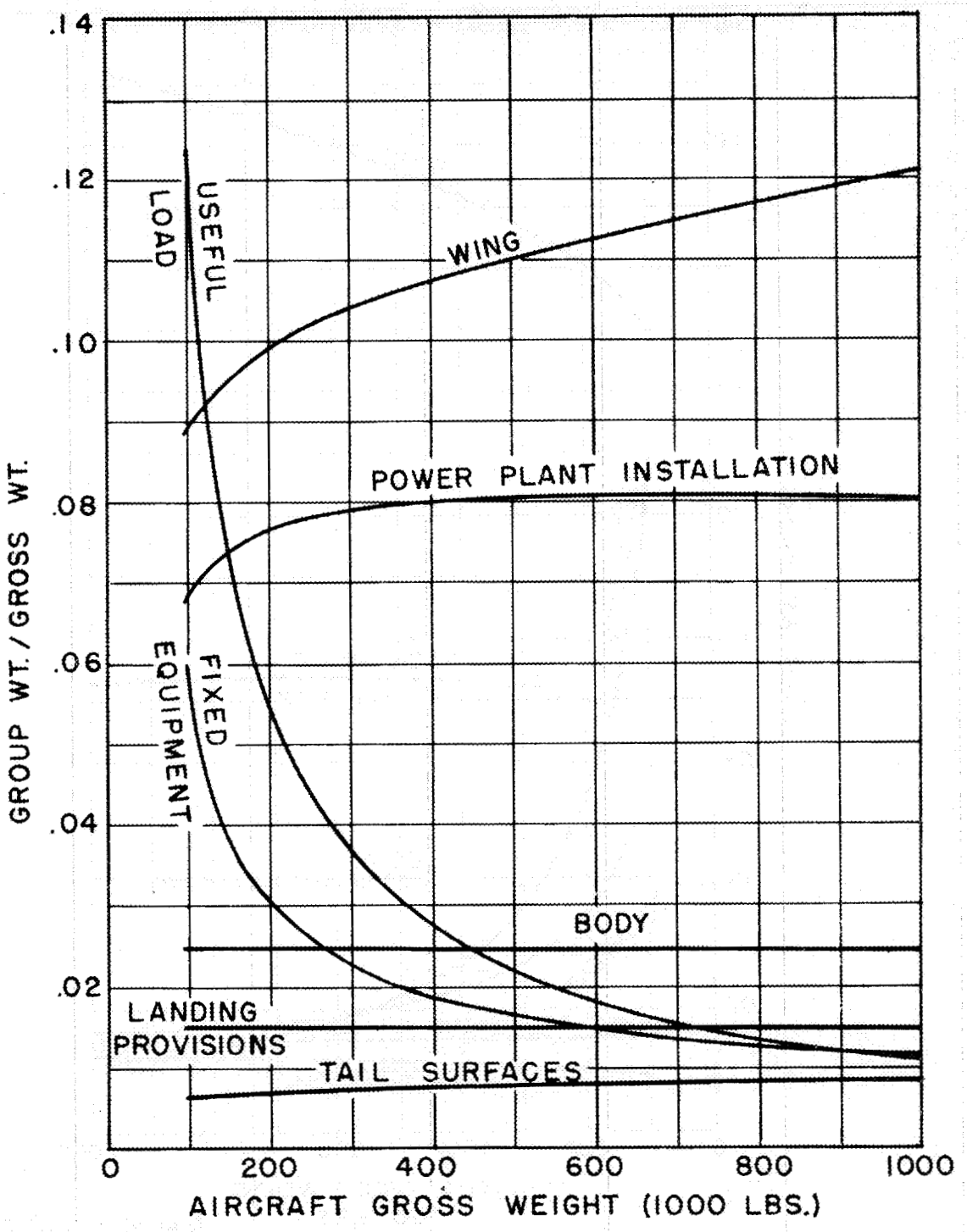
The corresponding formula for the thrust of the manned aircraft is of course

$$\frac{\text{Gross Weight of Manned Aircraft}}{15}$$

These formulas, together with allowable values of power plant weight given in Figure V-5, are used in calculating the results presented in Chapter VII.

The large saving in weight for the tug-tow arrangement which Chapter VII illustrates represents a much more sound base upon which to develop future plans. Low weight with margin left for growth eases every design problem and may well change the future status of nuclear-powered flight from the improbable category to the probable.

[redacted]



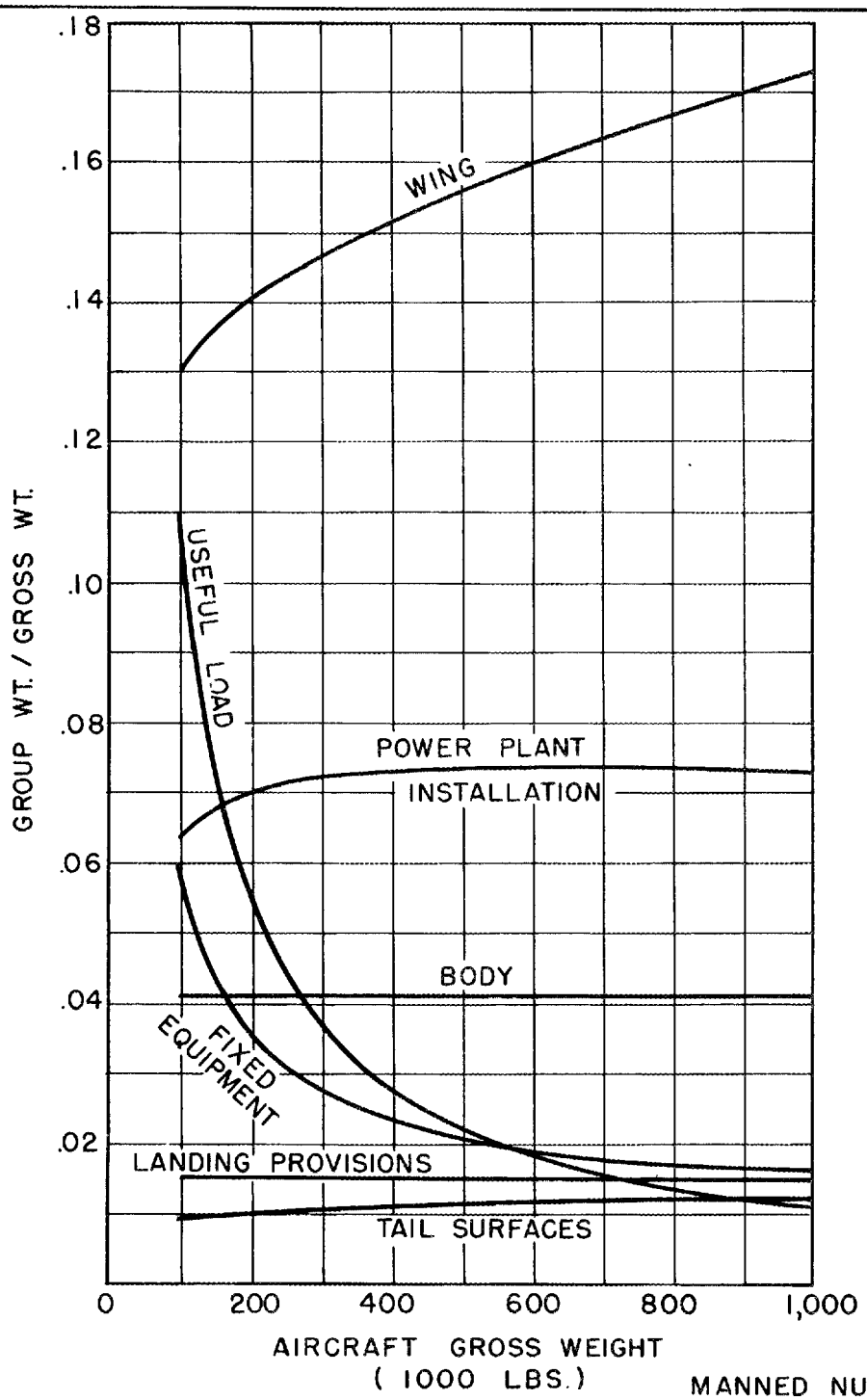
MANNED NUCLEAR AIRCRAFT AT 30,000 AND 40,000 FT. MACH NO. 0.90

FIGURE V-1

GROUP WT./GROSS WT.
VS.
GROSS WEIGHT

[redacted]

[redacted]



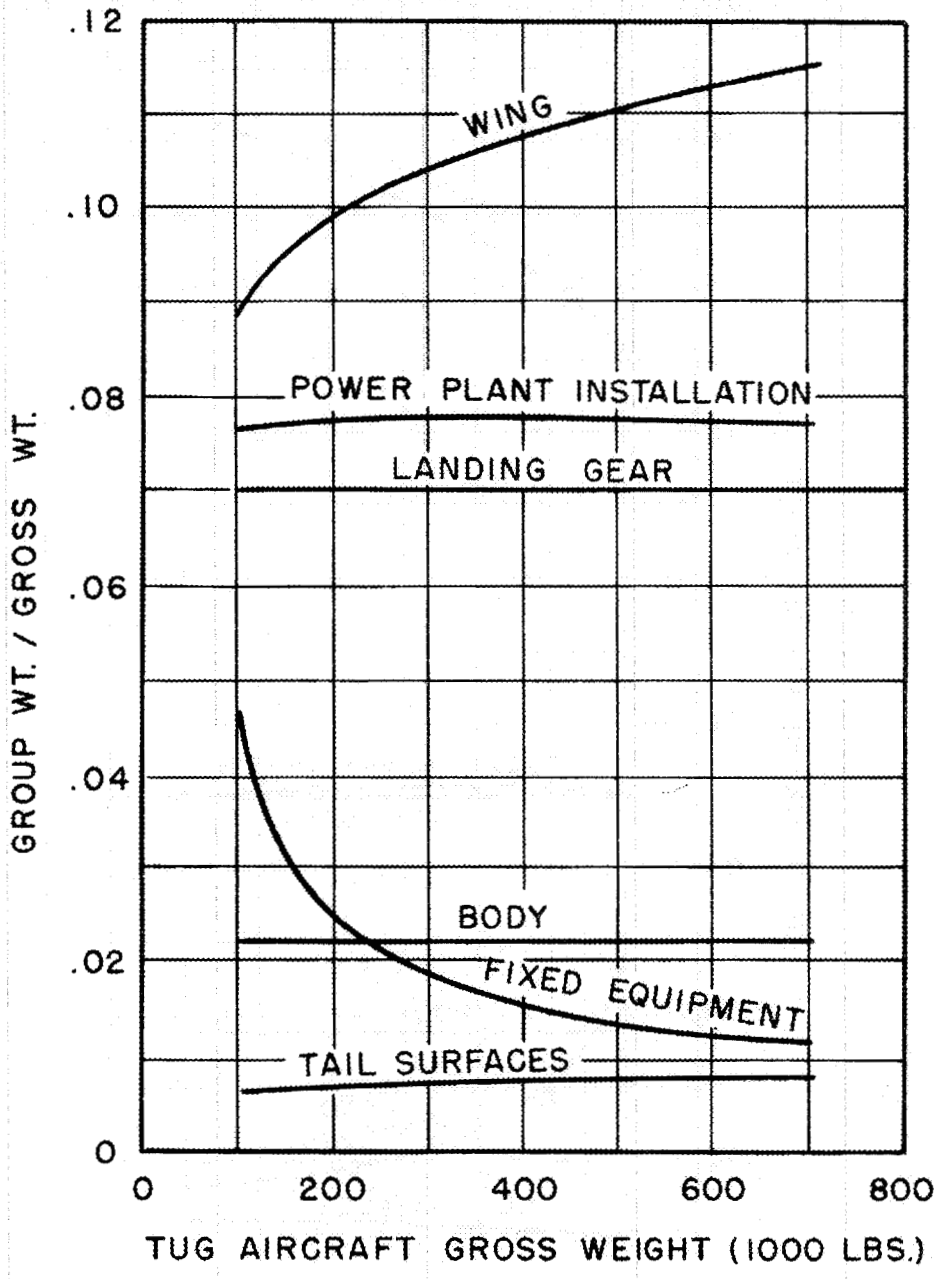
MANNED NUCLEAR AIRCRAFT
AT 50,000 FT.
MACH NO. 0.90

FIGURE V-2

GROUP WT. / GROSS WEIGHT
VS.
GROSS WEIGHT

[redacted]

[redacted]



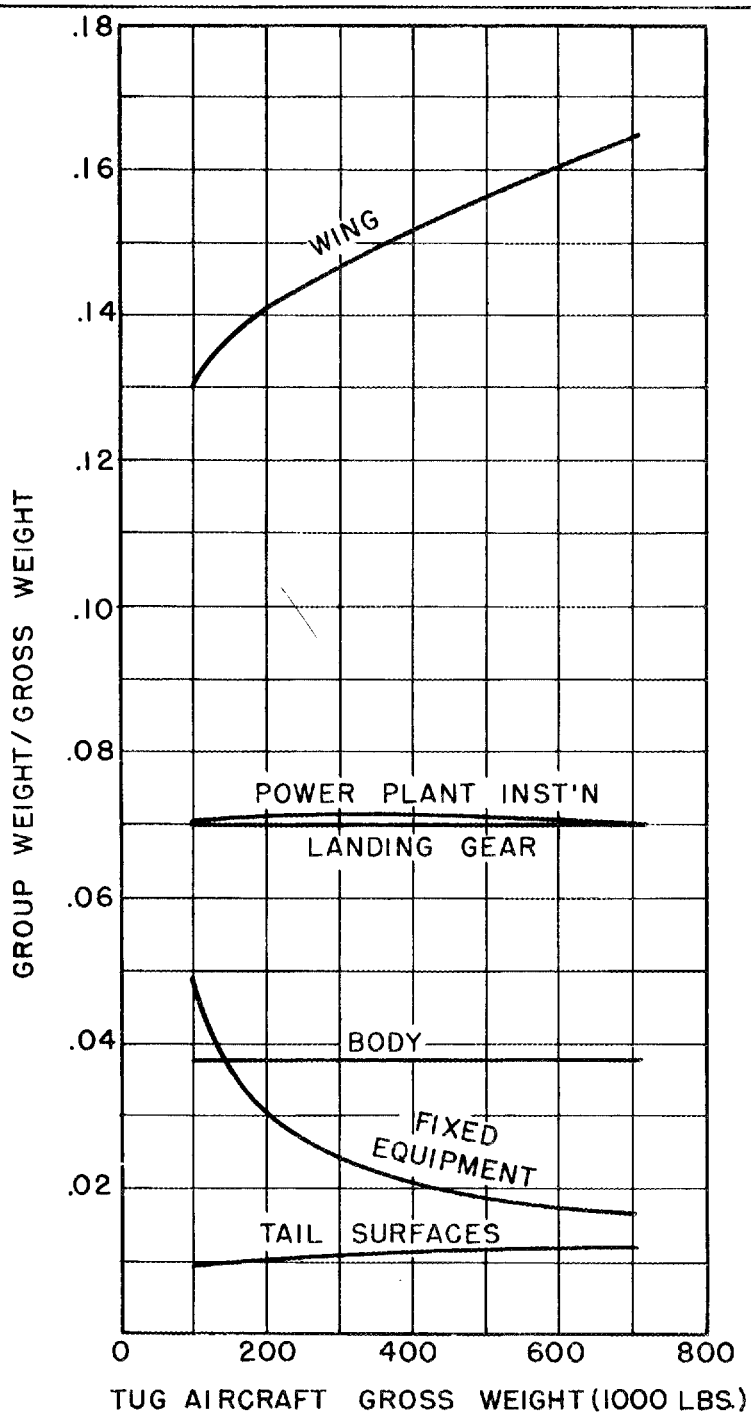
TUG NUCLEAR AIRCRAFT
AT 30,000 AND 40,000 FT.
MACH NO. 0.90

FIGURE V-3

GROUP WT./GROSS WT.
VS.
GROSS WEIGHT

[redacted]

[redacted]



TUG NUCLEAR AIRCRAFT
AT 50,000 FT.
MACH NO. 0.90

FIGURE V-4

GROUP WT./ GROSS WT.
VS.
GROSS WEIGHT

[redacted]

[redacted]

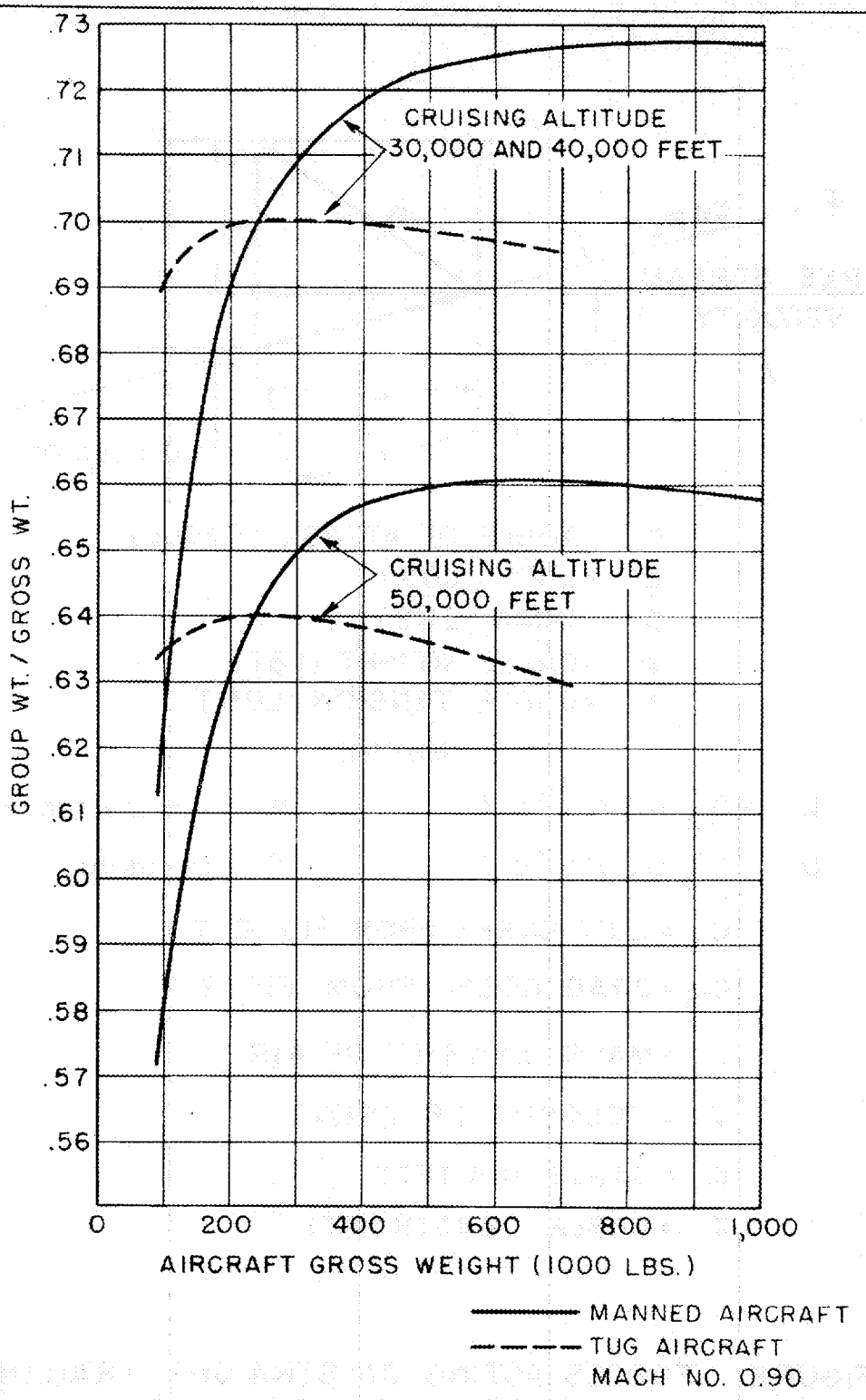
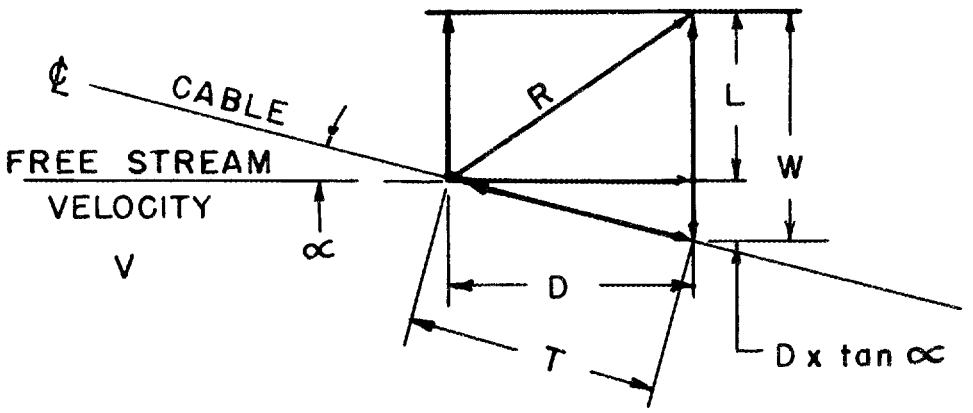


FIGURE V-5 ALLOWABLE POWER PLANT WT./GROSS WT. VS. GROSS WEIGHT

[redacted]



∝ = ANGLE OF ATTACK (CABLE)
L = LIFT (LBS.)
D = DRAG (LBS.)
W = CABLE WEIGHT (LBS.)
T = CABLE TENSION (LBS.)

