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RESEARCH MEMORANDUM

ANALYSIS OF THE TURBOJET ENGINE FOR PROPULSION OF SUPERSONIC BOMBERS

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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SUMMARY

An analysis was made to determine the effects of various turbojet-engine design and operating variables and combinations of these variables on the performance of a supersonic bomber. In order to evaluate the divers engine designs, two bomber flight plans were considered; to reduce bomber vulnerability, both plans specified supersonic flight over an assumed defended zone. The first flight plan involved cruise at a Mach number of 0.9 followed by a 500-mile dash at supersonic speeds to and from the target in the zone and return to base at a flight Mach number of 0.9. The second flight plan involved all-supersonic cruise.

The merit of the various engine designs and the relative importance of the various design parameters were evaluated chiefly by means of the total radius for both the flight plans. Take-off distance and maneuverability at the combat condition were also investigated. The effect on bomber performance of changes in the following variables was investigated: zone afterburner temperature, compressor pressure ratio, compressor efficiency, turbine-inlet temperature, turbine efficiency, air-handling capacity, engine specific weight, and exhaust-nozzle configuration. In order to produce a given thrust, a variation in zone afterburner temperature involves a change in engine size; the lower the zone afterburner temperature, the larger the engine size required. For the all-supersonic flight plan, the airplane radius is relatively insensitive to the zone afterburner temperature. For the flight plan involving a relatively large amount of subsonic flight, the total radius increases as the zone afterburner temperature is increased. Regardless of the amount of supersonic flight in the flight plan, the afterburner temperature in the combat zone should probably not exceed 3000° R, in order to ensure adequate thrust margins during take-off, climb, and acceleration.

Based on the assumption that engine specific weight is independent of compressor pressure ratio, the airplane radius for present-day

engine components is relatively insensitive to compressor pressure ratio. For the all-supersonic flight plan, compressor pressure ratios up to 9 give near-maximum performance. For the flight plan involving the 500-mile supersonic radius, a compressor pressure ratio of approximately 12 gives maximum airplane radius. Increases in turbine-inlet temperature produce gains in airplane radius, especially at high values of compressor pressure ratio. Substitution of a convergent exhaust nozzle for a convergent-divergent nozzle may penalize the bomber radius up to 24 percent, depending on the amount of supersonic cruise in the flight plan. The combination of several improvements to provide an "advanced" turbojet engine increased airplane radius about 36 percent, considerably more than the sum of the improvements attributable to individual component changes.

INTRODUCTION

The vulnerability of subsonic bombers to advanced interception techniques has promoted interest in airplanes capable of supersonic flight over enemy defended zones. In general, the mission of the supersonic bomber consists of take-off, subsonic cruise over friendly territory, supersonic dash over the defended zones to and from the target, and return to base at subsonic speeds. Turbojet engines for the propulsion of airplanes of this type must meet certain requirements for each phase of the flight plan. The engine must provide sufficient thrust to allow take-off from runways of a reasonable length. During cruise over friendly territory, low specific fuel consumption is of great importance. In order to obtain high speeds and maneuverability, the engine must provide sufficient thrust to allow acceleration through the transonic range and excess thrusts at high flight speeds. The turbojet engine, serving as a power plant for the bomber, can be designed to assure the aircraft adequate performance in each of these regions.

The investigation discussed in the present report is an analysis of the turbojet engine as the power plant for bombers capable of supersonic flight speeds. A similar analysis, in which the turbojet engine is considered for the propulsion of supersonic fighter aircraft, is presented in reference 1. Because of the continuing improvements being made in the components comprising the turbojet engine, the analytical study presented herein was made to ascertain the effects of these engine component changes on the performance of bombers and to evaluate their relative importance.

Two flight plans are considered in this report. The first consists of a subsonic cruise portion followed by 500 miles at supersonic speed over enemy territory; the pay load is then dropped and the bomber returns supersonically 500 miles and continues to base at a subsonic speed.

In the second flight plan, the entire cruise flight of the bomber is made at supersonic speeds. Both the subsonic and the supersonic flights were made on Breguet flight plans at the altitude for maximum range, that is, the one for which the product of drag-lift ratio and specific fuel consumption was a minimum. A representative airplane configuration was selected, and the effect on airplane performance of individual variations of the principal engine design variables was computed. The suitability of the various engine configurations and the relative importance of the different engine design variables are judged principally by the total bomber radius, generally for a fixed gross weight. Take-off distance and maneuverability were also investigated.

