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RESEARCH MEMORANDUM

for the

Air Research and Development Command, U. S. Air Force

RELATION OF TURBOJET PROPULSION SYSTEM DEVELOPMENT TO THE STRATEGIC BOMBER MISSION

By Addison M. Rothrock, Richard S. Cesaro, and Curtis L. Walker

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

A generalized analysis presents the effects of turbojet propulsion system development and fuel selection on ability of a strategic bomber to perform desired and minimum missions. The variation of bomber performance using a hydrocarbon, boron, or nuclear fuel is discussed. With chemical fuel, the effects of refueling are discussed. With nuclear fuel, the nuclear cruise-chemical dash bomber and the nuclear subsonic tug towing a chemically powered supersonic bomber are compared. The factors that determine bomber gross weight and bomber altitude are briefly discussed.

INTRODUCTION

In planning the defenses of the country, it is desirable to examine continually the technical likelihood of meeting operational requirements set by the strategic bomber mission. A technical evaluation should present the different methods by which these operational requirements can be met, or, if they cannot be met, offer a comparison of the alternatives that are currently technically possible. This report makes such an evaluation, and presents it in terms intended to be understandable to personnel not intimately familiar with the procedures of such analyses.

An examination is made of the interrelations between the basic variables that determine bomber range, gross weight, and flight altitude. Estimates are made of probable improvements that can be made in the propulsion factors, and of the resulting improvements in bomber range. To a certain extent, the current discussion is a continuation of that presented in reference 1.

*This analysis was made at the request of the U. S. Air Force and has been presented orally to various USAF groups.

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The data presented are based on generalized assumptions derived from current designs; therefore, the results do not represent specific design studies. For this reason, the information provides a generalized picture of the current technical feasibility of performing the strategic bomber mission and of the relative effectiveness of several approaches to the problem.

The range requirement of the strategic bomber is determined by the geographic distance from continental United States to Soviet industrial and military installation targets. Figure 1 presents data obtained from a presentation to the USAF by the Boeing Airplane Company, showing the percentage of the Soviet targets as a function of the great circle distance from Spokane, Washington or from Limestone, Maine. This distance is the "total radius" shown on the abscissa. The ordinate, supersonic dash radius, is the distance from a line 200 nautical miles outside the USSR to the specific target. This distance is presumably the flight distance under which the bomber could be under USSR land operated radar surveillance. The data show that if the bomber has a total radius of 5500 nautical miles, 100 percent coverage of the Soviet targets is obtained from the bases mentioned; and, furthermore, if 2000 of this 5500 nautical miles can be flown supersonicly, the supersonic dash will cover that part of the mission during which the bomber is subjected to ground operated enemy radar surveillance. This point on the graph, indicated by A, is the desired strategic bomber requirement as specified by the Air Force. At a total radius of 4000 nautical miles including 1000 nautical miles of supersonic dash, 50 percent of the Soviet targets could be covered. This second point, B, is the minimum requirement specified by the Air Force. The requirement is based on prestrike and poststrike portions of the flight being equal in regard to distance and speed.

The strategic bomber requirements will now be examined in relation to the propulsion system, based on what is currently believed to be technically feasible, and what may be feasible in the next 10 years.

FACTORS ESTABLISHING BOMBER RANGE

The radius of the bomber can be expressed by the Brequet equation, discussed in more detail in reference 1, in the form:

$$\text{Radius} = \frac{R}{2} = \frac{1}{2}\, h\eta_e \frac{L}{D}\, f\left(\frac{W_f}{W_{g,o}}\right) \tag{1a}$$

$$= \frac{1}{2}\, h\eta_e \frac{L}{D} \log_e \frac{1}{1 - \dfrac{W_f}{W_{g,o}}} \tag{1b}$$

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$$= \frac{1}{2}\, h\eta_e \frac{L}{D} \log_e \frac{1}{\dfrac{W_{af}}{W_{g,o}} + \dfrac{W_e}{W_{g,o}} + \dfrac{W_m}{W_{g,o}} + \dfrac{W_{f,Res}}{W_{g,o}}} \tag{1c}$$

in which

R — total range, i.e., twice the radius

h — heat of combustion of the fuel, expressed as the distance (nautical miles) over which unit thrust is produced if all the energy in unit weight of fuel is converted into thrust. For JP-4 fuel the value of h is 2400 nautical mile-pounds thrust per pound of fuel (18,500 Btu/lb)

η_e — over-all efficiency of the engine, expressed as work done (thrust X distance) divided by fuel energy consumed in doing this work. In relation to the more commonly used specific fuel consumption in terms of pounds of fuel per hour per pound of thrust, $\eta_e = \frac{V}{h \times sfc}$, where V is airplane velocity in knots and sfc is specific fuel consumption

L/D — lift-drag ratio of trimmed airplane in level flight, at the speed and altitude under consideration

$W_{g,o}$ — gross weight of airplane, start of cruise

W_e — weight of installed power plant

W_m — weight of military load (pilots, guidance system, armor, and armament)

W_f — fuel used during cruise

$W_{f,Res}$ — fuel remaining at end of cruise

W_{af} — weight of airframe, defined as $W_{g,o} - (W_e + W_m + W_{f,Res} + W_f)$

In equation (1a), the radius varies linearly with h, η_e, and L/D. Because of the logarithmic form of $f(W_f/W_{g,o})$, the variation of radius with $W_f/W_{g,o}$ is not linear (fig. 2). In this figure, two values of L/D are presented: a value of 20 representative of a bomber designed for subsonic flight, and a value of 5 representative of a bomber designed for flight at a speed of about $M = 2.0$. A value of $\eta_e = 0.20$ is used for both flight speeds; this choice is justified by the fact that engine efficiency is about the same for an engine at $M = 0.9$ without afterburner (or without afterburner operating) as for an engine at a flight speed of $M = 2.0$ with afterburner operating. 4221

A significant point of figure 2 is the rate of change of radius as $W_f/W_{g,o}$ is increased above 0.55 based on a value of η_e appropriate for chemical fuel.

The evolution of bombers since World War II may be traced by comparing their respective values of the parameters in the basic range equation. Table I gives the weight distribution of these bombers and, for information, gross weights and wing loadings (S is wing area, sq ft). In the time interval covered, ratio of airframe weight to gross weight has been cut about in half; this change accounts for an increase of about 0.22 in W_f/W_g (W_g is gross weight at take-off and W_f is total fuel load at take-off). Ratio of engine weight to gross weight has decreased from 0.18 to 0.10, allowing a further increase of 0.08 in W_f/W_g. Although the military load has varied in no regular manner, ratio of military load to gross weight has decreased from an average of about 0.20 to a value of about 0.08. As a cumulative result of these several decreases, ratio of fuel to gross weight has increased several fold.

Table II again lists the values of W_f/W_g for the bombers, together with the other variables given in equation (1a). In addition, the flight Mach numbers are listed. Dual performance is shown for the B-58; the values given are for all-subsonic or all-supersonic flight.

Since these bombers use conventional hydrocarbon fuels, the heat of combustion of the fuel is given as 2400 nautical mile-pounds of thrust per pound of fuel. The values of over-all engine efficiency show an increase as the reciprocating engine was developed, and then a decrease when the change was made to the turbojet engine. This lower efficiency at subsonic flight speeds is characteristic with the turbojet engine.

The values of airplane lift-drag ratio show a steady improvement through the B-52 airplane. Two values of L/D are shown for the B-58, again representing the subsonic and supersonic flight speeds. The subsonic value of L/D for the B-58 is low in comparison with that for the

B-52, mostly because of low aspect ratio. The value of L/D varies to a first approximation, as the square root of the aspect ratio, which is 8.6 for the B-56, and 2.1 for the B-58. This low aspect ratio is required structurally, because the wing must be thin for supersonic speed.

Allocation of reasonable quantities of fuel for take-off, climb, acceleration and reserve produces the values of flight radius given in the last column. Military specifications for these bombers may give radii somewhat different than those listed, depending on the particular mission. Comparing values for the B-24 through the B-52 indicates that increase in radius resulted for the most part from the increase in ratio of fuel weight to gross weight, the increase in L/D playing a secondary but important role.

For an airplane designed for supersonic flight, the low value of L/D for the subsonic portion of the flight considerably decreases the operational radius below that of the subsonic airplane. For all-supersonic flight, the further decrease in L/D further decreases the radius of action.

FACTORS ESTABLISHING BOMBER GROSS WEIGHT

Examination of the Brequet equation (1c) shows that the sum of the several ratios into which airplane weight has been distributed determines the ratio of fuel to gross weight, and so determines range. Weight of the military load is determined by the military mission to be performed. Once this value is set, the ratio $W_m/W_{g,o}$ can be decreased only by increasing airplane gross weight. Any decrease in the ratio $W_m/W_{g,o}$ so obtained can then be applied to increasing the value of $W_f/W_{g,o}$, and so increase range.

The relation of airplane gross weight to range for constant military load is shown in figure 3. The ratio of fuel weight to gross weight is increased by decreasing the ratio of $W_m/W_{g,o}$ from about 0.30 to 0.04. The curves show similar trends for the bomber and for the fighter.

Where range is of primary importance, the military load virtually determines the gross weight of the airplane. Figure 3 shows that for a reasonable range compromise, gross weight can be from 12.5 to 20 times the military load. Based on these figures and military loads for strategic bombers, to follow the B-58, varying from 25,000 to 40,000 pounds, bomber weight will be between 310,000 and 800,000 pounds.

FACTORS ESTABLISHING BOMBER ALTITUDE

The altitude at which the mission is to be flown, at a given Mach number, is determined when the following equation is satisfied:

$$\frac{W_e/W_g}{W_e/F} \times \frac{L}{D} = 1 \tag{2}$$

in which F is the engine net thrust produced at the flight altitude. 4221
Since under the condition of level flight the lift of the airplane is equal to the gross weight, the equation simply states that the thrust produced is equal to the airplane drag. As the airplane design altitude is increased, to fly at maximum L/D, wing area is increased in proportion to the decrease in ambient air density. This increase results in the fuselage area becoming a smaller proportion of the total area and consequently in increasing the L/D (fig. 4) (constructed from ref. 3, p. 23, fig. 21).

As altitude increases, the thrust produced decreases in proportion to the decrease in air density, if Reynolds number effects and effects of altitude on combustion efficiency are neglected. The required value of the engine weight to gross weight ratio varies, therefore, as $\frac{L/D}{\rho}$, where ρ is the density of the ambient air. The variation of ratio of engine weight to gross weight with change in altitude (neglecting the L/D effect) can be computed with considerable accuracy. A typical curve is shown in figure 5, for a flight speed of about M = 2.0.

The ratio of $W_{af}/W_{g,o}$ increases with increasing design altitude because of the larger wings required. The increase of this ratio with altitude cannot be estimated with the accuracy of the $W_e/W_{g,o}$ relation, because sufficient design studies have not been made, and the variation does not lend itself to precise mathematical treatment. An estimated curve is presented in figure 5. As altitude increases, the increases in $W_e/W_{g,o}$ and $W_{af}/W_{g,o}$ result in a decrease in $W_f/W_{g,o}$ as shown in the figure. The over-all effect of change in design altitude on range (fig. 5) results from the increase in L/D with design altitude (fig. 4) and the decrease in ratio of fuel weight to gross weight (fig. 5). The precise values indicated by the curve are not important, but the general shape of the curve is. It is seen that as design altitude increases, range passes through a maximum. With airplanes of current design, this maximum occurs at a flight altitude of about 55,000 to 60,000 feet, with an altitude at target of 7,500 to 10,000 feet higher. As engine specific weight and airframe weight in relation to gross weight are decreased, this maximum occurs at successively increasing altitudes.

IMPROVEMENTS IN RANGE FACTORS

The factors in equation (1b) will now be examined in relation to improvements that can be foreseen in the chemically powered bomber.