WHERE

$L = C_2 \, 1/2 \, \rho V^2 d \ell$ $\qquad$ $W = L + D \times \tan \alpha$

$D = C_D \, 1/2 \, \rho V^2 d \ell$ $\qquad$ $T = D/\cos \alpha$

C_L = LIFT COEFF. FROM FIG. V-7

C_D = DRAG COEFF. FROM FIG. V-7

ρ = MASS DENSITY OF AIR

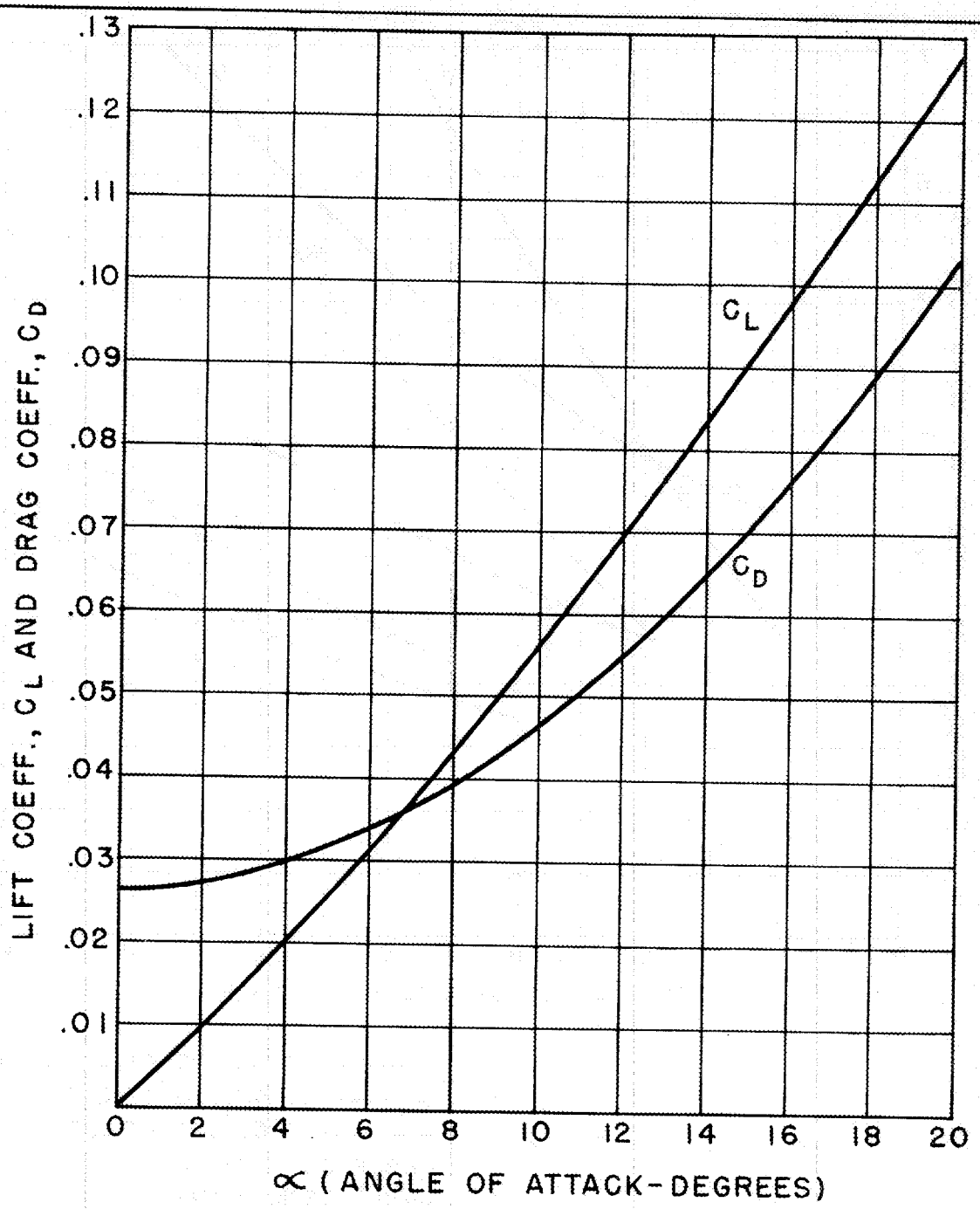
V = VELOCITY (FT./SEC.)

d = CABLE DIA. (FT.)

ℓ = CABLE LENGTH (FT.)

FIGURE V-6 FORCES ACTING ON STRAIGHT TRAILING CABLES IN A FREE AIRSTREAM

[redacted]

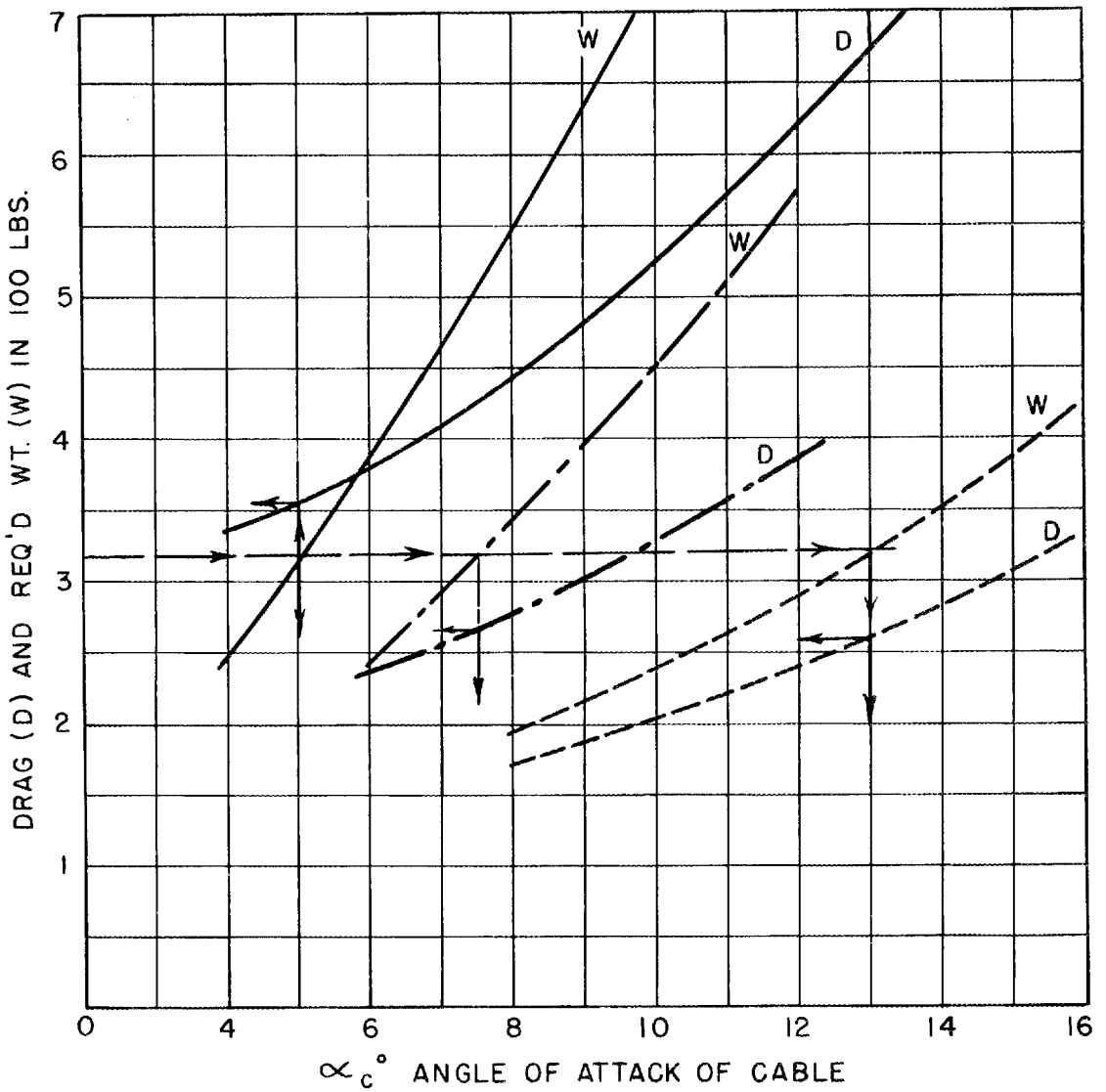


FROM H.D. ZIT, LUFTFAHRTFORSCHUNG-
VOL. 14, P. 307, FIG. 7 (1937)

FIGURE V-7 LIFT COEFF., C_L AND DRAG COEFF., C_D
VS.
ANGLE OF ATTACK-DEGREES

[redacted]

[redacted]



——— AT 30,000 FT. ALTITUDE
—·—·— AT 40,000 FT. ALT.
– – – – AT 50,000 FT. ALT.
CABLE DIA. 3/8 INCH
CABLE LENGTH 1000 FT.
MACH NO. 0.90

FIGURE V-8 DRAG AND REQ'D WEIGHT FOR SMOOTH CABLE VS. ANGLE OF ATTACK

[redacted]

Chapter VI

PROBLEMS OF FULL-SCALE TESTING

Much of the expense of the overall development program for nuclear-powered flight will be incurred in the period which begins when the preliminary design of a full-scale test-stand power plant has been decided upon and which ends with the first successful flights of a nuclear-powered plane. Many problems must be solved before this period arrives, but it is essential to consider the type of obstacles which can be expected and to get some idea of the order of magnitude of money and effort which it will entail.

In this review it has not been possible to draw on previous studies to any great extent. Planners in the field have been concerned primarily with much earlier phases of development, and have worked under contracts which did not call for the production of a flyable airplane. They have given consideration to the more remote steps involved, but not at all intensively, and have prepared hardly any written material.

Test Stand for Power Plant

The views of NEPA on the test stand for the power plant were obtained from conversations with their personnel, and some brief references in a report. (1) Their views may be summed up as follows:

The test stand will be designed for ultimate full-scale operation of both reactor and associated power plant. In the earliest stages, however, the reactor, though of full size, will be operated at low power, and will be cooled by a separately driven system (in their case an air blower, since they contemplate an open cycle). At some stage in the study of gradually increasing powers, the turbo-compressor units will be brought into use.

Such an order of development seems reasonable. In applying it to other than the open cycle, the question of using a cooling system specially designed for the test stand, or a system identical with that proposed for the airplane, arises. In principle, either system appears feasible, and the choice should be determined by considerations which cannot be evaluated until a late stage of the development.

There is a good deal of question as to the allowable location of the full-scale test stand. The reviews of various proposed reactor installations by the Safeguard Committee, together with conversations with two members of the Committee (2) suggests that a fairly remote area will be required. The power in the reactor is quite high; the reliability of the cooling must be regarded as questionable at the time when the unit begins to operate with turbo-compressor (or equivalent). Suitable areas in the continental United States can no doubt be found.

This question of location is made more acute by the necessity for providing considerable amounts of power, and in some cases of cooling,

at the test-stand site. The requirements differ for the open and the closed cycle. For the open cycle, air must be blown through the reactor. If this is to be done by a compressor system different from that in the final power plant, this compressor will require a large amount of power, since the full-scale compressors for the final power plant will consume at least 100,000 horsepower. (3) Alternatively, by recycling the air at high pressure and cooling it as it circulates, the power required can be reduced, but a large amount of cooling would be needed. Even if, in accordance with the proposal of NEPA, testing with the external compressor system is carried to only (say) one-fourth of full power, the power requirement will be a serious consideration in the selection of the site.

The closed-cycle power plant reactors could be tested separately from the power plant by providing adequate cooling for the heat-transfer fluid. This would call for either a fairly substantial water supply (one ton per second might be typical) or a large cooling tower to do the equivalent cooling. In this case the complete power plant, with the reactor simulated by some other source of heat, could be tested in some more convenient location, and not at the test-stand site. Eventually it will be essential to test the reactor and the power plant together on the ground.

The above considerations apply to testing at ground-level conditions. Testing the complete power plant on the ground at altitude conditions would involve a further very large expenditure either of power (if fresh air were used for cooling) or of cooling water (if the air were recirculated at low pressure). The coolers for the purpose would be in themselves a very bulky installation.

It will be obvious from these considerations that the cost of the test stand -- both capital and operating -- will be considerably higher than for any piles so far constructed or authorized. It should be noted that the pile itself is here the object of study, and that alterations and reconstruction of the original setup, or more probably the installation of entirely new models, are to be expected. It is quite certain that the cost would be measured in hundreds of millions of dollars.

The Chemically Fueled Prototype Plane

Whether or not the airplane to be driven by the nuclear power plant turns out to be beyond present size ranges, it is highly probable that it will be specially designed, and not a standard, tested unit. To put an experimental power plant in an experimental plane is to increase the chances of crashes very seriously -- and crashes will be costly. NEPA proposes to overcome this difficulty, in part at least, by the construction of prototype planes, as nearly identical as possible to the nuclear-powered design, on which flight characteristics, controls, and the like could be tested. For the open cycle they propose to use identical compressors and turbines, and simply to replace the reactor by a combustion chamber.

The concept of a chemically fueled prototype appears to be an important contribution to the economy of development of nuclear-powered flight,

even though it involves a notable increase in the design work on the airplane. Far from lengthening the time required for development, the prototype method should shorten it by permitting the correction of design errors in the airplane concurrently with the power plant development. It may be found practicable to make the chemically fueled prototype a somewhat smaller plane than the nuclear-powered design, at some further saving in money and time. In view of the importance of reducing the chances of a crash to a minimum, the advantages of this expedient should be carefully examined before it is adopted.

If the tug-tow combination is chosen, the development of the towed airplane, which should be relatively straightforward, could parallel other developments, but work on picking up and casting off the tow could well be done with the chemically powered prototype.

The Manufacture of the Airplane

At some stage in the operaiion of the test stand, it will be necessary to freeze the airplane design, make fairly full drawings of the nuclear-powered plane, adapt these for the chemically powered prototype, and then manufacture first the latter and then the former.

The difficulty of the task will depend considerably on the size of the plane. If it lies within the range of planes in production at the time of manufacture, the difficulty will presumably be no greater than for any plane of entirely new design. If the very large planes which are indicated for the manned type xre to be made, it may well be necessary to erect a special plant for the construction; and it must be expected that a series of bottlenecks will be encountered in the plants of component suppliers, because of the excessive size of the various elements.

Since the airframe may be expected to differ no more from conventional airplanes than these differ among themselves, some reasonable approximation of the cost of this item may be obtained. Assuming an original order for (say) two airplanes of 300,000 lbs. gross weight each, the cost of the airframes alone (exclusive of power plants) might be something like $70,000,000, and this supposes manufacturing methods suited to a small number of planes, not a large investment for mass production. If the airframe is larger than the largest in production at the time, the cost may be expected to go up disproportionately.

Airfield for Nuclear-Powered Planes

Even allowing for preliminary work with the chemically powered prototype, the flight testing of the nuclear-powered plane must be looked upon as exceptionally hazardous. Not only will the power plant be of a new type, but the effect of a crash will be more serious. It would presumably be comparable with the failure of any other pile, and would release intensely radioactive and poisonous materials, which can be expected to be deadly over an appreciable area. The immediate neighborhood of the crash will be uninhabitable for a long time. As in the case of other piles, mere interruption of the cooling of the reactor will produce these results, so that a relatively minor crash may be dangerous. Flight test-

ing must therefore be done in a very remote area. It is questionable whether any area in the continental United States will be acceptable, in view of the very considerable area required for planes traveling near the speed of sound. An island would seem to offer the most satisfactory solution.

The flight strips may have to be unusually long, since the landing speeds, as noted in Chapter V, will tend to be high. The installations at the site will be fairly extensive. Bringing the airplane to the airfield may be a big problem in itself. Normally, planes are flown from the factory; this procedure can hardly be considered in this case, and it may well be that the choice of factory and airfield location will be affected by the possibility of moving the plane from one to the other (possibly by carrier). Other factors to consider are the problems of removing the reactor from the plane at times, providing transportation for it to a processing plant, draining off coolant and general servicing of non-nuclear parts.

The Take-Off of the Nuclear-Powered Plane

If the tug-tow type of aircraft, with its lightly shielded reactor, is adopted, the airfield must incorporate special provisions to protect ground personnel from the radiations from the pile while the power plant is warming up. Presumably this will mean something like a heavy concrete hangar. Such provisions would not be needed with the manned plane, but if it employs the open cycle, it would require a stack to take care of the jet air, which would be radioactive.

Landing the Nuclear-Powered Plane

The main additional point requiring attention after landing a nuclear-powered aircraft is the fact that after the pile has been shut off so that fission no longer occurs, the radioactivity in the fission products will continue to generate gamma rays and heat at a considerable rate for a long time after landing. In the manned airplane the gamma rays will easily be cared for by the shield, and even if the unmanned plane close approach may be possible after a few hours. But in both cases the heat evolved will be (assuming a mission of some hours) 3% of full power ten seconds after shutting down, and 1% after an hour. Thereafter there is a further slow decline, but some cooling will be required on the landed plane for a long time after the main engines have stopped and cannot furnish the cooling. (4)

NEPA (5) has considered this problem, and expects to take care of it by a small closed-cycle cooling system, entirely independent of the main system, to go into effect on shut-down. No details of the system are available, and in the absence of any final design it would be difficult to provide any. In the closed cycles it might be enough to provide some auxiliary means of circulating the heat-transfer fluid through the reactor-exchanger system. In no case will it be possible to rely on natural convection, as is done in some existing piles.

Time Scale

No really significant estimate of the time required for the full-scale testing can be made. Apart from the obvious uncertainty as to the obstacles to be encountered in the various parts of the program, the time required will depend to a great extent on the effort expended, and the degree of the latter cannot readily be specified. However, two figures are presented to furnish some idea of what the time scale might be.

(a) NEPA, as a result of a conference with the member companies, (6) has recently prepared a tentative time scale. This calls for a period of 6½ years from the beginning of the test-stand power plant design to the completion of power tests. A further period of three years would elapse before a nuclear-powered aircraft was ready for flight test.

(b) A typical period for the development of a new airplane, or of a new aviation engine, is five years.

Conclusions

A review of the problems of full-scale testing does not disclose any difficulties which appear impossible of solution. It does present a picture of large expenditures, intensive design effort and careful planning, and the prospect of costly setbacks if failures are encountered.

REFERENCES

(1) NEPA Technical Report No. 4, Part I, Section 1.37 (1947).

(2) LP-100, Meeting with Dr. Manson Benedict; notes by E. W. Thiele, July 23, 1948.

(3) LP-144, Problems of Full-Scale Testing of Nuclear Power Plants for Aircraft (prepared from information submitted by NACA).

(4) K. Way and E. P. Wigner, "The Rate of Decay of Fission Products," Physical Review, vol. 73, p. 1318 (1948).

(5) NEPA Report No. 633 - 1 ER - 5, The NEPA Technical Program, Part I, p. 23 (1948).

(6) Report of Meeting, Board of Consultants and Member Companies, NEPA Project, August 11-12, 1948, Fig. 3.

Chapter VII

PERFORMANCE POSSIBILITIES OF NUCLEAR-POWERED AIRCRAFT

The integration of all the factors that have been discussed in Chapters II through VI into a specific aircraft design is an exceedingly complex problem. Many compromises must be effected. Uranium requirements must be balanced against reactor size, shield composition and dimensions against allowable radiation dosage for crew and payload, open against closed-cycle operation, and a host of other variables that affect ultimate aircraft performance must be weighed and evaluated.

For the present state of the art, no "optimum" solution of the problem is apparent. The best that can be done is to set up several hypothetical designs on the basis of known requirements, make certain basic assumptions, and then study the effects of the introduction of the several variables in turn. Such a procedure has been followed in this chapter.

Specifications for three "standard" manned aircraft have been chosen. The power plants are: The open-cycle air turbojet, the bismuth turbojet and the helium gas-turbine compressor-jet. For reasons outlined below, a series of curves of flyable shield thickness vs. airplane gross weight have been used to show the effect on aircraft performance of varying the critical design parameters.

As a final step a tabulation of twenty-five illustrative design combinations based on the curves is presented at the end of the chapter.

GENERAL DESIGN CONSIDERATIONS

The probable uses of nuclear-powered aircraft, as described in Chapter II, are for long-range bombing or reconnaissance missions. Aircraft for such missions must have high-speed and altitude performance, the minimum being a Mach number 0.9 and an altitude of 30,000 feet. The payload (bombs or photographic and radar-mapping equipment) is assumed to be approximately 10,000 pounds.

The shield requirement is of primary importance in determining the size and weight of the aircraft. Of the three different types of nuclear-powered aircraft which have been considered -- manned, pilotless and tug-tow -- the manned aircraft, with its crew in close proximity to the nuclear reactor, requires the thickest shield. For the protection of the crew in the towed aircraft, the nuclear-powered tug aircraft requires less than half the shield thickness required by the manned. The pilotless aircraft apparently needs roughly half of the shield thickness required by the manned, for the protection of the atomic bomb and the long-range guidance equipment.

Most of the illustrative examples in this chapter are based on a

shield thickness of 4 feet at specific gravity 6 for the manned plane, and of 2 feet at specific gravity 6 for the tug-tow combination and for the pilotless aircraft. These figures are considered to be fair estimates of a safe thickness at a shield specific gravity which is attainable using existing knowledge.

The discussion in this chapter of the effects on performance of varying certain parameters is centered on the manned aircraft type. The emphasis on the manned aircraft arises primarily from historical reasons and does not represent a judgment as to its merit relative to the tug-tow. Because of the advantagesof the tug-tow combination, resulting from the smaller required shield thickness, a detailed comparison of the manned and the tug-tow aircraft is made early in the chapter. For reasons discussed in Chapter II, the pilotless aircraft does not appear attractive, so it receives but slight attention throughout. All three types are illustrated in the list of possible designs at the end of the chapter.

The special features of nuclear-powered aircraft which affect the aerodynamic configuration and the structural analysis are the concentrated load of reactor and shield, and the relatively large heat exchangers required by some power plant cycles. These two features can be taken into account without fundamental departure from current aeronautical design practice. It is thus possible to make overall performance estimates by using known aircraft performance data without laying out specific nuclear-powered airplanes in detail.

The two parameters, lift-drag ratio and power plant weight as a percentage of gross weight, are the major aeronautical design factors which have affected the results in this chapter. No particular significance should be attached to the fact that a "delta" wing configuration was used in the aircraft calculations, since any of several configurations would give figures for the above two parameters within a few per cent of one another. When the time comes to design a plane for flight test, considerably more detailed aerodynamic design will be required.

The lift-drag ratio for the design analysis was selected as 15. This should be attainable at a Mach number of 0.9 without much difficulty. The percentage of power plant weight (including weight of shield) to the gross weight of the loaded plane was selected as 72% at 30,000 and 40,000 feet and 65% at 50,000 feet for all airplanes weighing more than 300,000 pounds. At lower gross weights the percentage drops off, as described in Chapter V.

The power plant studies of Chapter III have indicated at least three basic types of jet engines which seem to offer some promise for use in nuclear-powered aircraft: the open-cycle air turbojet, the bismuth turbojet, and the helium gas-turbine compressor-jet. Because of the widely differing nature of the integrated aircraft propelled by each of these power plants, separate performance analyses have been made for each.