The engine design variables considered were compressor pressure ratio, compressor and turbine efficiency, turbine-inlet temperature, afterburner temperature, engine specific weight, and engine air-handling capacity. Engines equipped with variable-area convergent and convergent-divergent exhaust nozzles were investigated. For the most part, changes in engine configuration were studied at a fixed take-off gross weight; the effect of changes in gross weight on radius was also determined.

ANALYSIS AND PROCEDURE

Flight Plans

The two flight plans considered are shown schematically in figure 1. These flight plans tend to cover the extremes in the percentage of flight at supersonic speeds that might be expected in the flight of a supersonic bomber. The first flight plan had a subsonic portion, followed by 500 miles at supersonic speeds to the target, 500 miles at supersonic speeds away from the target, and a final subsonic portion back to base. The subsonic portions were assumed to be flown at 0.9 Mach number. For the second flight plan, the entire flight to and from the target was assumed to be flown at supersonic speeds. In general, a supersonic Mach number of 2 was assumed. It was further assumed that all cruise flight followed a Breguet flight plan, and that in all instances the initial altitude was chosen to give maximum range for the particular flight plan being considered. It was assumed that 5 percent of the original fuel load was maintained in the tanks at the end of the flight as reserve. In general, the amount of fuel burned and the distance covered during take-off, climb, and acceleration were neglected. (Calculations indicated that the results obtained from the analysis were not significantly affected by neglecting these compensating factors.)

Airplane Configuration

For all calculations, except those in which the effect of gross weight was investigated, the airplane take-off gross weight was assumed to be 150,000 pounds, of which 23 percent was structure, including crew and electronic equipment, 5000 pounds was bomb load, and the remainder was distributed among engines, fuel, and fuel tanks. Although a delta wing with aspect ratio of 2 and thickness-chord ratio of 0.05 was chosen for this analysis, the results, insofar as the relative effects of the various engine parameters on airplane performance are 2913
concerned, are not especially critical to the particular wing type chosen. The take-off wing loading was assumed to be 100 pounds per square foot. The fuselage, for a gross weight of 150,000 pounds, was assumed to have a maximum diameter of 8 feet. For other values of gross weight, the fuselage maximum diameter was assumed to vary as the cube root of the gross weight. For all cases, the over-all fineness ratio of the fuselage was 12. The engines were assumed to be mounted in nacelles, one on either side of the fuselage. The over-all fineness ratio of the nacelles was 9. The structure to gross weight ratio was decreased to 0.225 for a gross weight of 200,000 pounds and increased to 0.25 for a gross weight of 100,000 pounds.

Airplane drag was computed by summing the drag of wing, tail, fuselage, and nacelles. It was assumed that the entire wing area, including that blanketed by the fuselage, was effective in producing lift; whereas, only the exposed wing area was assumed to produce drag. The tail drag was assumed to be 20 percent of the wing zero-lift drag. The airplane lift-drag ratios obtained for a representative configuration were approximately 11 at a Mach number of 0.9 and about 5 at a Mach number of 2. Further details regarding the aerodynamic assumptions are presented in the appendix.

Procedure and Engine Operation

The analysis was an off-design study, in that the engines were assumed to be designed for sea-level static conditions and then operated, at all flight Mach numbers and altitudes excepting subsonic cruise, at constant mechanical speed. The performance at the combat-zone Mach numbers was therefore off-design with respect to compressor aerodynamic speed and associated air flow and compressor efficiency. For the part-subsonic flight plan, off-design studies were necessary to establish engine performance at subsonic speeds for various thrust levels, because engines providing sufficient thrust to fly at the supersonic-zone Mach number had to be throttled to fly at the subsonic cruise Mach number and the most favorable altitude.

Basic values of the various engine design variables were assumed, and the effects of departures from these values were determined by varying each while holding all others fixed. The effect of the afterburner temperature used in the supersonic portion of the flight (zone afterburner temperature) was first investigated. Then, the effect on airplane performance of substituting a convergent nozzle for a convergent-divergent nozzle was determined. Next, the effect of compressor pressure ratio and compressor efficiency was evaluated. The engine specific weight was changed, and its importance determined. In turn, the turbine-inlet temperature and air-handling capacity were increased, and their contribution to improved bomber performance assessed. After the effect of the foregoing changes had been determined individually, several of the advantageous improvements that seemed realizable in the foreseeable future were combined into what is called the "advanced engine." The effect of changes in compressor pressure ratio and afterburner temperature on bomber performance with the advanced engine was determined, as was performance with an advanced engine having no afterburner.