Ratio of fuel weight to gross weight. - The ratio of fuel weight to gross weight will be increased as technical progress permits the ratios of airframe weight and engine weight to gross weight to be decreased, except as these decreases are used to increase altitude. As shown in table I, considerable progress has been made in decreasing these ratios; it is difficult to estimate the additional progress that will be made. With current values for $W_{af}/W_{g,o}$ of about 0.23 and for $W_e/W_{g,o}$ of about 0.10, a decrease of 10 percent in each would increase the value of $W_f/W_{g,o}$ from 0.58 to 0.61 giving an increase in range of 8 percent.

Fuel heat of combustion. - A convenient means of presenting the fuel picture in relation to heat of combustion is to plot the heat of combustion of the elements as a function of their atomic numbers (fig. 6). Heats of combustion in excess of the current value of JP-4, 2400 nautical-mile pounds per pound (approximately 18,500 Btu/lb), can be obtained by substituting lithium, beryllium, or boron for the carbon of hydrocarbons, or by eliminating these elements entirely and using hydrogen.

Lithium is not enough better than carbon to be of much interest. Beryllium is much rarer, and is more toxic than boron, which leaves the boron-hydrides of major interest. Pentaborane (B_5H_9) has a heat of combustion of 29,000 Btu per pound. The development of the boron fuels under the code name of Zip is being actively sponsored by the Department of Defense. A fuel consisting of a combination of boron-hydride and hydrocarbon with an estimated heat of combustion of 25,000 to 26,000 Btu is being produced in laboratory quantities. If this fuel can be used in place of JP-4, a range increase of 40 percent (26,000/18,500 = 1.40) will be realized. However, the combustion products, boron oxide, tend to deposit as a solid in the combustor and on the turbine stator blades; intensive research is needed on this problem. Without going into details, current research indicates that combustion of Zip fuel in the afterburner causes less trouble than combustion in the primary combustor. Use of Zip fuel in the afterburner only will increase range about 25 percent, for that portion of the flight in which the afterburner is used.

Research and development on Zip fuel is currently limited in scope because of the small quantities of the fuel that have been available. Based on present recommedndations of the Department of the Navy and Department of the Air Force, sufficient fuel should be available in about 2 years to permit an adequate attack on the problem of exhaust product deposits. In the mean time, interesting laboratory results on full scale engines are being obtained with the limited fuel quantities now available.

Hydrogen as a fuel would give a heat of combustion 2.75 times that of JP-4, and would present no major engine operation problems. In fact, because of its combustibility, hydrogen has good combustion efficiency at altitudes much in excess of those currently being used. The principal disadvantage of hydrogen is its low density; in liquid form hydrogen is only one-tenth as dense as JP-4, and extremely low temperatures are required to maintain the liquid form. In comparison with a quantity of JP-4 of given energy content, an equivalent amount of hydrogen would weigh 0.4 as much, but would occupy four times the volume. This lower density, with consequent larger fuel tanks, has led to consideration of hydrogen primarily for altitudes above 70,000 feet and generally for radii of action less than that required for the strategic bomber mission. Current interest in this fuel is for flights at considerably higher altitudes, with particular emphasis on the reconnaissance mission. The Department of the Air Force, in conjunction with the NACA, is conducting an accelerated program on the use of hydrogen. Use of hydrogen is discussed more fully in references 2 and 3.

Summarizing the chemical fuel picture: Zip fuels may well increase potential range of the strategic bomber by 25 percent of the portion of the flight in which an afterburner is used. A potential range increase of 40 percent will be realized if the boron oxide deposit problem is solved, permitting full use of Zip fuel. The low density of hydrogen makes it of current interest as a fuel to be used at quite high altitudes; decision on its use as a long-range fuel must await additional research.

If the strategic bomber is powered with nuclear fuel instead of chemical fuel, the value of h becomes many orders of magnitude greater than that for chemical fuels. In this case, because range is sufficiently greater than that required by the bomber mission, other factors, notably nuclear radiation effects on the crew, determine the time the airplane can stay in the air and so determine the radius.

Engine efficiency. - Over-all efficiency of the turbojet engine is primarily a function of pressure ratio of the engine, combustion temperatures (i.e., turbine-inlet temperature and afterburner temperature, if an afterburner is used), and airplane flight speed. In the present discussion, the effect of pressure ratio will not be considered. In figure 7, engine efficiency is shown as a function of combustion temperatures and airplane speed. In the figure, T_4 is the turbine-inlet temperature. These data are taken from reference 1, which contains a discussion of the assumptions used in determining the curves. The data show that engine efficiency can be increased effectively either by increasing flight speed or by eliminating the afterburner.

Figure 8 shows the effect of turbine-inlet temperature on relative specific engine weight, at an altitude of 35,000 feet, at different

airplane flight speeds. As flight speed is increased, specific weight of the engine decreases. As turbine-inlet temperature is increased from the current value of 1540° F to a value of 2040° F, specific weight of the nonafterburner engine becomes appreciably closer to that of the afterburner engine. The data presented in figure 8 typify the amount of specific weight improvement that can be made in the nonafterburner engine through increasing turbine-inlet temperature. Specific engine designs will deviate from relative values shown, but the general trend will remain. Research to permit higher turbine-inlet temperatures
4221 through use of proper materials or through turbine cooling is progressing, and an upper limit of 2000° F can be considered as technically feasible. As specific engine weights are reduced through reduction in amount of metal in the engine, increase in rate of air flow through the engine, and increase in turbine-inlet temperature, the generic differences in engine weight and in engine frontal area between afterburner and nonafterburner engines will be reduced.

Airplane lift-drag ratio. - The last factor to be discussed in the Brequet equation is the lift-drag ratio of the airplane. Current values and future trends are more difficult to assess for L/D than for fuel heat of combustion or engine efficiency.

A representative curve of the effect of airplane speed on maximum lift-drag ratio is shown in figure 9. The curve for the all-subsonic airplane shows a peak value of 22, which corresponds to the value for the B-52, given in table II. The curve for the subsonic cruise-supersonic dash airplane shows a value of 14 at subsonic speeds and a value of about 5.5 above $M = 2.0$. The specific values will vary as design altitude of the airplane is varied (fig. 4) and as configuration of the airplane is varied. No particular brief is held for the curves as drawn, except that the values are representative of current practice.

For a given airplane, lift-drag ratio is a function of the altitude at which the airplane is flying and of wing-loading, as well as of airplane speed. Figure 10 illustrates this variation. The maximum values of these curves correspond to the maximum lift-drag ratios as given in figure 9. In calculations to be presented later, it is assumed that the airplane is flying at maximum lift-drag ratio. The effect of flight speed on the altitude at which maximum L/D occurs is important. These curves indicate the extent to which reduction in wing loading increases altitude for maximum L/D. Figure 10, therefore, illustrates the reason for following the Brequet cruise-climb; as fuel is consumed and the wing loading reduced, altitude is increased to maintain maximum L/D.

STRATEGIC BOMBER PERFORMANCE

Performance estimates for strategic bombers are now discussed, utilizing the previously developed values of the factors in the Brequet range equation. This performance is discussed first for JP-4 fuel and later for Zip fuel. The subsonic radius (M = 0.9) indicated in figure 11 by point C is determined by using the following values in equation (1b):

$$h = 2400 \text{ nautical mile-pounds per pound of fuel (JP-4)}$$

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$$\eta_e = 0.20 \text{ (afterburner not operating)}$$

$$L/D = 14$$

$$\frac{W_f}{W_{g,o}} = 0.60$$

Point D, assumed to be at M = 2.0 is next determined for an all-supersonic mission (total radius equal to supersonic dash radius) with the same airplane. In this case, the values used are

$$h = 2400 \text{ nautical mile-pounds per pound of fuel (JP-4)}$$

$$\eta_e = 0.22 \text{ (afterburner operating)}$$

$$L/D = 5.5$$

$$\frac{W_f}{W_{g,o}} = 0.60$$

Weight distribution in the bomber of the above example might be as follows:

$$\frac{W_{af}}{W_{g,o}} = 0.225$$

$$\frac{W_{e}}{W_{g,o}} = 0.100$$

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$$\frac{W_{m}}{W_{g,o}} = 0.050$$

$$\frac{W_{f}}{W_{g,o}} = 0.600$$

$$\frac{W_{f,Res}}{W_{g,o}} = \underline{0.025}$$

$$\text{Total} \quad W_{g,o} = 1.000$$

The all-subsonic radius is shown as 3100 nautical miles and the all-supersonic radius as 1300 nautical miles. For an analysis of this type, the points C and D can be joined by a straight line. The line C - D then represents the trade-off between total radius and supersonic dash for the airplane. The distance covered during the climb and acceleration from the subsonic cruise condition to the supersonic dash condition is included in the ordinate value. Point E, for instance, represents a total radius of 2000 nautical miles of which 800 is at a supersonic speed of $M = 2.0$. Reference 4 presents a detailed discussion of the distance covered during the accleration and climb. Figure 11 further shows that current technical capability without refueling or floating wing tips (line C - D) does not meet the minimum (point B) or desired (point A) strategic bomber requirement.

In figure 12, curve C - D of figure 11 has been reproduced, and the several values of engine efficiency, airplane lift-drag ratio, and fuel to gross weight ratio used in determining the two ends of different curves are indicated. For the line F - G, point G is computed for a ratio of fuel to gross weight 10 percent (0.06) higher than the value used for point C. The equivalent increase in L/D for the same change in radius is from a value of 14.0 to a value of 16.5. An equivalent increase in engine efficiency would be from 0.200 to 0.235. Similar computations are made for the all-supersonic dash (point F) and the two points are joined by the dashed line F - G.

As discussed previously, engine efficiency in the supersonic region can be improved by eliminating the afterburner or by increasing flight velocity. A new value of engine efficiency of 0.38 is chosen representing approximately the value obtained either by eliminating the afterburner at a flight speed of $M = 2.0$ or by increasing the flight speed to $M = 3.0$ with the afterburner operating. In computing the all supersonic radius (point I) with this value of engine efficiency, an L/D decrease from 5.5 to 5.0 or 6.5 to 6.0 is assumed, to allow for additional drag resulting from the larger frontal area of the nonafterburner engine or from the higher flight speed. The line I - G represents about the most optimistic range improvement that can currently be considered with hydrocarbon fuel. 4221

As the slope of the line through G (fig. 11) is increased, more consideration can be given to all-supersonic flight. For example, points H and I in the figure give the same total radius of action but represent different flight paths, as shown in figure 13. In this comparison, the all-supersonic flight with a nonafterburner engine is based on no decrease in fuel weight to account for the greater weight of engine required for the nonafterburner engine (or for the afterburner engine designed for the higher dash Mach number). Therefore, the all-supersonic airplane, point I (fig. 12), with nonafterburner engines is designed for a lower altitude than is the subsonic cruise-supersonic dash airplane, point H. As the difference between the supersonic dash-radii for the two cases is decreased, the difference in altitude between the airplane with the nonafterburner engines and the airplane with the afterburner engines is decreased. The question of all-supersonic flight is discussed in more detail in reference 5.

Bomber range with one refueling at 2500 nautical miles is indicated in figure 14. For convenience, the bomber performance indicated in figure 12 is repeated. The line J - K represents the refueled performance, with refueling just before the start of the supersonic dash (prestrike refueling) or at the end of the dash (poststrike refueling). The airplane is considerably lighter at the target with poststrike refueling than with prestrike refueling. The on-target altitude is, therefore, about 7,500 to 10,000 feet higher with poststrike refueling than with prestrike refueling. Figure 14 shows that with refueling at 2500 nautical miles, current feasible bomber performance includes the minimum strategic bomber requirement. Based on the discussion of figure 12, the area in which the minimum requirement now falls is considered to be optimistic bomber performance.

Floating wing-tip tanks that are currently being considered will give approximately the same bomber radius increase (900 to 1250 nautical miles) as obtained with one refueling at 2500 nautical miles. The floating wing-tip tanks increase the subsonic cruise lift-drag ratio because of the increase in aspect ratio, and also increase the ratio of fuel to gross weight.

