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RESEARCH MEMORANDUM

PRELIMINARY ANALYSIS OF THREE CYCLES FOR NUCLEAR PROPULSION OF AIRCRAFT

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PRELIMINARY ANALYSIS OF THREE CYCLES FOR NUCLEAR PROPULSION OF AIRCRAFT*

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SUMMARY

A preliminary study was made of the feasibility of three cycles for nuclear propulsion of aircraft: a direct-air turbojet, a binary liquid-metal turbojet, and a helium-compressor jet.

The analysis indicated that all three cycles may tentatively be considered feasible for flight at a Mach number of 0.9 and altitudes up to at least 50,000 feet. The direct-air and helium cycles resulted in heavier aircraft to carry a specified pay load than did the liquid-metal cycle.

The liquid-metal cycle appeared feasible for flight at a Mach number of 1.5 and altitudes up to 50,000 feet (maximum altitude investigated at Mach number of 1.5). With the shields considered, however, the direct-air and helium cycles resulted in considerably heavier aircraft than for the liquid-metal cycle at this Mach number.

The relative advantage of the liquid-metal cycle, as indicated by the minimum airplane weight required to carry a specified pay load, became greater as the flight speed and altitude increased and as the reactor-wall temperature decreased.

INTRODUCTION

Analyses are being made at the NACA Lewis laboratory to determine the feasibility of various cycles for nuclear propulsion of aircraft. These studies are intended to reveal the propulsion cycle, or cycles, that will result in the lightest airplane to carry a specified pay load, and an effort is also being made to define the optimum engine

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design conditions at various flight speeds and altitudes. Some of the results that have been obtained to date are presented herein where the performance of three cycles is considered. These cycles are a direct-air turbojet, a liquid-metal turbojet, and a helium-compressor jet. 1403

Inasmuch as the feasibility studies are incomplete, the present results must be considered as preliminary and therefore subject to additional refinement and revision. Further, it is realized that analysis alone cannot yield a conclusive answer as to the most feasible cycle. For example, in the present analysis, the existence of suitable reactor materials for operation at the various temperatures, heat-release rates, and coolant flow rates is implicitly assumed. A considerable amount of experimental and development work will be required for an accurate indication of the limitations imposed by materials on reactor operating conditions, which will permit a more realistic evaluation of the feasibility of the various cycles.

DESCRIPTION OF CYCLES

Direct-air turbojet. - The components of the direct-air turbojet are schematically illustrated in figure 1. Air enters through an inlet diffuser and passes through the compressor into the reactor where it is heated by contact with the walls of the reactor flow passages. From the reactor, the air expands through the turbine and the exhaust nozzle as in the conventional turbojet. Inasmuch as the optimum performance of this system occurs at relatively high compressor pressure ratios, an intercooler was included between compressor stages.

Liquid-metal turbojet. - The liquid-metal turbojet is shown in figure 2. It is a binary system in which the liquid-metal coolant is pumped through the reactor and the heat exchanger. Air enters a diffuser, passes through the compressor and heat exchanger, and then expands through the turbine and the nozzle.

For the present study, the liquid-metal coolant was assumed to be lithium.

Helium-compressor jet. - In the helium-compressor jet illustrated in figure 3, helium at high pressure circulates through reactor, turbine, heat exchanger, and compressor. The excess turbine power over that required for compression of the helium is delivered through gears to an air compressor in an open cycle. In the open part of the cycle, air enters through a diffuser and passes through the air compressor into the heat exchanger. From the heat exchanger, the air expands through a nozzle to provide the propulsive thrust.

Only the compressor jet arrangement will be considered herein. A few calculations have been made, however, for two other configurations. In one of these configurations, the excess power from the helium turbine was delivered to a conventional propeller. In the other configuration, the helium turbine supplied only enough power to drive the helium compressor. The air compressor was driven by an air turbine located downstream of the heat exchanger. The configuration shown in figure 3 resulted in slightly higher values of thrust per unit engine weight than the other two at the conditions investigated.

ASSUMPTIONS

Engine components. - In the analysis of the three cycles, the efficiencies and the weights of the various engine components, such as compressors, turbines, pumps, and heat exchangers are, insofar as possible, representative of the best current practice. Some of the pertinent assumptions regarding component performance for the three cycles are listed in the following table:

	Direct air	Liquid metal	Helium
Diffuser recovery factor (ratio of actual to theoretical total pressure):			
Mach number, 0.9	0.96	0.96	0.96
Mach number, 1.5	.95	.95	.95
Compressor small-stage efficiency	.88	.88	----
Compressor adiabatic efficiency:			
Air	----	----	.85
Helium	----	----	.85
Turbine adiabatic efficiency	.90	.90	.90
Exhaust-nozzle velocity coefficient	.96	.97	.97
Intercooler cooling effectiveness	.50	----	----

Reactors. - The reactors were, in general, assumed to be cylinders of equal length and diameter. A minimum length and diameter of 3 feet (based on nuclear considerations) was set for the gas-cooled reactors, otherwise the size was determined by heat-transfer limitations, and was allowed to vary as necessary to obtain the required engine thrust.

Preliminary calculations for the liquid-metal cooled reactor indicated that a minimum diameter of 2.5 feet, as limited by nuclear considerations, might be feasible. Because of the uncertainty of the actual nuclear limitation on reactor size, however, calculations were made for both 2.5- and 3.0-foot reactor diameters. With each of these sizes, the heat release was varied as necessary to obtain the required engine thrust. The use of a constant reactor size, irrespective of heat release, was possible because, in general, heat-transfer considerations did not appear to impose a size limitation on the liquid-metal cooled reactor, as was the case for the gas-cooled reactors.

A few calculations were also made for a split-flow gas-cooled reactor at operating conditions where the straight-through gas-cooled reactors were large. In the split-flow arrangement, the reactor is cut by a transverse gap midway between the ends. The coolant flows into this gap, through the reactor, and out both ends. The split-flow arrangement permits a specified flow area to be obtained with a smaller core diameter, and hence with a lighter shield, than the single-pass cylindrical reactor.

Shield weights. - Two types of shield were considered. These are the integral, or unit, shield and the separated shield.

The integral shield is essentially a thick jacket of high-density material surrounding the reactor. For most of the calculations, the weight of the integral shield was based on a specific gravity of 8.0 and a thickness of 2.5 feet. A few calculations were also made to show the effect of varying the shield thickness on airplane weight with the liquid-metal cycle.

The separated shield, which was considered only for the air-cooled reactor, consists essentially of a jacket of a relatively low-density material surrounding the reactor and a separate lead compartment for the airplane crew. For the present investigation, the weight of the separated shield was based on the following configuration: The reactor was assumed to be surrounded by 4 inches of lead, which in turn was surrounded by 4 feet of water and a steel container. The crew compartment was assumed to be a hollow lead cylinder closed on the end facing the reactor. The length and the diameter of the crew compartment were taken as 18 and 7 feet, respectively, and the weight was approximately 100,000 pounds.

The shield weights for the gas-cooled reactors were increased to account for ducting the coolant into and out of the reactor.

For the present analysis, the integral and separated shields for the air-cooled reactor have the same weight for a reactor length and diameter of 3 feet. For smaller reactors, the integral shield is lighter and for larger reactors the separated shield is lighter.

Airplane structure weight. - The airplane structure weight was assumed to be 35 percent of the gross weight. This fraction of the gross weight represents a mean value obtained from a survey of current large aircraft. The ratio of structure to gross weight will probably vary with airplane and power-plant design, and possibly with airplane size and design flight condition. More refined values for the structure weight would, however, require detailed design studies, which were beyond the scope of the present preliminary study.

Airplane lift-drag ratio. - The airplane lift-drag ratio was assumed to vary with flight Mach number, as shown in figure 4. This curve was obtained from a survey of existing data and represents what is felt to be a reasonable mean. The present analysis was made for flight Mach numbers of 0.9 and 1.5 for which the lift-drag ratios are 18 and 9, respectively.

No limit was placed on the airplane wing loading and no consideration has been given to take-off or landing limitations.

METHODS

The evaluation of the three cycles considered in the analysis is, in general, on the basis of the airplane gross weight required to carry a pay load of 20,000 pounds.

The performance of each cycle was considered optimum at a given flight speed, altitude, and reactor-wall temperature when the gross weight to carry the specified pay load was a minimum.

The performance of the direct-air cycle was optimized at each flight condition and reactor-wall temperature by varying the mass flow rate of air in the reactor through a range of values at each of several compressor pressure ratios. The combination of reactor air flow (or reactor pressure drop) and compressor pressure ratio resulting in the minimum airplane weight was considered to be the optimum engine design condition.

The performance of the liquid-metal cycle was optimized by varying the heat-exchanger pressure drop and compressor pressure ratio through a range of values at each flight condition and reactor-wall temperature in much the same manner as for the direct-air cycle.