The performance was computed for an initial gross weight of 150,000 pounds and a combat-zone Mach number of 2. The effect of changes in the gross weight was computed for the basic and advanced engines and for both flight plans. Two bombers, one with basic engines and one with advanced engines, were flown over a range of Mach numbers from 0.9 to 2.5 to determine the effect of flight Mach number on the radius of the bomber.

The three critical thrust regions through which the airplane must pass and in which it must have sufficient thrust to give satisfactory performance are (1) at take-off, (2) in the transonic region, and (3) in the combat zone. In this analysis, the drag of the airplane in the combat zone determined the engine thrust requirement. It was assumed that the engines were operated at rated rotational speed and rated turbine-inlet temperature in the combat zone. For the subsonic cruise portions of the flight, the afterburner was turned off and the fuel flow and rotational speed were reduced to provide the required thrust.

As the flight Mach number and altitude varied, the compressor-inlet temperature and, hence, equivalent engine speed varied. The variations in compressor efficiency and air flow with equivalent engine speed given in reference 2 and in appendix C of reference 1 were used to calculate engine performance during flight. These variations were assumed independent of rated compressor pressure ratio. The compressor pressure ratio also varied with equivalent engine speed or flight condition and was computed from the air flow and turbine-inlet temperature with the assumption that the turbine nozzles were choked. Throughout the report, all values cited for compressor pressure ratio are those occurring at rated sea-level static conditions.

The engine-inlet diffuser was assumed to be of the spike type with a translating spike. The total-pressure recovery ratio is shown in figure 14 as a function of flight Mach number.

Range of Engine Variables

The effects on airplane performance of independently varying the following engine design variables over the indicated ranges were computed:

Compressor pressure ratio at sea level, zero ram	5 to 15
[1]Maximum compressor adiabatic efficiency	0.85 and 0.88
[1]Turbine efficiency	0.85 and 0.90
Equivalent rated air flow per unit of compressor frontal area (area swept by compressor tip), (lb/sec)/sq ft	18 to 36
Turbine-inlet temperature, °R	2000 and 2500
Zone afterburner temperature, °R	turbine outlet to 3500
Engine specific weight, lb/sq ft of compressor frontal area	400 to 800
Engine size, lb of air/(sec)(lb gross weight)	0.002 to 0.008

The following variables were held at the given constant values:

Primary-combustor efficiency	0.95
Ratio of primary-combustor total-pressure loss to inlet total pressure	0.05
Ratio of primary-combustor diameter to afterburner diameter . .	1.00
Ratio of inner to outer diameter of primary combustor	0.40
Ratio of outer diameter of primary combustor to compressor tip diameter	1.177
Ratio of afterburner friction pressure drop to inlet dynamic head	2
Afterburner combustion efficiency	0.85

For most of the calculations, use of a completely variable-area convergent-divergent exhaust nozzle was assumed. The effects on performance of using a variable-area convergent nozzle were computed for one particular engine configuration.

RESULTS AND DISCUSSION

Basic Engine

The first results of the analysis show the effect, for both flight plans, of changes in the basic engine on the bomber relative radius.

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[1]The compressor and turbine efficiencies were assumed paired, the low turbine efficiency being used with the low compressor efficiency.

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The basic engine is an afterburning engine with a rated sea-level compressor pressure ratio of 7, a maximum adiabatic compressor efficiency of 0.85, a turbine-inlet temperature of 2000° R, a turbine adiabatic efficiency of 0.85, an air-handling capacity of 27 pounds per second per square foot of compressor area, an engine specific weight of 650 pounds per square foot of compressor area, and a completely variable-area convergent-divergent nozzle.

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Effects of combat-zone afterburner temperature. - The effect of changes in zone afterburner temperature on bomber relative radius is shown in figure 2(a) for both flight plans considered. The relative radius plotted in figure 2(a) and many of the succeeding figures is computed by dividing the bomber radius for the engine configuration and flight plan under consideration by the maximum radius obtainable with the basic engine flying the all-supersonic flight plan. The lower curve of figure 2(a) represents the performance of a series of bombers having various zone afterburner temperatures and, hence, various engine sizes, flying an all-supersonic flight plan. The upper curve represents the performance of the same series of bombers flying the flight plan involving a 500-mile supersonic radius to and from the target, the remainder of the flight being made at subsonic speed. For both subsonic and supersonic flight, the airplane was flown at that altitude giving maximum range.