The ratio of tanker-gross weight to bomber gross weight for a prestrike refueling is as shown in figure 15. The turboprop tanker has lower gross weight because of higher engine efficiency. In figure 16 comparative curves are shown for poststrike and for prestrike refueling. Each curve represents a tanker weight midway between that of a turboprop and that of a turbojet tanker. For a refueling at 2500 nautical miles, tanker weight is about equal to bomber weight for prestrike refueling and about half the bomber weight for poststrike refueling. If the tanker has a ratio of fuel to gross weight of 0.55 (instead of the value 0.65 used in fig. 16), the ratio of tanker to bomber weight is about 25 percent greater than the values shown.

For the data presented in figure 16, weight of the bomber is considered constant and a series of tankers is assumed. In practice, the military load and the gross weight of the tanker may be fixed; then, as the refueling takes place at successively greater distances, the fuel to be transferred to the bomber is successively decreased. Successively smaller bombers are then assumed, since the transferred fuel must equal the fuel to be consumed by the bomber from the refueling point to its home base. With constant military load, this decrease in bomber size means the ratio of fuel weight to gross weight decreases, with a subsequent decrease in range following the refueling. Figure 17 shows that the increase in refueling distance just about offsets the decrease in bomber range. Therefore, it is concluded that if weights of the tanker and of the bomber military load are fixed, total radius of operation is more or less independent of bomber gross weight, although the refueling distance increases as bomber gross weight is decreased. The figure also shows the effect of the ratio of fuel weight to tanker gross weight $W_f/W_{g,t}$, on the total radius.

Figure 18 shows that with both prestrike and poststrike refuelings at 3000 nautical miles, the desired strategic bomber requirement is just included in the optimistic area of estimated technical feasiblity. Figure 18 indicates the difficulty of meeting this requirement with JP-4 fuel even considering two refuelings.

Hydrocarbon fuel plus Zip fuel. - As mentioned previously, there is reasonable expectancy that Zip fuel can be burned satisfactorily in afterburners. The effective heat of combustion of the combination fuel for that period of operation in which the afterburner is used is then increased about 25 percent; the effective value of h becomes 3000 nautical mile-pounds per pound of fuel. Figure 18 is redrawn (fig. 19) with a 25-percent increase in the supersonic dash radius; the use of the nonafterburner engines is not considered. Comparison of figure 19 with figure 18 shows that use of Zip in the afterburner has slightly improved range potential.

Zip fuel. - The results obtained if Zip can be burned throughout the engine are shown in figure 20. For this case, because Zip is used ahead of the turbine as well as in the afterburner, the nonafterburner engine can be considered for the supersonic dash. The situation is now greatly improved. Slightly more than the minimum strategic bomber requirement is achieved with one refueling at 2500 nautical miles. The requirement may be achieved without refueling, if optimistic bomber performance is assumed. The desired performance, point A, falls within the area of estimated technical feasibility with two refuelings.

Chemical fuel summary. - Airplane operational data for the chemical fuels are summarized in table III. For each bracket, the upper tabulation is the performance using the more conservative values of fuel to gross weight ratio, engine efficiency, and airplane lift-drag ratio. The lower tabulation represents the optimistic values for these same variables. The table clearly brings out the value of using Zip fuel throughout the engine.

NUCLEAR FUEL

All nuclear power. - With nuclear fuel, the heat of combustion is sufficiently great so that radii in excess of 10,000 miles are feasible as far as the term $h\eta_e$ is concerned (eq. (1)). The determining factor is the speed and altitude at which the airplane can fly. If the flight is all under nuclear power (referred to as all nuclear), except for a chemical powered take-off and landing, the weight distribution of the airplane may be about as follows:

$$\frac{W_{af}}{W_{g,o}} = 0.275$$

$$\frac{W_e}{W_{g,o}} = 0.600$$

$$\frac{W_m}{W_{g,o}} = 0.050$$

$$\frac{W_{f,chem}}{W_{g,o}} = \underline{0.075}$$

$$\text{Total} \quad W_{g,o} = 1.000$$

In this case, for a 30,000 pound military load, gross weight of the airplane is 600,000 pounds.

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The question becomes one of determining, within the nuclear propulsion system, specific weight limitations and airplane lift-drag ratio limitations, the altitude and flight speeds that can be realized. In this case, equation (2) applies. With current design weights of nuclear propulsion systems, the feasible altitude at subsonic speeds is in the order of 40,000 feet. Supersonic flight on all nuclear power does not at this time appear feasible, although modest improvements in either the propulsion system specific weight or in the airplane lift-drag ratios can change this picture.

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To achieve higher combat altitudes or flight speeds, chemical power is considered in conjunction with nuclear power. Two systems are currently being considered: (1) the subsonic nuclear cruise-supersonic chemical dash airplane and (2) a nuclear subsonic tug airplane towing a chemically powered supersonic airplane.

Nuclear cruise - chemical dash. - The airplane with nuclear cruise and chemical dash has a radius in excess of 10,000 miles. By cruising at an altitude of 20,000 to 30,000 feet the ratio of engine weight to gross weight is reduced to about half the value of 0.60 considered in the all-nuclear powered plane. The 0.30 of the gross weight so saved can be carried as chemical fuel for the supersonic dash. Using the following values,

$$h = 2400 \text{ nautical mile-pounds per pound of fuel}$$

$$\eta = 0.23$$

$$L/D = 5.5$$

$$\frac{W_f}{W_{g,o}} = 0.30$$

radius of the dash, equation (1b), becomes 540 nautical miles.

The effect of increasing turbine-inlet gas temperature on supersonic dash radius is shown in figure 21. The supersonic dash is plotted against turbine-inlet temperature, because turbine-inlet temperature is the variable that shows the greatest likelihood of resulting in increased thrust per pound of propulsion system weight. The data show that supersonic dash radii, including distance for climb and acceleration, of about 500 nautical miles can be expected from the current technical knowledge. The possibility of getting dash radii of much greater than this value will require considerable progress in achieving higher turbine-inlet temperatures.

[redacted]

Nuclear tug-chemical tow. - With the nuclear cruise-chemical dash airplane, sufficient thrust must be produced during the chemically powered dash to carry the nuclear propulsion system. As a means of increasing the supersonic dash radius, a nuclear powered subsonic tug towing a chemically powered supersonic dash airplane is considered. The effect of the chemical dash radius on the gross weight of the nuclear tug and of the chemical tow is shown in figure 22. Two engine efficiencies, 0.23 and 0.38, are assumed for the chemically powered plane to represent the conditions considered in figure 7. The subsonic cruise radius is in excess of 10,000 nautical miles. The tug-tow arrangement requires a poststrike contact to be made. The supersonic dash with this tug-tow combination is about twice that for the nuclear cruise-chemical dash airplane.

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The results with the nuclear powered bomber are summarized in table IV. Where two values are shown, the value to the right represents the optimistic estimate.

CONCLUSIONS

1. The analysis presented herein indicates that the desired strategic bomber requirement of 5500 nautical mile total radius including a 2000 nautical mile supersonic dash requires the use of Zip fuel in place of the current hydrocarbon fuel, and two refuelings at 3500 nautical miles - prestrike and poststrike - unless optimistic results are achieved in regard to airplane lift-drag ratio or fuel to gross weight ratio. If optimistic results are achieved, the mission can be achieved with two refuelings at 2800 nautical miles using a hydrocarbon fuel or with one refueling at 2800 nautical miles using Zip fuel.

2. For the minimum strategic bomber requirement (4000 nautical mile total radius of which 1000 nautical miles is supersonic dash), assuming conservative values for airplane lift-drag ratio or ratio of fuel to gross weight, two refuelings at about 2000 nautical miles will be required with a hydrocarbon fuel or one with Zip fuel. For optimistic assumptions of these variables one refueling with the hydrocarbon fuel or no refueling with Zip fuel will suffice.

3. In general, Zip fuel permits the mission to be performed with one less refueling than with hydrocarbon fuel. However, use of Zip fuel throughout the engine will require an extensive research program.

4. With refueling at 2500 nautical miles, the tanker will have a gross weight about equal to the gross weight of the bomber for a prestrike refueling and about half the weight for a poststrike refueling.

5. Regardless of the airplane used, gross weight at take-off will be 12.5 to 20 times military load unless appreciable loss in range is to be taken. The value of 12.5 gives a range 0.92 of that with the value of 20.

6. If, in a refueled mission, gross weight of the tanker and the weight of the military load are fixed, total radius of action is nearly independent of the bomber weight.

4221 7. The use of floating wing-tip tanks, because of increased aspect ratio and increased ratio of fuel to gross weight, allows a radius increase of 900 to 1250 nautical miles.

8. An all nuclear powered supersonic strategic bomber does not appear feasible at this time, but modest improvements in either ratio of thrust to propulsion system weight or in airplane L/D can change this picture.

9. A dash radius on the order of 500 to 1000 nautical miles is possible for an airplane with nuclear subsonic cruise and chemical supersonic ($M = 2.0$ to 3.0) dash at a combat altitude of about 60,000 feet. A dash radius on the order of 1000 to 2000 nautical miles is possible if a tug-tow combination is used. In each case, the higher figure represents an optimistic estimate.

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TABLE I. - PERFORMANCE DESIGN TREND

AIRPLANE	AIRPLANE GROSS WEIGHT LB, $W_{g,o}$	RATIO TO AIRPLANE GROSS WEIGHT AIRFRAME WEIGHT, $W_{a,f}/W_{g,o}$	ENGINE WEIGHT, $W_e/W_{g,o}$	MILITARY WEIGHT, $W_m/W_{g,o}$	FUEL WEIGHT, $W_f/W_{g,o}$	WING LOADING, $W_{g,o}/S$
B-24	56,000	0.443	0.179	0.255	0.123	38.0
B-29	105,000	.432	.193	.133	.242	81.4
B-50	164,500	.275	.171	.234	.320	100.5
B-36	370,000	.335	.105	.058	.502	77.5
B-47	200,000	.381	.109	.085	.425	133.2
B-52	450,000	.212	.097	.099	.592	112.5
B-58	147,000	.226	.104	.05	.620	100.0

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TABLE II. - PERFORMANCE DESIGN TREND

AIRPLANE	FLIGHT MACH NO, M	FUEL HEATING VALUE, h, $\frac{\text{NAUT MI}}{\text{LB}}$	OVER-ALL ENGINE EFFICIENCY, η_e, PERCENT	TRIMMED LIFT-DRAG RATIO, L/D	FUEL WEIGHT, $W_{f,o}/W_{g,o}$	RADIUS, NAUT MI
B-24	0.3	2400	24	13.0	0.123	440
B-29	.47	2400	28	17.0	.242	1400
B-50	.5	2400	28	16.6	.320	1890
B-36	.6	2400	28	19.4	.502	3920
B-47	.78	2400	18	17.7	.425	2080
B-52	.73	2400	20	22.0	.592	3400
B-58	.90 2.0	2400	20 20	12.0 5.0	.620	1970 820

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TABLE III. - CHEMICAL STRATEGIC BOMBER

	DASH AT $M_a = 2.0 - 3.0$	65,000 FT TARGET ALT.
	4000 / 1000 (MIN)	5500 / 2000 (REQUIRED)
JP-4	2 REFUEL @ 2200 N.M.	———
	1 REFUEL @ 1600 N.M.	2 REFUEL @ 2800 N.M.
ZIP	1 REFUEL @ 2000 N.M.	2 REFUEL @ 3500 N.M.
	NO REFUEL	1 REFUEL @ 2800 N.M.