The performance of the helium cycle was optimized in a manner similar to that for the other cycles, except that several additional variables were considered. These were the helium compressor pressure ratio, the pressure drop of both helium and air in the heat exchanger, the warming and cooling effectiveness of the heat exchanger, and the ratio of helium flow to air flow.

RESULTS AND DISCUSSION

Comparison of Cycles at Mach Number of 0.9

Effect of compressor pressure ratio. - For a given flight condition and turbine-inlet temperature, the engine thrust and weight, and hence the airplane weight, is a function of compressor pressure ratio and a pressure ratio exists for which the airplane weight required to carry a specified pay load is a minimum. This relation is illustrated in figure 5 where the airplane gross weight required to carry a pay load of 20,000 pounds is plotted against compressor pressure ratio for each of the three cycles. The curves are for flight at a Mach number of 0.9 and an altitude of 40,000 feet. The reactor-wall temperature is 2300° R and the turbine-inlet temperature is 2100° R.

The gross weight for the air cycle has a minimum value of about 525,000 pounds at a pressure ratio of about 44. At the point of minimum weight, the reactor diameter is 3.25 feet, and it increases to about 5 feet at a pressure ratio of 20. The curve is based on the use of an integral-type shield having a thickness of 2.5 feet and a specific gravity of 8.0. Inasmuch as the reactor size at the point of minimum gross weight is only slightly greater than that for which the integral and separated shields weigh the same, the use of the separated shield would result in a decrease in gross weight of only 5 percent.

The optimum pressure ratio for the air cycle decreases as the reactor-wall temperature is decreased and as the flight speed is increased. For example, although not shown in figure 5, a reduction of wall temperature to 2000° R decreases the optimum pressure ratio to about 30 for the flight conditions shown. At the same altitude and temperature as indicated in the figure, an increase in flight Mach number to 1.5 reduces the optimum pressure ratio to about 15. At constant Mach number and temperature, the optimum pressure ratio increases slightly with altitude, at least for altitudes up to 50,000 feet.

The curve for the helium cycle represents the variation of helium compressor pressure ratio. The minimum gross weight is about 550,000 pounds and occurs at a pressure ratio of about 3. The corresponding air compressor pressure ratio remains nearly constant at about 2, as indicated by the vertical line. The reactor diameter is 3 feet at the minimum point.

The liquid-metal cycle gives the lightest airplane at the conditions shown, with a minimum gross weight of approximately 330,000 pounds at a pressure ratio of about 10. Use of the air cycle results in an airplane that is over 50 percent heavier than that with the liquid-metal cycle at the same flight condition. Also much higher pressure ratios are required with their attendant development problems. The pressure ratio for the liquid-metal cycle is in the range of current practice, and the cycle is relatively insensitive to changes in pressure ratio. For example, the gross weight at a pressure ratio of 5 is only 4 percent greater than the minimum.

Because the reactor length and diameter for the liquid-metal cycle are constant at 2.5 feet, the integral type of shield was used. Increasing the reactor size to 3 feet would increase the gross weight to about 390,000 pounds, which is still appreciably lighter than with the gas cycles at the flight condition considered.

The optimum pressure ratio for the liquid-metal cycle is relatively insensitive to changes in altitude and temperature. It decreases, however, with increasing flight Mach number, being about 5 at a Mach number of 1.5.

Effect of reactor-wall temperature. - A comparison of the performance of the three cycles as affected by reactor-wall temperature is made in figure 6. Airplane gross weight, for 20,000 pounds pay load, is plotted against reactor-wall temperature at a flight Mach number of 0.9 and an altitude of 40,000 feet. The turbine-inlet temperature is 200° R below the reactor-wall temperature.

At the point where the two curves for the air cycle meet, the integral and separated shields weigh the same. The reactor length and the diameter at this point are 3 feet, which, as previously stated, was the minimum size considered for the gas-cooled reactors. The corresponding wall temperature is slightly less than 2400° R and the gross weight is about 500,000 pounds. As the wall temperature is decreased, the reactor size and airplane gross weight increase quite rapidly. At a wall temperature of 2000° R, the airplane would weigh well over 1,000,000 pounds with the integral-type shield, and about 650,000 pounds with the separated shield. Calculations for the split-flow-type reactor core with an integral shield indicate that at 2000° R the weight might be further reduced to about 530,000 pounds.

A single point is indicated for the helium cycle, which is slightly heavier than the air cycle at the same operating condition.

Curves for two reactor sizes are shown for the liquid-metal cycle. Figure 6 indicates that the required weights with the liquid-metal cycle are again less than with the air cycle and also that the airplane weight increases less rapidly with a reduction in reactor-wall temperature. Inasmuch as the size of the liquid-metal cooled reactor remains constant over the range of wall temperatures shown, the increase in airplane weight with reduction in wall temperature is due primarily to the required increase in engine weight.

Effect of altitude. - The performance of the cycles as affected by altitude is compared in figure 7. The curves are for 0.9 Mach number and reactor-wall and turbine-inlet temperatures of 2300° and 2100° R, respectively. Curves representing both the integral and separated shields are again shown for the air cycle. The point at which the curves meet again corresponds to the minimum reactor size of 3 feet. The altitude at this point is slightly less than 40,000 feet.

With the integral-type shield, the required airplane weight for the direct-air cycle increases extremely rapidly as the altitude increases above 40,000 feet. The weight increase with altitude is much less with the separated shield; however, even with the separated shield, the airplane would weigh about 750,000 pounds at 50,000 feet. The use of a split-flow reactor with an integral shield further reduces the weight at 50,000 feet to about 600,000 pounds.

The point for the helium cycle from figure 6 is again included, and indicates that the helium cycle is slightly heavier than the air cycle at the same altitude.

Curves for the two reactor sizes are included for the liquid-metal cycle. Figure 7 indicates that the required weights with the liquid-metal cycle are less than with the air cycle, and increase less rapidly as the altitude is increased. For each reactor size, the increase in airplane weight with altitude is due to the required increase in engine weight. The gross weights with the liquid-metal cycle appear reasonable even at an altitude of 70,000 feet, whereas with the other cycles the weights would be prohibitively large.

Effect of pay load. - The results that have been presented thus far are for a constant pay load of 20,000 pounds. This value is, however, somewhat arbitrary and figure 8 is included to illustrate the increase in gross weight required as the pay load is increased above 20,000 pounds. The curves are for flight at a Mach number of 0.9 and 40,000 feet. The reactor-wall and turbine-inlet temperatures are 2300° and 2100° R, respectively.

For the air and helium cycles, the reactor size increases with increased pay load. For the liquid-metal cycle, the reactor length and diameter are constant at 2.5 feet but the reactor heat release per unit volume increases with increased pay load.

Increasing the pay load from 20,000 to 40,000 pounds, or by 100 percent, increases the airplane gross weight approximately 12 percent for each cycle. Roughly the same percentage increase in gross weight would be expected at other flight conditions and temperatures. Closer comparison of the two curves for the air cycle shows the increasing advantage of the separated shield in saving weight as the pay load is increased.

Direct Air and Helium Cycles at Mach Number of 1.5

The results presented thus far have been for flight at a Mach number of 0.9. Preliminary analyses were also made for a Mach number of 1.5; the results indicated that airplanes powered by the direct-air or helium cycles might weigh on the order of 600,000 to 1,000,000 pounds. If reactor-shield weights can be materially reduced from the values used in this preliminary study, the airplane weights for the air and helium cycles can be greatly decreased. Further study of these two cycles is required for more accurate evaluation of their feasibility at supersonic flight speeds.

Liquid-Metal Cycle at Mach Number of 1.5 and Further Evaluation at Mach Number of 0.9

The liquid-metal cycle appeared more promising at a Mach number of 1.5 than the direct-air and helium cycles, and some of the results are presented in figures 9 to 11 with additional results for a Mach number of 0.9.

Effect of reactor-wall temperature, altitude, and Mach number. - The effect of reactor-wall temperature on the gross weight required to carry a pay load of 20,000 pounds with the liquid-metal cycle is illustrated in figure 9(a). Curves are shown for three altitudes at a flight Mach number of 0.9. The turbine-inlet temperature is 200° R below the reactor-wall temperature. The curves are based on a 2.5-foot reactor and an integral shield.

The relative insensitivity of the cycle to reduction in temperature level for altitudes up to 50,000 feet is indicated by the figure. At 30,000 feet, a reduction in wall temperature from 2500° to 1700° R increases the gross weight from about 300,000 to 340,000 pounds, or by 13 percent. At 50,000 feet, the corresponding weight increase would be from 350,000 to 440,000 pounds, or about 25 percent. At 70,000 feet, the weight is about 490,000 pounds at 2500° R, and increases fairly rapidly as the wall temperature is decreased.

The effect of temperature on the performance of the liquid-metal cycle is further illustrated in figure 9(b). The coordinates are the same as before, but in this case the curves are for a flight Mach number of 1.5, and the maximum altitude considered is 50,000 feet.