A change in zone afterburner temperature requires a change in engine size in order to sustain constant-speed supersonic flight, an increase in afterburner temperature necessitating a decrease in engine size. The relation between engine size (as expressed in terms of air flow per unit airplane gross weight) and zone afterburner temperature is not unique but is dependent on the flight altitude. The relation between zone afterburner temperature and engine size for the altitude giving maximum supersonic range is shown in figure 2(b). An engine that handles almost 0.007 pound of air per second per pound of airplane take-off gross weight is required for flight at a Mach number of 2 at the most favorable altitude if the afterburner is inoperative; whereas, an engine size of 0.0025 is sufficient with a zone afterburner temperature of about 3500° R.

As the engine size and weight decrease, with increased zone afterburner temperature, more fuel can be carried for a given airplane take-off gross weight, and the airplane drag is decreased because of the reduced size of the nacelles. Both the increased fuel on board and the decreased drag increase the bomber radius (fig. 2(a)). Detrimental to increased radius is the increased specific fuel consumption accompanying increased zone afterburner temperature. Increasing the zone afterburner temperature and decreasing the engine size result in increased radius of the bomber flying all supersonically until the zone afterburner temperature reaches 2500° R. For design afterburner temperatures above 2500° R, the engine size and weight are not reduced sufficiently to compensate the increased specific fuel consumption, and the airplane radius decreases. In many of the subsequent figures, an engine size corresponding to 0.004 pound of air per second per pound of airplane take-off gross weight is assumed, corresponding to a zone afterburner temperature of about 2500° R. 2913

Factors comparable to those that influenced the relation between relative radius and afterburner temperature for the all-supersonic flight plan are involved in the relation illustrated in the upper curve of figure 2(a). For the very low values of zone afterburner temperature, the engine is so large that it must be throttled back for subsonic cruise to a thrust level at which the specific fuel consumption is greater than the minimum. As the zone afterburner temperature is increased, the specific fuel consumption for subsonic cruise decreases and stays near its minimum value, the engine weight decreases, the fuel load increases, and the drag decreases. All the previously mentioned factors tend to produce an increased radius at subsonic speed. As a result, in contradistinction to the change in relative radius with zone afterburner temperature for the all-supersonic flight plan, wherein relative radius maximizes at some intermediate afterburner temperature, the total radius for the flight plan involving subsonic and 500 miles of supersonic cruise increases continuously with increasing afterburner temperature in the combat zone. At a zone afterburner temperature corresponding to maximum radius for the all-supersonic flight plan, the relative radius for the 500-mile supersonic flight plan is 1.228.

The results of figure 2(a) indicate that the most desirable zone afterburner temperature is dependent on the amount of supersonic flight involved in the flight plan. For all-supersonic flight, the radius is relatively insensitive to afterburner temperature, and the most favorable afterburner temperature increases as the amount of supersonic flight decreases. For an airplane that might be expected to fly both flight plans considered in figure 2(a), a good compromise value of zone afterburner temperature would be about 3000° R, corresponding to an engine size of approximately 0.003 pound of air flow per second per pound of airplane gross weight.

The influence on radius of the amount of fuel consumed and the distance covered in take-off, climb, and acceleration was not considered for the data of figure 2(a). However, when these factors were considered, it was found that, for zone afterburner temperatures up to 3000° R, the curves obtained were nearly

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coincident with those of figure 2(a). For zone afterburner temperatures above 3000° R, the engine size and thrust margins became so small that the airplane encountered difficulty in take-off, climb, and acceleration. Take-off distance is shown as a function of zone afterburner temperature in figure 2(c). The take-off distance was computed with the engines assumed to be operating at an afterburner temperature of 3500° R during the take-off operation. For high values of zone afterburner temperature, which correspond to small engine sizes, thrust margins are small and long take-off distances result. If take-off distances less than 5000 feet are required, the combat-zone afterburner temperature must be limited to 3000° R or less. Thus, the afterburner temperatues for maximum range on the all-supersonic flight plan and for acceptable take-off distance are nearly the same, and no compromise of take-off distance is required to obtain maximum all-supersonic range. The thrust margin in the transonic-speed range was also investigated for several afterburner temperatures in the range of interest. All airplanes investigated had sufficient thrust to accelerate through this second critical region.