An important factor in the integrated design is the balance between the fuel (usually considered to be uranium 235) requirement and the size

of the reactor required to remove the heat. A nominal uranium investment of 100 pounds has been selected for most of the cases treated in this chapter. The necessary balance can be provided for each design by utilizing a graph of the type of Figure VII-0 on which is plotted the diameter of the reactor versus the free-flow ratio. In Chapter IV it was shown that the reactor diameter increases with increasing free-flow ratio when the moderating material and uranium investment are fixed, in a manner indicated by Curve I of Figure VII-0. In Chapter III the free-flow ratio was for illustrative purposes fixed at 0.4, and the diameter of the reactor was determined by heat transfer requirements, regardless of the uranium investment. However, the heat transfer requirements for a square-cylinder reactor can be met by other combinations of reactor diameter, D, and free-flow ratio, FFR, (all other things excepting fuel investment and tube diameter remaining unchanged) according to the relation

$$D^2 \text{ (FFR)} = \text{Constant}$$

as illustrated by Curve 2 of Figure VII-0. Where the line dictated by heat transfer requirements crosses the line corresponding to the chosen uranium investment and chosen moderator, the diameter and free-flow ratio satisfy both heat transfer and nuclear requirements. (Point A on the chart)

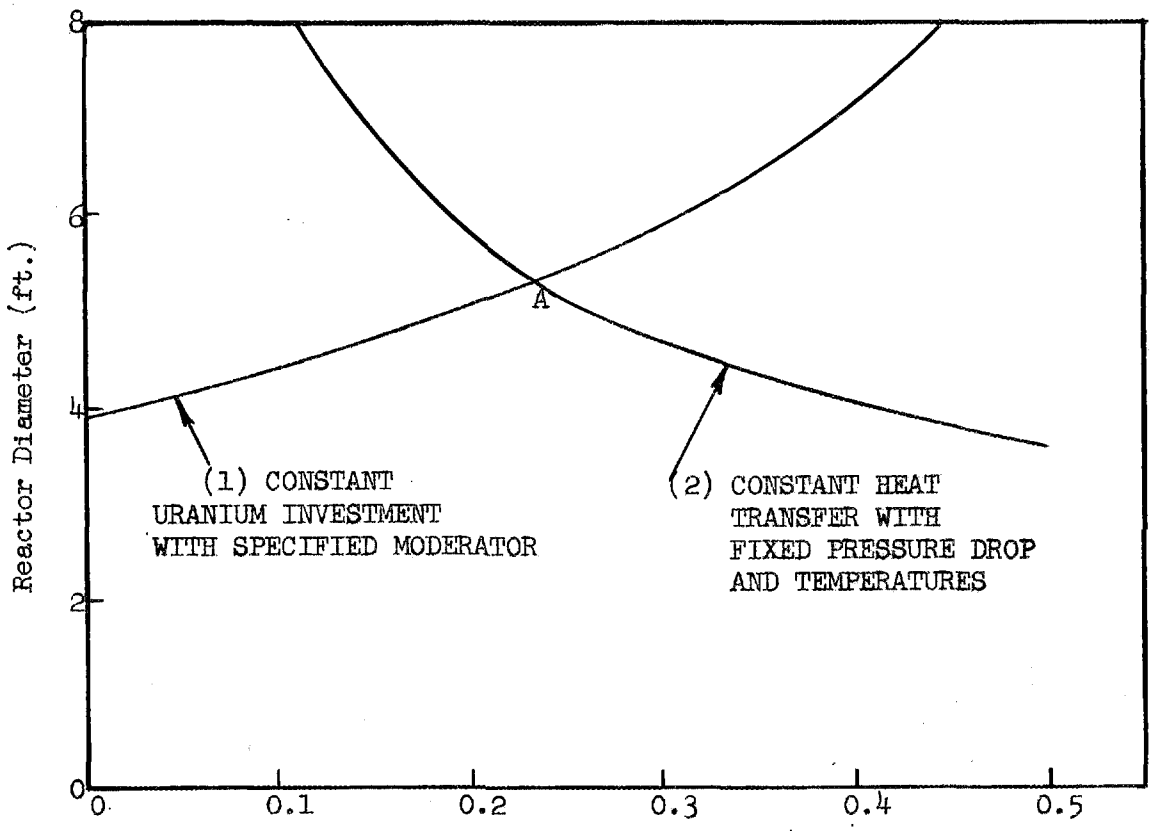


FIGURE VII-0

THE METHOD OF PRESENTING THE RESULTS

Graphs of flyable shield thickness *versus* gross weight of airplane have been selected for presenting the results of design studies, for comparing the types of aircraft and power plants, and for showing the effect on aircraft performance of varying critical design parameters.

The flyable shield thickness is defined as the thickness of shield whose weight is just equal to the load which can be carried by the aircraft after all other necessary components have been taken into account. It is described more fully in the following account of the steps which were taken in the design study calculations.

After the type of aircraft and power plant, and the operating altitude and speed are selected, the weight which can be allowed for the reactor, the power plant proper and the shield can be calculated for any selected gross weight of airplane. This is the weight which remains after the weights of the payload, the aircraft structure, the crew and the fixed equipment are subtracted from the gross weight. The weight of the power plant and reactor components can be determined by the thrust required to fly the plane with its previously selected gross weight and assumed lift-drag ratio, as described in Chapter III. When the reactor and power plant weights are subtracted from that which had been allowed for the reactor, the power plant and the shield, the remaining weight is that which can be allowed for the shield. The flyable shield thickness is then calculated by disposing this allowed weight of shield material around the reactor whose size has already been determined from the combined considerations of uranium investment, heat transfer, and power plant performance.

The method of calculating the shield weight which results from a certain thickness of shield requires some explanation. In Chapter IV certain shield weights are estimated on an idealized assumption that the reactor is spherical and that the shield is a spherical shell around it. Calculations of weight for a given thickness based on this assumption result in minimum shield weights for a given reactor diameter. In the present chapter the shield weights are estimated on a more conservative assumption that the reactor is a cylinder and that the shield around it is also cylindrical. The assumption of a cylinder results in heavier shields for a given shield thickness than are calculated from the spherical assumption.

The section on shields in Chapter IV discusses this problem. For present purposes it may be adequate to say that (1) the true answer will have to wait for detailed studies of the shape and thickness of shielding which a specific reactor design will require and (2) the cylindrical assumption used in this chapter may result in a closer approximation to the right shield weights at a given thickness for reactors cooled by air or helium in which large gas ducts run through the shield, but is probably too conservative for reactors which are cooled by liquid metals.

The nuclear-powered plane design which results from calculations of flyable shield thickness is practical only if the thickness will give

satisfactory radiation protection to the most sensitive element in the system whether it be a man or a piece of equipment. The flyable shield thickness increases with an increase in gross weight of the airplane if all design parameters are held fixed. This means that, for any design, a minimum gross weight can be found at or above which the aircraft can carry a sufficiently thick shield to provide the necessary radiation protection. A simplifying assumption is involved that the shield thickness for radiation protection does not increase significantly with an increase in the power output of the reactor. Because of the exponential attenuation of radiation in shield material, resulting in only a few inches increase in thickness for a ten-fold increase in power, this is a fair preliminary assumption.

The graphs of flyable shield thickness *versus* airplane gross weight are plotted for shield thicknesses from one to five feet and for gross weights from zero to one million pounds. For purposes of orientation the maximum take-off weight, including gasoline and load, of the B-29 is about 150,000 pounds, of the B-36 is about 300,000 pounds and of the Hughes flying boat is about 450,000 pounds. A weight of one million pounds was selected as maximum because it is thought that practical aircraft are unlikely to exceed this.

The uncertainties in the efficiencies of shielding material make necessary the above described manner of presentation of the results. Although figures of 4 feet at specific gravity 6 for the manned plane and 2 feet at specific gravity 6 for the tug-tow and pilotless plane are used most often in the examples in this chapter, other values, perhaps lower, may prove to be desirable as a result of a shielding development program. The graphs are usable for any thickness between 1 and 5 feet, which certainly covers the range of interest. Furthermore, correction factors can easily be calculated if it is desired to convert the assumption of a cylindrical shield to one involving some other shape.

In comparing curves of flyable shield thickness *versus* airplane gross weight, the uppermost curve usually represents the best performance since it gives the lowest minimum gross weight for a given shield thickness. There are, however, some notable exceptions to this rule, which will be pointed out where they occur.

THE THREE STANDARD AIRCRAFT

Table VII-1 presents the design parameters for three aircraft, driven by different types of power plants. These will be referred to throughout as the "standard" aircraft. These arbitrarily selected designs will be used as points of departure in showing the effect on performance of varying separately each of several critical design parameters. They are not necessarily the optimum nor the recommended designs.

COMPARISON OF THE OPEN-CYCLE TURBOJET, THE BISMUTH TURBOJET AND THE HELIUM GAS-TURBINE COMPRESSOR-JET

Figure VII-1 compares the three power plants, all with a moderator of graphite. With the exception of the moderators for the bismuth turbojet and for the helium compressor-jet, the curves are for the standard aircraft listed above. Figure VII-2 compares the three power plants, all with a moderator of beryllia. The curves are for the standard aircraft with the exception of the moderator for the open-cycle air turbojet.

On these two graphs the bismuth turbojet is superior since a smaller plane is capable of carrying a shield of given thickness. The gross weight of the manned plane (shield thickness of 4 feet with specific gravity 6) propelled by a bismuth turbojet is 525,000 pounds with a moderator of beryllia, and 925,000 pounds with a moderator of graphite. The helium gas-turbine compressor-jet is intermediate, with a manned plane weighing 650,000 pounds using beryllia, and slightly greater than 1,000,000 pounds with graphite. The open-cycle air turbojet is poorest, with a manned plane weighing 900,000 pounds using beryllia, and considerably more than 1,000,000 pounds with graphite.

It is interesting to note from these curves that pilotless aircraft (shield thickness of 2 feet with specific gravity 6) can be designed for a minimum weight varying from 100,000 pounds for the most favorable case to 300,000 pounds for the worst.

In Figures VII-1 and VII-2, each point on each curve corresponds to a different reactor diameter and free-flow ratio, the values of which were determined by the procedure previously described on the basis of the curves of reactor diameter versus free-flow ratio derived in Chapter IV. Curves of reactor diameter so determined, as a function of the aircraft gross weight corresponding to the curves of Figures VII-1 and VII-2 are shown in Figures VII-1a and VII-2a. These curves make clear the reason that for the same gross weight aircraft, beryllia moderated reactors lead to greater flyable shield thicknesses than do graphite-moderated reactors; the reactor sizes involved are considerably smaller for the former. The curves also show why the bismuth turbojet leads to the greatest flyable shield thicknesses.

The maximum reactor-wall temperature used in the standard open-cycle air turbojet aircraft is much higher (2500°F) than those for the standard aircraft powered by either the bismuth turbojet (1840°F) or the helium gas-turbine compressor-jet (1830°F). In addition there are many other points which make the three cases not directly comparable. Because of these differences it would be a serious mistake to assume that the simple comparison given above gives the whole story. There are many pros and cons for each of these systems, such as vulnerability, state of development, structural nature, material requirements, temperature requirements and power plant frontal area required, details of which are discussed in Chapters III, IV, and V.

TABLE VII - I

THE THREE STANDARD AIRCRAFT

	Open-Cycle Air Turbojet	Bismuth Turbojet	Helium Gas-Turbine Compressor-jet
Moderator	Graphite	Beryllia	Beryllia
Maximum Reactor Wall Temperature	2500°F	1840°F	1830°F
Turbine Inlet Temperature	1800°F	1500°F	1540°F
Coolant Temperature Leaving Reactor	-	1650°F	-
Coolant Temperature Entering Reactor	-	1180°F	-
Reactor Inlet Pressure	-	-	1000 psi
Reactor Pressure Drop	30% of Inlet Pressure	-	-
Air Compressor Pressure Ratio	40	6	-
Coolant Pumping Power	-	3% of thrust power	-
Component Efficiencies - Foreseeable			
Air: Diffuser	90%	90%	90%
Air: Compressor	82%	83.5%	85%
Air: Turbine	87%	87%	-
Air: Nozzle	90%	90%	92%
Binary Fluid: Compressor	-	-	85%
Binary Fluid: Turbine	-	-	88%

Nominal Uranium 235 Investment	100 lbs. for all three cases
Shield Specific Gravity	6 " " " "
Reflector Thickness	0 inches " " " "
Design Altitude	30,000 feet " " " "
Lift to Drag Ratio	15 " " " "
Mach Number	0.9 " " " "
Type of Aircraft	Manned " " " "

COMPARISON OF THE TUG-TOW AND THE MANNED AIRCRAFT

Figures VII-3, VII-4, and VII-5 compare the tug-tow combination with the manned aircraft for each of the three standard cases. In these comparisons the gross weight of the tug and not the total weight of tug plus towed plane is plotted. These are cases where the uppermost curves do not represent the best performance. The reason is that, if a shield of approximately 4 feet thickness is required for the manned aircraft, a shield of only approximately 2 feet thickness or even less is required for the tug-tow aircraft.

For the open-cycle turbojet with the above shield assumptions, a manned aircraft carrying a four-foot shield exceeds considerably 1,000,000 pounds, whereas the tug aircraft of the tug-tow combination with its two-foot shield weighs 325,000 pounds, slightly more than the B-36.

For the bismuth turbojet a manned aircraft weighs 525,000 pounds, whereas the tug aircraft weighs only 150,000 pounds, about the same as a B-29.

For the helium compressor-jet a manned aircraft weighs 650,000 pounds, whereas the tug aircraft weighs only 150,000 pounds.

Incidentally, the reason that the curve of the tug-tow combination falls below the one for the manned aircraft is that the tug must devote part of its power plant thrust to overcoming the drags of the cable and of the towed plane. Thus, less thrust is available for flying the shield. A figure of 5000 pounds thrust has been used in these calculations as that needed to overcome 1000 pounds drag from a 3000-foot cable plus 4000 pounds drag from a 40,000-pound towed plane with a lift-drag ratio of 10. The method of arriving at these figures is described in Chapter V.

EFFECTS OF VARIATIONS OF PARAMETERS FOR THE OPEN-CYCLE AIR TURBOJET

Effect of Change of Moderator Material - Figure VII-6

Beryllia has better nuclear-moderating properties than does graphit This fact permits the use of a smaller reactor using beryllia, with a consequent saving in shield weight. The advantage in the use of beryllia is shown by Figure VII-6 to be considerable, since a manned plane weighs 900,000 pounds using beryllia, whereas it weighs considerably more than 1,000,000 using graphite.

A fast reactor does not appear on these graphs of an air-cooled unit since such a reactor containing only 100 pounds of fuel is too small for the removal of the amount of heat required to produce the necessary thrusts (except for very small thrusts, which are of no interest because practically no shield could be carried).

Graphite has been chosen as the moderator for the standard open-

cycle air turbojet example. However, it is possible that beryllia will be more satisfactory at temperatures well above 1800°F even though it is not entirely satisfactory around 1800°F. This would be the case if it were shown that beryllia developed sufficient creep at 2500°F to absorb thermal stresses. At 1800°F beryllia definitely does not creep sufficiently to withstand thermal stresses. Hence, structural metals must be used for support.

Effect of Variation of Uranium Investment - Figure VII-7

The flyable shield thickness for a given aircraft gross weight is not very sensitive to quite a wide variation in uranium investment. A variation from the figure of 100 pounds selected for the standard aircraft up to 200 pounds or down to 50 pounds changes the aircraft gross weight by less than 50,000 pounds.

It should be noted that the figures quoted for uranium investment are those which would produce the power, without any consideration of the deleterious effects of fission product poisons and structural materials and the desire to get uniform radial heat distribution and high temperatures in the reactor. When all these factors are considered the uranium investment could well be twice the figures used above.

Effect of Variation of Shield Specific Gravity - Figure VII-8

The curves for shield specific gravity are also exceptional in that the lower curves represent better performance. This is due to the fact that the protecting power of a shield depends upon the product of shield thickness and specific gravity so that a thickness of 2.4 feet of specific gravity 10, or a thickness of 1.6 feet of specific gravity 15 is equivalent in radiation protection to 4.0 feet thickness of specific gravity 6, the standard case. To show this relationship better on the graphs, points which represent shielding equivalent to 4.0 feet of specific gravity 6 are put on the two lower curves.

The curves show that shield specific gravity very strongly affects the gross weight of the aircraft. For example, a manned plane, with a shield of specific gravity of 6 and a thickness of 4 feet, weighs more than 1,000,000 pounds. With a shield specific gravity of 10 and a thickness of 2.4 feet, the manned plane weighs 750,000 pounds, and with a shield specific gravity of 15 and a thickness of 1.6 feet, the manned plane weighs 550,000 pounds.

A shield specific gravity of 6 was chosen for the standard aircraft because it could be attained with only a small amount of development effort. A shield specific gravity of 10 might be feasible with considerable development. A shield specific gravity of 15 is probably close to an upper limit because it approaches that of the heaviest pure materials known, most of which are extremely scarce in nature. The curves, however, emphasize the desirability of a program aimed at developing shields of high specific gravity.

Effect of Variation of Reflector Thickness - Figure VII-9

This effect is not great, as can be seen from the curves which are plotted for a shield specific gravity of both 6 and 10. The effect that does appear is purely a geometrical one, since the curves were obtained by keeping the reactor fixed in characteristics, and surrounding it by a reflector which in turn was surrounded by the shield. This method of calculating the effect of adding a reflector is somewhat misleading. Actually the shield required for a given radiation protection would not be so thick if a reflector is used because the reflector itself acts to a considerable extent as a shield. Moreover, the reflector reduces the uranium requirement and produces a more uniform temperature distribution in the reactor. Reflector materials such as beryllia and graphite act as heat insulators as well.

All of these factors favoring the use of a reflector compensate largely for the adverse geometrical factor. Thus the effect of adding a reflector is relatively small, justifying the simplifying choice of no reflector for most of the calculations in this chapter.

Effect of Variation of Maximum Reactor Wall Temperature - Figure VII-10

The curves show very little difference in performance for the three reactor wall temperatures of 2300°F, 2500°F, and 2700°F. At temperatures lower than these the performance begins to decrease markedly. It is clear that there is only slight improvement possible in the performance of an open-cycle air turbojet due to an increase in the maximum reactor wall temperature above 2300°F when the turbine inlet temperature has been fixed at 1800°F. On the other hand, if it is at all possible to employ a reactor wall temperature of 2300°F, it probably will not be difficult to increase this temperature to 2500°F or 2700°F because the materials which might be used at 2300°F will probably be non-metallic materials with still higher temperature potentialities.

Effect of Variation of Turbine Inlet Temperature - Figure VII-11

Curves are plotted for turbine inlet temperatures of 2100°F, 1800°F and 1500°F. A comparison of these curves shows that considerable gain in performance will be obtained by an increase in turbine inlet temperature from 1500°F to 1800°F. However, an increase in this temperature above 1800°F results in little gain in performance if the reactor wall temperature is limited to 2500°F.

1800°F was chosen for the standard aircraft because it is thought probable that metallic turbine blade materials can be developed within five years that will withstand turbine inlet temperatures of 1800°F at present day working stresses. Presently available turbine blades are operating up to 1500°F. If turbine inlet temperatures are raised to and above 1800°F, it will be necessary to resort to different types of materials than are now used, or to special methods of cooling the blades.

Effect of Turbine Inlet Temperature at Lower Maximum Reactor Wall Temperature (1830°F) - Figure VII-12

This graph has been included because a drop in the maximum reactor wall temperature from 2500°F to the region of 1800°F will open up many possibilities in reactor materials. With this lower maximum reactor wall temperature, it is clear that improvement of turbine blades from the present maximum temperature levels around 1500°F to a temperature of about 1650°F would be desirable. If the reactor wall temperature is fixed at 1800°F, further improvement in temperature performance of the turbine blades would not greatly improve the overall performance of the aircraft.

The performance of the open-cycle air turbojet is very poor with this maximum reactor wall temperature of 1830°F. A manned plane with a four-foot shield would be quite impracticable. A pilotless aircraft with a shield thickness of 2 feet would weigh between 300,000 and 400,000 pounds.

Effect of Variation in Altitude - Figure VII-13

The performance of the open-cycle air turbojet is extremely sensitive to the choice of operating altitude as is shown by a comparison of the curves for 30,000, 40,000, and 50,000 feet. For example, a pilotless aircraft with a shield of 2 feet thickness at 30,000 ft. weighs 275,000 pounds; at 40,000 feet it weighs 350,000 pounds, whereas at 50,000 feet it weighs 950,000 pounds.

Variation of Compressor Pressure Ratio - Figure VII-14

For a nuclear-powered turbojet it is desirable to employ compressor pressure ratios greater than those for a chemically fueled turbojet because high pressure ratios increase the air density in the reactor and also make the cycle less sensitive to large pressure drops in the reactor; both these effects are in the direction of reducing the size of the reactor. The curves indicate, however, that the pressure ratio does not have a critical effect over a fairly wide range of values, say from 20 to 60.

A compressor pressure ratio of 40 has never been achieved in practice and will require considerable development to achieve it in a reasonable time. But since this design parameter can be freely chosen by the designer, a value lower than the optimum, say perhaps 20, would probably be used in order to simplify some of the compressor and intercooler development problems. As is shown by the curves, such a choice would not be seriously detrimental to the overall performance.

Effect of Variation of Reactor Pressure Drop - Figure VII-15

The curves which are plotted for pressure drops of 20%, 30%, and 40% of the total pressure at the entrance of the reactor show that the results are insensitive to this design factor as long as it is in the

neighborhood of 30%. The curves indicate that 30% is near an optimum. Reactor pressure drop is another design factor which can be chosen freely by the designer, so it is worthwhile optimizing it.

Effect of Variation of Component Efficiencies - Figure VII-16

A marked gain in performance results from an improvement in power plant component efficiencies. The efficiencies used for the standard aircraft are those thought to be obtainable after several years of intensive development. They are: diffuser efficiency - 90%, compressor efficiency - 82%, turbine efficiency - 87%, and nozzle efficiency - 90%. It is shown by the curves that considerable gain in performance will be realized by such a development. Presently obtainable efficiencies are taken as: diffuser efficiency - 70%, compressor efficiency - 78%, turbine efficiency - 83%, and nozzle efficiency - 90%. A curve for 100% efficiency for all components has been included to indicate the theoretical limit on the possible improvement in performance along these lines.

Effect of Variation of the Lift-Drag Ratio - Figure VII-17

The performance would benefit from an increase in the lift-drag ratio resulting from improvements in aerodynamics. A lift-drag ratio of 15 has been chosen for the standard aircraft since this is obtainable today with careful aerodynamic design. A curve for a ratio of 18 is also shown since it is not unreasonable to expect improvements of this magnitude. A curve for a lift-drag ratio of 25 is also shown. Attainment of such a lift-drag ratio would require novel developments such as the practical realization of boundary layer control.

It should be noted that any improvement in the lift-drag ratio obtainable with a chemically propelled aircraft can probably be incorporated into a nuclear-propelled aircraft design.

Comparison of the Turbojet with the Turboprop - Figure VII-18

In this comparison the turboprop and turbojet engines operate under the same conditions. The two curves for the turbojet and turboprop at the lift-drag ratio of 15 show that, for a propeller efficiency of 80%, the turboprop has a slightly better overall performance. This slightly improved performance is not particularly significant and would probably not justify the additional mechanical complications in the gearing and the high-speed propeller of the turbojet. Moreover, the cases are not directly comparable inasmuch as the lift-drag ratio for the turboprop would not be as high as that for the turbojet because of the adverse effect of the propeller slipstream on the wings. For this reason a curve for the turboprop with a lift-drag ratio of 12 is also shown. Comparing on this basis, the turboprop has a slightly poorer overall performance.

Incidently, considerable development will be required to obtain a propeller efficiency of 80% at a flight Mach number of 0.9 as assumed for the turboprop in the above computations, hence the estimates of turboprop performance are optimistic in this respect.

A big advantage of the turbojet over the turboprop is revealed as the speed is increased above a Mach number of 0.9 into the transonic and supersonic regions. The propeller efficiency drops off markedly, leaving the turbojet vastly superior. On the other hand, at speeds below about Mach number 0.6 the turboprop will be significantly better than the turbojet.

Comparison of the Manned Plane with the Tug-Tow at the Two Altitudes of 30,000 ft. and 50,000 ft. - Figure VII-19

As was pointed out earlier, the tug-tow combination requires a shield thickness of less than half that required by the manned aircraft, perhaps only 2 feet as compared with 4 feet for the manned plane. Thus, the curve for the tug-tow, although slightly below that of the manned plane, actually represents a much superior performance. For example, at 30,000 feet a tug plane of 325,000 pounds would be satisfactory for the tug-tow combination whereas a manned aircraft would weigh considerably in excess of 1,000,000 pounds.