As for the low Mach number, the effect of temperatures on gross weight is relatively small for the range of temperatures shown. At 50,000 feet, a reduction in reactor-wall temperature from 2500° to 2000° R increases the gross weight from 380,000 to 450,000 pounds, or 18 percent. Further reduction in temperature with the corresponding decrease in thrust would cause the engine, and hence the airplane, weight to increase quite rapidly at this flight condition. 1403

For any temperature and altitude, the weights are heavier at a flight Mach number of 1.5 than at 0.9. The increase in airplane weight with flight speed is due primarily to increased engine weight inasmuch as the same reactor size, and hence the same shield weight, was used at both Mach numbers. The increased engine weight is due mainly to the greater thrust requirement resulting from the lower lift-drag ratio at the higher flight speed.

Reactor heat-release rate. - In figures 9(a) and 9(b), the reactor length and diameter were constant at 2.5 feet so that changes in altitude, temperature, or flight speed resulted in a change in the reactor heat release per unit volume. Some idea of these heat-release requirements can be obtained from figure 10. Gross weight, for a pay load of 20,000 pounds, is plotted against reactor heat release in kilowatts per cubic inch of reactor volume. The figure is for reactor-wall and turbine-inlet temperatures of 2300° and 2100° R, respectively, and curves are shown for two altitudes at each of two flight Mach numbers. The right end of each constant altitude curve represents the heat release required from a 2.5-foot reactor, and the left end represents the requirement for a 3-foot reactor. Increasing the reactor diameter from 2.5 to 3.0 feet at constant flight Mach number and altitude might be considered to correspond to a reduction in thermal stresses, a decrease in reactor free-flow area, a decrease in liquid-metal velocity, and attendant erosion and corrosion rate, or a combination of these changes. The benefits of the decrease in heat-release rate are obtained, however, at the expense of a heavier airplane.

At a Mach number of 1.5 and 30,000 feet, the required heat release decreases from about 13 to 8.5 kilowatts per cubic inch as the reactor core size is increased from 2.5 to 3.0 feet. The corresponding gross weight increases from about 340,000 to 380,000 pounds, or about 12 percent. At 50,000 feet, the heat release decreases from about 15.5 to 10 kilowatts per cubic inch for the same increase in reactor size. The gross weight increases from 400,000 to 450,000 pounds, which is again about 12 percent.

At a Mach number of 0.9, the required heat releases are much lower, varying from about 3.5 to 6 kilowatts per cubic inch for the range of conditions shown. The corresponding heat-release rates for the air-cooled reactor are of the order of 3 to 4 kilowatts per cubic inch.

<u>Effect of shield thickness</u>. - The results shown thus far for the liquid-metal cycle have been based on the use of the integral-type shield having a specific gravity of 8.0 and a thickness of 2.5 feet. Inasmuch as some uncertainty exists as to the actual thickness that will be required to achieve satisfactory attenuation, figure 11 is included to illustrate the change in gross weight that might be expected to accompany a change in shield-thickness requirements. Gross weight required to carry a pay load of 20,000 pounds is plotted against shield thickness for two reactor sizes. The plot is for flight at 1.5 Mach number, and 50,000 feet. The reactor-wall and turbine-inlet temperatures are 2300° and 2100° R, respectively. The shield specific gravity is 8.0.

The desirability, with respect to decreased airplane weight, of keeping the shield thickness small is illustrated in figure 11. For example, with either reactor size, an increase in shield thicknes from 2 to 3 feet practically doubles the airplane weight.

SUMMARY OF RESULTS

The results of this preliminary study of the feasibility, for nuclear propulsion of aircraft, of a direct-air turbojet, a liquid-metal turbojet, and a helium-compressor jet may be summarized as follows:

1. All three cycles appeared feasible for flight at a Mach number of 0.9, and altitudes up to at least 50,000 feet. The direct-air and helium cycles, however, resulted in somewhat heavier aircraft than the liquid-metal cycle.

2. The liquid-metal cycle appeared feasible for flight at a Mach number of 1.5 and altitudes up to 50,000 feet, which was the highest altitude considered at this Mach number. With the shields considered in this preliminary analysis, the direct-air and helium cycles resulted in much heavier aircraft than the liquid-metal cycle at a Mach number of 1.5. If shield weights can be materially reduced from the values used herein, the airplane weights with these cycles can be appreciably decreased.

3. The relative advantage of the liquid-metal cycle, as indicated by minimum airplane weight to carry a specified pay load, became greater as the flight speed and altitude was increased and as the reactor-wall temperature was decreased.

4. Further analysis as well as experimental and development work are required to obtain a more accurate and realistic indication of the relative feasibility of these cycles.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics, 1403
Cleveland, Ohio, August 10, 1950.

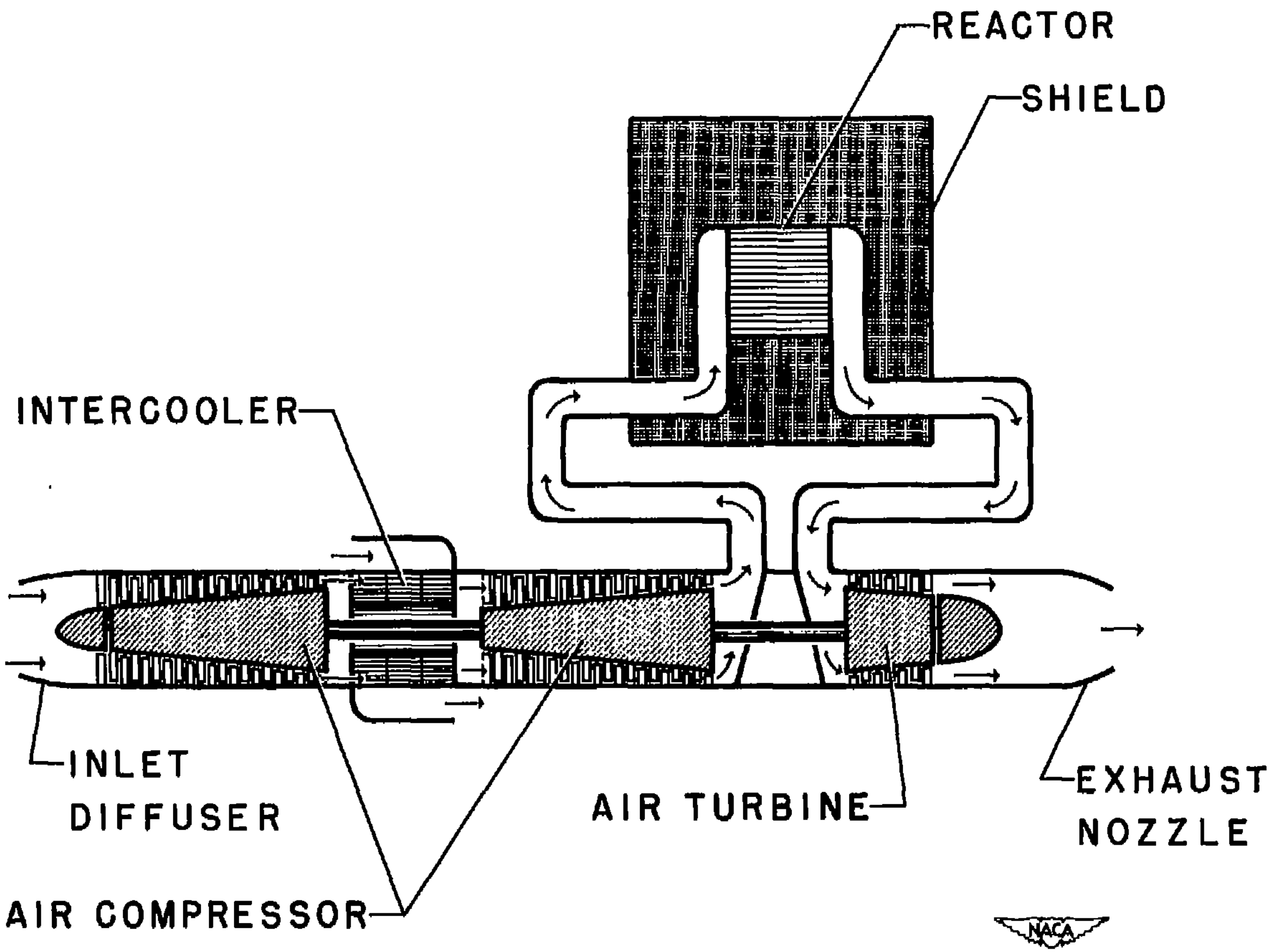

Figure 1. - Schematic diagram of direct-air cycle.