Maneuverability in the combat zone, expressed as the normal load factor the airplane can sustain without loss of speed or altitude, is shown in figure 2(d). A load factor of unity corresponds to level, unaccelerated flight. The data of figure 2(d) are all for a maneuvering afterburner temperature of 3500° R; and, therefore, for a zone afterburner temperature of 3500° R, a load factor of unity is obtained. As the zone afterburner temperature is decreased, the maneuverability or normal load factor increases. Maneuverability at a zone afterburner temperature of 3500° R could be obtained by increasing the afterburner temperature above 3500° R during maneuvering, by the use of some other thrust-augmentation method, or by decreasing the flight altitude. Any maneuvering would result in a decrease in airplane radius. For a zone afterburner temperature of 3000° R, a normal load factor of approximately 1.3 can be obtained, at a flight Mach number of 2, by increasing the afterburner temperature to 3500° R.

Inasmuch as the radius for the all-supersonic flight plan was relatively insensitive to zone afterburner temperature, airplane performance for the all-supersonic flight plan was computed for the case of an engine without an afterburner. Because of the absence of the afterburner, it was assumed that the engine specific weight was decreased 16 percent. This nonafterburning-engine performance is shown in figures 2(a), (c), and (d) by the symbols. While the radius obtained for the nonafterburning case is nearly equal to the maximum radius obtainable for afterburning engines, the take-off distance is rather great (about 6600 ft, fig. 2(c)). The maneuverability or normal load factor is unity at the design Mach number and altitude.

From consideration of the effect of zone afterburner temperature on airplane performance it may be concluded that, for the basic engine, typical of current turbojet engines, an afterburner is desirable. If the flight plan is all supersonic, the airplane radius is relatively insensitive to afterburner temperature; although, if the engine has no afterburner, the take-off, climb, and acceleration performance is impaired. As the amount of subsonic flight in the flight plan increases, the zone afterburner temperature giving maximum radius increases. 2913

Effect of convergent nozzle. - The foregoing results were obtained with the engines assumed equipped with convergent-divergent nozzles of variable throat and variable expansion ratio. The substitution of a simple convergent nozzle with variable throat puts a great penalty on the range of the supersonic bomber, as shown in figure 3, where the radius of the bomber is plotted against the combat-zone afterburner temperature. At an afterburner temperature of 2500° R, there is approximately a 13-percent decrease in the radius of the bomber for the flight plan involving only 500 miles of supersonic radius. At the same afterburner temperature, there is a 24-percent decrease in radius for the bomber flying all supersonically at a Mach number of 2.

The engines for a given thrust requirement are considerably larger for a given afterburner temperature when a convergent nozzle is used in place of a convergent-divergent nozzle. Accordingly, when the afterburner temperature is increased to reduce the engine size and weight, more of an advantage in relative radius is obtainable with the convergent nozzle than with the convergent-divergent nozzle.

The previously discussed results have all been based on the assumption of an ideal, infinitely variable, convergent-divergent nozzle. If the losses in a recently investigated translating-plug-type nozzle are included, the radius for the all-supersonic flight plan is decreased about 13 percent, the radius for the flight plan involving 500 miles of supersonic flight is decreased approximately 10 percent, and the take-off distance is increased 2 percent.

Effect of rated compressor pressure ratio and compressor and turbine efficiencies. - The effect of changes in rated compressor pressure ratio and maximum compressor and turbine efficiencies on the relative radius of the bomber is shown in figure 4(a). For ease in calculation, a fixed engine size of 0.004 pound of air per second per pound of gross weight at take-off was assumed, with a resulting zone afterburner temperature of approximately 2500° R. As the pressure ratio and the efficiency of the compressor and turbine were changed, a small adjustment in afterburner temperature was necessary to match the airplane drag and the available thrust at the altitude for best range. Other values for the engine with a convergent-divergent nozzle were a turbine-inlet temperature of 2000° R, an air-handling capacity of 27 pounds per second per square foot of compressor area, an engine specific weight of 650 pounds per square foot of compressor area, and a supersonic Mach number in the combat zone of 2. The value of compressor pressure ratio shown on the abscissa is the pressure ratio of the engine at rated speed under sea-level static conditions.

For an engine equipped with a compressor having a maximum efficiency of 0.85 and a constant turbine efficiency of the same value, the relative radius for the 500-mile supersonic flight plan is relatively insensitive to changes in rated compressor pressure ratio from 7 to 15. For the bomber flying at supersonic speed throughout its entire flight, any rated compressor pressure ratio between 4 and 10 gives nearly maximum radius. The results of these calculations and the conclusions drawn from the data are all based on the supposition that the engine specific weight is unaffected by changes in compressor pressure ratio. Should this not be the case, the effect of compressor pressure ratio must be evaluated with the variation in engine specific weight with compressor pressure ratio considered. Such an evaluation can be made through the application of one of the succeeding figures.