Comparison of the Manned Plane with the Tug-Tow at Lower Maximum Reactor Wall Temperatures - Figure VII-20

These curves show that with a reactor wall temperature of only 1830°F the tug aircraft of the tug-tow combination weighs 650,000 pounds, whereas a manned aircraft is impracticable.

EFFECTS OF VARIATIONS OF PARAMETERS FOR THE BISMUTH TURBOJET

Effect of Change of Moderator Material - Figure VII-21

Beryllia has better nuclear-moderating properties than has graphite. This permits the use of a smaller reactor, with a consequent saving in shield weight. As was the case for the open-cycle air turbojet, the advantage in the use of beryllia is considerable, as shown by a comparison of the curves of Figure VII-21. A manned plane with beryllia would weigh 525,000 pounds, whereas with graphite it would weigh 900,000 pounds.

A curve is plotted for a fast reactor, using plutonium as the fissionable material. Also, an extremely short curve is plotted for a fast reactor using uranium 235 as the fissionable material. These curves are terminated at the upper end because the reactor using only 100 pounds of fissionable material becomes too small for the removal of the power. It can be seen that the use of fast reactors might improve the performance of the aircraft considerably in those ranges of power plant size which permit the removal of the heat. For example, manned planes may be flyable in the 300,000-pound region and tug-tow aircraft in the 100,000-pound region using fast reactors. It should be noted that an increase in the size of a fast reactor in order to go to larger aircraft gross weights will require an additional investment of fissionable material. This tendency is opposite to that encountered in the moderated reactors.

Beryllia has been chosen in preference to graphite as the moderator for the standard bismuth turbojet because of its superior moderating properties and because the reactor temperatures which yield reasonable aircraft gross weights with the bismuth turbojet cycle are sufficiently low to permit consideration of the use of beryllia supported by structural metals. The favorable characteristics of the bismuth cycle at low temperatures and the consequent possibility of bringing beryllia into the picture as a moderating material give the bismuth cycle a considerable advantage.

Effect of Variation of Uranium Investment - Figure VII-22

The performance of the bismuth turbojet aircraft, like the open-cycle turbojet aircraft, is not very sensitive to a large variation in uranium investment. As is shown by the curves, a variation from an investment of 200 pounds to an investment of 20 pounds changes the gross weight of the manned aircraft only from about 500,000 pounds to 650,000 pounds. It should be noted that the effect of varying the uranium investment is even less in a graphite-moderated pile because graphite-moderated reactors change less in size than do beryllia reactors for a given change in uranium investment.

Effect of Variation of Shield Specific Gravity - Figure VII-23

The remarks made concerning the effect of shield specific gravity on the open-cycle air turbojet (Figure VII-8) apply equally well here. For this case of the bismuth turbojet, a manned aircraft using a shield of specific gravity 6 weighs 525,000 pounds; with specific gravity 10 an equivalent shield gives a plane which weighs 250,000 pounds; and with specific gravity 15 the plane weighs 175,000 pounds.

Effect of Variation of Reflector Thickness - Figure VII-24

The remarks made concerning the effect of variation of reflector thickness on the open-cycle air turbojet (Figure VII-9) apply equally well to the bismuth turbojet.

Effect of Variation of Compressor Pressure Ratio - Figure VII-25

It is not necessary to have high pressure ratios for the turbojet with the bismuth cycle because the air does not pass through the reactor. Hence the compressor pressure ratios are in the range of those used for chemically fueled turbojets. As is shown, the overall performance of the aircraft is not sensitive to variation in the compressor pressure ratio, with practically no change in performance for any value between 4 and 12. Since this is a design parameter which can be freely chosen by the designer, a value of perhaps 6, which could be easily reached using present day practice, might be selected. For this reason a compressor pressure ratio of 6 has been used in the standard case.

Effect of Variation of Bismuth Pumping Power - Figure VII-26

The curves show that the overall performance is not very sensitive

to a variation in bismuth pumping power of from 1 to 4% of the thrust power. A value of 3% is close to the optimum so this has been chosen for the standard case. This design characteristic can also be selected by the designer.

Effect of Variation of Bismuth Temperature - Figure VII-27

The curves are plotted for several different values of bismuth temperatures into and out of the reactor. A comparison of these curves shows that the overall performance of the aircraft is approximately the same for the combinations of temperatures shown. These temperatures are probably near the optimum.

Effect of Variation of Turbine Inlet Temperature at Low Maximum Reactor Wall Temperature - Figure VII-28

The curves plotted on the graph are not for the standard case inasmuch as the maximum reactor wall temperature has been lowered to 1370°F. A comparison of the curves show that there is very little effect in varying turbine inlet temperature from 1000 to 1200°F..

An important characteristic of the bismuth turbojet cycle is illustrated by comparison of the curves in this figure with the standard case in which the maximum wall temperature is 1840°F (Figure VII-27). Such a comparison indicates that the performance of the aircraft is not greatly affected by a change of maximum reactor wall temperature from 1840 to 1370°F since the gross weight of the manned aircraft with a four-foot shield increases only from 525,000 pounds to 650,000 pounds. Such a decrease in temperature of the reactor would greatly alleviate the reactor materials problems. However, reference to Chapter III will show that the frontal area of the machine may be excessive with such low temperatures. Furthermore, the performance would be very sensitive to even minor reductions in the values assumed for the component efficiencies. Hence, it may well prove that the minimum practicable value for the reactor wall temperature will be of the order of 1500°F.

Effect of Variation of Design Altitude - Figure VII-29

The bismuth turbojet is considerably less sensitive than the open-cycle turbojet to change in design altitudes as is shown by comparison of this figure with Figure VII-13. This is due to the fact that the air does not pass through the reactor of the bismuth turbojet. Hence, as the design altitude is increased it is not necessary to increase the reactor size, although it is still necessary to increase the size of the air machinery and heat exchanger. Even so there is still a marked effect of design altitude on the aircraft performance, since a change from 30,000 to 50,000 feet entails an increase in gross weight of the manned aircraft from 525,000 pounds to 950,000 pounds.

Comparison of the Manned Aircraft with the Tug-Tow at the Two Altitudes of 30,000 Ft. and 50,000 Ft. - Figure VII-30

As pointed out earlier, the tug-tow combination requires a shield

thickness of not more than one half that of the manned plane. At 30,000 feet a manned plane with a four-foot shield weighs 525,000 pounds, whereas the tug aircraft of the tug-tow combination with a 2-foot shield weighs only 125,000 pounds. At 50,000 feet the manned aircraft weighs 950,000 pounds, whereas the tug aircraft weighs 250,000 pounds.

Comparison of the Manned Aircraft with the Tug-Tow at Lower Maximum Reactor Wall Temperature - Figure VII-31

These curves are plotted for a maximum reactor wall temperature of 1370°F and a turbine inlet temperature of 1100°F. They show that at a design altitude of 30,000 feet a manned aircraft weighs 625,000 pounds, whereas the tug aircraft weighs only 150,000 pounds. At 50,000 feet the manned aircraft is entirely impracticable, whereas the tug-tow weighs 550,000 pounds. As pointed out previously, there may be reasons why temperatures this low are not satisfactory.

EFFECT OF VARIATIONS OF PARAMETERS FOR THE HELIUM GAS-TURBINE COMPRESSOR-JET

Effect of Change in Moderating Material - Figure VII-32

The remarks made concerning the effect of a change in moderating material for the bismuth turbojet (Figure VII-21) apply equally well here with the exception of those on the fast reactors. No fast reactors are included in the analysis of the helium gas-turbine compressor-jet because a gas-cooled fast reactor does not seem to be practicable from the standpoint of uranium inventory, except at extremely small thrusts.

A manned plane with beryllia would weigh 650,000 pounds, whereas with graphite it would weigh over 1,000,000 pounds.

Effect of Variation of Uranium Investment - Figure VII-33

The discussion on the effect on the bismuth turbojet performance of varying the uranium investment (Figure VII-22) applies equally well here. For the helium compressor-jet a change in uranium investment from 200 pounds to 20 pounds increases the manned aircraft weight from 625,000 pounds to 850,000 pounds.

Effect of Variation of Shield Specific Gravity - Figure VII-34

The remarks concerning the effect of shield specific gravity on the open-cycle air turbojet (Figure VII-8) apply equally well here. For the standard helium compressor-jet, a manned aircraft using a shield specific gravity of 6 weighs 650,000 pounds, an equivalent shield of 10 weighs 325,000 pounds, and one of 15 weighs 200,000 pounds.

Effect of Variation of Reflector Thickness - Figure VII-35

The comments on the effect of varying reflector thickness on the

performance of the open-cycle air turbojet (Figure VII-9) apply here. For the standard helium compressor-jet, an addition of a reflector of 8 inches thickness results in an increase in the gross weight of a manned plane from 650,000 pounds to 1,000,000 pounds.

Effect of Variation of Pressure of Helium Entering the Reactor - Figure VII-36

The overall performance of the aircraft is markedly improved by an increase in the pressure of helium in the cycle. For example, a manned plane weighs considerably in excess of 1,000,000 pounds if helium is used at 250 pounds per square inch, whereas at 2000 pounds per square inch the manned plane weighs only 575,000 pounds. This improvement is due to the fact that the increase in helium pressure permits a reduction in the reactor size with a consequent reduction in shield weight. 1000 pounds per square inch has been chosen for the standard aircraft.

Effect of Variation of Maximum Reactor Wall Temperature - Figure VII-37

The curves show that the overall performance of the aircraft is not greatly affected by a variation of maximum reactor wall temperature over a range from 1600°F to 2000°F for the fixed turbine inlet temperature of 1540°F.

Effect of Variation of Turbine Inlet Temperature - Figure VII-38

The curves show that the overall performance of the aircraft is not greatly affected by a variation of turbine inlet temperature from 1340°F to 1800°F when the reactor wall temperature is limited to 1830°F. There is evidently an optimum turbine inlet temperature which is not far from the 1540°F which has been chosen as the standard case. It is to be noted that this temperature is almost attainable in practice using present day techniques.

Effect of Variation of Altitude - Figure VII-39

The remarks concerning the effect of design altitude on the performance of the bismuth turbojet apply equally well here. For the helium gas-turbine compressor-jet a change from 30,000 feet to 50,000 feet means an increase in manned aircraft weight from 675,000 pounds to considerably more than 1,000,000 pounds.

Effect of Variation of Altitude at a Higher Maximum Reactor Wall Temperature - (2500°F) - Figure VII-40

These curves show that the effects of altitude variation on the overall performance are similar to those described in the previous case. A comparison of these curves with the curves of Figure VII-39 shows that an increase in the maximum wall temperature from the standard case of 1830°F to 2500°F results in a reduction of aircraft gross weight of only about 75,000 pounds.

Comparison of the Manned Aircraft with the Tug-Tow at Altitudes of 30,000 Feet and 50,000 Feet - Figure VII-41

At 30,000 feet a manned plane using the helium cycle with a standard four-foot shield weighs 650,000 pounds, whereas the tug aircraft of the tug-tow combination with a two-foot shield weighs only 150,000 pounds. At 50,000 feet the manned aircraft weighs in excess of 1,000,000 pounds, whereas the tug aircraft weighs only 325,000 pounds.

Comparison of Manned Aircraft with Tug-Tow at Higher Maximum Reactor Wall Temperature - Figure VII-42

These curves show the same general effect as in the previous case.

EXAMPLES OF DESIGN POSSIBILITIES

Twenty-five illustrative examples of integrated aircraft designs are listed in Table VII-2. The data for these examples are taken in part from the curves of the previous section. In addition there are more detailed specifications of some components. For convenience in reading this table an asterisk (*) has been placed against the factor which distinguishes each design from its neighbors.

TABLE VII-2

EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	1	2	3
Type of Airplane	Manned	Manned	Manned
Type of Power Plant	Air TJ	Air TJ	Air TJ
Airplane (Tug) Gross Weight, lb.	X	890,000	X
Altitude, ft.	30,000	30,000	30,000
Wing Area, sq. ft.	X	15,300	X
Shield Thickness, ft.	4	4	4
Shield Specific Gravity	6	6	6
Moderator	C	Be*	C
Uranium 235 Investment, lb.	100	100	50*
Reactor Core Length, ft.	X	3.28	X
Reactor Core Diameter, ft.	X	4.10	X
Free-Flow Ratio	X	.46	X
Reactor Coolant Tube Diameter, in.	X	0.316	X
Number of Tubes	X	11,200	X
Maximum Reactor Wall Temperature, °F	2500	2500	2500
Reactor Heat Output Rate, kw.	X	395,000	X
Reactor plus Shield Length, ft.	X	12.10	X
Reactor plus Shield Diameter, ft.	X	12.10	X
Total Thrust at Altitude, lb.	X	59,000	X
Total Air Flow at Altitude, lb./sec.	X	1,370	X
Air Compressor Frontal Area, sq. ft.	X	180	X
Air Compressor Pressure Ratio	40	40	40
Air Compressor Intercooler	yes	yes	yes
Air Compressor Efficiency, %	82	82	82
Air Temperature at Turbine Inlet, °F	1800	1800	1800
Air Turbine Efficiency, %	87	87	87
Jet Nozzle Efficiency, %	90	90	90
Air Radiator Frontal Area, sq. ft.	---	---	---
Air Radiator Tube Diameter, in.	---	---	---
Air Radiator Tube Length, ft.	---	---	---
Number of Radiator Tubes	---	---	---
Bismuth Flow Rate, lb./sec.	---	---	---
Helium Flow Rate, lb./sec.	---	---	---
Helium Pressure at Compressor Inlet, psi	---	---	---
Helium Pressure at Compressor Exit, psi	---	---	---
Helium Temperature at Turbine Inlet, °F	---	---	---
Helium Turbine Efficiency, %	---	---	---
Helium Compressor Efficiency, %	---	---	---
Weight of Payload including Crew, lb.	11,150	11,150	11,150
Weight of Airframe and Landing Gear, lb.	X	231,000	X
Weight of Power Plant Machinery, lb.	X	145,000	X
Weight of Reactor plus Shield, lb.	X	503,000	X
Mach Number	0.9 for all cases		
Towed Plane Gross Weight	40,000 lb. for all cases		
Drag of Towed Plane and Cable	5,000 lb. for all cases		

X -- Gross wt. >1,000,000 lb.
Be -- Beryllia
C -- Graphite
Air TJ -- Open-cycle turbojet
Bi TJ -- Bismuth turbojet
He CJ -- Helium compressor-jet

EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	4	5	6	7
Type of Plane	Manned	Pilotless	Tug-Tow	Tug-Tow
Type of Power Plant	Air TJ	Air TJ	Air TJ	Air TJ
Gross Wt., lb.	750,000	490,000	320,000	660,000
Altitude, ft.	30,000	30,000	30,000	30,000
Wing Area, sq. ft.	12,900	8,500	5,500	11,400
Shield Thickness, ft.	2.4	2.0	2.0	2.0
Shield Sp. Gravity	10*	6	6	6
Moderator	C	C	C	C
Uranium 235, lb.	100	100	100	100
Reactor Core L., ft.	4.24	4.48	3.78	4.96
Reactor Core Dia., ft.	5.30	5.60	4.72	6.20
Free-Flow Ratio	0.233	0.270	0.153	0.33
Tube Dia., in.	0.378	0.315	0.343	0.343
Number of Tubes	6630	12,300	4,200	15,500
Max. R. Wall Temp., °F	2500	1832*	2500	1832*
R. Heat Output, kw.	335,000	294,000	176,000	441,000
R. + Shield L., ft.	10.10	9.60	8.72	10.20
R. + Shield Dia., ft.	10.10	9.60	8.72	10.20
Total Thrust, lb.	50,000	32,600	26,300	49,000
Total Air Flow, lb./sec.	1160	1440	612	2170
Air Comp. F. Area, sq. ft.	150	185	80	280
Air Comp. Press.Ratio	40	40	40	40
Air Comp. Intercooler	yes	yes	yes	yes
Air Comp. Effic., %	82	82	82	82
Air Turbine Temp., °F	1800	1500	1800	1500
Air Turbine Effic., %	87	87	87	87
Jet Nozzle Effic., %	90	90	90	90
Air Rad. F. Area, sq. ft.	-------	-------	-------	-------
Air Rad. Tube Dia., in.	-------	-------	-------	-------
Air Rad. Tube L., ft.	-------	-------	-------	-------
Number Rad. Tubes	-------	-------	-------	-------
Bi Flow Rate, lb./sec.	-------	-------	-------	-------
He Flow Rate, lb./sec.	-------	-------	-------	-------
He Press. Comp. In., psi	-------	-------	-------	-------
He Press. Comp. Ex., psi	-------	-------	-------	-------
He Turbine Temp., °F	-------	-------	-------	-------
He Turbine Effic., %	-------	-------	-------	-------
He Comp. Effic., %	-------	-------	-------	-------
Wt. Payload, lb.	11,150	11,150	0	0
Wt. Airframe, lb.	193,000	125,000	96,000	201,000
Wt. Power Plant, lb.	123,000	151,000	65,000	227,000
Wt. Reactor - Sh., lb.	423,000	203,000	159,000	232,000
Mach Number	0.9 for all cases			
Towed Plane Gross Weight	40,000 lb. for all cases			
Drag of Towed Plane and Cable	5,000 lb. for all cases			

X -- Gross wt. > 1,000,000 lb. Air TJ -- Open-cycle turbojet
Be -- Beryllia Bi TJ -- Bismuth turbojet
C -- Graphite He CJ -- Helium compressor-jet

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EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	8	9	10	11
Type of Plane	Tug-Tow	Manned	Manned	Manned
Type of Power Plant	Air TJ	Bi TJ	Bi TJ	Bi TJ
Gross Wt., lb.	X	520,000	910,000	940,000
Altitude, ft.	50,000*	30,000	30,000	50,000*
Wing Area, sq. ft.	X	9,000	15,700	26,000
Shield Thickness, ft.	2.0	4	4	4
Shield Sp. Gravity	6	6	6	6
Moderator	C	Be	C*	Be
Uranium 235, lb.	100	100	100	100
Reactor Core L., ft.	X	2.03	4.09	2.20
Reactor Core Dia., ft.	X	2.03	4.09	2.20
Free-Flow Ratio	X	0.170	0.075	0.237
Tube Dia., in.	X	0.28	0.45	0.29
Number of Tubes	X	1,290	890	1,970
Max. R. Wall Temp., °F	2500	1840	1840	1840
R. Heat Output, kw.	X	242,000	429,000	400,000
R. + Shield L., ft.	X	10.44	12.91	10.64
R. + Shield Dia., ft.	X	10.03	12.09	10.20
Total Thrust, lb.	X	34,600	61,000	63,000
Total Air Flow, lb./sec.	X	839	1480	1390
Air Comp. F. Area, sq. ft.	X	81.0	143	330
Air Comp. Press. Ratio	40	6	6	6
Air Comp. Intercooler	yes	No	No	No
Air Comp. Effic., %	82	83.5	83.5	83.5
Air Turbine Temp., °F	1800	1500	1500	1500
Air Turbine Effic., %	87	87	87	87
Jet Nozzle Effic., %	90	90	90	90
Air Rad. F. Area, sq. ft.	-------	61.5	108	251
Air Rad. Tube Dia., in.	-------	0.125	0.125	0.125
Air Rad. Tube L., ft.	-------	1.62	1.62	1.62
Number Rad. Tubes	-------	362,000	635,000	1,475,000
Bi Flow Rate, lb./sec.	-------	12,700	22,400	21,000
He Flow Rate, lb./sec.	-------	-------	-------	-------
He Press. Comp. In., psi	-------	-------	-------	-------
He Press. Comp. Ex., psi	-------	-------	-------	-------
He Turbine Temp., °F	-------	-------	-------	-------
He Turbine Effic., %	-------	-------	-------	-------
He Comp. Effic., %	-------	-------	-------	-------
Wt. Payload, lb.	0	11,150	11,150	11,150
Wt. Airframe, lb.	X	133,000	237,000	312,000
Wt. Power Plant, lb.	X	72,200	127,000	294,000
Wt. Reactor Sh., lb.	X	303,700	535,000	323,000
Mach Number	0.9 for all cases			
Towed Plane Gross Weight	40,000 lb. for all cases			
Drag of Towed Plane and Cable	5,000 lb. for all cases			

X -- Gross wt. > 1,000,000 lb.
Be -- Beryllia
C -- Graphite

Air TJ -- Open-cycle turbojet
Bi TJ -- Bismuth turbojet
He CJ -- Helium compressor-jet

[redacted]

EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	12	13	14	15
Type of Plane	Manned	Manned	Manned	Pilotless
Type of Power Plant	Bi TJ	Bi TJ	Bi TJ	Bi TJ
Gross Wt., lb.	650,000	300,000	260,000	110,000
Altitude, ft.	30,000	30,000	30,000	30,000
Wing Area, sq. ft.	11,200	5,200	4,500	1,900
Shield Thickness, ft.	4	3.75*	2.4	2
Shield Sp. Gravity	6	6	10*	6
Moderator	Be	Fast (Pu)*	Be	Be
Uranium 235, lb.	20*	100	100	100
Reactor Core L., ft.	2.85	0.84	1.87	1.80
Reactor Core Dia., ft.	2.85	0.84	1.87	1.80
Free-Flow Ratio	0.105	0.59	0.102	0.077
Tube Dia., in.	0.36	0.19	0.26	0.26
Number of Tubes	950	1660	760	530
Max. R. Wall Temp., °F	1840	1840	1840	1840
R. Heat Output, kw.	302,000	140,000	121,000	86,300
R. + Shield L., ft.	11.42	8.51	7.00	6.16
R. + Shield Dia., ft.	10.85	8.34	6.63	5.80
Total Thrust, lb.	43,000	20,000	17,300	12,300
Total Air Flow, lb./sec.	1.040	485	420	298
Air Comp. F. Area, sq. ft.	100	47	40	29
Air Comp. Press. Ratio	6	6	6	6
Air Comp. Intercooler	No	No	No	No
Air Comp. Effic., %	83.5	83.5	83.5	83.5
Air Turbine Temp., °F	1500	1500	1500	1500
Air Turbine Effic., %	87	87	87	87
Jet Nozzle Effic., %	90	90	90	90
Air Rad. F. Area, sq. ft.	76	36	31	22
Air Rad. Tube Dia., in.	0.125	0.125	0.125	0.125
Air Rad. Tube L., ft.	1.62	1.62	1.62	1.62
Number Rad. Tubes	440,000	210,000	180,000	130,000
Bi Flow Rate, lb./sec.	15,800	7,360	6,360	4,530
He Flow Rate, lb./sec.	-------	-------	-------	-------
He Press. Comp. In., psi	-------	-------	-------	-------
He Press. Comp. Ex., psi	-------	-------	-------	-------
He Turbine Temp., °F	-------	-------	-------	-------
He Turbine Effic., %	-------	-------	-------	-------
He Comp. Effic., %	-------	-------	-------	-------
Wt. Payload, lb.	11,150	11,150	11,150	11,150
Wt. Airframe, lb.	166,000	76,000	66,000	30,000
Wt. Power Plant, lb.	89,800	41,800	36,200	25,700
Wt. Reactor Sh., lb.	383,000	171,000	146,700	44,000
Mach Number	0.9 for all cases			
Towed Plane Gross Weight	40,000 lb. for all cases			
Drag of Towed Plane and Cable	5,000 lb. for all cases			