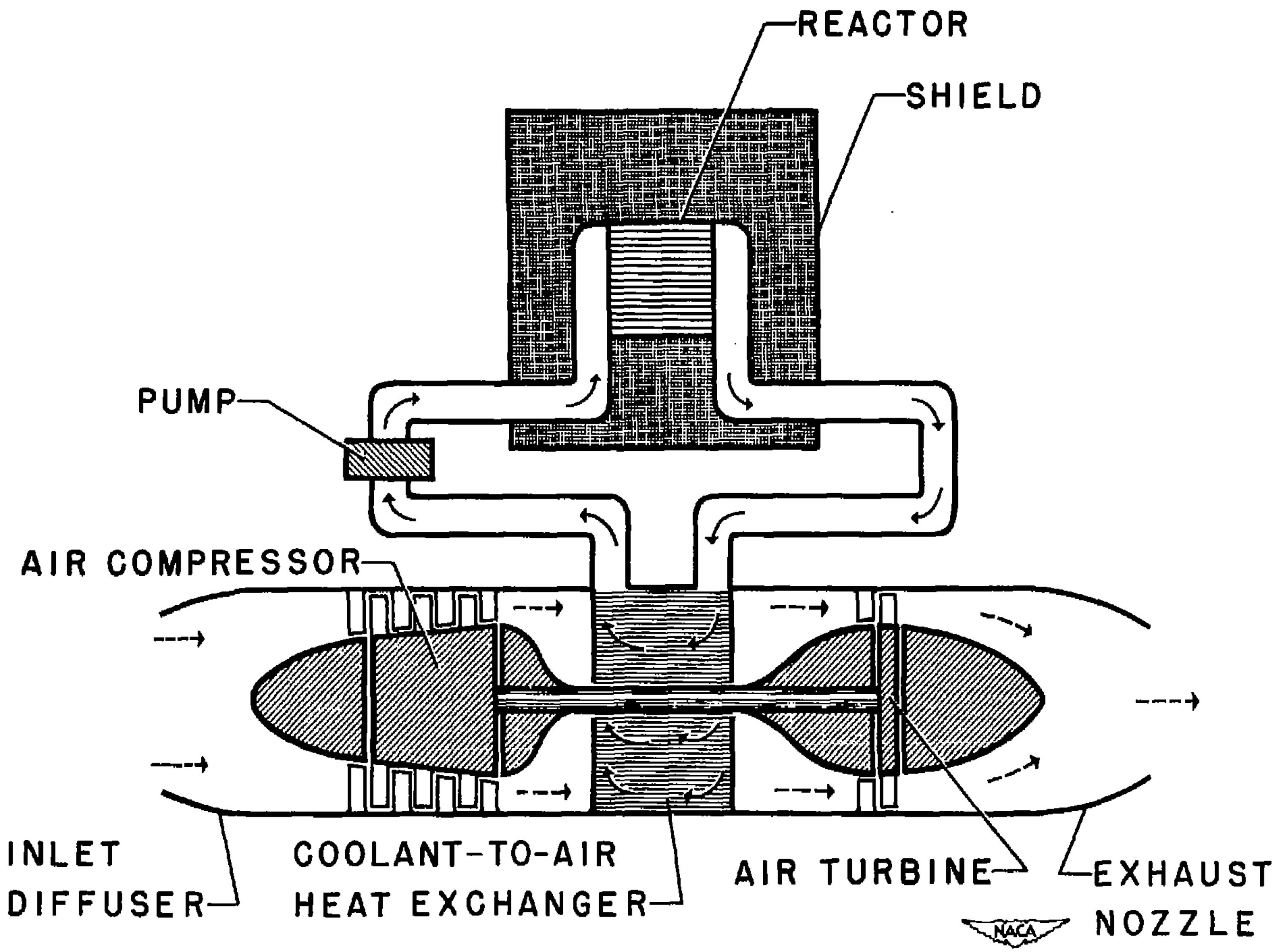

Figure 2. - Schematic diagram of liquid-metal cycle.

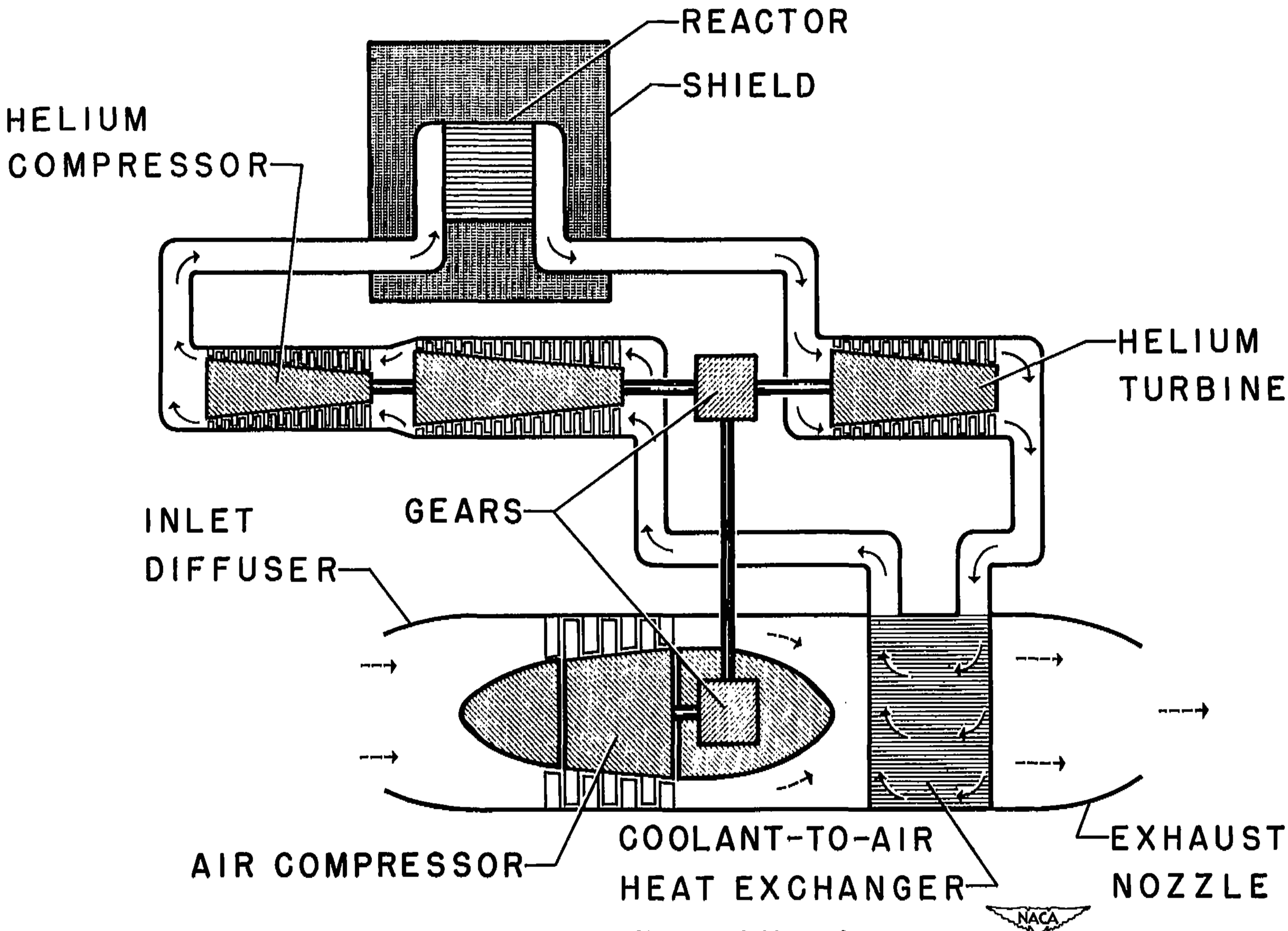

Figure 3. - Schematic diagram of helium cycle.