When the component efficiencies are raised to a maxium of 0.88 for the compressor and 0.90 for the turbine, the compressor pressure ratio for maximum radius increases somewhat, although the effect is small for the all-supersonic flight plan. For this flight plan, the increase in component efficiencies increases the radius approximately 8 percent. The same increase in component efficiencies produced increases in radius for the 500-mile supersonic flight plan of 14 and 21 percent at rated compressor pressure ratios of 7 and 12, respectively.

Another factor to be considered in the selection of compressor pressure ratio and the evaluation of benefits to be derived from increased component efficiencies is their effect on combustor velocities. Both primary-combustor and afterburner velocities are shown as functions of compressor pressure ratio in figure 4(b). Afterburner velocities are well below 500 feet per second for all compressor pressure ratios and component efficiencies considered. A pressure ratio of at least 7 is required with the lower efficiency components in order to keep the primary-combustor inlet velocity below 100 feet per second. The data shown in figure 4 indicate that, for an engine with current component efficiencies, a compressor pressure ratio of about 7 gives nearly maximum radius for the all-supersonic flight plan. For the flight plan incorporating 500 miles of supersonic flight, a pressure ratio of 10 gives nearly maximum radius. These results are based on the assumption that engine specific weight is independent of compressor pressure ratio.

Effect of engine specific weight. - In the long-range bomber under consideration, the gross weight is held fixed and any decrease in weight of structure, engine, or fuel tank results in an increase in fuel-carrying capability. The effect of exchanging engine weight for fuel is shown in figure 5, where the bomber relative radius is plotted against rated compressor pressure ratio for the two flight plans considered and for engine specific weights of 400, 600, and 800 pounds per square foot of compressor area. Other constants for this data are engine size of 0.004 pound of air per second per pound of take-off gross weight, compressor efficiency of 0.85, turbine efficiency of 0.85, turbine-inlet temperature 2000° R, an ideal convergent-divergent exhaust nozzle, and air-handling capacity of 27 pounds per second per square foot of compressor area.

A decrease in engine specific weight from 600 to 400 pounds per square foot at a compressor pressure ratio of 7 increases the relative radius for each flight plan approximately 8.5 percent. For an engine specific weight of 650 pounds per square foot of compressor area, the engine weighs approximately 10 percent of the gross weight in the bomber, as compared with approximately 27 percent in the fighter of reference 1. Therefore, engine specific weight is considerably less important for the bomber than for the local defense fighter of reference 1. The large difference in the relative engine weight between the bomber and the fighter of reference 1 is chiefly the result of the greater maneuverability requirements of the fighter at the high-speed, high-altitude flight conditions. These large differences in engine weight indicate that the bomber engine specific weight may be relatively high and still not jeopardize performance. Specifically, if an increase of 3 points in efficiency of the compressor and 5 points in the efficiency of the turbine can be made with less than a 66-percent increase in engine specific weight, such a change is advantageous in providing increased radius for the bomber flying the flight plan involving 500 miles at supersonic speeds.

Effect of turbine-inlet temperature. - By increasing the turbine-inlet temperature, the same work can be extracted from the turbine with a decreased turbine pressure drop. With the turbine pressure drop diminished, the pressure ratio across the exhaust nozzle is increased, with the result, for a given exhaust-gas temperature, of an increase in engine thrust. The effect of increased turbine-inlet temperature on bomber performance is shown in figure 6(a), where relative radius for both bomber flight plans is plotted against rated compressor pressure ratio for two turbine-inlet temperatures. Two-percent bleed from the compressor was used to cool the turbine at a turbine-inlet temperature of 2500° R, in accordance with the analysis of reference 3. The bleed air, after cooling the turbine, was returned to the tail pipe.

For the flight plan with the 500-mile supersonic radius, the increase in turbine-inlet temperature from 2000° to 2500° R has a relatively slight effect on radius for rated compressor pressure ratios up to about 9. For higher pressure ratios the effect is increased, and a compressor pressure ratio of 12 and a turbine-inlet temperature of 2500° R provide a radius approximately 7 percent greater than the maximum attainable for a turbine-inlet temperature of 2000° R.

2913 For the all-supersonic flight plan, the effect of increased turbine-inlet temperature is greater at all values of compressor pressure ratio. The compressor pressure ratio giving maximum radius increases from about 7 at a turbine-inlet temperature of 2000° R to approximately 12 at a turbine-inlet temperature of 2500° R. The maximum radius at a turbine-inlet temperature of 2500° R is 20 percent greater than the maximum at a turbine-inlet temperature of 2000° R.