GROSS WT:		
NO REFUEL	450,000 LB.	1 AIRPLANE
1 REFUEL*	675,000 — 900,000 LB.	2 AIRPLANES
2 REFUEL*	1,125,000 LB.	3 AIRPLANES

*REFUEL @ 2500 NAUT MI

NOTE: FLOATING WING PANELS ≒ 1 REFUELING

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TABLE IV. - NUCLEAR STRATEGIC BOMBER

	CHEM. DASH	TUG-TOW
DASH RADIUS, NAUT MI	500 - 1000	1200 - 2000
DASH, M_a	2.0 - 3.0	2.0 - 3.0
COMBAT ALT, FT	60,000	60,000
GROSS WT, LB	600,000	320,000 + 320,000

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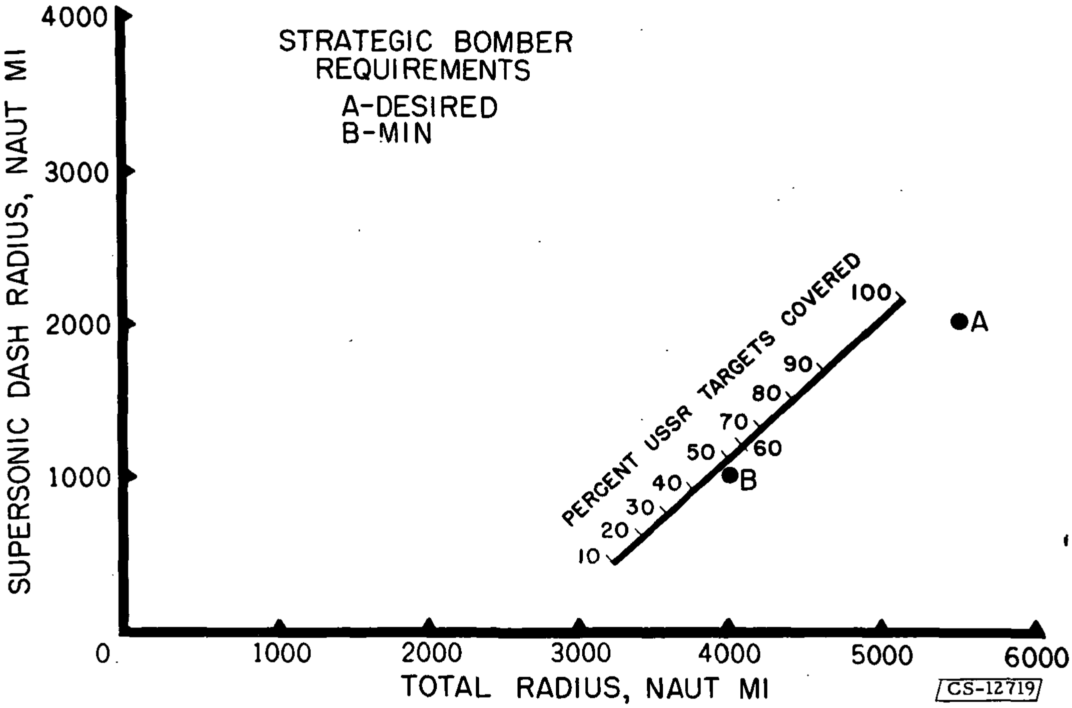

Figure 1. - Supersonic dash vs total radius.

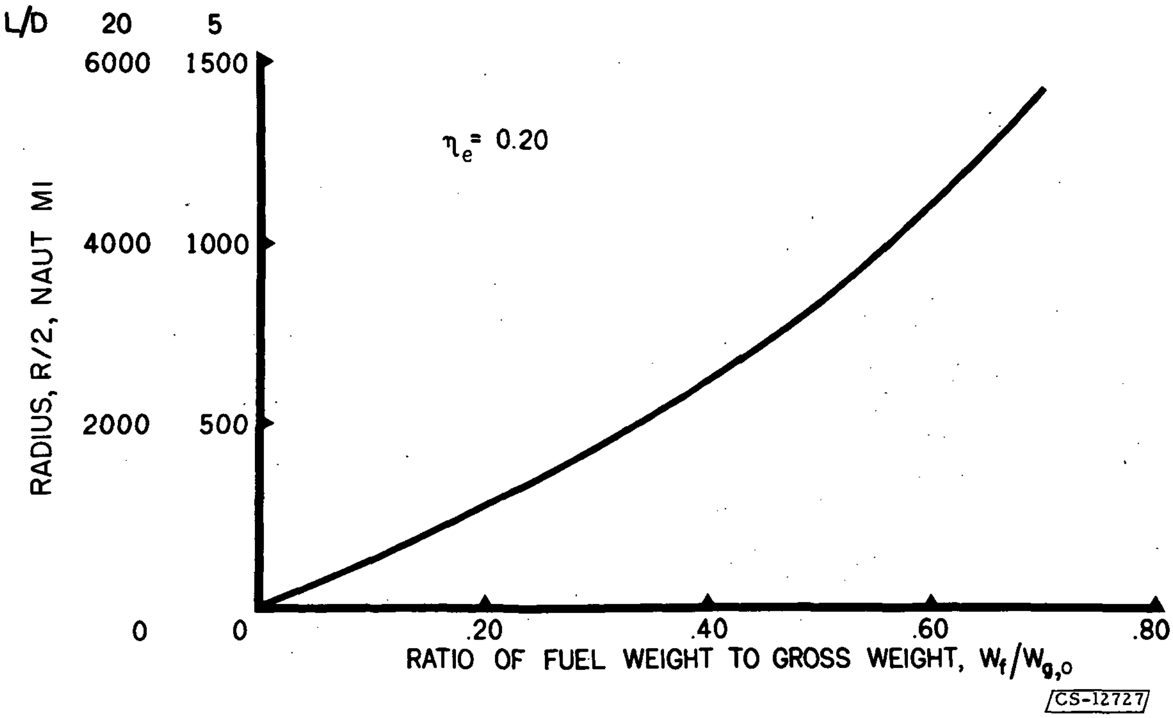

Figure 2. - Relation of fuel quantity to radius of action.

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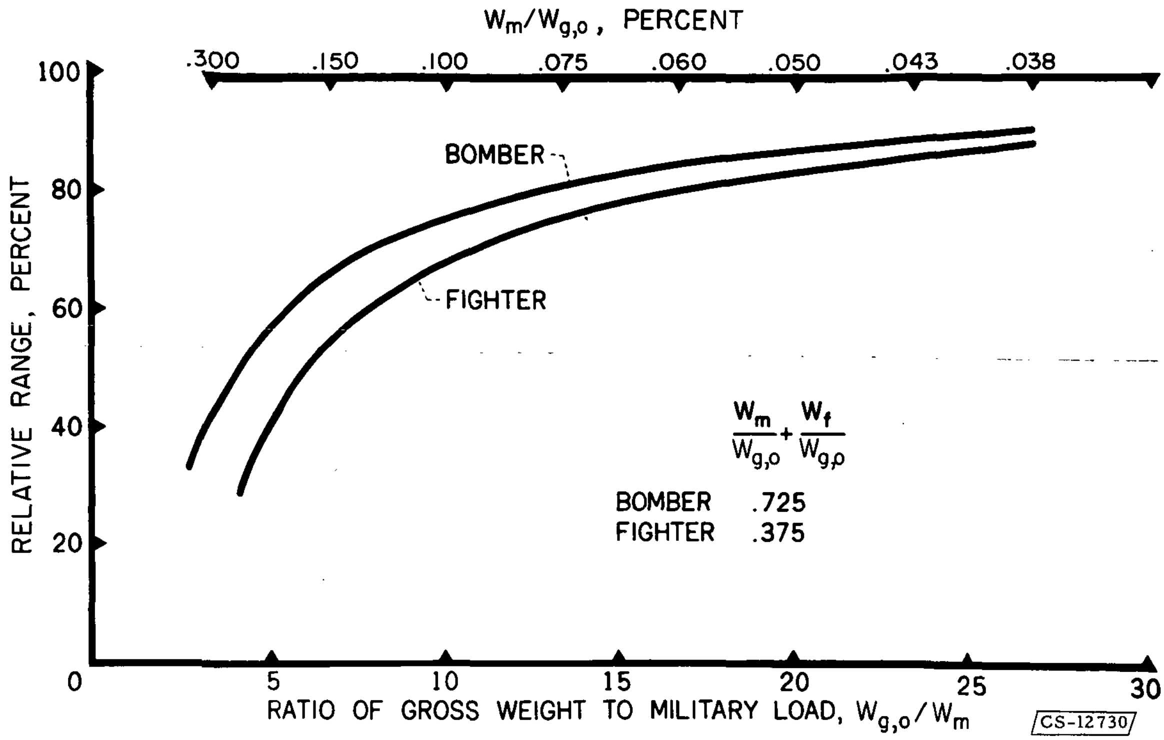

Figure 3. - Relation of airplane gross weight to range.

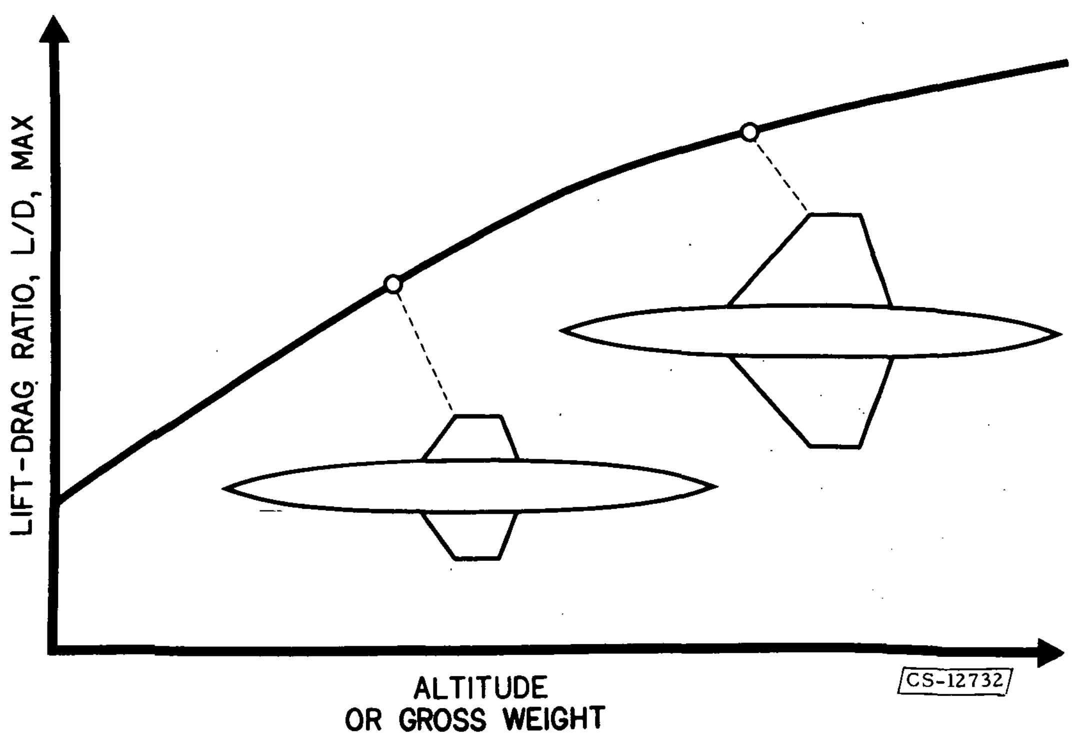

Figure 4. - Effect of altitude on lift-drag ratio.