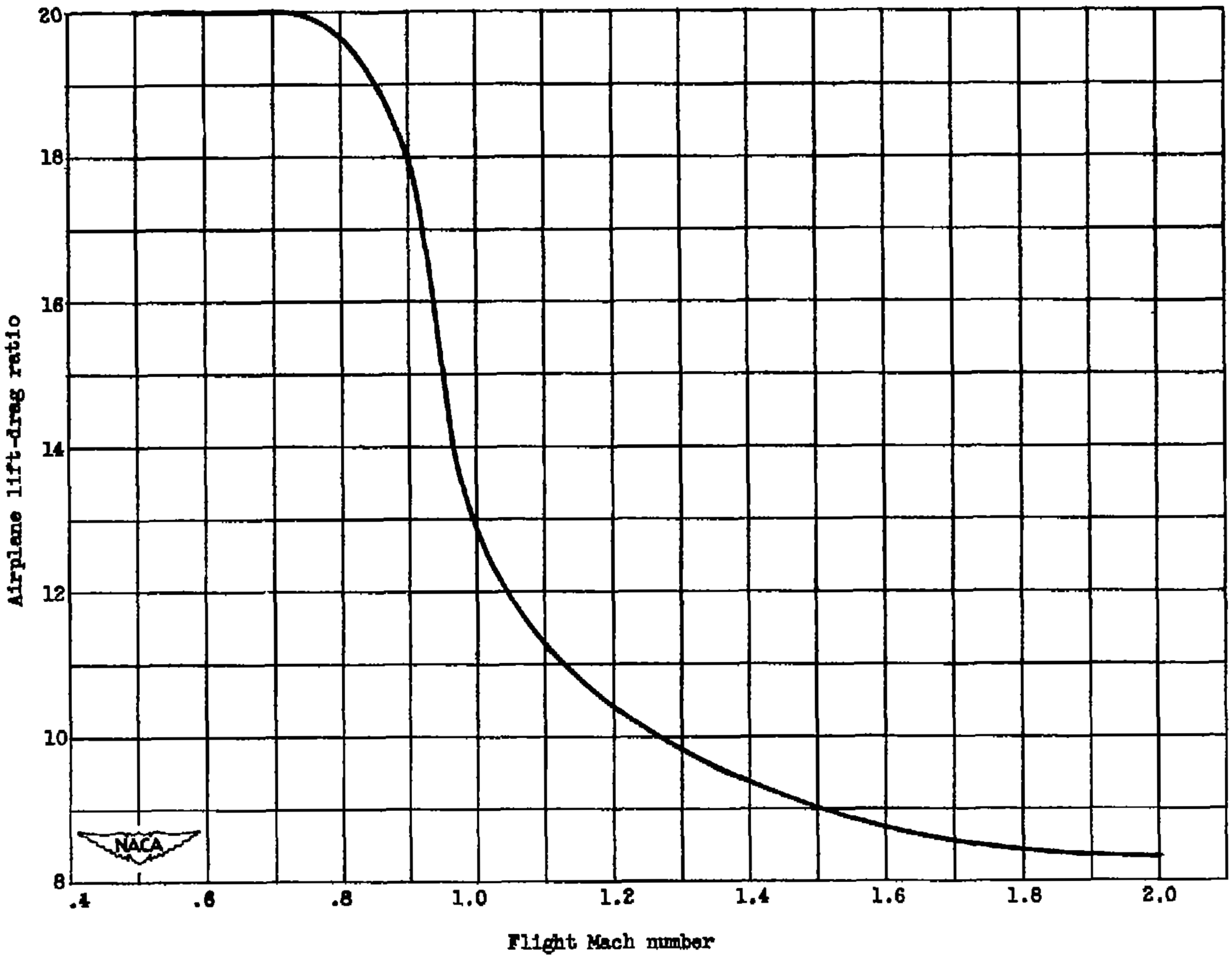

Figure 4. - Variation of airplane lift-drag ratio with flight Mach number.

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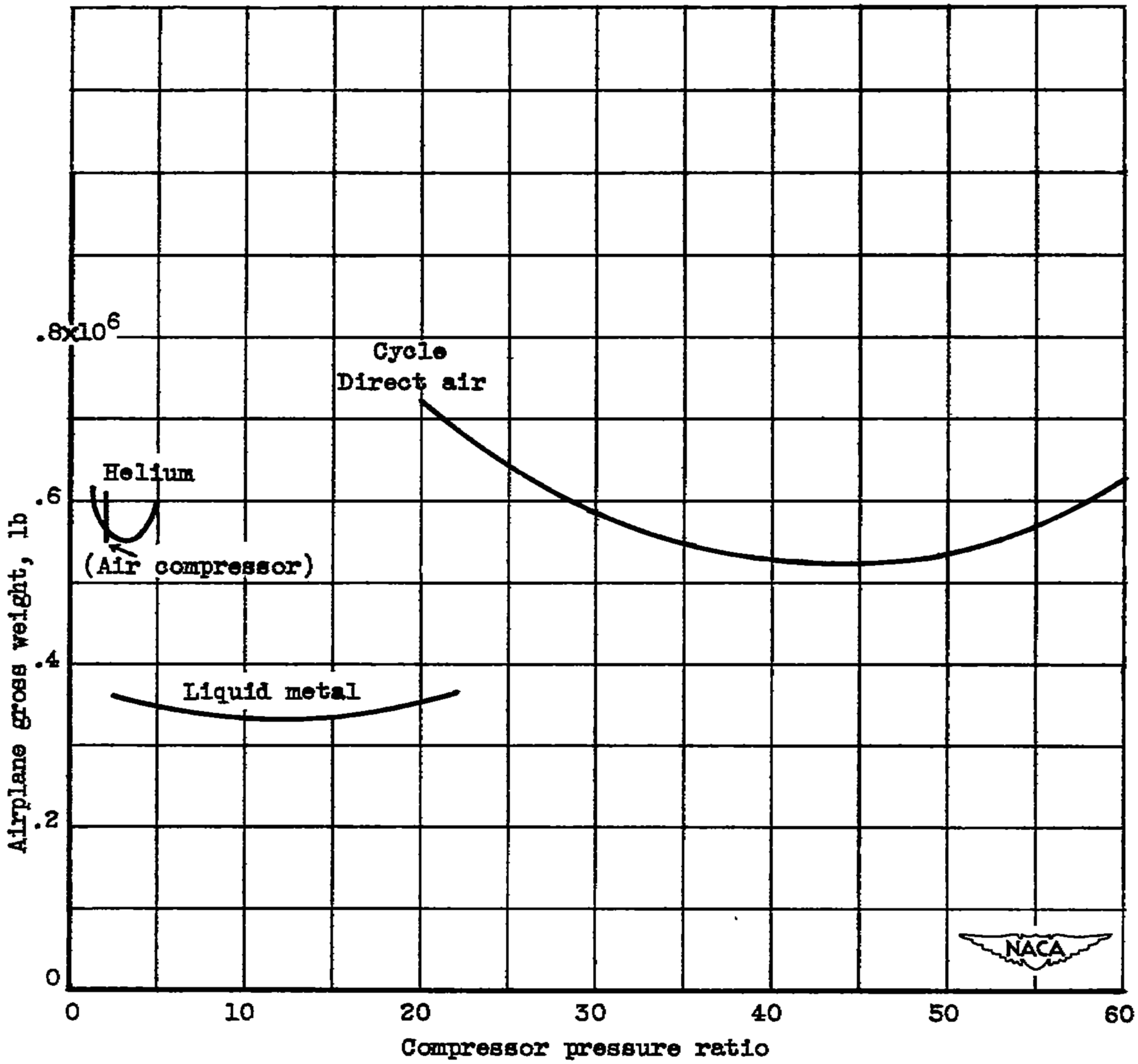


Figure 5. - Effect of compressor pressure ratio on airplane gross weight for three cycles. Flight Mach number, 0.9; altitude, 40,000 feet; reactor-wall temperature, 2300° R; turbine-inlet temperature, 2100° R; pay load, 20,000 pounds; reactor length-to-diameter ratio, 1.0; shield specific gravity, 8.0; shield thickness, 2.5 feet.