The combined effects of changes in turbine-inlet temperature and compressor pressure ratio on the afterburner-inlet velocity are shown in figure 6(b). For compressor pressure ratios of interest for a supersonic bomber, between 7 and 12, there is little effect of the turbine-inlet temperatures investigated on the afterburner-inlet velocities, and the velocities are all low enough to afford efficient combustion.

<u>Effect of air-handling capacity</u>. - An increase in the air-handling capacity improves the performance of the airplane for two reasons: (1) the increase reduces the drag of the engine nacelles, and (2) a decrease in the area for a given air flow will decrease the engine weight proportionately, because engine weight is assumed to be directly proportional to the engine compressor area. In the present investigation, all values of air-handling capacity cited are the values at rated sea-level static conditions.

The effect of air-handling capacity on the radius of the bomber is shown in figure 7(a) for three rated compressor pressure ratios and the two flight plans under consideration. The other engine variables are the same as for the basic engine, and the engine size corresponds to 0.004 pound of air per second per pound of take-off gross weight, with a zone afterburner temperature of approximately 2500° R. The increase in radius with increased air-handling capacity is about the same for the three pressure ratios investigated. Increasing the compressor air flow from 27 to 33 pounds per second per square foot increases the radius for the flight plan involving the 500-mile supersonic radius approximately 8 percent; for the all-supersonic flight plan, the increase is about 11 percent. Relative radius increased at a reduced rate with an increase in air-handling capacity, because a given increment of air-handling capacity has less effect on both the airplane drag and the engine weight at high air-handling capacities.

Increasing the air-handling capacity has an adverse effect on velocities throughout the engine. Figure 7(b) indicates that, for a rated compressor pressure ratio of 7, the air-handling capacity is limited to about 27 pounds per square foot of compressor frontal area to maintain the primary-combustor inlet velocity below 100 feet per second. At a pressure ratio of 12, a velocity of a 100 feet per second is not reached until the air-handling capacity of the engine has exceeded 40 pounds per second per square foot of compressor area. For air-handling capacities below 36 pounds per second per square foot, the afterburner-inlet velocity is less than 500 feet per second. 2913

Advanced Engine

The foregoing sections of this report indicate the effects of independent changes in afterburner temperature, component efficiency, turbine-inlet temperature, compressor pressure ratio, type of exhaust nozzle, air-handling capacity, and engine specific weight on the performance of the supersonic bomber. Several of the previously discussed beneficial effects are combined in this section into what might be called an "advanced engine," that is, an engine realizing future scheduled developments in the performance of several of the components. The advanced engine had a maximum compressor efficiency of 0.88, a turbine-inlet temperature of 2500° R, a turbine efficiency of 0.90, a convergent-divergent nozzle, and an air-handling capacity of 33 pounds per second per square foot. The rated compressor pressure ratio and zone afterburner temperature were varied to determine the effect of changes in them on the relative radius of a bomber equipped with advanced engines.

<u>Effects of rated compressor pressure ratio and engine specific weight</u>. - The effects of rated compressor pressure ratio and engine specific weight on the radius of the supersonic bomber are shown in figure 8 for an engine size of 0.004 pound of air per second per pound of gross weight. As previously illustrated for a turbine-inlet temperature of 2500° R, the data in this figure show a continuous increase in relative radius with increasing rated compressor pressure ratio for the flight plan involving 500 miles of supersonic radius. An increase in compressor pressure ratio from 7 to 12 at a fixed engine weight of 600 pounds per square foot results in an increase in the relative radius of more than 13 percent. There is much less effect for the all-supersonic flight plan, and little increase in radius can be gained by increasing the rated compressor pressure ratio above 9.

All these results were calculated on the basis of engine specific weight invariant with compressor pressure ratio. The effect of engine specific weight is smaller for the advanced engine than for the basic engine, because the advanced engine weighs less for a given thrust. For example, a 33-percent increase in engine weight decreases the relative radius 6 percent. This compares with a 9-percent change in radius for the same change in specific weight of the basic engine.

Effect of combat-zone afterburner temperature. - The effect of afterburner temperature on bomber radius when the bomber is equipped with an advanced afterburning engine having a rated compressor pressure ratio of 12 is shown in figure 9. Performance with basic engines having a compressor pressure ratio of 7 is also included for comparison. For the advanced engine and the flight plan involving 500 miles of supersonic cruise, the total radius is relatively insensitive to the afterburner temperature used in the combat zone. For the all-supersonic flight plan, the nonoperative afterburner provides maximum radius. With the advanced engine the gains afforded by afterburning, that is, smaller engines and reduced airplane drag, do not balance the increased fuel consumption of the afterburner, inasmuch as the engines are relatively small.