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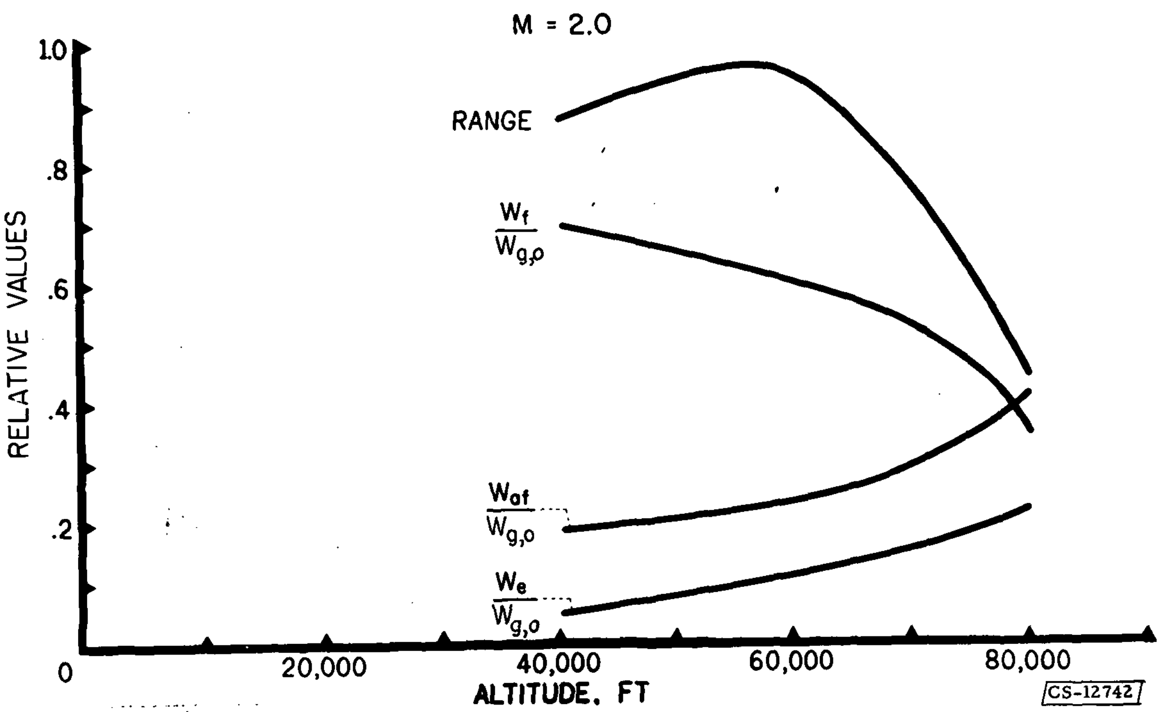

Figure 5. - Relation of altitude to airplane weight distribution.

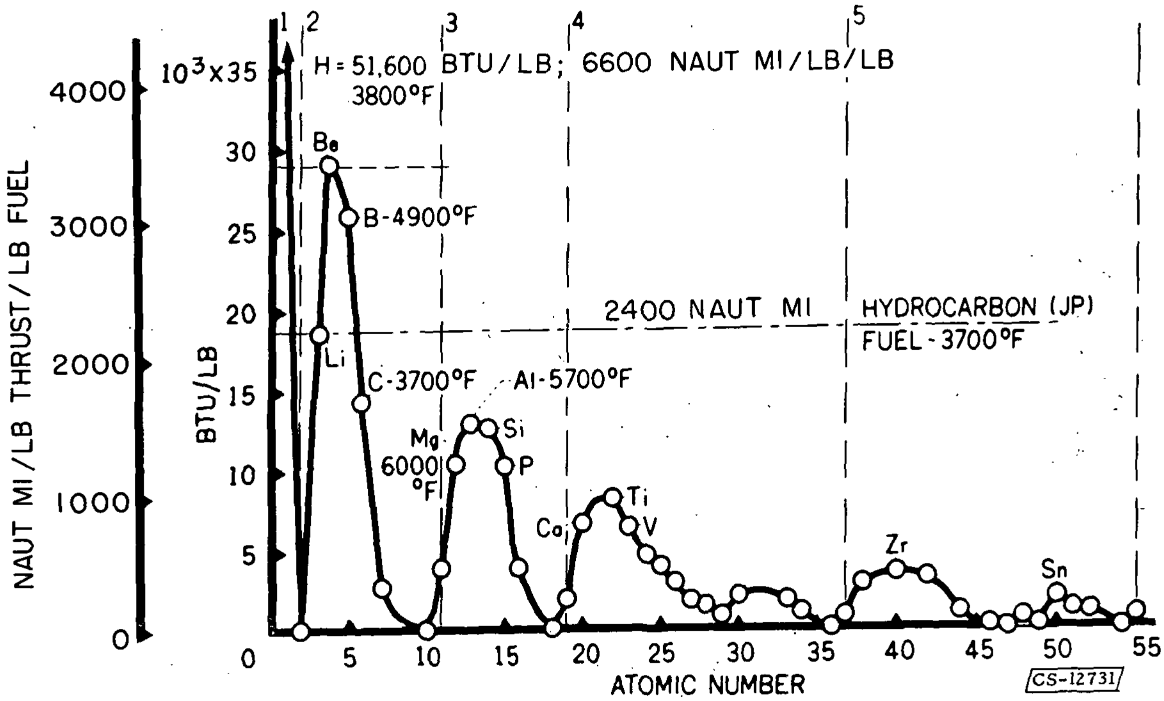

Figure 6. - Heat of combustion of elements.

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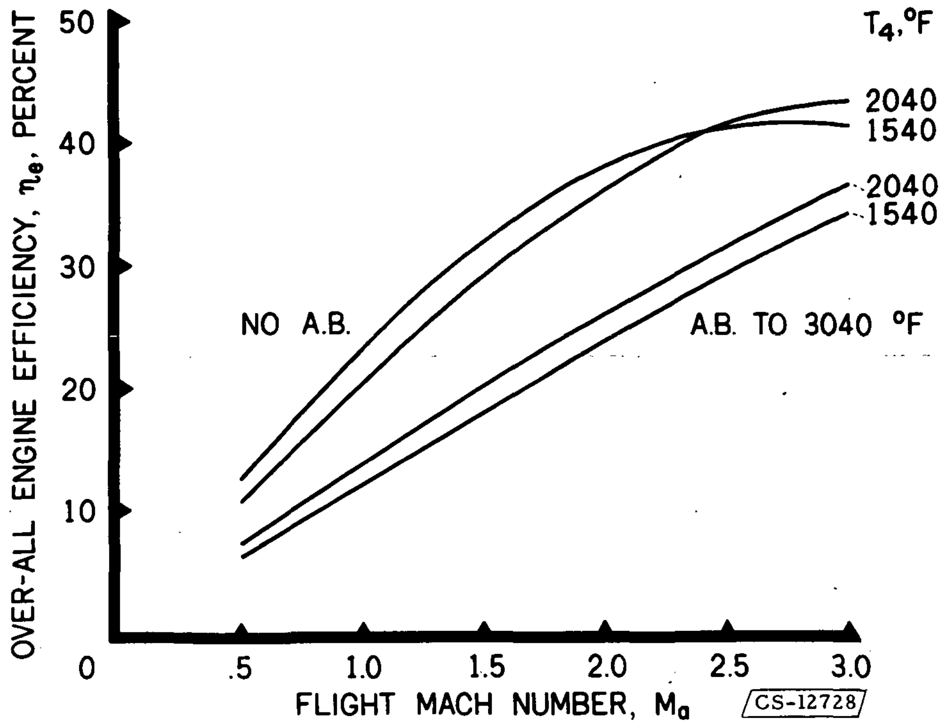

Figure 7. - Effect of Mach number on engine efficiencies with and without afterburner altitude 35,000 feet and above.

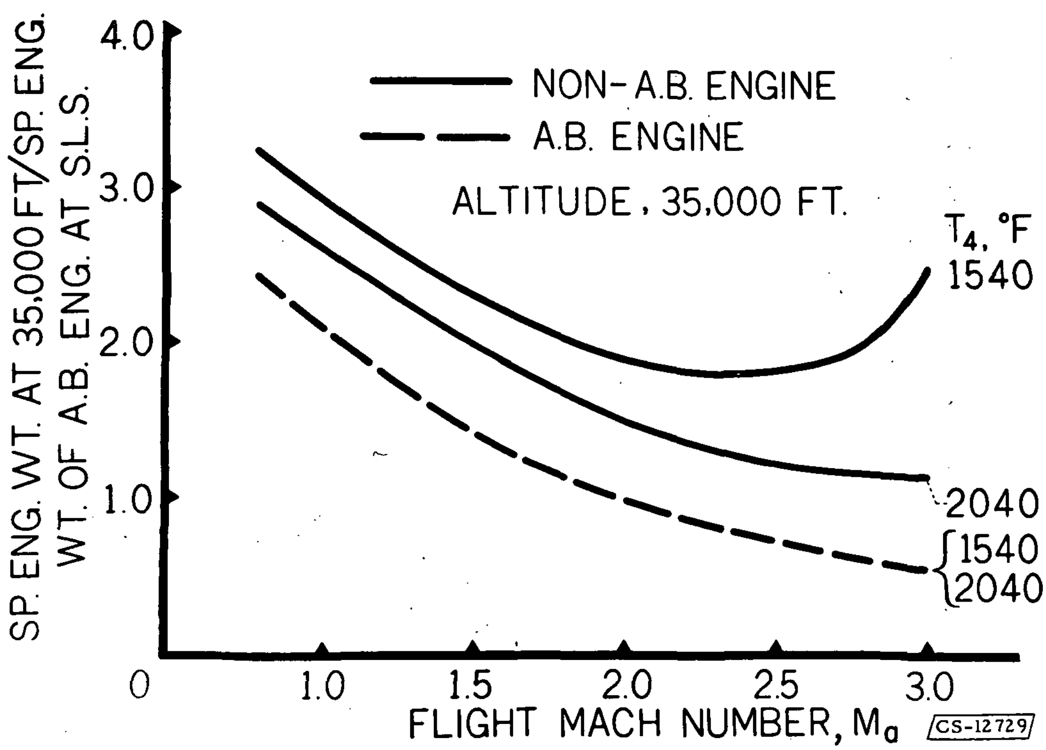

Figure 8. - Relation between flight speed and specific engine weight.

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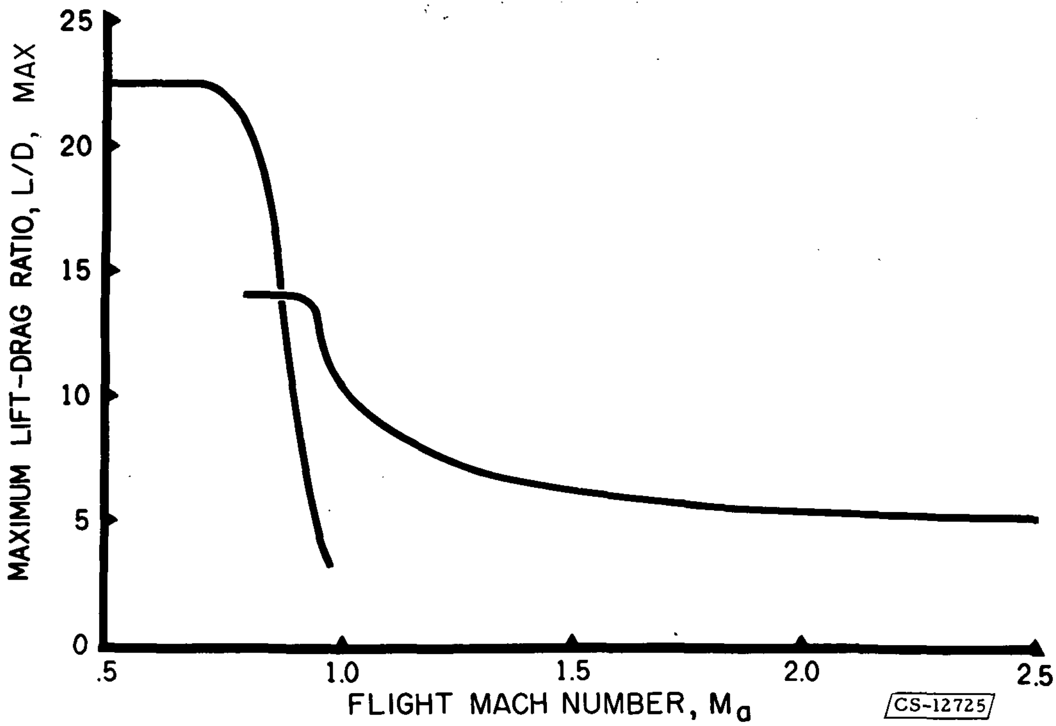

Figure 9. - Typical maximum lift/drag ratios.