X -- Gross wt. > 1,000,000 lb.
Be -- Beryllia
C -- Graphite

Air TJ -- Open-cycle turbojet
Bi TJ -- Bismuth turbojet
He CJ -- Helium compressor-jet

[redacted]

EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	16	17	18	19
Type of Plane	Tug-Tow	Tug-Tow	Tug-Tow	Manned
Type of Power Plant	Bi TJ	Bi TJ	Bi TJ	He CJ
Gross Wt., lb.	120,000	260,000	150,000	660,000
Altitude, ft.	30,000	50,000*	30,000	30,000
Wing Area, sq. ft.	2,100	7,200	2,600	11,400
Shield Thickness, ft.	2	2	2	4
Shield Sp. Gravity	6	6	6	6
Moderator	Be	Be	Be	Be
Uranium 235, lb.	100	100	100	100
Reactor Core L., ft.	1.81	1.90	1.97	2.74
Reactor Core Dia., ft.	1.81	1.90	1.97	2.74
Free-Flow Ratio	0.081	0.118	0.148	0.29
Tube Dia., in.	0.26	0.26	0.22	0.177
Number of Tubes	560	910	1,710	9,650
Max. R. Wall Temp., °F	1840	1840	1370*	1832
R. Heat Output, kw.	91,100	143,000	96,600	302,000
R. + Shield L., ft.	6.17	6.28	6.37	11.29
R. + Shield Dia., ft.	5.81	5.90	5.97	10.74
Total Thrust, lb.	13,000	22,300	15,000	44,000
Total Air Flow, lb./sec.	315	540	611	1,670
Air Comp. F. Area, sq. ft.	30	117	54	161
Air Comp. Press. Ratio	6	6	6	1.99
Air Comp. Intercooler	No	No	No	No
Air Comp. Effic., %	83.5	83.5	83.5	85
Air Turbine Temp., °F	1500	1500	1100	-------
Air Turbine Effic., %	87	87	87	-------
Jet Nozzle Effic., %	90	90	90	92
Air Rad. F. Area, sq. ft.	23	89	37	310
Air Rad. Tube Dia., in.	0.125	0.125	0.125	0.125
Air Rad. Tube L., ft.	1.62	1.62	1.22	2.47
Number Rad. Tubes	135,000	520,000	220,000	1,820,000
Bi Flow Rate, lb./sec.	4,780	7,460	10,000	-------
He Flow Rate, lb./sec.	-------	-------	-------	250
He Press. Comp. In., psi	-------	-------	-------	333
He Press. Comp. Ex., psi	-------	-------	-------	1,000
He Turbine Temp., °F	-------	-------	-------	1540
He Turbine Effic., %	-------	-------	-------	88
He Comp. Effic., %	-------	-------	-------	85
Wt. Payload, lb.	0	0	0	11,150
Wt. Airframe, lb.	42,000	94,000	49,000	169,000
Wt. Power Plant, lb.	27,200	104,000	44,600	100,000
Wt. Reactor - Sh., lb.	51,000	62,000	56,400	380,000
Mach Number	0.9 for all cases			
Towed Plane Gross Weight	40,000 lb. for all cases			
Drag of Towed Plane and Cable	5,000 lb. for all cases			

X -- Gross wt. > 1,000,000 lb.
Be -- Beryllia
C -- Graphite

Air TJ -- Open-cycle turbojet
Bi TJ -- Bismuth turbojet
He CJ -- Helium compressor-jet

[redacted]

EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	20	21	22	23
Type of Plane	Manned	Manned	Manned	Tug-Tow
Type of Power Plant	He CJ	He CJ	He CJ	He CJ
Gross Wt., lb.	X	320,000	850,000	130,000
Altitude, ft.	30,000	30,000	30,000	30,000
Wing Area, sq. ft.	X	5,500	14,700	2,200
Shield Thickness, ft.	4	2.4	4	2
Shield Sp. Gravity	6	10*	6	6
Moderator	C*	Be	Be	Be
Uranium 235, lb.	100	100	20*	100
Reactor Core L., ft.	X	2.30	3.70	1.90
Reactor Core Dia., ft.	X	2.30	3.70	1.90
Free-Flow Ratio	X	0.202	0.203	0.090
Tube Dia., in.	X	0.158	0.224	0.224
Number of Tubes	X	6,160	8,000	930
Max. R. Wall Temp., °F	1832	1832	1832	2500*
R. Heat Output, kw.	X	146,000	391,000	93,000
R. + Shield L., ft.	X	7.56	12.44	6.28
R. + Shield Dia., ft.	X	7.10	11.70	5.90
Total Thrust, lb.	X	21,300	57,000	13,700
Total Air Flow, lb./sec.	X	800	2,160	490
Air Comp. F. Area, sq. ft.	X	78	210	47
Air Comp. Press. Ratio	X	1.99	1.99	2.27
Air Comp. Intercooler	No	No	No	No
Air Comp. Effic., %	85	85	85	85
Air Turbine Temp., °F	-------	-------	-------	-------
Air Turbine Effic., %	-------	-------	-------	-------
Jet Nozzle Effic., %	92	92	92	92
Air Rad. F. Area, sq. ft.	X	150	400	87
Air Rad. Tube Dia., in.	0.125	0.125	0.125	0.125
Air Rad. Tube L., ft.	X	2.47	2.47	2.48
Number Rad. Tubes	X	880,000	2,450,000	510,000
Bi Flow Rate, lb./sec.	-------	-------	-------	-------
He Flow Rate, lb./sec.	X	120	324	63
He Press. Comp. In., psi	X	333	333	250
He Press. Comp. Ex., psi	X	1,000	1,000	1,000
He Turbine Temp., °F	1540	1540	1540	1800
He Turbine Effic., %	88	88	88	88
He Comp. Effic., %	85	85	85	85
Wt. Payload, lb.	11,150	11,150	11,150	0
Wt. Airframe, lb.	X	81,000	220,000	44,000
Wt. Power Plant, lb.	X	50,000	132,000	29,800
Wt. Reactor - Sh., lb.	X	178,000	487,000	56,000
Mach Number		0.9 for all cases		
Towed Plane Gross Weight		40,000 lb. for all cases		
Drag of Towed Plane and Cable		5,000 lb. for all cases		

X -- Gross wt. > 1,000,000 lb.
Be -- Beryllia
C -- Graphite

Air TJ -- Open-cycle turbojet
Bi TJ -- Bismuth turbojet
He CJ -- Helium compressor-jet

EXAMPLES OF NUCLEAR-POWERED AIRCRAFT DESIGN POSSIBILITIES

Design Number	24	25
Type of Plane	Tug-Tow	Tug-Tow
Type of Power Plant	He CJ	He CJ
Gross Wt., lb.	150,000	330,000
Altitude, ft.	30,000	50,000*
Wing Area, sq. ft.	2,600	9,100
Shield Thickness, ft.	2	2
Shield Sp. Gravity	6	6
Moderator	Be	Be
Uranium 235, lb.	100	100
Reactor Core L., ft.	2.15	2.37
Reactor Core Dia., ft.	2.15	2.37
Free-Flow Ratio	0.168	0.217
Tube Dia., in.	0.157	0.154
Number of Tubes	4,540	7,380
Max. R. Wall Temp., °F	1832	1832
R. Heat Output, kw.	103,000	168,000
R. + Shield L., ft.	6.58	6.84
R. + Shield Dia., ft.	6.15	6.37
Total Thrust, lb.	15,000	27,000
Total Air Flow, lb./sec.	570	930
Air Comp. F. Area, sq. ft.	55	220
Air Comp. Press. Ratio	1.99	1.99
Air Comp. Intercooler	No	No
Air Comp. Effic., %	85	85
Air Turbine Temp., °F	-------	-------
Air Turbine Effic., %	-------	-------
Jet Nozzle Effic., %	92	92
Air Rad. F. Area. sq. ft.	104	420
Air Rad. Tube Dia., in.	0.125	0.125
Air Rad. Tube L., ft.	2.47	2.47
Number Rad. Tubes	610,000	2,470,000
Bi Flow Rate, lb./sec.	-------	-------
He Flow Rate, lb./sec.	86	139
He Press. Comp. In., psi	333	333
He Press. Comp. Ex., psi	1,000	1,000
He Turbine Temp., °F	1540	1540
He Turbine Effic., %	88	88
He Comp. Effic., %	85	85
Wt. Payload, lb.	0	0
Wt. Airframe, lb.	49,000	120,000
Wt. Power Plant, lb.	36,000	135,000
Wt. Reactor - Sh., lb.	65,000	75,000
Mach Number	0.9 for all cases	
Towed Plane Gross Weight	40,000 lb. for all cases	
Drag of Towed Plane and Cable	5,000 lb. for all cases	

X -- Gross wt. 1,000,000 lb.
Be -- Beryllia
C -- Graphite

Air TJ -- Open-cycle turbojet
Bi TJ -- Bismuth turbojet
He CJ -- Helium compressor-jet

[redacted]

FIGURE VII-1, VII-1A, VII-2, VII-2A

COMPARISON OF STANDARD POWER PLANTS

IN MANNED AIRPLANES

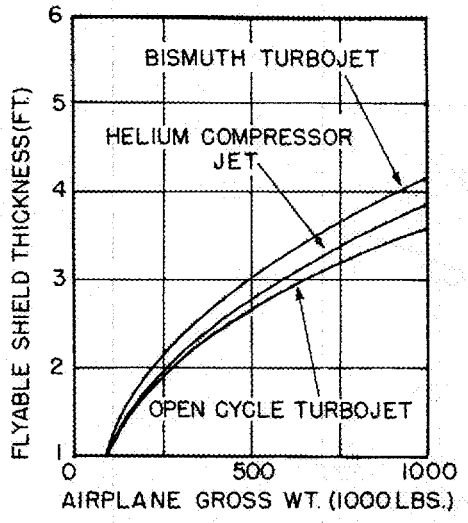


FIGURE VII-1 VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WT. USING GRAPHITE MODERATOR

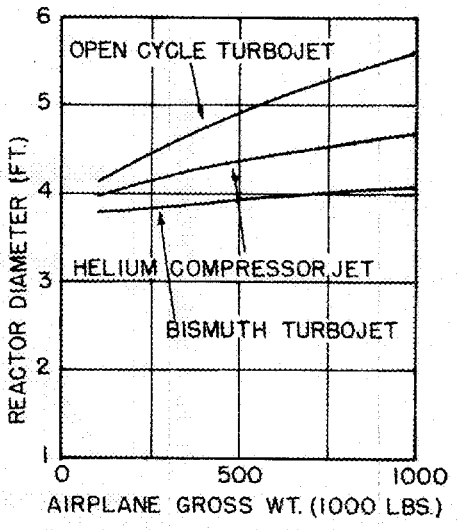


FIGURE VII-1A VARIATION OF REACTOR DIAMETER WITH AIRCRAFT GROSS WT. USING GRAPHITE MODERATOR

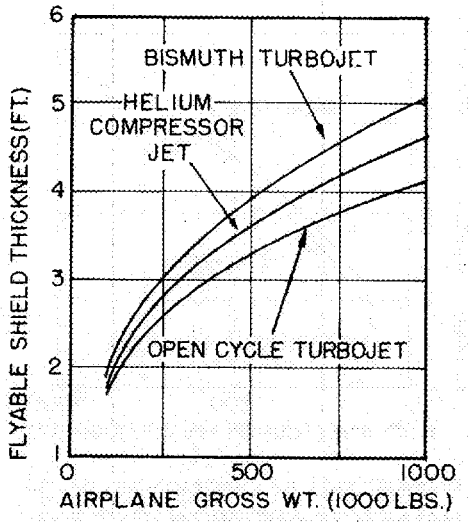


FIGURE VII-2 VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WT. USING BERYLLIA MODERATOR

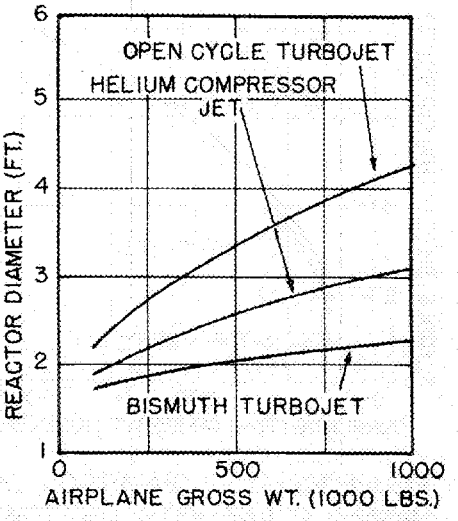


FIGURE VII-2A VARIATION OF REACTOR DIAMETER WITH AIRCRAFT GROSS WT. BERYLLIA MODERATOR

[redacted]

[redacted]

FIGURES VII-3, VII-4, VII-5
VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR MANNED AND TUG-TOW AIRPLANES WITH STANDARD POWER PLANTS

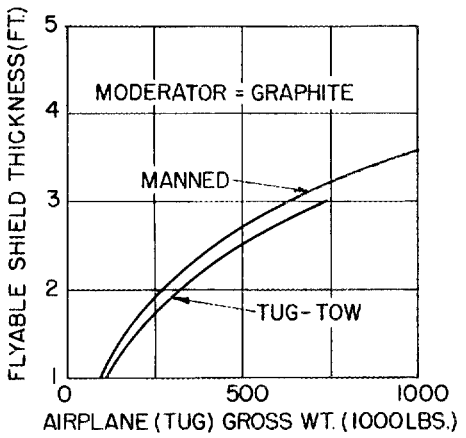


FIGURE VII-3 COMPARISON OF MANNED PLANE WITH TUG-TOW FOR OPEN CYCLE TURBOJET

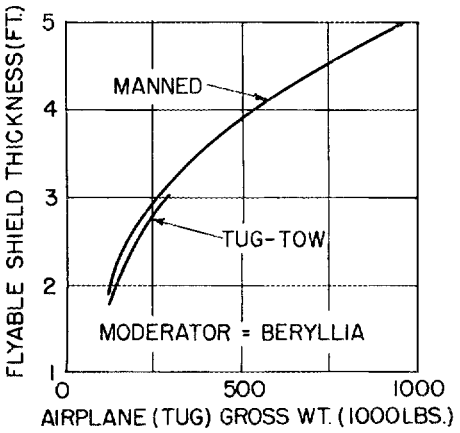


FIGURE VII-4 COMPARISON OF MANNED PLANE WITH TUG-TOW FOR BISMUTH TURBOJET

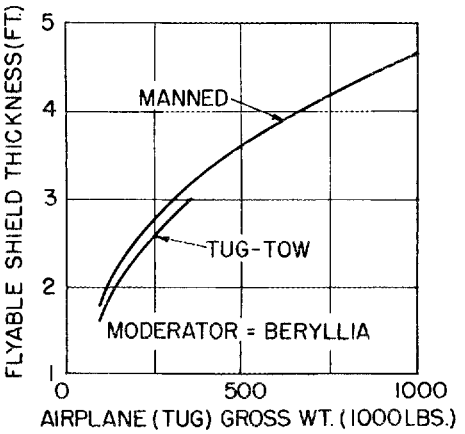


FIGURE VII-5 COMPARISON OF MANNED PLANE WITH TUG-TOW FOR HELIUM COMPRESSOR JET

[redacted]

[redacted]

FIGURES VII-6, VII-7, VII-8, VII-9
OPEN CYCLE AIR TURBOJET
VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

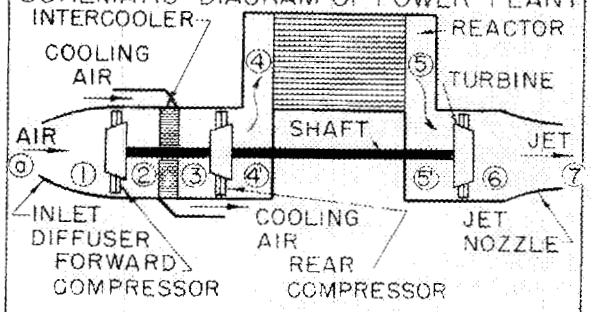
FLIGHT MACH NUMBER:	0.9
LIFT-DRAG RATIO:	15
ALTITUDE:	30,000 FT.
MAX. REACTOR WALL TEMP.:	2500 °F
TURBINE INLET TEMP.:	1800 °F
REACTOR PRESSURE DROP:	30 %
COMPRESSOR PRESSURE RATIO:	40

EFFICIENCIES: $\eta_D = 90$; $\eta_C = 82$; $\eta_T = 87$; $\eta_N = 90$

SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS

MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



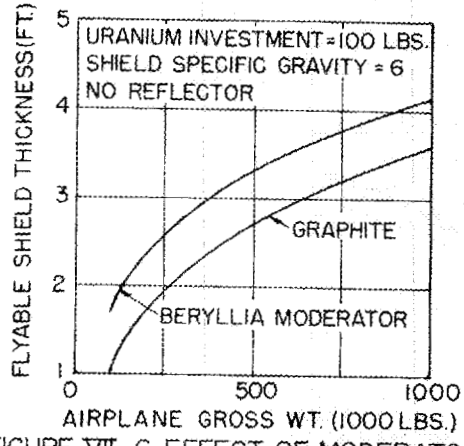


FIGURE VII-6 EFFECT OF MODERATOR

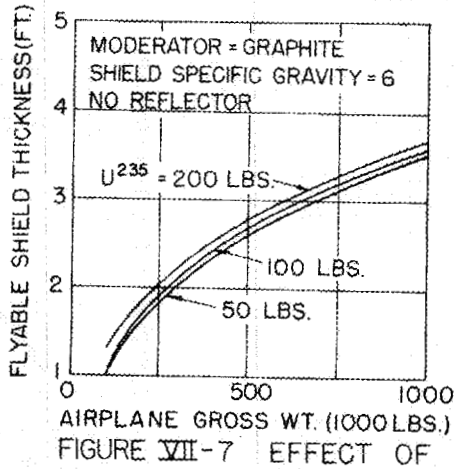


FIGURE VII-7 EFFECT OF URANIUM INVESTMENT

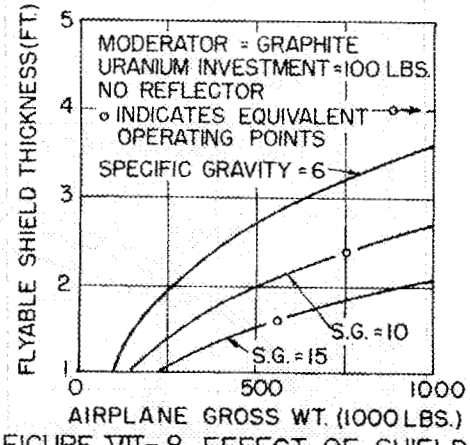


FIGURE VII-8 EFFECT OF SHIELD SPECIFIC GRAVITY

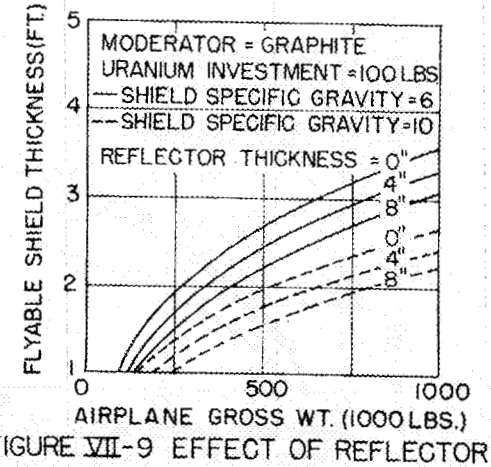


FIGURE VII-9 EFFECT OF REFLECTOR

[redacted]

[redacted]

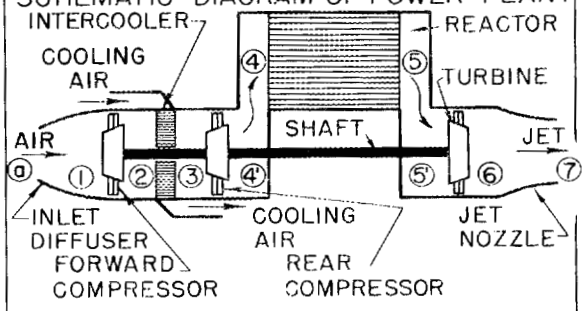
FIGURES VII-10, VII-11, VII-12
OPEN CYCLE AIR TURBOJET
VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
SHIELD SPECIFIC GRAVITY:	6
LIFT-DRAG RATIO:	15
ALTITUDE:	30,000 FT.
REACTOR PRESSURE DROP:	30 %
COMPRESSOR PRESSURE RATIO:	40
EFFICIENCIES:	$\eta_D = 90$; $\eta_C = 82$; $\eta_T = 87$; $\eta_N = 90$
URANIUM INVESTMENT:	100 LBS.
MODERATOR	GRAPHITE

SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS
MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



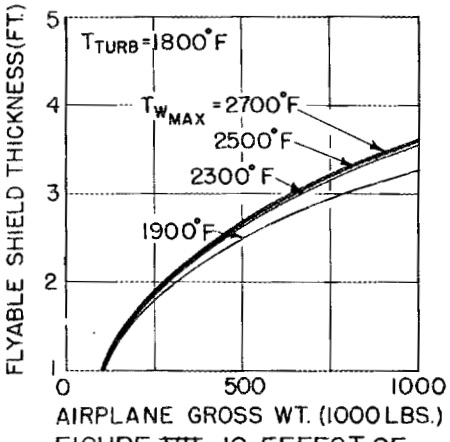


FIGURE VII-10 EFFECT OF MAX. REACTOR WALL TEMP.

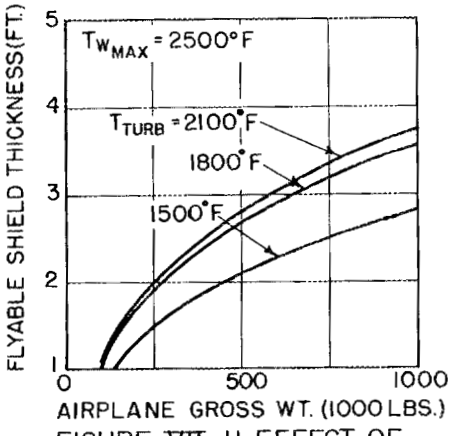


FIGURE VII-11 EFFECT OF TURBINE INLET TEMP.