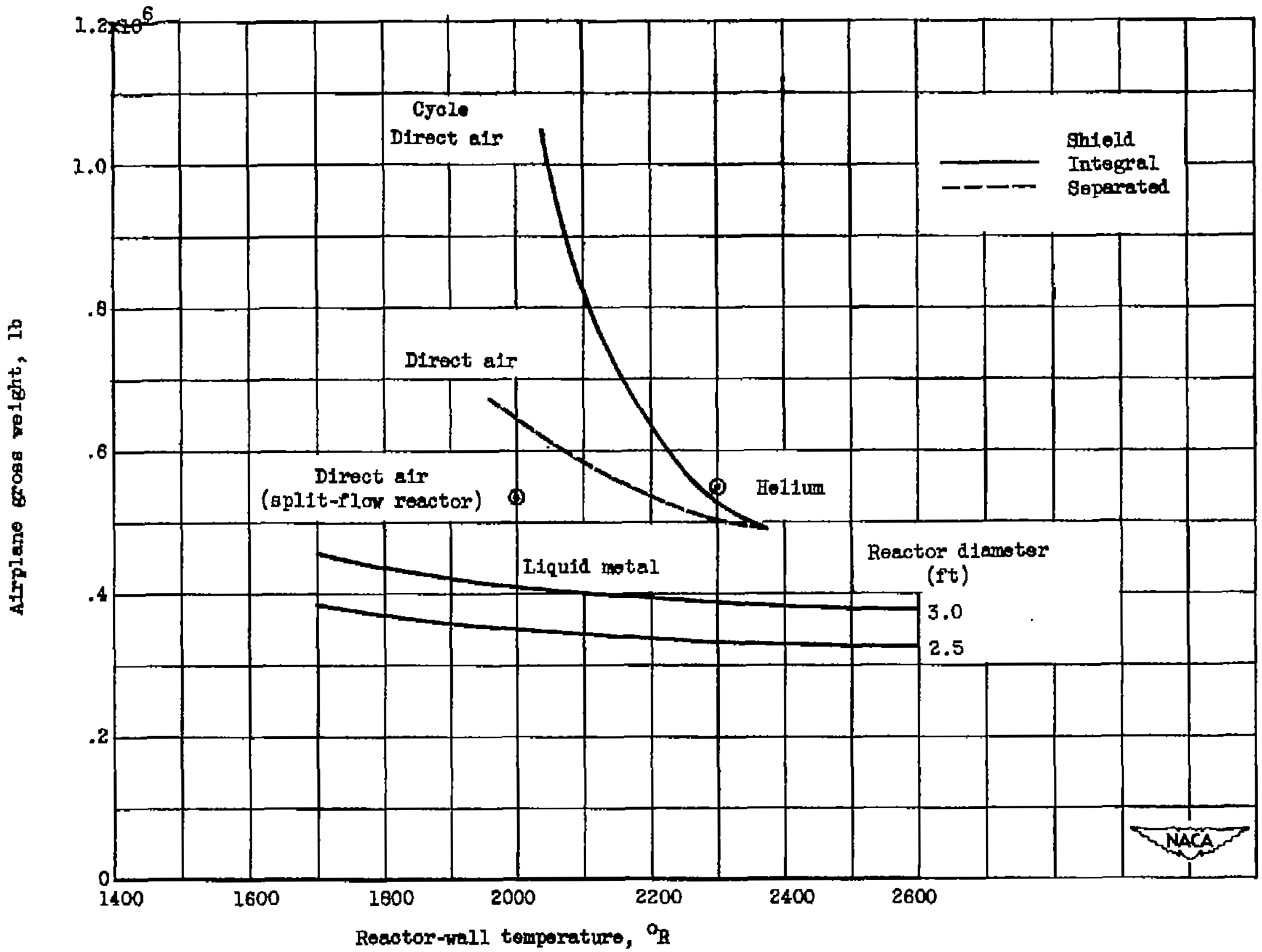


Figure 6. - Effect of reactor-wall temperature on airplane gross weight for three cycles. Flight Mach number, 0.9; altitude, 40,000 feet; reactor-wall temperature minus turbine-inlet temperature; 200° R; pay load, 20,000 pounds; integral-shield specific gravity, 8.0; integral-shield thickness, 2,5 feet.

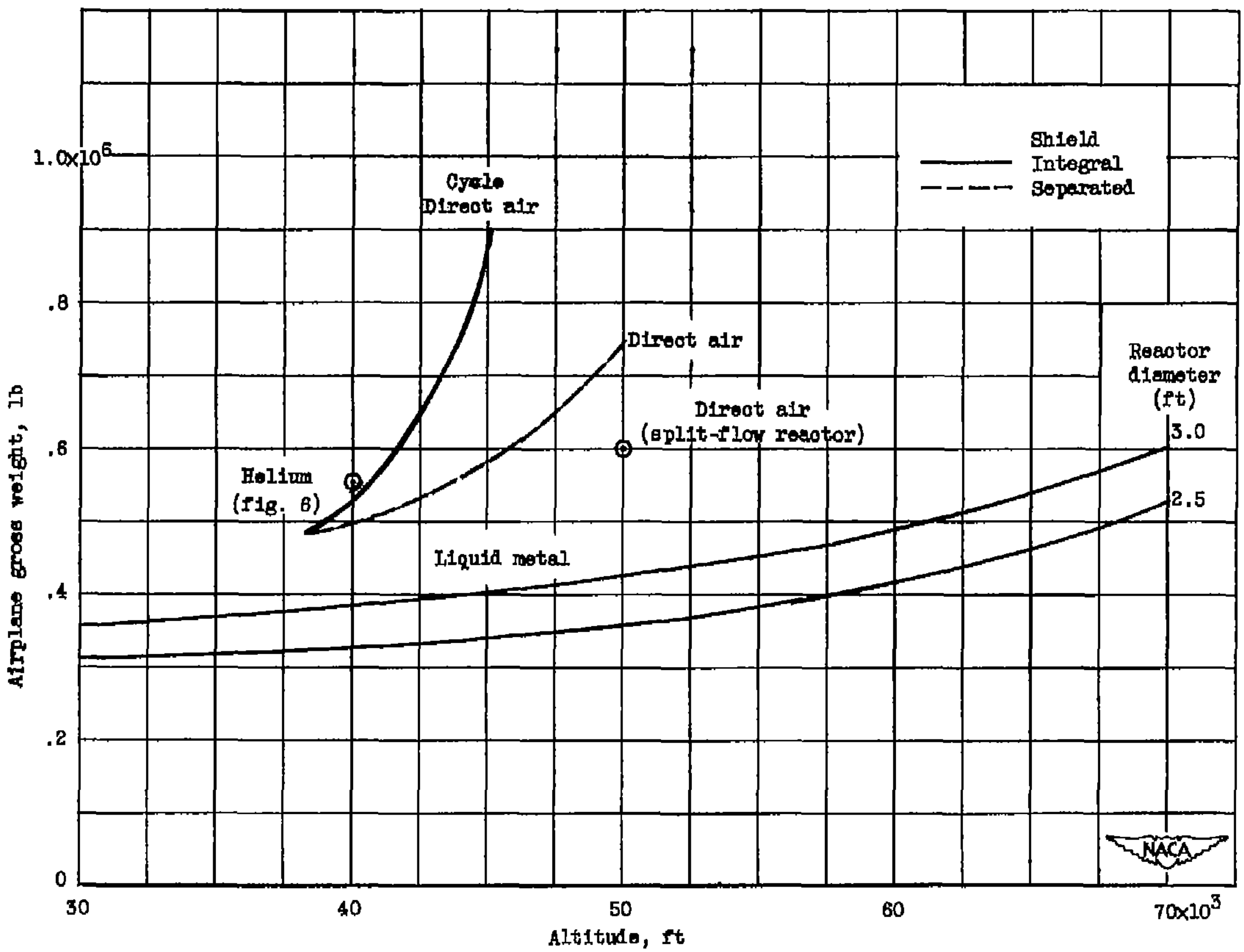


Figure 7. - Effect of altitude on airplane gross weight for three cycles. Flight Mach number, 0.9; reactor-wall temperature, 2300° R; turbine-inlet temperature, 2100° R; pay load, 20,000 pounds; integral-shield specific gravity, 8.0; integral-shield thickness, 2.5 feet.