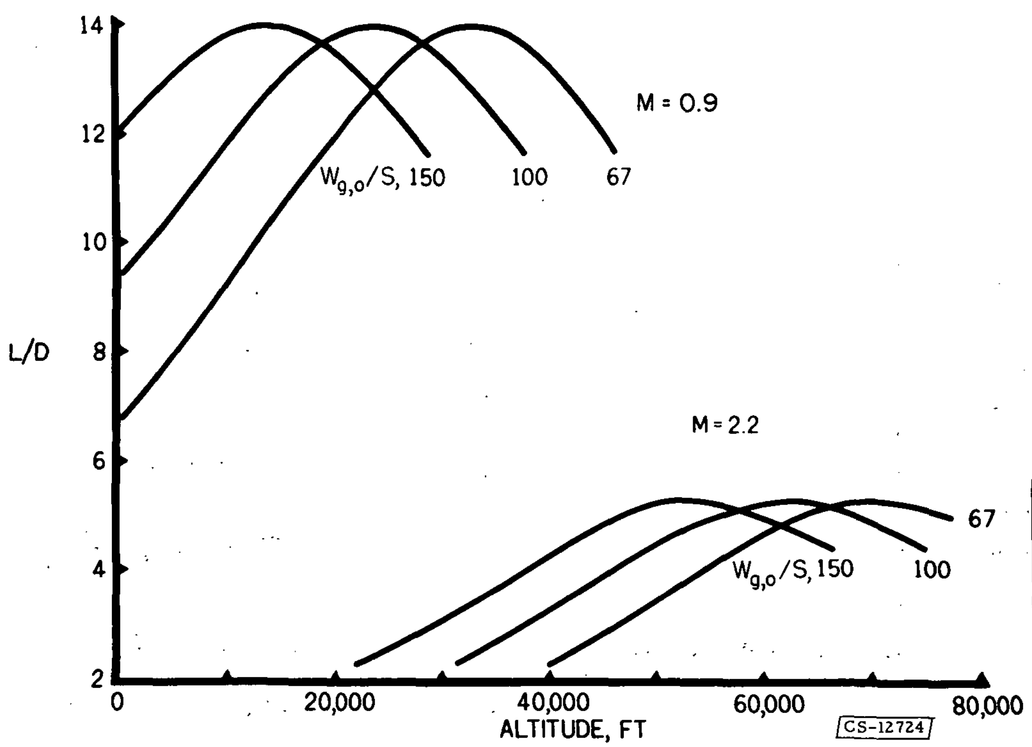

Figure 10. - Relation of altitude to lift-drag ratio.

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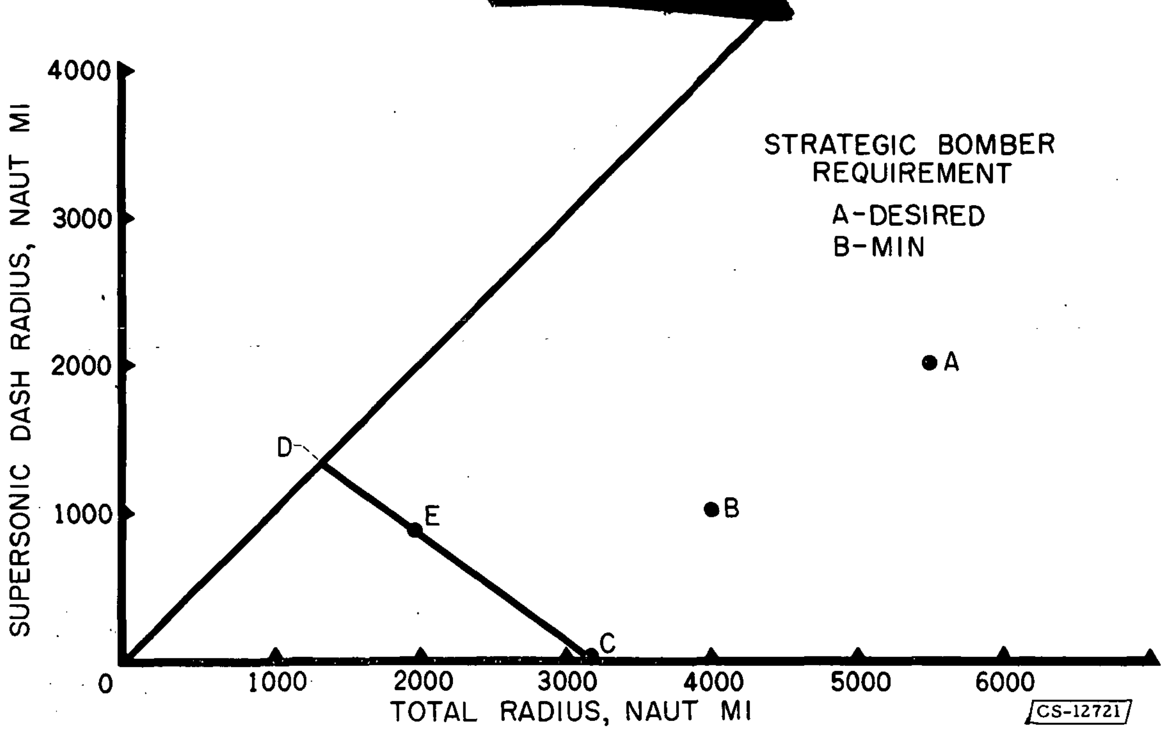

Figure 11. - Supersonic dash vs total radius using JP-4.

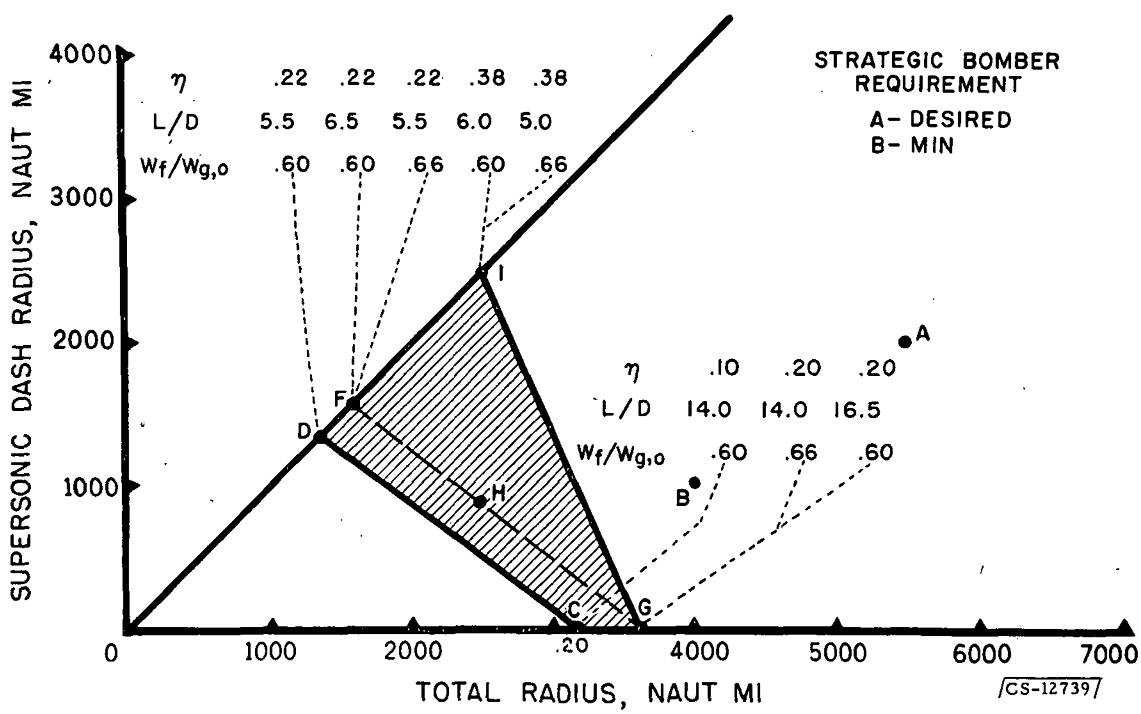

Figure 12. - Supersonic dash vs total radius using JP-4.

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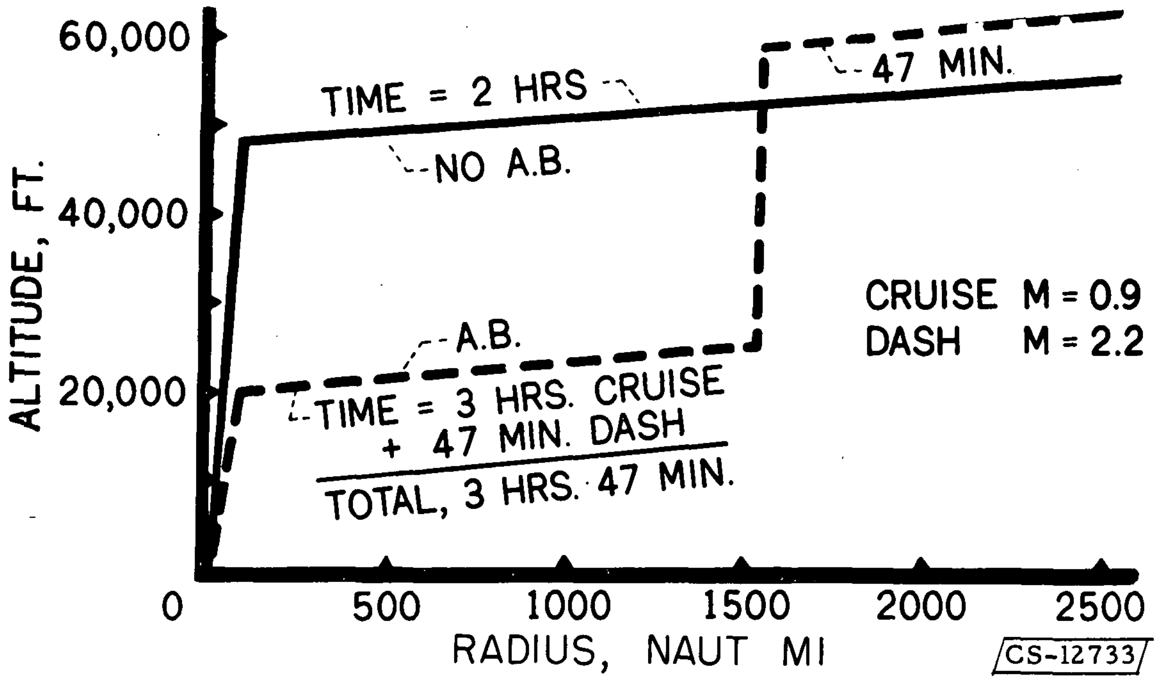

Figure 13. - Alternate bomber flight plans.

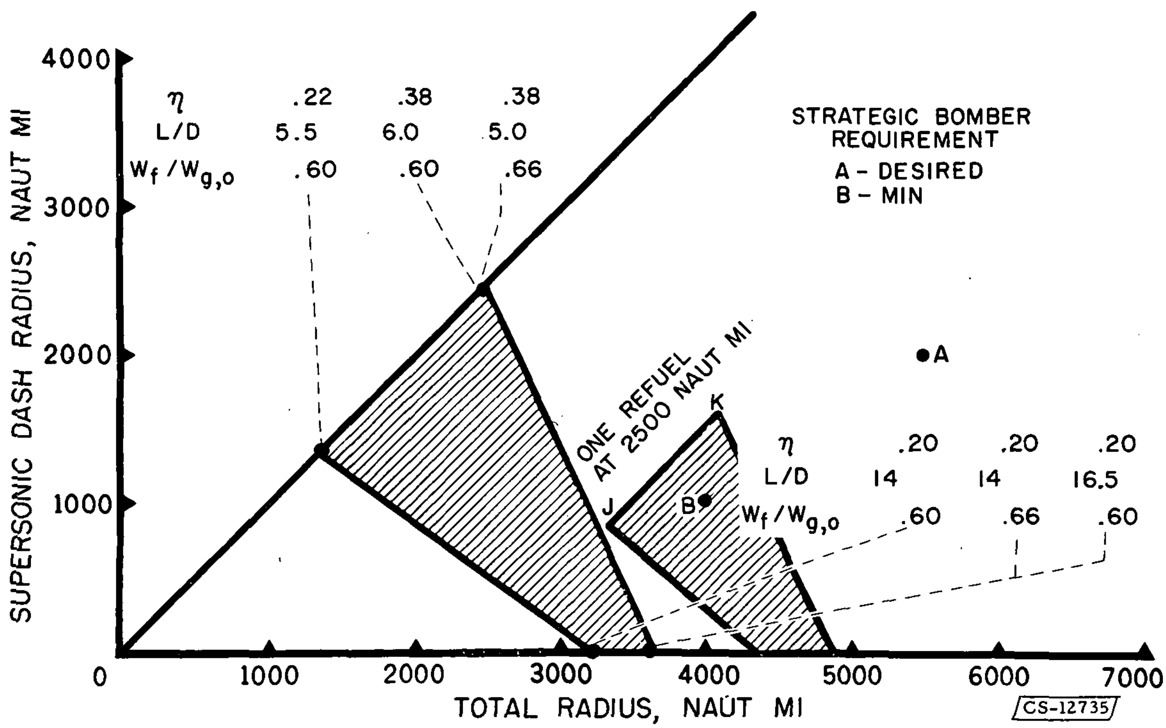

Figure 14. - Supersonic dash vs total radius using JP-4.