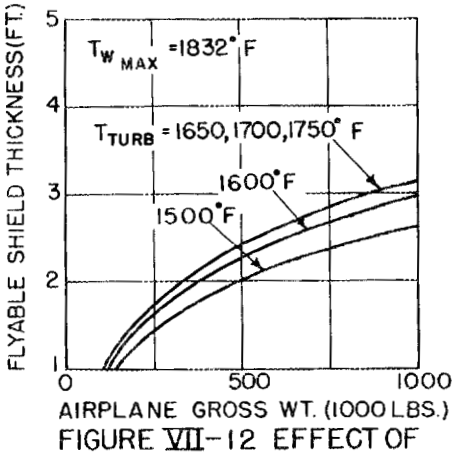


FIGURE VII-12 EFFECT OF TURBINE INLET TEMP.

[redacted]

FIGURES VII-13, VII-14, VII-15, VII-16

OPEN CYCLE AIR TURBOJET

VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

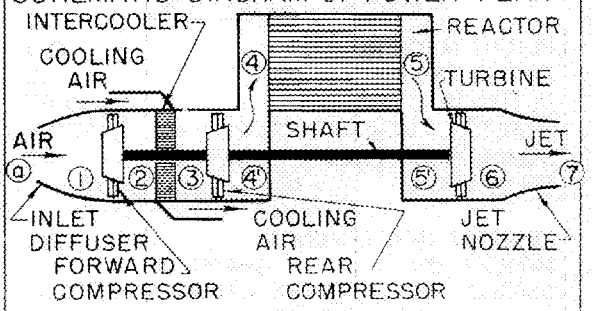
DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
SHIELD SPECIFIC GRAVITY:	6
LIFT-DRAG RATIO:	15
MAX. REACTOR WALL TEMP.:	2500 °F
TURBINE INLET TEMP.:	1800 °F
URANIUM INVESTMENT:	100 LBS.
MODERATOR:	GRAPHITE

SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS

MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



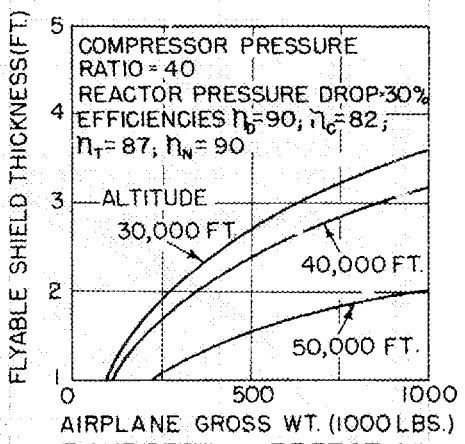


FIGURE VII-13 EFFECT OF ALTITUDE

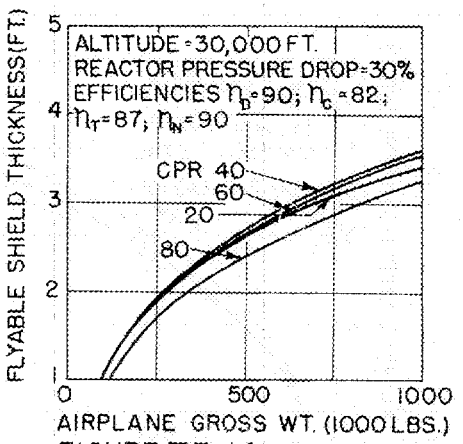


FIGURE VII-14 EFFECT OF COMPRESSOR PRESSURE RATIO

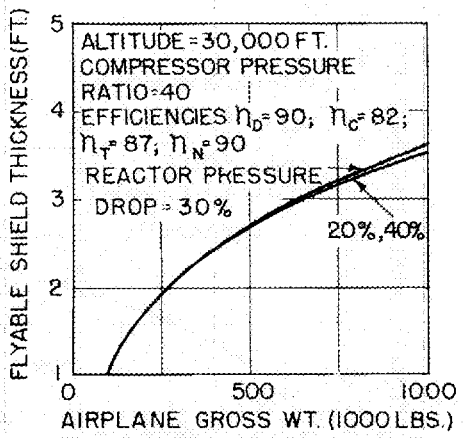


FIGURE VII-15 EFFECT OF REACTOR PRESSURE DROP

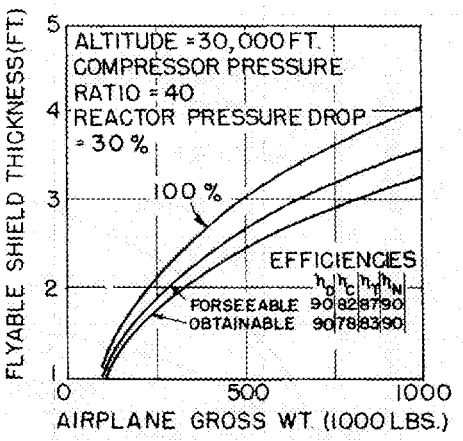


FIGURE VII-16 EFFECT OF COMPONENT EFFICIENCIES

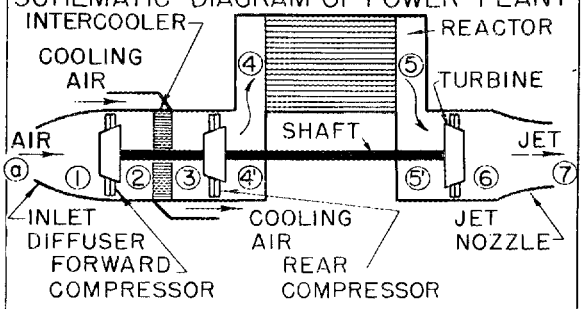
[redacted]

FIGURES VII-17, VII-18, VII-19, VII-20
OPEN CYCLE AIR TURBOJET
VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
SHIELD SPECIFIC GRAVITY:	6
ALTITUDE:	30,000 FT.
REACTOR PRESSURE DROP:	30%
COMPRESSOR PRESSURE RATIO:	40
EFFICIENCIES:	$\eta_D = 90$; $\eta_C = 82$; $\eta_T = 87$; $\eta_N = 90$
URANIUM INVESTMENT:	100 LBS.
MODERATOR	GRAPHITE
SINGLE PASS, SQUARE CYLINDER REACTOR WITH HEADERS	

SCHEMATIC DIAGRAM OF POWER PLANT



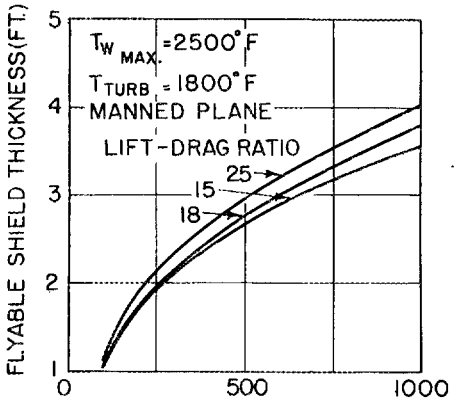


FIGURE VII-17 EFFECT OF LIFT-DRAG RATIO

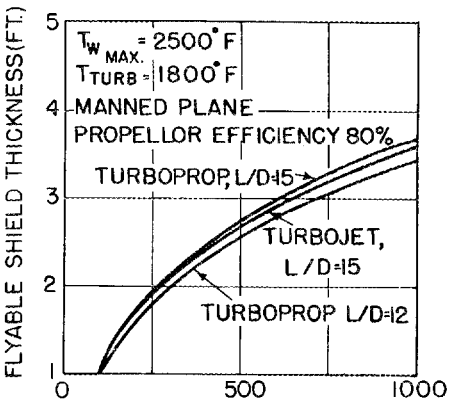


FIGURE III-18 COMPARISON OF TURBOJET WITH TURBOPROP

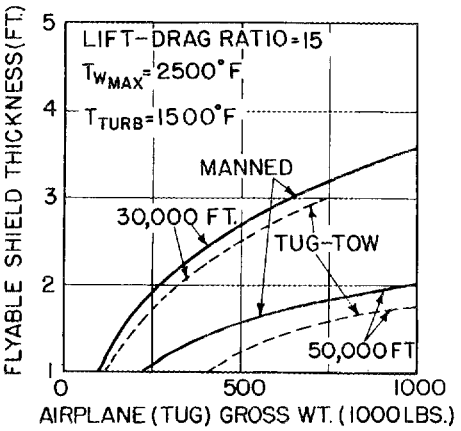


FIGURE VII-19 COMPARISON OF MANNED PLANE WITH TUG-TOW

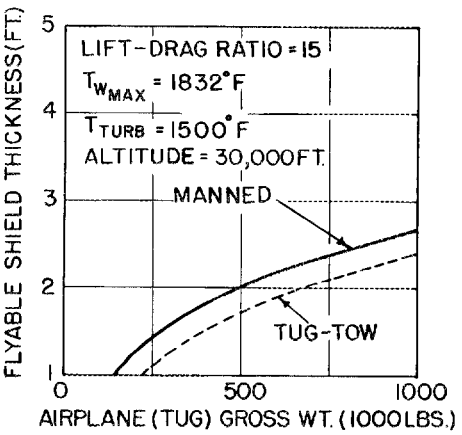


FIGURE VII-20 COMPARISON OF MANNED PLANE WITH TUG-TOW

[redacted]

[redacted]

FIGURES VII-21, VII-22, VII-23, VII-24

BISMUTH TURBOJET

VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

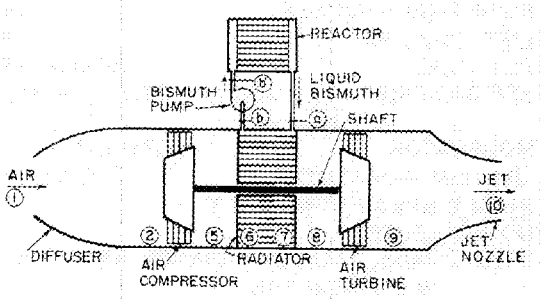
FLIGHT MACH NUMBER:	0.9
LIFT-DRAG RATIO:	15
ALTITUDE:	30,000 FT.
REACTOR WALL TEMP.:	1840 °F
TURBINE INLET TEMP.:	1500 °F
BISMUTH TEMPS.:	1650°-1180 °F
COMPRESSOR PRESSURE RATIO:	6

EFFICIENCIES: $\eta_D = 90; \eta_C = 83.5; \eta_T = 87; \eta_N = 90$

SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS

MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



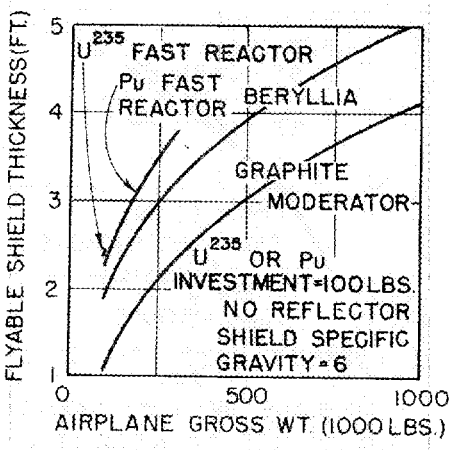


FIGURE VII-21 EFFECT OF MODERATOR

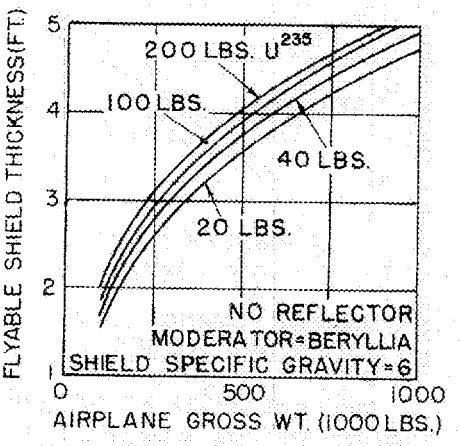


FIGURE VII-22 EFFECT OF URANIUM INVESTMENT

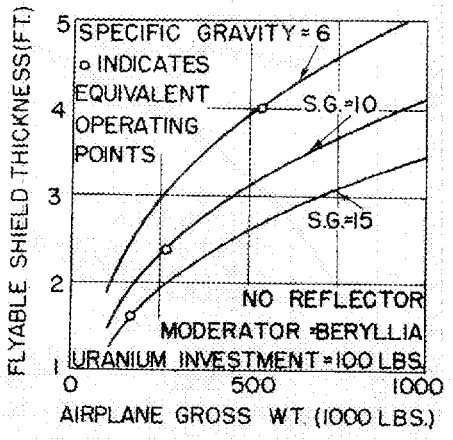


FIGURE VII-23 EFFECT OF SHIELD SPECIFIC GRAVITY

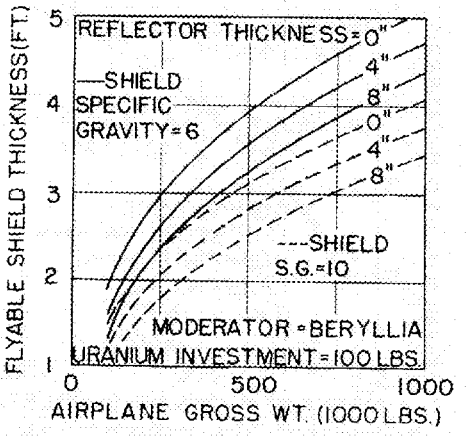


FIGURE VII-24 EFFECT OF REFLECTOR

[redacted]

[redacted]

FIGURES VII-25, VII-26, VII-27, VII-28

BISMUTH TURBOJET

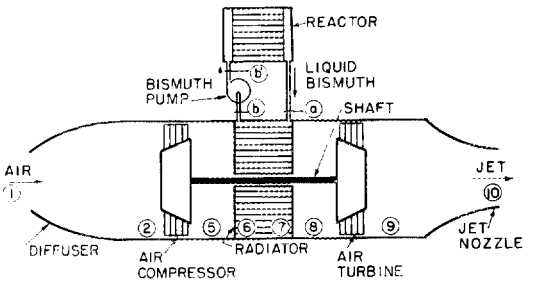
VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
LIFT-DRAG RATIO:	15
ALTITUDE:	30,000 FT.
EFFICIENCIES:	$\eta_D = 90; \eta_C = 83.5; \eta_T = 37; \eta_N = 90;$
MODERATOR	BERYLLIA
URANIUM INVESTMENT	100 LBS.
SHIELD SPECIFIC GRAVITY	6

NO REFLECTOR
SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS
MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



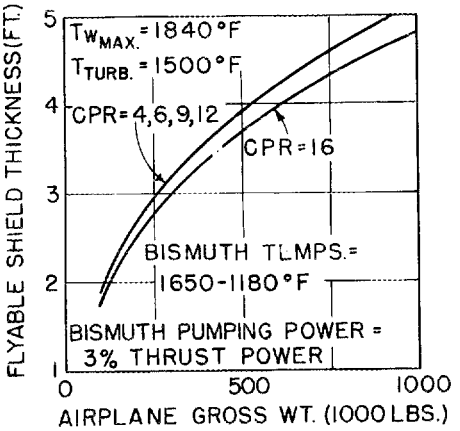


FIGURE VII-25 EFFECT OF COMPRESSOR PRESSURE RATIO

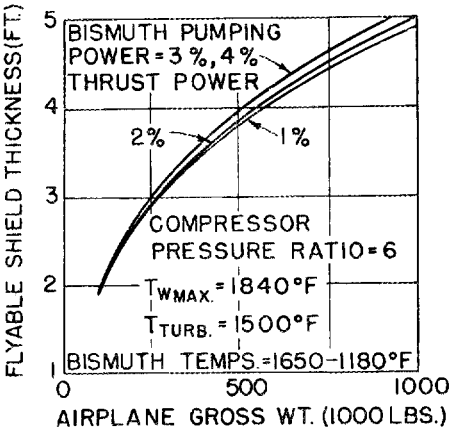


FIGURE VII-26 EFFECT OF BISMUTH PUMPING POWER

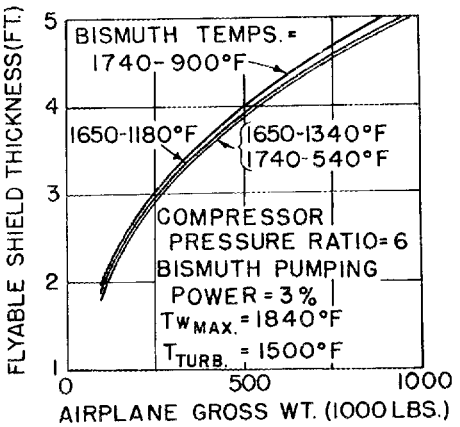


FIGURE VII-27 EFFECT OF BISMUTH TEMPERATURE

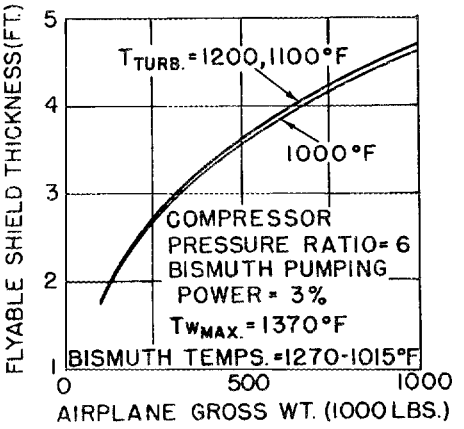


FIGURE VII-28 EFFECT OF TURBINE INLET TEMPERATURE

[redacted]

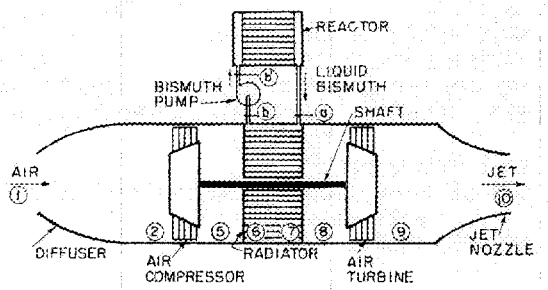
FIGURES VII-29, VII-30, VII-31
VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NUMBER:	0.9
LIFT-DRAG RATIO:	15
COMPRESSOR PRESSURE RATIO:	6
EFFICIENCIES:	$\eta_D = 90, \eta_C = 83.5, \eta_T = 87, \eta_N = 90$
MODERATOR:	BERYLLIA
URANIUM INVESTMENT:	100 LBS
SHIELD SPECIFIC GRAVITY:	6

NO REFLECTOR
SINGLE PASS, SQUARE CYLINDER REACTOR WITH HEADERS

SCHEMATIC DIAGRAM OF POWER PLANT



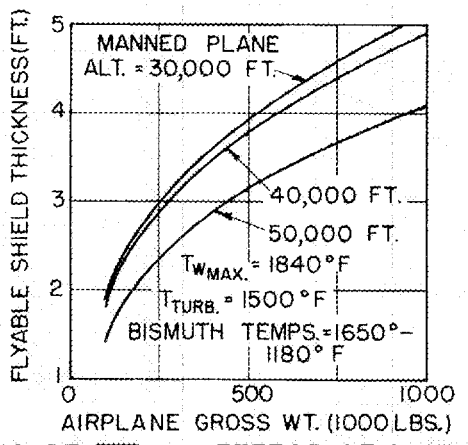


FIGURE VII-29 EFFECT OF ALTITUDE

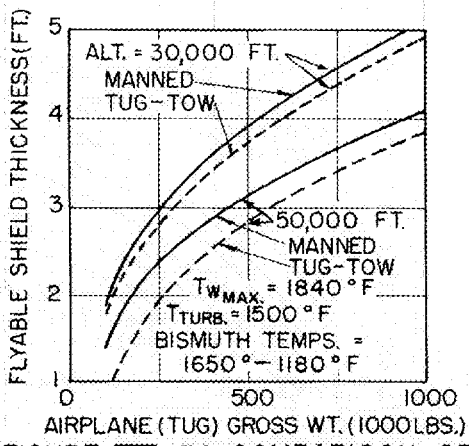


FIGURE VII-30 COMPARISON OF MANNED PLANE WITH TUG-TOW

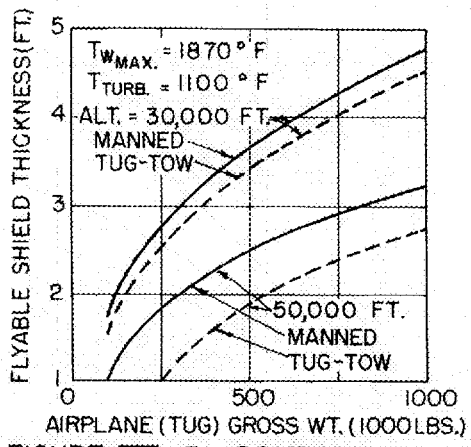


FIGURE VII-31 COMPARISON OF MANNED PLANE WITH TUG-TOW

[redacted]

FIGURES VII-32 VII-33 VII-34 VII-35

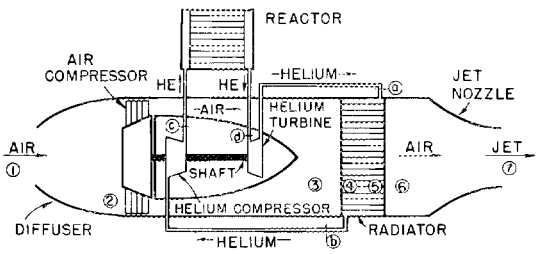
HELIUM GAS-TURBINE COMPRESSOR-JET

VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NUMBER: 0.9
ALTITUDE: 30,000 FT.
MAX. REACTOR WALL TEMP.: 1832 °F
TURBINE INLET TEMP.: 1540 °F
HELIUM PRESSURE IN REACTOR: 1000 PSIA
EFFICIENCIES: $\eta_D = 90$; $\eta_{C(AIR)} = 85$; $\eta_{C(HE)} = 85$
$\eta_N = 92$
LIFT-DRAG RATIO: 15
SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS
MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



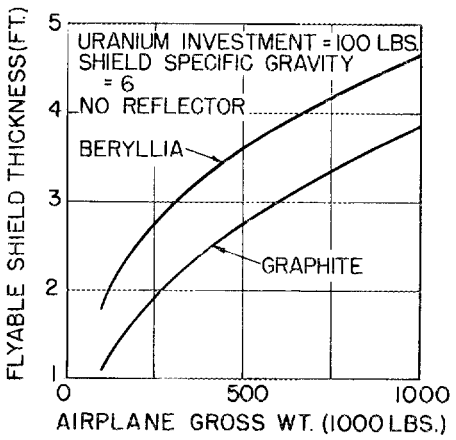


FIGURE VII-32 EFFECT OF MODERATOR

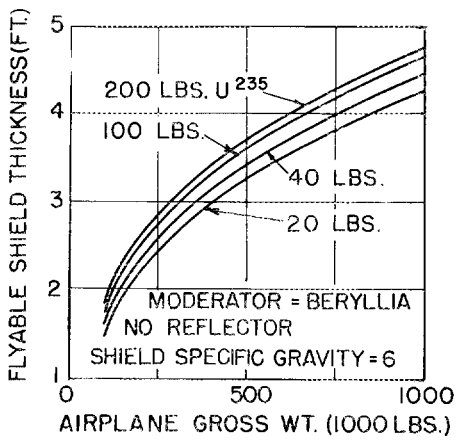


FIGURE VII-33 EFFECT OF URANIUM INVESTMENT

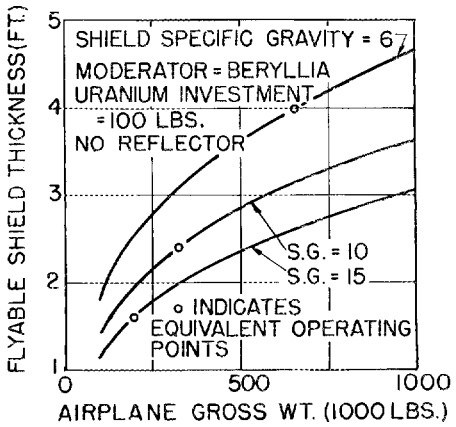


FIGURE VII-34 EFFECT OF SHIELD SPECIFIC GRAVITY

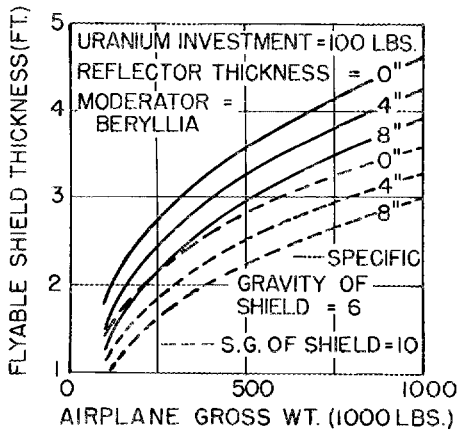


FIGURE VII-35 EFFECT OF REFLECTOR

[redacted]