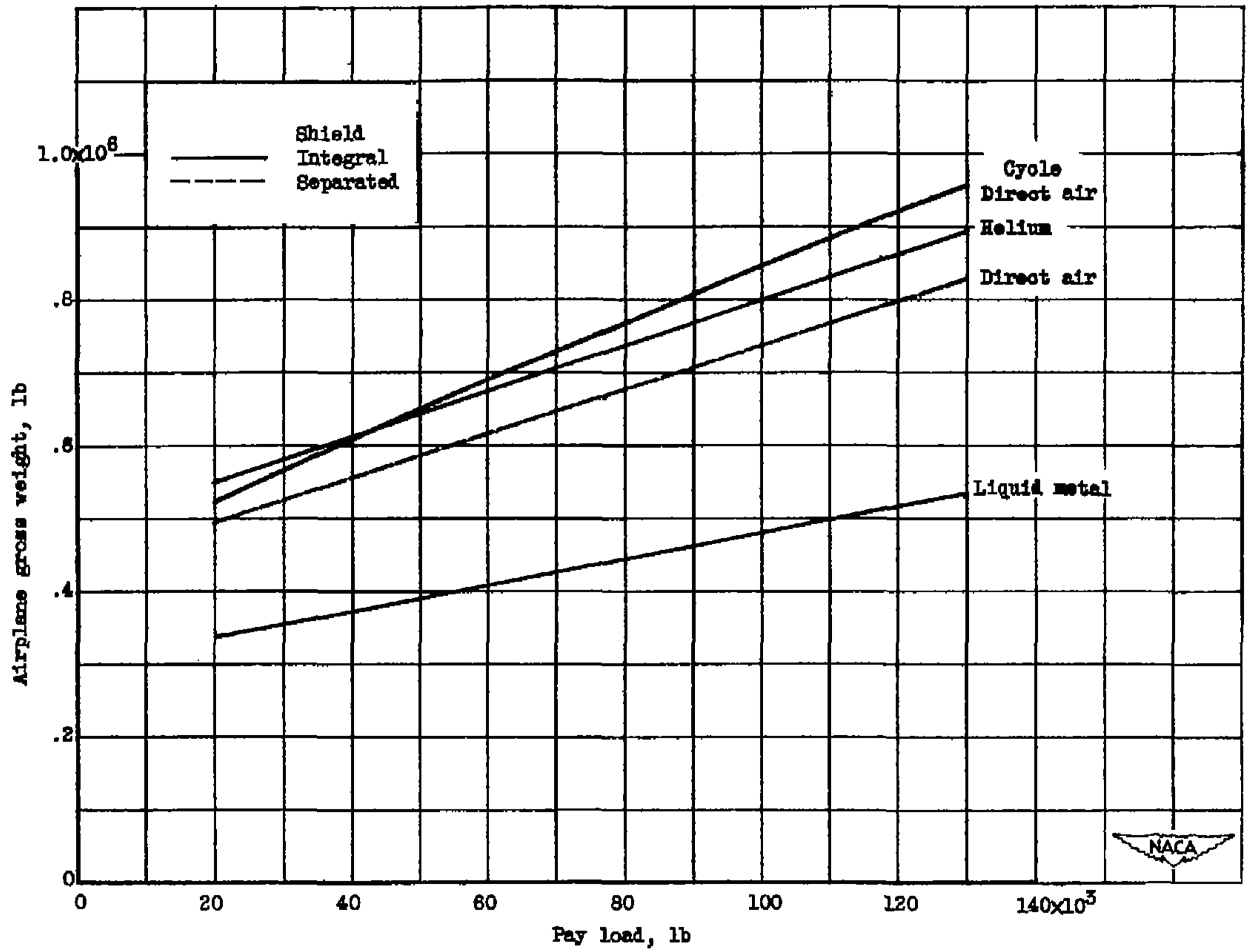
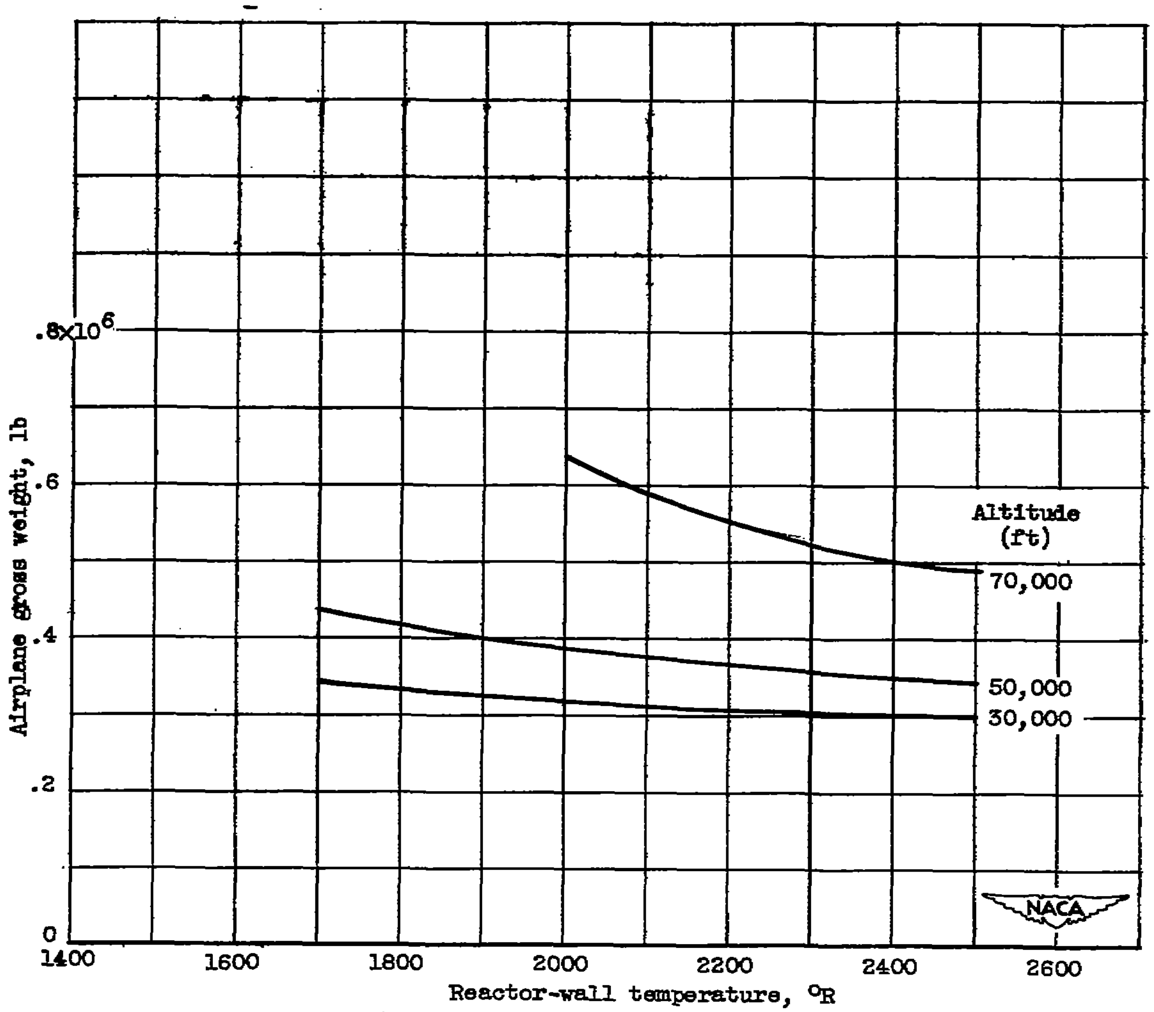
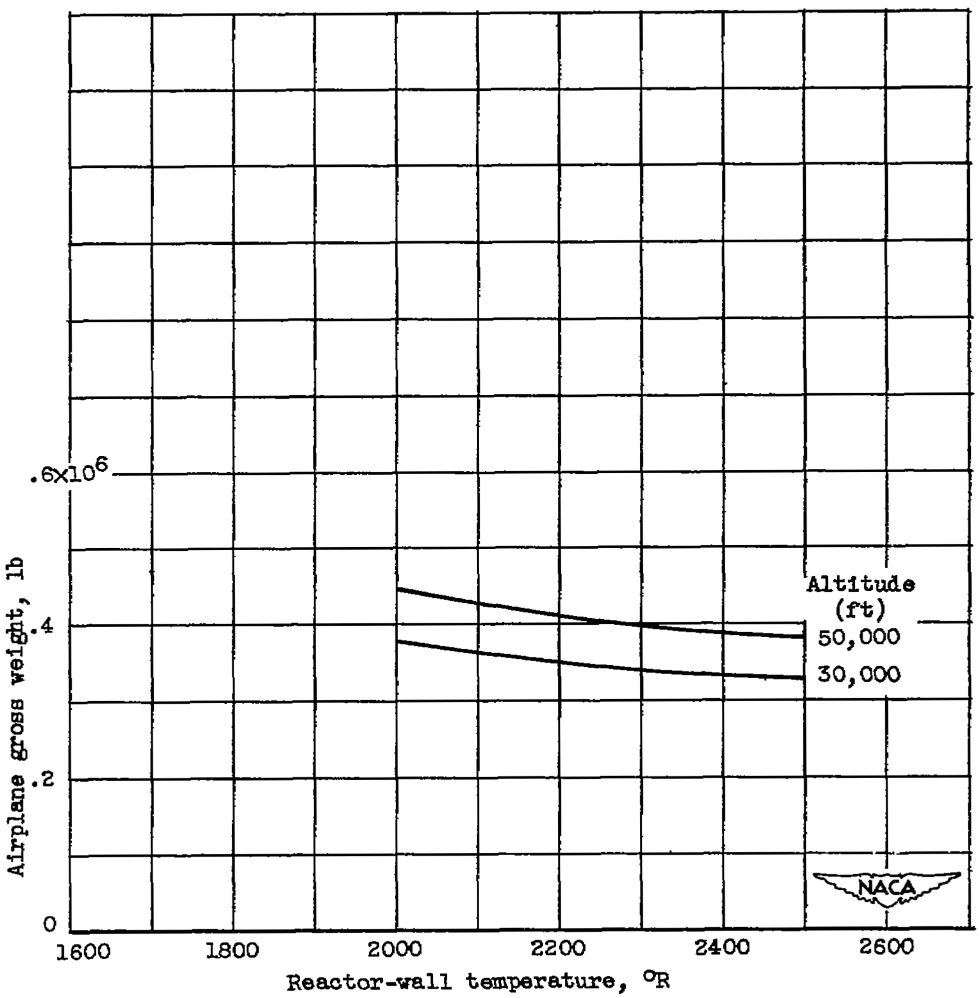


Figure 8. - Effect of pay load on airplane gross weight for three cycles. Flight Mach number, 0.9; altitude, 40,000 feet; reactor-wall temperature, 2300° R; turbine-inlet temperature, 2100° R; reactor length-to-diameter ratio, 1.0; integral-shield specific gravity, 8.0; integral-shield thickness, 2.5 feet.