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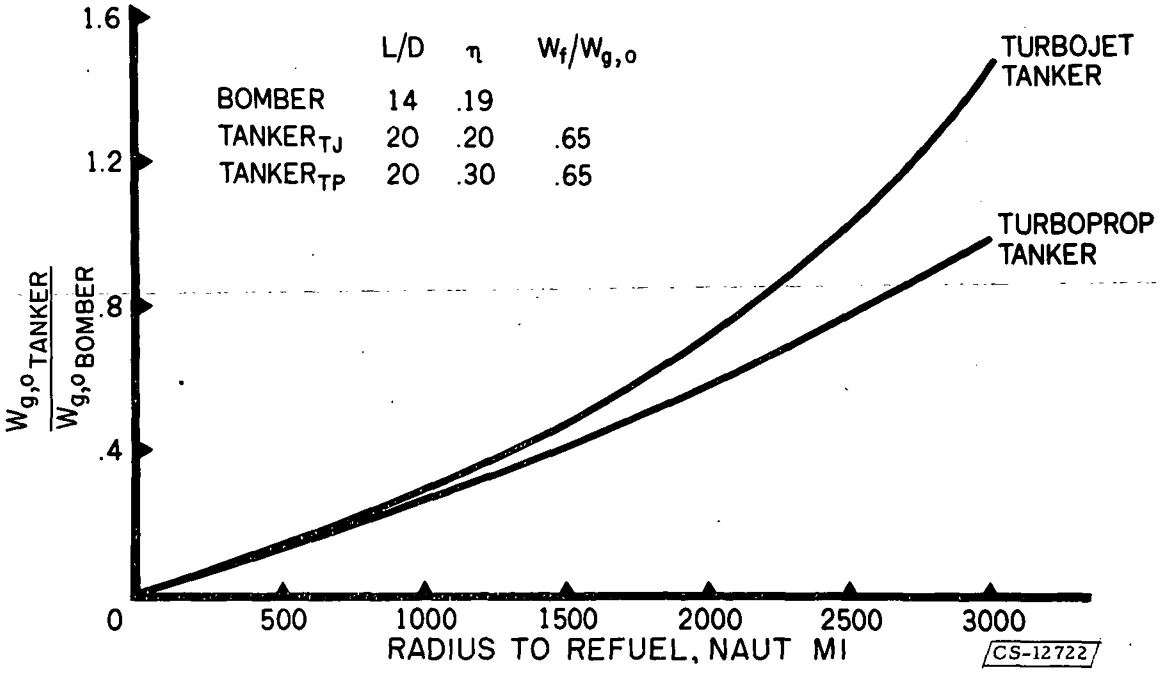

Figure 15. - Relation of tanker weight to bomber weight.

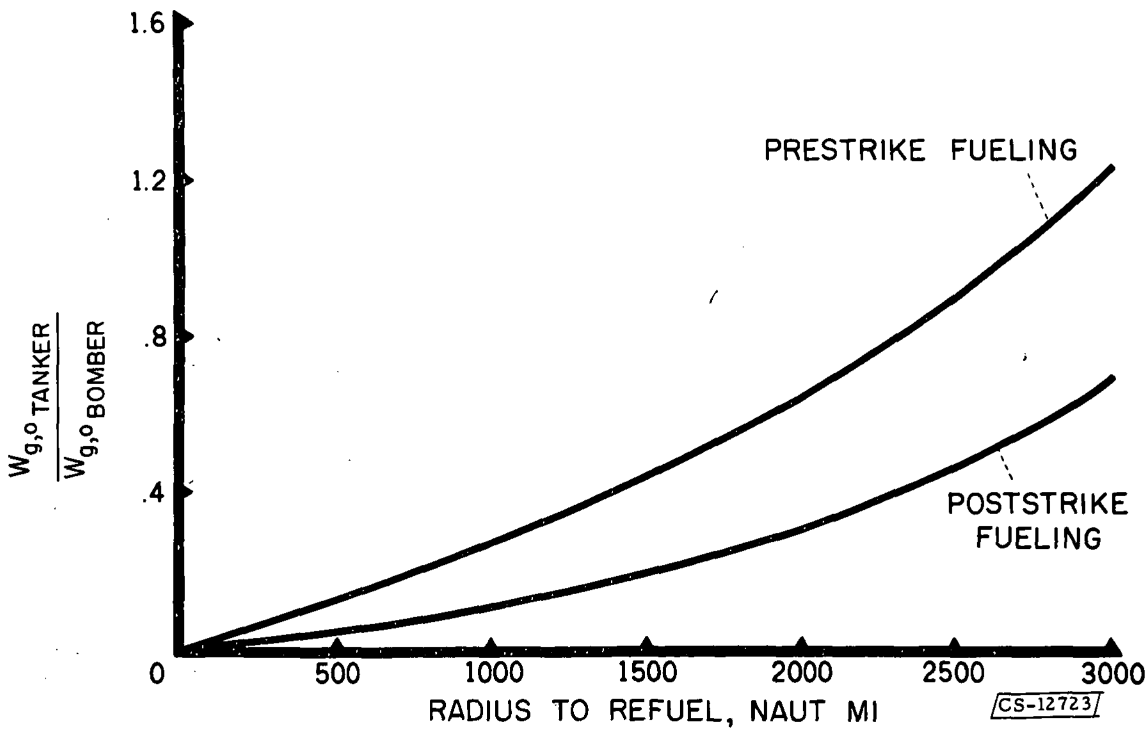

Figure 16. - Relation of tanker weight to bomber weight.

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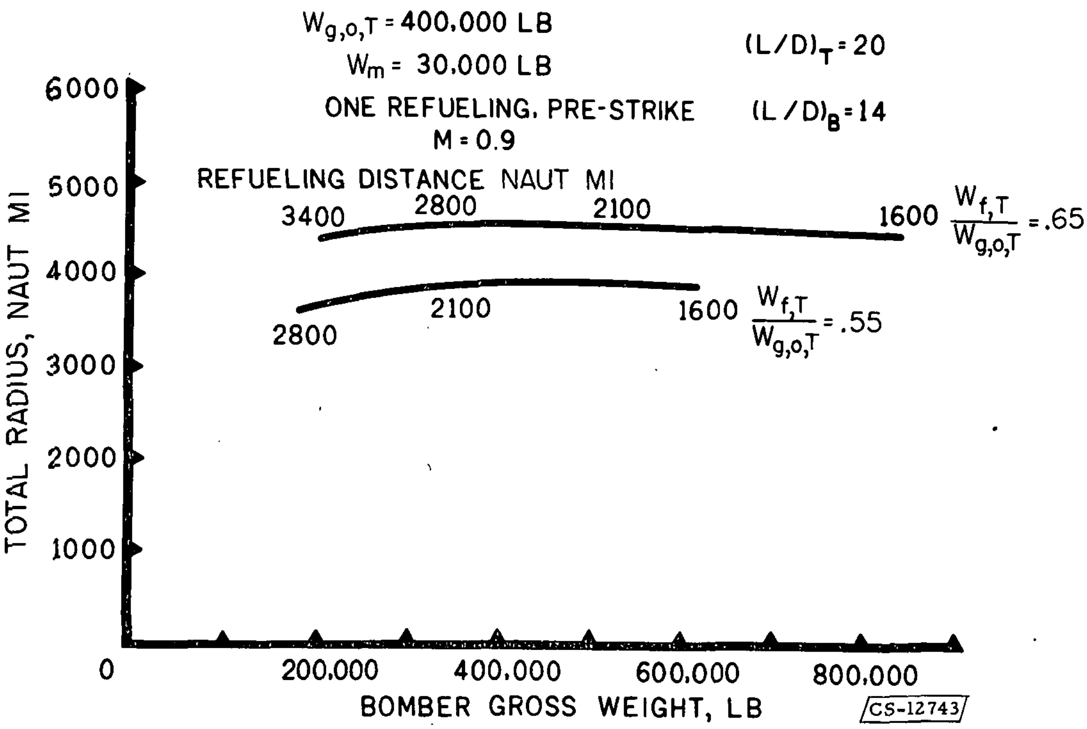

Figure 17. - Relation of bomber gross weight to total radius for constant tanker gross weight.

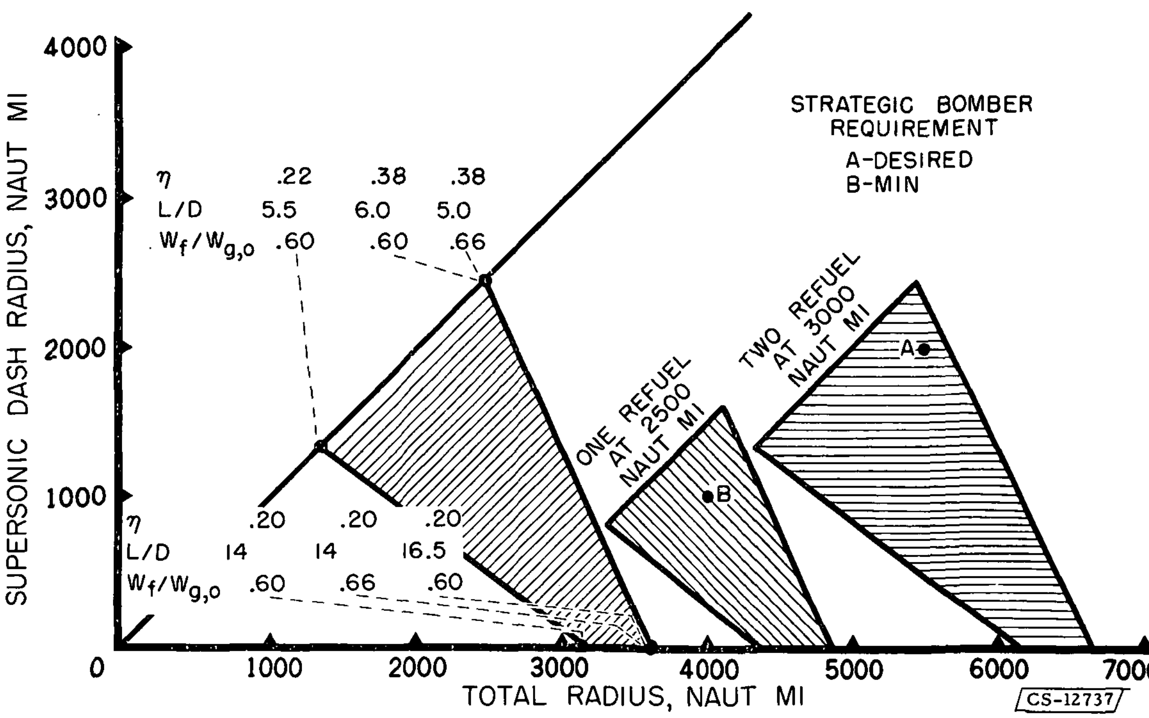

Figure 18. - Supersonic dash vs total radius using JP-4.