[redacted]

FIGURES VII-36, VII-37, VII-38

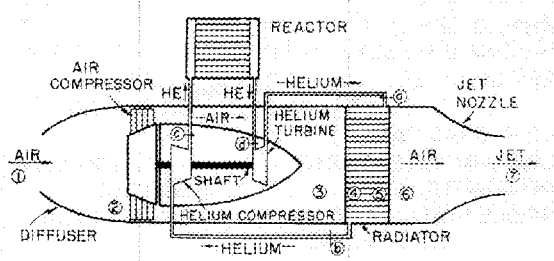
HELIUM GAS-TURBINE COMPRESSOR-JET

VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NUMBER: 0.9
SHIELD SPECIFIC GRAVITY: 6
ALTITUDE: 30,000 FT.
EFFICIENCIES: $\eta_D = 90$, $\eta_{C(AIR)} = 85$, $\eta_{C(HE)} = 85$, $\eta_T = 88$, $\eta_N = 92$
LIFT-DRAG RATIO: 15
URANIUM INVESTMENT: 100 LBS.
MODERATOR: BERYLLIA
SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS
MANNED PLANE

SCHEMATIC DIAGRAM OF POWER PLANT



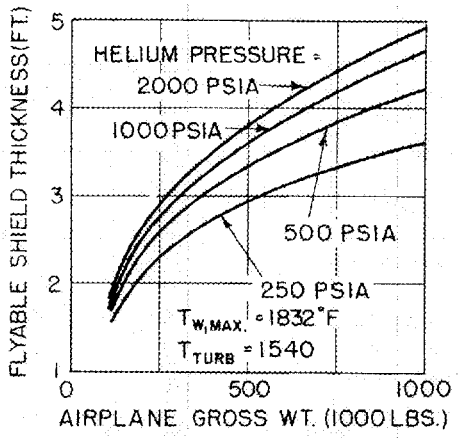


FIGURE VII-36 EFFECT OF HELIUM PRESSURE IN REACTOR

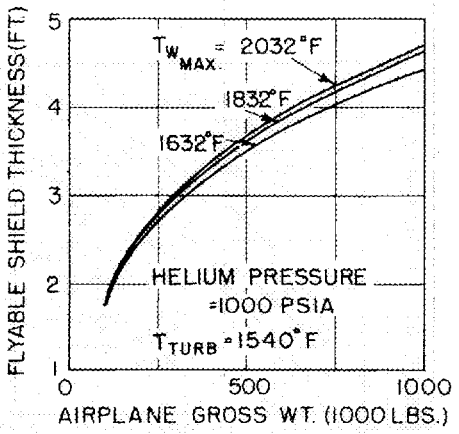


FIGURE VII-37 EFFECT OF MAXIMUM REACTOR WALL TEMP.

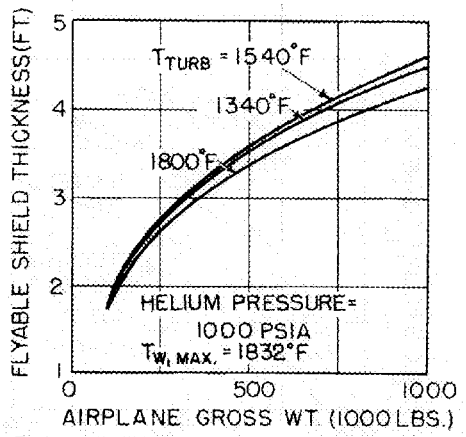


FIGURE VII-38 EFFECT OF TURBINE INLET HELIUM TEMP.

[redacted]

[redacted]

FIGURES VII-39, VII-40, VII-41, VII-42

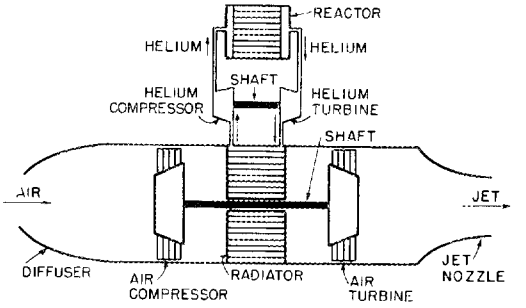
HELIUM GAS TURBINE COMPRESSOR JET

VARIATION OF FLYABLE SHIELD THICKNESS WITH AIRCRAFT GROSS WEIGHT FOR VARIOUS OPERATING CONDITIONS

DESIGN CONDITIONS

FLIGHT MACH NO.: 0.9
SHIELD SPECIFIC GRAVITY: 6
HELIUM PRESSURE IN REACTOR: 1000 PSIA
EFFICIENCIES: $\eta_D = 90$; $\eta_{C(AIR)} = 85$; $\eta_{C(HE)} = 85$; $\eta_T = 88$; $\eta_N = 92$
LIFT-DRAG RATIO: 15
URANIUM INVESTMENT: 100 LBS.
MODERATOR: BERYLLIA
SINGLE-PASS, SQUARE CYLINDER REACTOR WITH HEADERS

SCHEMATIC DIAGRAM OF POWER PLANT



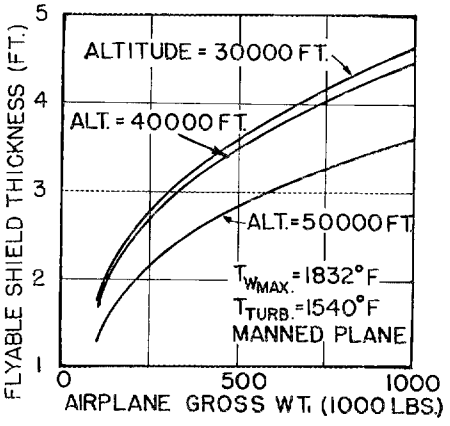


FIGURE-VII-39 EFFECT OF ALTITUDE

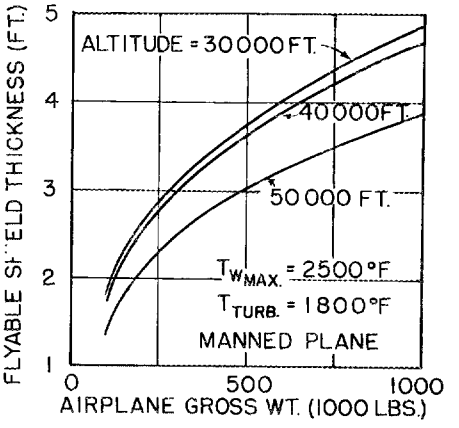


FIGURE-VII-40 EFFECT OF ALTITUDE

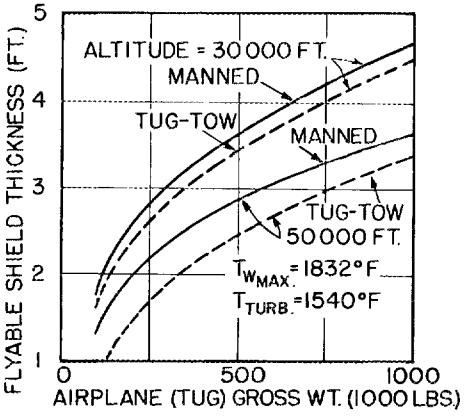


FIGURE VII-41 COMPARISON OF MANNED PLANE WITH TUG-TOW

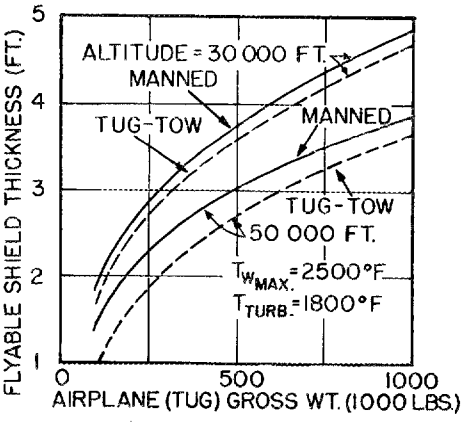


FIGURE VII-42 COMPARISON OF MANNED PLANE WITH TUG-TOW

[redacted]

APPENDICES

Appendix A

TECHNICAL PERSONNEL OF LEXINGTON PROJECT

Bevans, Rowland S.	M.I.T., Guided Missiles Program
Bloom, David S.	M.I.T., Research Associate, Metallurgy
Brown, Gordon S. (Cons.)	M.I.T., Professor, Electrical Engineering
Cooley, William C.	M.I.T., Graduate Student, Mechanical Engineering
Deutsch, Martin (Cons.)	M.I.T., Assistant Professor, Physics
Drew, Thomas B.	Columbia University, Head of Department, Chemical Engineering
Feld, Bernard T.	M.I.T., Assistant Professor, Physics
Frisch, David H. (Cons.)	M.I.T., Research Associate, Physics
Friedman, Francis L.	M.I.T., Research Associate, Physics
(Geller, Roman)	National Bureau of Standards, Ceramics
Gilliland, Edwin R. (Cons.)	M.I.T., Professor, Chemical Engineering
Goertzel, Gerald	Nuclear Development Associates
Goodman, Clark	M.I.T., Associate Professor, Physics
Grant, Nicholas J.	M.I.T., Associate Professor, Metallurgy
Gruber, Alan R.	Harvard, Fellow, Mechanical Engineering
Gurinsky, David H.	Brookhaven, Metallurgy
Hunter, Lloyd P.	Westinghouse Electric, (Oak Ridge National Laboratory, Materials)
Johnston, S. Paul (Cons.)	Institute of Aeronautical Sciences, Director
Kasschau, Kenneth	Wright Aeronautical (Oak Ridge National Laboratory, Power Pile Division)
Kaufmann, Albert R.	M.I.T., Associate Professor, Metallurgy
(Keirn, Col. Donald J.)	Atomic Energy Commission, Military Applications
Klein, James H.	M.I.T., Graduate Student, Chemical Engineering
Leverett, Miles (Cons.)	Humble Oil Company, (formerly Oak Ridge National Laboratory)
Loomis, F. Wheeler	University of Illinois, Head of Department, Physics
Marion, Charles P.	M.I.T., Graduate Student, Chemical Engineering
Mickley, Harold S.	M.I.T., Assistant Professor, Chemical Engineering

Millikan, Clark B. (Cons.)	California Institute of Technology, Professor, Aeronautical Engineering
Newell, Joseph H. (Cons.)	M.I.T., Professor, Aeronautical Engineering
Nordheim, Lothar	Duke, Professor, Physics
Norton, Frederick H.	M.I.T., Professor, Metallurgy
Ober, Shatswell (Cons.)	M.I.T., Professor, Aeronautical Engineering
(Old, Bruce)	Atomic Energy Commission, Chief Metallurgist
Redding, Edward M.	Kettering Foundation, Director (formerly North American Aviation, Aerophysics Lab.)
(Rothrock, Addison M.)	National Advisory Committee for Aeronautics, Assistant Director (on loan to AEC)
Shapiro, Ascher H.	M.I.T., Associate Professor, Mechanical Engineering
Shoults, D. Roy (Cons.)	Glenn L. Martin Company, Vice President, Engineering
Smith, Chester W.	General Electric, Turbine & Compressor Design
Soodak, Harry	M.I.T., Research Associate, Physics
Stever, H. Guyford	M.I.T., Assistant Professor, Aeronautical Engineering
Taylor, Philip B. (Cons.)	Sanderson & Porter, Vice President
Thiele, Ernest W.	Standard Oil Company (Indiana), Assistant Director of Research
van Zelm, Willem D.	Glenn L. Martin Company, Chief, Preliminary Design
Welton, Theodore A.	M.I.T., Research Associate, Physics
Whitman, Walter G.	M.I.T., Head of Department, Chemical Engineering
Wiesner, Jerome B. (Cons.)	M.I.T., Associate Professor, Electrical Engineering
Wigner, Eugene P. (Cons.)	Princeton, Professor, Physics
Young, Gale J.	Nuclear Development Associates
Zacharias, Jerrold R.	M.I.T., Professor, Director, Laboratory Nuclear Science and Engineering

(Cons.) following a name indicates the person was a Consultant.

Parentheses about names indicate Atomic Energy Commission personnel who gave a substantial amount of time to the Lexington Project.

Appendix B

CONTACTS MADE DURING LEXINGTON PROJECT

The following is a list of organizations and individuals contacted in the course of the Lexington Project to determine what work has been accomplished and is in progress in the field of nuclear-powered flight and related problems.

The listing is alphabetical by organization. The contacts are subdivided into two groups: (1) visits by Lexington Project personnel and (2) visits to the Lexington Project by others. In the case of visits elsewhere, the names of the Lexington Project personnel making the trip are listed together with the names of certain people contacted. The visitors to the project are listed. An abbreviated title is given for the main subject of the visit, and reference is made to the pertinent Lexington Project reports on these contacts.

A. C. SPARK PLUG CO.

Visitor:

July 6	"Ceramics" T. G. McDougal	LP-74

ATOMIC ENERGY COMMISSION

Visitors:

May 12 - 13	"Orientation Meeting" See NEPA	LP-73
May 19	"Air Force Interest in Nuclear-Powered Flight" See USAF	LP-0
July 1 - 2	J. B. Fisk, George L. Weil	
Aug. 12	"General Policy" I. I. Rabi	
Aug. 20	"Tentative Conclusions and Discussion" R. F. Bacher, J. M. Dunford, J. B. Fisk, W. E. Kelley, Brig. Gen. J. McCormack, J. K. Pickard, S. T. Pike, I. I. Rabi, Hartley Rowe, R. S. Warner, C. L. Wilson	LP-133 LP-135
Aug. 30	"Discussion" D. E. Lilienthal, C. L. Wilson	

ATOMIC ENERGY COMMISSION (Continued)

Visits:

June 1	"Orientation" Bevans, Bloom, Cooley, Feld, Friedman, Goodman, Grant, Kaufmann, Keirn, Klein, Marion, Mickley, Old, Redding, Rothrock, Shapiro, Smith, Soodak, Stever, Thiele, Welton, Whitman, Zacharias; with R. F. Bacher and J. R. Oppenheimer	LP-6
Aug. 27	"Discussion of Conclusions" Feld and Whitman; to see Enrico Fermi	

AEC REACTOR SAFEGUARD COMMITTEE

Visit:

July 13	"Reactor Safety Discussion" E. Teller; with Friedman, Nordheim, Whitman, Zacharias	

Visitor:

July 23	"Reactor Safety Discussion" Manson Benedict	LP-100

ALFRED UNIVERSITY, N. Y. COLLEGE OF CERAMICS

Visitor:

July 9	"Reactions in Solid State Metal Oxides" V. D. Frechette	LP-83

APPLIED PHYSICS LABORATORY, JOHNS HOPKINS UNIVERSITY

Visitor:

July 21	"Nuclear-Powered Flight" J. T. McClure	LP-104

ARGONNE NATIONAL LABORATORY

Visits:

July 21 - 22	"Argonne Policy and Reactor Program" "Properties of Liquid Na-K" Drew, Feld, Whitman, Young, Zacharias; to see Farrington Daniels H. M. Hilberry, O. C. Simpson and W. H. Zinn et al.	LP-33 LP-33b

BATTELLE MEMORIAL INSTITUTE

Visits:

June 23	"Materials" Bloom, Gurinsky, Kaufmann	LP-54

BELL TELEPHONE CO.

Visitor:

July 9	"Policy" W. A. McNair	LP-86

BOEING AIRCRAFT CO.

Visitor:

Aug. 16	"Airplanes" Holden Withington	LP-126

BROOKHAVEN NATIONAL LABORATORY

Visits:

July 21	"Brookhaven Materials-Testing Facilities" Hunter	LP-66
Aug. 4	"Materials" Bloom, Grant	

CALIFORNIA INSTITUTE OF TECHNOLOGY

Visit:

July 14	"Ceramics" Pol Duwez	LP-94

CARBON AND CARBIDE CHEMICALS CORP.

Visit:

July 30	"Reactors" Clifford Beck	

CHAMPION SPARK PLUG CO.

Visitor:

June 25	"Ceramics" H. F. Royal	LP-50

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DOW CHEMICAL CO.

Visitor:

July 15	"Nuclear-Powered Aircraft" J. J. Grebe	LP-113

FANSTEEL CORP., CHICAGO

Visits:

June 30	"Materials" Bloom, Grant	LP-56

GENERAL ELECTRIC CO., SCHENECTADY

Visits:

July 21	"G. E. Policy and Reactor Program" Lexington Personnel: Loomis, Rothrock, Stever, Thiele, Whitman, Young; to visit Harvey Brooks, Zay Jeffries, K. H. Kingdon, B. R. Prentice, C. G. Suits, and H. A. Winne	LP-82

G. E. KNOLLS LABORATORY

Visits:

July 12	"Liquid Metal Cooled Piles" "Materials" Bloom, Grant, Gurinsky, Hunter, Kaufmann, Mickley	LP-84 LP-93

HANFORD, WASH.

Visits:

June 15 - 16	"Hanford Facilities" Kaufmann	LP-58
Aug. 24 - 25	"Shielding" Feld	LP-145

LOS ALAMOS, N. M.

Visits:

July 11 - 14	"Questions and Answers on Trip to Los Alamos" Friedman, Keirn, Nordheim, Whitman, Zacharias	LP-88

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LOS ALAMOS, N. M. (Continued)

Visits:

Aug. 8 - 19	Nordheim	

MASSACHUSETTS INSTITUTE OF TECHNOLOGY

Visitors:

June 18	"Ducted Fans" W. R. Hawthorne	LP-30
July 23	"Materials" H. C. Hottel	
July 28	"Reactors and Shielding" Martin Deutsch, D. H. Frisch	
Aug. 2	"Aircraft Design" Shatswell Ober	LP-110
Aug. 10	"Aircraft Design" Shatswell Ober	

MIT - SUPERSONIC LABORATORY

Visitors:

Aug. 24	"Supersonic Flight and Cable Drag" Dalimil Kybal, F. E. Goddard, Jr.	

METAL HYDRIDES CORP.

Visit:

Sept. 16	Bloom; to see T. R. P. Gibb	

MILITARY LIAISON COMMITTEE

Visitor:

Aug. 20	"Discussion of Tentative Conclusions" Adm. W. S. Parsons	LP-133

NACA, LANGLEY AERONAUTICAL LABORATORY

Visitors:

June 15	"Propellers" E. C. Draley	LP-26
July 27	"Power Plants" E. J. Manganiello	

NACA, LANGLEY AERONAUTICAL LABORATORY

Visitors:

Aug. 10	"Power Plants" E. J. Manganiello, A. Silverstein	LP-119

Visits:

June 29	"Materials" Bloom, Gurinsky, Kaufmann	LP-55
June 30	"Power Plants" Bevans, Cooley, Goertzel, Redding, Shapiro, Smith	LP-60

NACA, LEWIS FLIGHT PROPULSION LABORATORY

Visitor:

June 10 - 11	"Aircraft Power Plants" Benjamin Pinkel	LP-20

NACA, MATERIALS CONFERENCE, ALLISON MOTORS, INDIANAPOLIS

Visit:

June 29	"Materials Conference" Grant	LP-53

NACA, WASHINGTON, D. C.

Visits:

July 6 - 7	"NACA and Nuclear-Powered Flight" Keirn, Loomis, Rothrock, Stever. Whitman; to see Hugh Drydon	LP-88b

NATIONAL CARBON CO.

Visitor:

July 6	"Graphite" H. G. MacPherson	LP-59

NEPA, OAK RIDGE

Visitors:

May 12 - 13	"Orientation Meeting" W. Burke, AEC; A. Kalitinsky, NEPA; Col. D. J. Keirn, AEC; B. S. Old, AEC; R. C. Palmer, Fairchild; A. M. Rothrock, AEC; G. Simmons, NEPA; T. A. Sims, NEPA	LP-73
May 19	"Air Force Interest in Nuclear-Powered Flight" See U. S. Air Force	LP-0
June 17	"NEPA Policy" R. C. Palmer, T. A. Sims, Carlton Ward	LP-38
June 21	"NEPA Designs" R. E. Adams, Kenneth Cooper, G. E. Cremer, Andrew Kalitinsky, A. Mason, David Poole	LP-31
June 29	F. B. Kaufmann	
Aug. 9	"Future Policy for NEPA" A. Kalitinsky, G. Simmons, T. A. Sims	LP-118
Sept. 10	Andrew Kalitinsky	
Sept 13	F. B. Kaufmann, R. C. Palmer	

Visits:

June 1 - 5

Lexington Personnel: Bevans, Bloom, Cooley, Feld, Friedman, Goodman, Grant, Kaufmann, Keirn, Klein, Marion, Mickley, Old, Redding, Rothrock, Shapiro, Smith, Soodak, Stever, Thiele, Welton, Whitman, Zacharias

Talks were given by NEPA as follows:

"Objectives of the NEPA Project" Gordon Simmons, Jr.	LP-5
"Aircraft Requirements" George Knopf	LP-3

NEPA, OAK RIDGE (Continued)

Visits:

June 1 - 5	Talks were given by NEPA as follows: (Continued)	
	"Engineering Approach" Andrew Kalitinsky	LP-1
	"Analysis Approach" Kenneth Cooper	LP-2
	"Cycle Analysis" David Poole	LP-7
	"Heat Transmission" A. C. Durland	LP-4
	"Reactivity" A. O. Mooneyham	LP-39
	"Radiation Protection" Murray Lesser	LP-40
	"Materials and Chemistry" Walter Koshuba, A. J. Miller, W. J. O'Leary	LP-41
	"Experiment and Physics" J. K. Bair, Frank Coneybear	LP-42
	"Controls and Instrumentation" E. S. Bettis	LP-43
	"Aircraft Design" R. E. Adams	LP-9
	"Power Plant Design" A. Mason	LP-8
June 16 - 18	"NEPA Symposium on Metals" Bloom	LP-52
June 28 - 29	"Further Discussion of NEPA Designs" Friedman, Hunter, Mickley, Smith, van Zelm	LP-57

NORTH AMERICAN AVIATION INC.