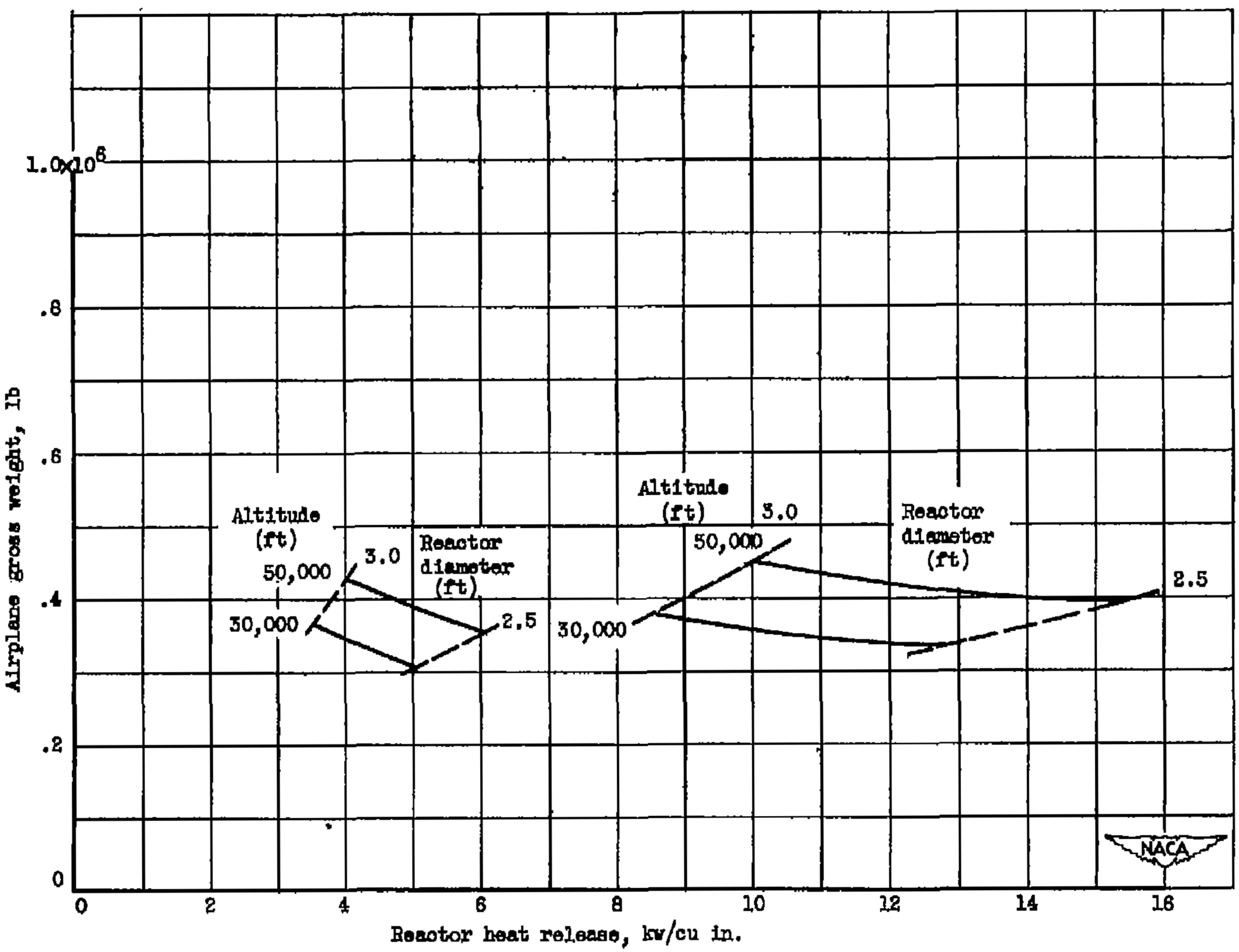
(a) Flight Mach number, 0.9.

Figure 9. - Effect of reactor-wall temperature on airplane gross weight at various altitudes for liquid-metal cycle. Reactor-wall temperature minus turbine-inlet temperature, 200° R; pay load, 20,000 pounds; reactor diameter, 2.5 feet; reactor length, 2.5 feet; shield specific gravity, 8.0; shield thickness, 2.5 feet.



(b) Flight Mach number, 1.5.

Figure 9. - Concluded. Effect of reactor-wall temperature on airplane gross weight at various altitudes for liquid-metal cycle. Reactor-wall temperature minus turbine-inlet temperature, 200° R; pay load, 20,000 pounds; reactor diameter, 2.5 feet; reactor length, 2.5 feet; shield specific gravity, 8.0; shield thickness, 2.5 feet.



(a) Flight Mach number, 0.9.

(b) Flight Mach number, 1.5.

Figure 10. - Effect of reactor heat release on airplane gross weight with the liquid-metal cycle for various flight Mach numbers, altitudes, and reactor diameters. Reactor-wall temperature, 2300° R; turbine-inlet temperature, 2100° R; pay load, 20,000 pounds; reactor length-to-diameter ratio, 1.0; shield specific gravity, 8.0; shield thickness, 2.5 feet.

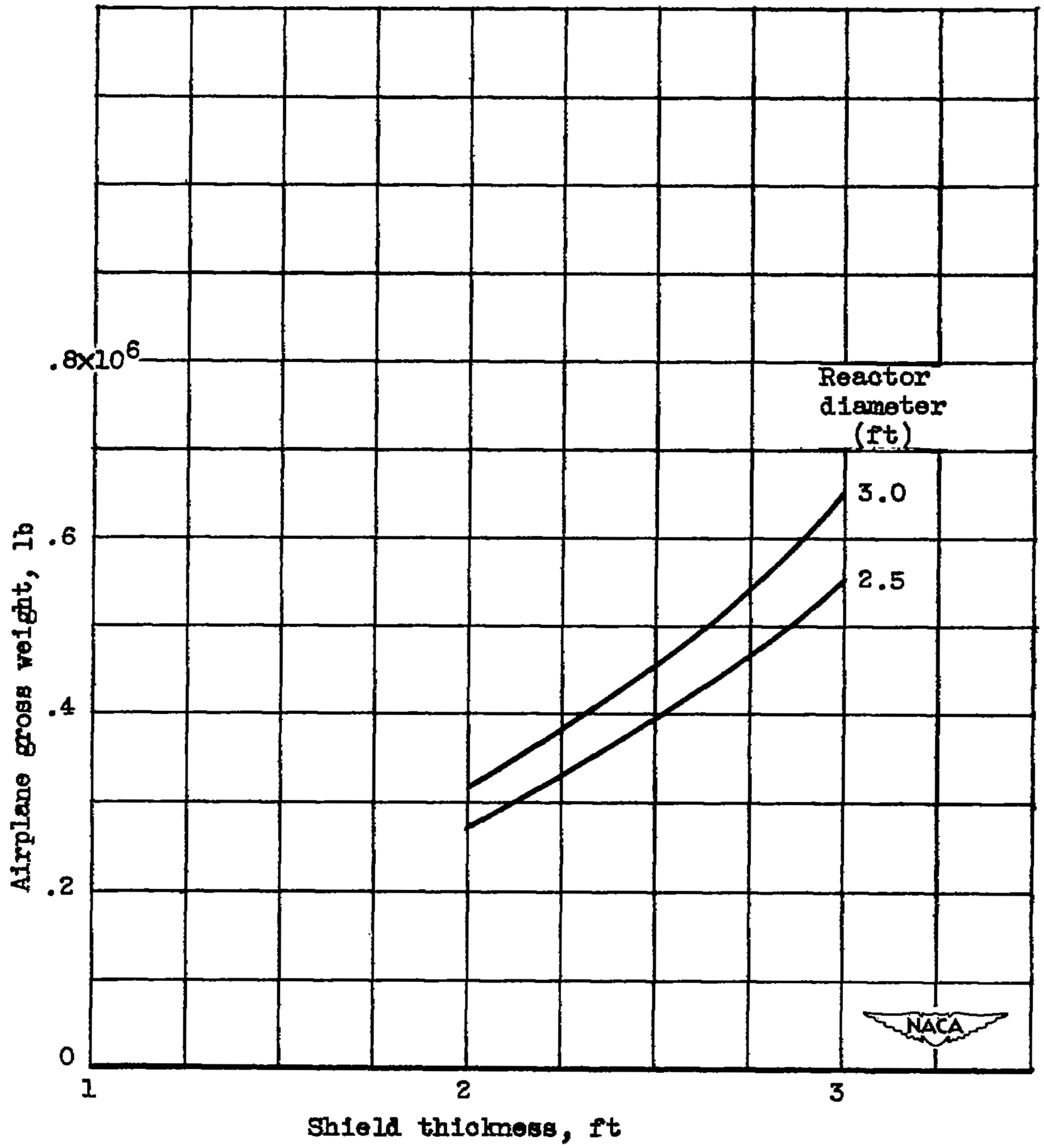


Figure 11. - Effect of shield thickness on airplane gross weight for liquid-metal cycle with reactor diameters of 2.5 and 3.0 feet. Flight Mach number, 1.5; altitude, 50,000 feet; reactor wall temperature, 2300° R; turbine-inlet temperature, 2100° R; pay load, 20,000 pounds; reactor length-to-diameter ratio, 1.0; shield specific gravity, 8.0.