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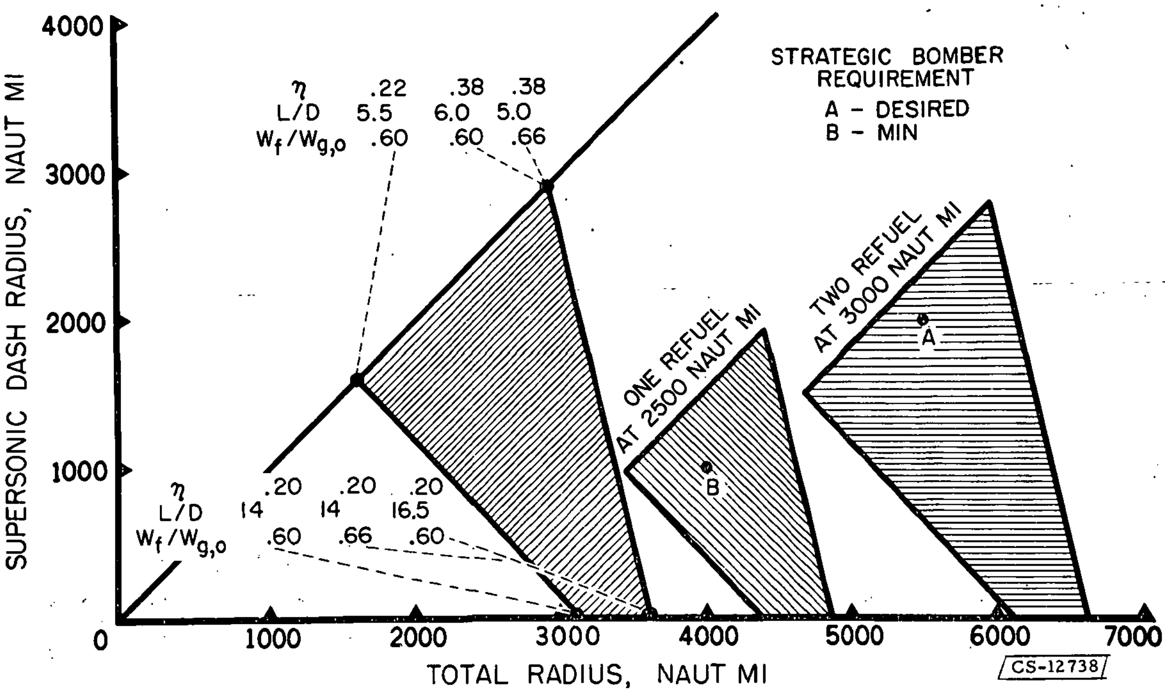

Figure 19. - Supersonic dash vs total radius using JP-4 and zip.

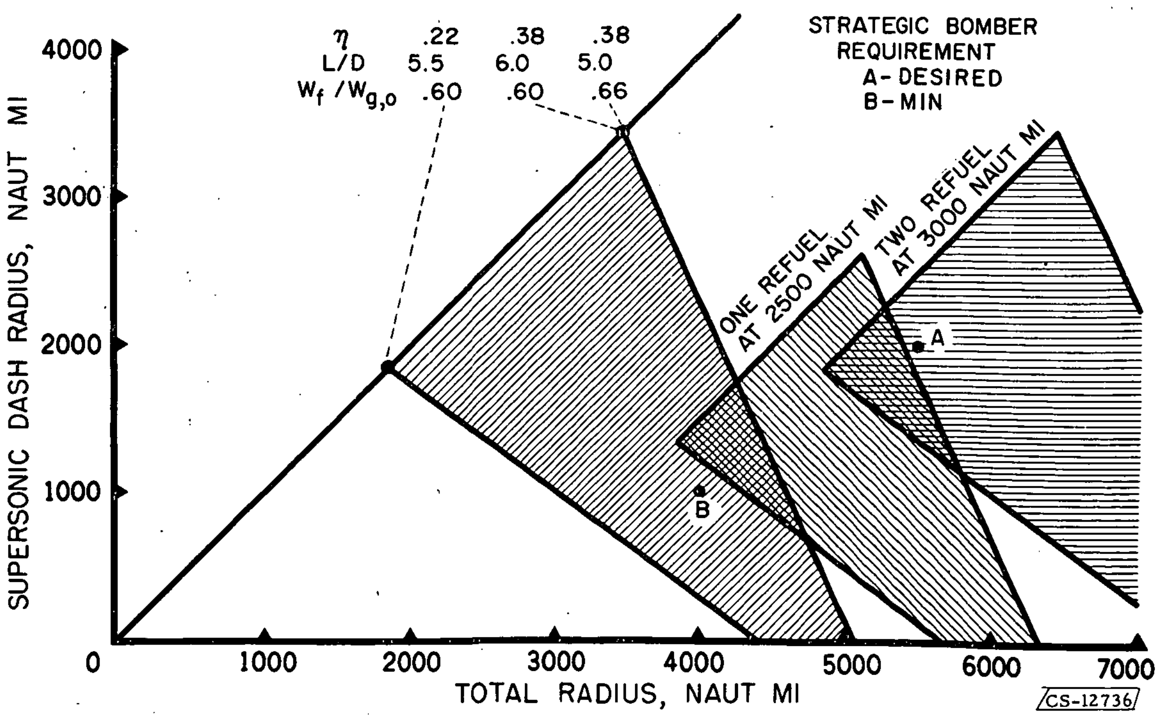

Figure 20. - Supersonic dash vs total radius using zip.

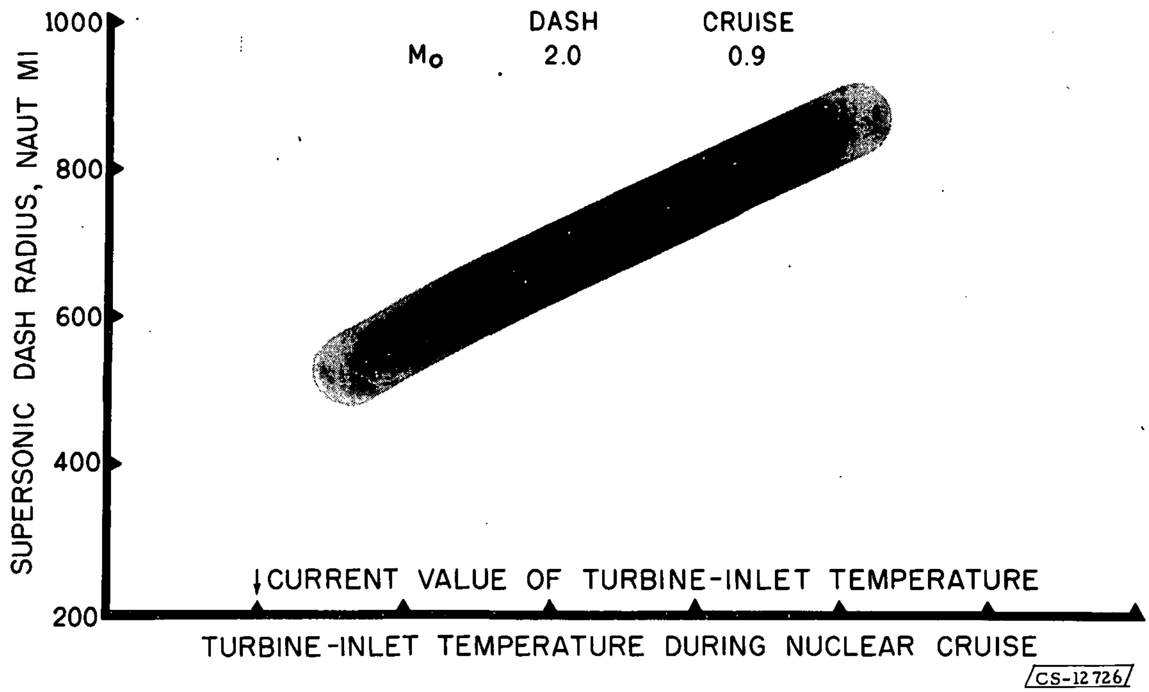

Figure 21. - Effect of turbine inlet temperature on supersonic dash radius.

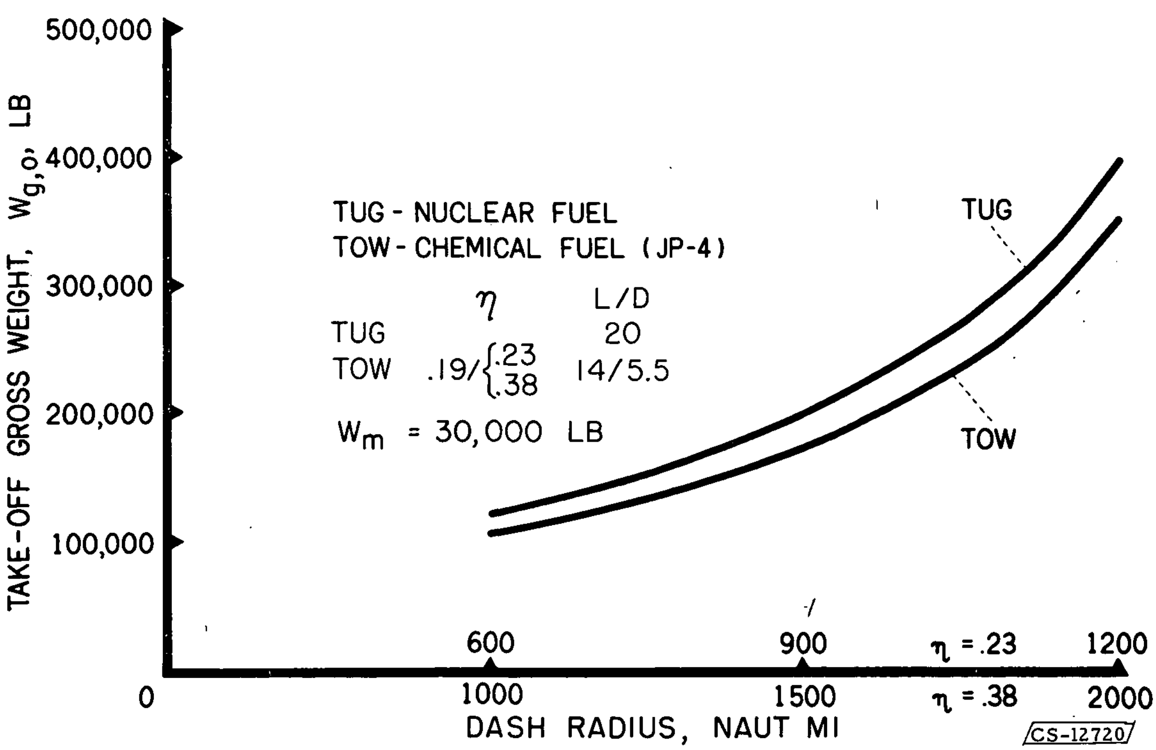

Figure 22. - Relation between dash radius and gross weight of tug-tow.

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Engines, Turbo-Jet 3.1.3

Nuclear Energy Systems 3.1.10

Engine Types, Comparison 3.1.12

Fuels - Relation to Engine Performance 3.4.3

Rothrock, Addison M., Cesaro, Richard S., and Walker, Curtis L.

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RELATION OF TURBOJET PROPULSION SYSTEM DEVELOPMENT
TO THE STRATEGIC BOMBER MISSION

Abstract

A generalized analysis presents the effects of turbojet propulsion system development and fuel selection on ability of a strategic bomber to perform desired and minimum missions. The variation of bomber performance using a hydrocarbon, boron, or nuclear fuel is discussed. With chemical fuel, the effects of refueling are discussed. With nuclear fuel, the nuclear cruise-chemical dash bomber and the nuclear subsonic tug towing a chemically powered supersonic bomber are compared. The factors that determine bomber gross weight and bomber altitude are briefly discussed.

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