Visitors:

July 1 - 2	"Nuclear-Powered Flight" "Power Plants" F. W. Brown, Arthur Ford, M. A. Greenfield, Harry Pearlman, C. Starr, Stanley Thompson	LP-63 LP-64

F. H. NORTON CO.

Visitor:

July 7	"Ceramics" L. G. Milligan	LP-76

OHIO STATE UNIVERSITY

Visitor:

July 8	"Ceramets" A. R. Blackburn, G. A. Bole, F. H. McRitchie	LP-90

OAK RIDGE NATIONAL LABORATORY

Visitors:

June 23	"Radiation Damage" Sidney Siegel	LP-49
July 28 - 29	"Shielding" E. P. Blizzard	
Aug. 9 - 10	"Controls" H. W. Newson	
Sept. 8	"Reactors" Alvin Weinberg	

Visits:

June 29	"Biological Effects of Radiation" Friedman	LP-67
July 19 - 20	"Materials" Hunter	LP-62
Aug. 2 - 3	"Materials" Hunter	LP-111
Aug. 16 - 17	"Materials" Hunter	LP-128

RAND

Visitors:

June 14	"Discussion of RAND" Frank Collbohm	LP-47
July 1 - 2	"Nuclear-Powered Flight" "Power Plants" R. W. Dayton, E. H. Plesset, M. S. Plesset, K. M. Salter, Jr.	LP-63 LP-64

RUTGERS UNIVERSITY

Visitor:

July 9	"Ceramic Coatings" Harry Whitaker at MIT	LP-80

SURGEON GENERAL'S OFFICE, WASHINGTON, D. C.

Visitor:

July 20	"Biological Tolerances for Radiation" J. P. Cooney	

TURBINE POWER PLANTS MFG. CO.

Visitor:

June 25	"High Pressure Compressors" Nathan Price	LP-35

UNITED AIRCRAFT CORP.

Visitor:

Aug. 18	"Aircraft Power Plants" L. S. Hobbs	LP-129

U. S. AIR FORCE, WASHINGTON, D. C.

Visitors:

May 12 - 13	"Orientation Meeting" See NEPA	LP-73
May 19	"Plans for Organization at Lexington" Capt. A. L. Baird, U.S.N. Bu. Aer.; Maj. Gen. L.C. Craigie, USAF; Maj. Gen. A. R. Crawford, AMC; Col. D. J. Keirn, AEC; Brig. Gen. J. McCormack, AEC; R. C. Palmer, Fairchild; A. M. Rothrock, AEC; T. A. Sims, NEPA; Col. F. L. Wassell, AMC	LP-0

U. S. AIR FORCE, WASHINGTON, D. C. (Continued)

Visitors:

Aug. 16 - 17	"Airplanes" M. C. Demler	LP-127

Visits:

July 6 - 7	"Strategy" Johnston, Keirn, Loomis, Rothrock, Stever, Whitman of Lexington and Weil of AEC; to see Col. Armstrong, Comm. Brown, Lt. Col. Carlisle, Maj. Gen. Craigie, Col. Demler, Adm. Lonquist, Brig. Gen. Ritchie, Maj. Gen. Slater	LP-88a

U. S. AIR FORCE, WRIGHT FIELD

Visitor:

Aug. 16 - 17	"Airplanes" C. E. Reichert, Col. R. L. Wassell	LP-127

Visit:

Aug. 25	"Materials" A. Kaufmann	

U. S. CONGRESS, ATOMIC ENERGY COMMITTEE

Visitor:

July 8	E. L. Heller

U. S. NAVY, BUREAU OF SHIPS, WASHINGTON, D. C.

Visitor:

July 20	"Naval Nuclear Power Plants" Capt. H. G. Rickover	

Visit:

June 9	"Powder Metallurgical Techniques" Grant	LP-18

Appendix C

THE PLANNED SCHEDULE OF THE LEXINGTON PROJECT

The Lexington Project had to accomplish its mission in the short space of four months. It may be of interest to indicate the planned schedule by reproducing the only general policy memorandum which was issued during the life of the project.

June 1, 1948

MEMO TO STAFF - LEXINGTON PROJECT

The Atomic Energy Commission has asked M.I.T. to evaluate the feasibility of propelling aircraft by nuclear energy and to make recommendations for guiding the Commission's activities in this field. M.I.T. has consented. The scope of the agreement is as follows:

"The Contractor hereby agrees to use its best efforts to furnish or obtain the necessary personnel and facilities and do all things necessary to perform with the utmost dispatch the following:

"(a) Conduct a review of work which has been done or which is being done in the United States relevant to the propulsion of aircraft by nuclear energy

"(b) Evaluate the possibilities of nuclear propulsion of aircraft from a broad scientific and technical standpoint

"(c) Identify the important elements which should constitute a program for accomplishing the application of nuclear energy to the purpose of aircraft propulsion

"(d) Estimate a time scale or schedule for accomplishment of the application of nuclear energy to the purpose of aircraft propulsion

"(e) Make such other studies and review regarding nuclear propulsion of aircraft as may be agreed upon by the parties from time to time

"(f) Prepare and submit to the Commission in the shortest possible time after performance of the studies and investigations called for by this contract a report summarizing the Contractor's activities, findings and conclusions."

This is our task.

The first problem was to assemble in the short time between late April, when the decision was reached, and the first of June a group of

[redacted]

scientists and engineers who would be capable of fulfilling this responsibility. The group as now constituted is listed on the attached sheet. Some are on a full-time basis, some are half-time and others are consultants. About half are from M.I.T. There will doubtless be some additions as we start to function, but the major recruiting phase is over.

I believe that this emergency assignment may best be handled by close cooperation within the group, working to integrate the findings from the diverse investigations which we will make into the conclusions and recommendations of the finished report. Most of us are relatively uninformed on some of the most difficult questions which will be encountered. It would seem advisable to avoid prejudgment of the whole issue during the early phases.

The time schedule on the task is insistent. The Commission needs an early answer. Also, most of us come from academic posts and could not participate extensively after the fall term opens. We, therefore, shall turn in the report by September 30.

I estimate that meeting a final deadline of September 30 involves:

By September 1 - the final report is drafted, requiring only editing and printing

By August 1 - all component investigations have been completed and reported -- the overall conclusions and recommendations should be reasonably evident

By July 15 - all initial investigations should be completed.

A matter which the group might well discuss at its initial meetings in Lexington is the form of the final report. As a suggestion, the component studies might constitute separate reports, to be prepared by the specialists in the particular fields. With these as a background, the general report could then be reasonably concise.

We have accepted this assignment as a call to national service which we could not well refuse. We may well hope that our performance will at the same time bring long-range benefit to the science and engineering of nuclear energy.

Walter G. Whitman

[redacted]

[redacted]

Appendix D

NUCLEAR-POWERED FLIGHT INVESTIGATIONS

THE NEPA PROJECT

Because of the activity of NEPA (Nuclear Energy for Propulsion of Aircraft) in the field of atomic propulsion for aircraft, the Lexington Project has had frequent contacts with NEPA personnel during the summer (1948). The Lexington staff visited NEPA at Oak Ridge on four occasions, the first of which was for two and a half days in the first week of June and involved about thirty people from Lexington. Members of the NEPA organization have made six visits to the Lexington Project and have been thoroughly cooperative.

The following brief description is based upon information supplied to the Lexington Project by NEPA and represents an interpretation of NEPA's present plans and thinking.

The NEPA Project is an operation of the Fairchild Engine and Airplane Corporation, which is carrying out its program under a direct contract with the U. S. Air Force. The contract was awarded on May 26, 1946, under an agreement which established Fairchild as the responsible directing agency of the Project, and involved the participation of other aircraft power plant concerns as Member Companies. The purpose and scope of the project was twofold: (1) feasibility investigations and research leading toward the adaptation of nuclear energy to propulsion of aircraft, and (2) education of the aircraft engine industry in the field of nuclear science and its adaptation to aeronautical propulsion.

The educational function received first emphasis. A library of technical information pertinent to NEPA's problem was organized principally from Atomic Energy Commission reports. The first major objective was to create a manual of pile engineering, which has grown to about twenty-five volumes but is not yet published. The second objective was to present instruction and information programs. These have included periodic seminars, lecture series and symposia on such subjects as nuclear chemistry, physics, pile theory, heat transfer, mathematics, metallurgy and aircraft engineering. NEPA is planning a continuous program of research with study or teaching for its staff.

The major technical objective of NEPA is a reactor suitable for aircraft propulsion. Cycle analysis, one of NEPA's first activities, has resulted in the selection of the open-cycle, turbojet type of propulsion for the following stated reasons: (1) aircraft for this type are now well developed; (2) the art of designing and constructing turbojet machinery is considered to be fairly advanced; (3) the turbojet is said to require the lowest reactor temperature of the various jet-propulsion engines, and (4) the closed cycle is considered a dead end for supersonic applications. Thus, the present practical objective of NEPA's engineering approach is the design, construction, and development of a test-stand turbojet power plant.

[redacted]

NEPA has based its plans on three types of airplanes: inhabited subsonic, inhabited supersonic, and uninhabited supersonic. The first study undertaken was on the uninhabited supersonic plane, without a shield. Design point calculations have yielded a Mach number of 1.5 at 40,000 feet altitude with a payload of 10,000 pounds and gross weight of 87,000 pounds. The reactor would use enriched uranium 235 in graphite moderator and would give a heat release of 132,000 kw. The power plant consists of a reactor and four turbo-compressor units. Studies of inhabited aircraft are now being initiated.

NEPA has studied shielding, reactivity and reactor control, materials and material development, heat transfer, and aircraft design. Pile construction problems have also been considered. NEPA believes that the most uncertain aspects of its present program are those relating to thermal stresses in and coatings for the reactor fuel elements. Some sort of coating is regarded as necessary to prevent the combustion of carbon with air at the reactor temperature and the escape of fission products into the air.

The future program planned by NEPA is outlined in a recent report. (1) This represents their considered expectations rather than guaranteed progress. At the end of about a ten-year development period, a complete integrated power plant is expected to be available.

An immediate requirement stated by NEPA is a site for its critical experiments, 225 kg (500 lb.) of fissionable material for these experiments and facilities for reprocessing such material. At Oak Ridge additional space and housing allotments are regarded as essential.

NEPA feels that its work would be greatly aided if the Atomic Energy Commission and the U. S. Air Force develop "joint sponsorship and interest" in the Project. Specifically, they favor a policy-level exchange of correspondence to establish this cooperation, which already exists in fair measure at the working level. The representatives of NEPA have expressed their belief that the control of an integrated effort, with undivided responsibility for the development of a nuclear aircraft power plant, should reside in an organization with the objectives represented by NEPA.

NORTH AMERICAN AVIATION INC.

In January, 1946, the Aerophysics Laboratory of North American Aviation Inc. undertook as Project MX-770 a guided missile study for the United States Air Force. This program included a preliminary investigation of a nuclear-powered rocket, the results of which are summarized in a North American Aviation Report. (1) The conclusion reached was that nuclear rockets were feasible provided marked advances could be made in the materials field.

On the basis of this preliminary report, an extensive investigation of nuclear-powered rockets and ramjets was then undertaken with a staff of about fifteen full-time technical personnel, with another fifteen

working half-time. Another North American Aviation report (2) summarizes this work. The Project examined the engineering feasibility of the application of nuclear energy to long-range, supersonic rockets and ramjets. This analysis indicated that a 10,000-mile rocket missile, nuclear-powered and hydrogen-propelled, could be designed and constructed provided a graphite-moderated reactor could be built to operate at 5700°F. The estimated gross weight of the missile was about 100,000 pounds and the useful payload was 8000 pounds. The reactor considered was unshielded. With only slight modifications and without the payload, this rocket could escape from the gravitational field of the earth. Several propellant fluids were investigated; of these hydrogen appeared to be the best.

An analysis of the components of this nuclear, hydrogen-propelled, 10,000-mile missile was made, including problems of structural arrangements, aerodynamic stability, steering, controls, turbines and pumps for the propellant, and nuclear reactor design. Various techniques of impregnating graphite with uranium were considered. Tantalum carbide coatings to protect the graphite from hydrogen at high operating tempertures were developed and tested.

The Project also considered a nuclear-powered ramjet with an unshielded reactor of uranium impregnated with beryllium oxide, operating at approximately 3600°F. Such a nuclear-powered ramjet, carrying an 8000-pound payload, would have a gross weight of 17,600 pounds and could fly almost indefinitely at a speed of 2000 miles per hour.

On the assumption that these reactors could be built, and neglecting the cost of fissionable material, the comparative construction cost of nuclear-powered vs. chemically powered ramjets and rockets was investigated. With the same payload and performance characteristics, for ranges of over 4000 miles, the nuclear-powered vehicles were cheaper. This same North American Aviation report (2) contains a recommended research and development program for the next five years and emphasizes the necessity for basic investigation of materials at high temperatures. The program does not include any work on shielding.

PROJECT RAND

Project RAND was set up by the U. S. Air Force after the war, and at present is administered by Wright Field. Its purpose is to make studies of all aspects of intercontinental warfare. The Project was initiated early in 1946 at the Douglas Aircraft Corporation with the assistance of Consolidated Vultee, Northrop Aircraft, North American Aviation, and several other agencies. Practically all of its work in the field of nuclear-powered aircraft has been done in cooperation or under subcontracts with North American Aviation Inc. In August, 1947, a month-long conference of scientists from various parts of the United States was held at RAND to study nuclear power. The most important outcome of this conference was a considerable clarification of the problem in the fields of uncertainty connected with nuclear power. A large number of calculations and studies was made at this conference. (19-26) A

joint study by RAND and North American was made on a complete nuclear-powered turbojet power plant design.(11-14) RAND has now almost completed its study, pending accumulation of more adequate experimental data on shielding and materials.

At the end of 1947, North American was directed by the Air Force to terminate its nuclear investigations under Project MX-770. The work was then financed by the company for several months until the Atomic Energy Commission gave North American a contract covering reactor and related material investigations. Laboratory investigations at high temperatures on heat transfer, coatings and properties of graphite were initiated early in the program.(1)(2) Arrangements were made with the University of California at Berkeley to use a cyclotron for bombarding graphite and other materials with alpha particles and deuterons to simulate the effects of high neutron fluxes. Experiments have been carried out at both high and low temperatures with encouraging results.(3)(4)(5) In addition, North American has made several other applied reactor problem studies (6-14) and has also developed a preliminary analysis technique applicable to nuclear-powered aircraft, (18) the purpose of which is to optimize the principal parameters of nuclear power plant design.

THE NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

The National Advisory Committee for Aeronautics was established by the Congress in 1915 to be the Government's aeronautical research agency. The enabling act charges the NACA with "study of the problems of flight with a view to their practical application". The NACA maintains a small headquarters staff in Washington and owns and operates three laboratories -- at Langley Field, Virginia, at Moffet Field, California and at Cleveland, Ohio. The total NACA staff is about 6500. Since 1945 the NACA has been studying the application of nuclear energy to aircraft propulsion. Members of the Lexington Project staff have made visits to the Lewis Flight Propulsion Laboratory (Cleveland, Ohio) of the NACA to discuss these studies. Members of the NACA staff have also visited Lexington.

The NACA investigations on nuclear energy power plants for aircraft have been chiefly in the fields of analysis of power plant cycles suitable for the application of nuclear energy, heat-transfer research and high temperature materials research. The purposes of these researches are to provide the National Military Establishment and the aircraft industry with information on which to base the development of a power plant for nuclear-powered aircraft. No particular power plant cycle has been recommended as yet, the overall purpose of the analysis being at present to determine the advantages and disadvantages of the different cycles. The analyses include the open and closed cycles with the emphasis on optimizing each cycle.

Experimental researches in heat transfer are extending the existing data on heat-transfer coefficients to the range of temperatures and heat flows that would be experienced in a nuclear reactor for an aircraft power plant. Liquids, gases, metals and refractories are being investigated.

The work of the NACA on high-temperature materials is conducted on metals, ceramics and ceramets and is expected to provide information which will permit the use of higher temperatures in the turbine engines and will afford greater reliability of the highly heated engine parts. Some consideration has been given to the problem of shielding and to the effects of radioactivity on materials that might be used in an aircraft reactor.

To further its knowledge in the study of nuclear energy the NACA has established a working agreement with the Atomic Energy Commission through the Oak Ridge and Argonne National Laboratories, and has assigned members of its staff to these laboratories. Conferences between the NEPA and NACA staffs are also held to insure that the two organizations benefit from each other's work.

APPLIED PHYSICS LABORATORY

The Applied Physics Laboratory of the Johns Hopkins University was formed at the outbreak of the war to undertake research and development work on fire control, radar, and radio proximity fuses for the Office of Scientific Research and Development and the United States Naval Bureau of Ordnance. In late 1944 the Laboratory entered the guided missiles field primarily to develop a ramjet-propelled, supersonic, antiaircraft radar-guided missile for the Bureau of Ordnance. This is now the main project. At present the central laboratory at Silver Springs coordinates the work of about twenty-five industrial and academic institutions engaged on this guided missile program and associated problems.

In June, 1946, a committee of seventeen members headed by A. E. Ruark was formed within the Laboratory to consider the general problem of air vehicles driven by nuclear power. They wished to determine whether nuclear-powered aircraft were feasible and whether these might be competitive with the chemically fueled, guided missiles they had under consideration. The report of the Committee (1) was written without the use of any classified information from the Manhattan Engineering District of nuclear power. With the exception of a small investigation on shielding no further work on nuclear power has been done by this group.

Several different schemes for the utilization of nuclear energy for propulsion were considered, but it was decided that the only feasible methods were those which extracted the energy in the form of heat to run more or less conventional propulsive power plants. Calculations were made on the basis of unshielded, cylindrical open-cycle thermal reactors, utilizing uranium 235 or plutonium uniformly mixed with graphite or beryllium oxide as moderators. Both convective and radiative heating of the coolant were investigated. The Laboratory concluded that only convective heating was feasible.

The materials question, which in all cases constituted the limiting design factor and the chief research problem, was investigated only briefly. Little more was accomplished than to define the necessary advances in this field. Because this group felt the uncertainties in the report (due to the non-use of Manhattan Engineering District

classified information) to be quite great, such problems as aerodynamic design, power plants, navigation, etc. were only lightly considered. The nuclear aspects were examined more thoroughly, using estimates where information was either classified or unavailable.

This group concluded that the nuclear turbojet offered the most promise for future consideration and decided that it would be competitive on a weight basis with chemically fueled aircraft at ranges of over 4000 miles. Estimated design specifications for a missile to carry a 12,000-pound payload at Mach number 0.9 were: gross weight, 46,000 pounds; weight of uranium 235, 50 pounds; unshielded reactor diameter, 7 feet, length, 6 feet, weight, 13,500 pounds; structural weight, 18,200 pounds; turbine inlet temperature, 1600°F; average reactor wall temperature 1950°F, and thrust 4000 pounds. The engineering and materials problems of this design were not investigated sufficiently to prove feasibility. For an unshielded nuclear ramjet operating at Mach number 2, it was estimated that a beryllium oxide-moderated reactor, 8 feet long and 7 feet in diameter with a free-flow ratio of 0.556 would be required. The reactor wall temperature would be 4000°F.

The feasibility of rockets was also studied, and the conclusion reached that these offered a much more difficult problem than the turbojet or ramjet. Hydrogen appeared to be the most desirable working fluid because of its low molecular weight. However, since hydrogen has a low liquid density, a large tankage volume (and consequently a large tankage weight) would be required. A nuclear rocket would not be feasible with gas temperatures below 2000°F. With wall temperatures of 4900°F the specific impulse would be about three times as high as the usual chemical rockets.

The Applied Physics Laboratory estimated that a study program of at least two and a half years would be required to investigate further the basic problems of nuclear-powered flight, particularly those concerning materials of reactors and power plants.

REFERENCES

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(1) Report of Meeting, Board of Consultants and Member Companies, NEPA Project, Oak Ridge, Tennessee, August 11-12, 1948.

Many other communications were received from NEPA -- references will be found elsewhere in the report.

North American Aviation, Inc.

(1) Yockey, H. P., Dixon, T. F. "A Preliminary Study on the Use of Nuclear Power in Rocket Missiles", NAA Report No. NA-46-574, July 1, 1946.

(2) NAA Report NA-47-15, February 11, 1947, "Feasibility of Nuclear Powered Rockets and Ram Jets", prepared by Staff of Aerophysics Laboratory, C. Starr -- Group Leader.

Project RAND

(1) Pearlman, H. "Studies on Nuclear Reactors -- High Temperature Chemical Behavior of Reactor Materials", Report AL-326, December 1, 1947.

(2) Malmstrom, C. R. "Studies on Nuclear Reactors -- Tensile Strength and Creep of Graphite Up to the Sublimation Point", Report AL-327, December 1, 1947.

(3) Perkins, W. E. "Studies on Nuclear Reactors -- Effects of Cyclotron Bombardment on the Physical Properties of Graphite", Report AL-328, November 15, 1947.

(4) Brown, F. W. "Studies on Nuclear Reactors. 4. Theory of Ionization and Atomic Displacement Produced by Fast Particles in Graphite. 1. Preliminary Considerations", Report AL-329, February 1, 1948.

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(6) Starr, C. "Studies on Nuclear Reactors. 8. Elementary Theory of Homogeneous Epithermal Nuclear Reactors", Report AL-331, March 1, 1948.

(7) Greenfield, M. A. "Studies on Nuclear Reactors. 9. Temperature Stability of an Epithermal Reactor", Report AL-332, March 1, 1948.

(8) Malmstrom, C. R. "Studies on Nuclear Reactors. 10. Experimental Method for the Determination of Resistivity Change in Graphite Bombarded with Deuterons and Alpha Particles", Report AL-333.

(9) Greenfield, M. A. "Studies on Nuclear Reactors. 6. Power Developed by Decay of Fission Fragments", Report AL-335, November 1, 1947.

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(12) Salter, R. M., Jr. "A Parametric Study of Nuclear Power Plant Factors Applicable to Aircraft Propulsion", Report AL-338, Volume 1.

(13) Same: Report AL-338, Appendix 2, Volume II.

(14) Same: Report AL-338, Appendix 3, Volume III.

(15) Starr, C., Beaupre, M. A., Salter, R. M., Vinson, W. D. "Very Long Range Bomber Study", Report AL-339, January 5, 1948.

(16) Thompson, A. S. "Thermal Stress in Power-Producing Element", Report AL-348, March 1, 1948.

(17) Brown, F. W., and Greenfield, M. A. "Analysis of Feasibility of Long Range Guidance Systems", Report AL-349, November 1, 1947.

(18) Starr, C. "Method for Preliminary Analysis of Nuclear Powered Aircraft", Report NAA-SR-22, August 17, 1948.

(19) Serber, R. "Use of Atomic Power for Rockets", Report RAD-2, July 5, 1946.

(20) Alvarez, L. W. "Notes on the Polonium Engine", Report RAD-31, August 16, 1946.

(21) McMillan, E. M. "Atomic Drive for Missiles", Report RAD-32, August 16, 1946.

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