From a comparison of maximum radii in all cases, it can be seen from figure 9 that, for both flight plans, the advanced engine provides increases in radius of about 36 percent over that for the basic engine. These maximum radii occur at widely differing values of zone afterburner temperature. The increase in radius is greater than the sum of the increases attributable to the individual changes.

Comparison of advanced engines with and without afterburner. - Because radius for both flight plans is near maximum when cruising in the combat zone with the afterburner inoperative, an investigation was made to determine the effect of compressor pressure ratio on bomber performance for an advanced engine without the afterburner. The engine specific weight was reduced 16 percent to account for the removal of the afterburner. The results are shown for the two different flight plans in figure 10. For both flight plans the radius increases continuously with an increase in rated compressor pressure ratio. The foregoing results were computed with the assumption of a constant engine specific weight, and the engine size was selected for the altitude giving maximum supersonic range.

A comparison of figures 9 and 10 indicates that, for the advanced engine (compressor pressure ratio of 12), removing the afterburner provides a radius for the 500-mile supersonic flight plan only 4 percent less than the maximum afterburning radius. For the all-supersonic flight plan, the maximum radius, which was shown in figure 9 to occur with the afterburner inoperative, is further increased about 3.5 percent by removing the afterburner. A further comparison of the bomber powered by advanced afterburning and nonafterburning engines was made to determine the effect on take-off distance and maneuverability. The afterburning engine was sized so as to require an afterburner temperature of 2800° R in the combat zone and was operated at an afterburner temperature of 3500° R during take-off and maneuvering. Such a choice favored best performance for the 500-mile supersonic flight plan at the expense of all-supersonic range, take-off distance, and maneuverability. The lower thrust per pound of air flow obtained from the engine without the afterburner required an engine size 43 percent larger than that for the engine with the afterburner. The airplane take-off distance and maneuverability are as follows:

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Engine	No afterburner	Afterburner
Take-off distance, ft	4300	4600
Maneuverability, normal load factor	1.0	1.4

Effect of Flight Mach Number

The effect of flight Mach number on the relative radius was investigated for two different bombers, one equipped with the basic engine and the other with an advanced engine. The engine sizes were 0.004 and 0.003 pound of air per second per pound of take-off gross weight, respectively. The entire cruise portion of the flight was made at constant Mach number. The results are shown in figure 11. The change in relative radius with flight Mach number is the same for both bombers. Maximum range is afforded at a Mach number of 0.9. Maximum range at supersonic speeds occurs at a Mach number slightly above 2.0.

At a Mach number of 2.0, a bomber equipped with advanced engines will fly 27 percent farther than one using the basic engine. The specific fuel consumption of the advanced engine is about 14 percent less than that of the basic engine. The 25-percent reduction in engine weight increases the amount of fuel on board and adds another 11 percent to the radius. The improved lift-drag ratio due to smaller engines accounts for the remaining 2-percent improvement in radius. In the subsonic region, at a Mach number of 0.9, the relative range is about 43 percent better for the advanced engine. The greater improvement in the subsonic region is due principally to the fact that the specific fuel consumption for the advanced engine with a rated compressor pressure ratio of 12 and a turbine-inlet temperature of 2500° R is about 25 percent less during throttled operation than for the basic engine. The remainder of the improvement is due to the decreased engine weight and slightly improved lift-drag ratio.

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Effect of Bomber Gross Weight

The effect of gross weight on the relative radius of the airplane powered by a basic and by an advanced afterburning engine is compared in figure 12. The basic engine size was held constant at 0.004 pound of air per second per pound of airplane gross weight. The advanced engine was sized so that no afterburning would be required in the combat zone. The results of figure 12 indicate a consistent increase in relative radius as the bomber gross weight is increased. The rate of increase of the relative radius decreases, however, with increasing gross weight, inasmuch as the bomb load becomes a decreasing percentage of the gross weight. When the gross weight is doubled from 100,000 to 200,000 pounds, the total radius for the bomber flying the 500-mile supersonic flight plan increases 13 percent for both engines. The effect of an engine parameter such as compressor pressure ratio or afterburner temperature on airplane performance was shown in reference 1 to be relatively independent of airplane gross weight. Therefore, although presented in this report for an airplane gross weight of 150,000 pounds, the results are applicable to airplanes having somewhat different gross weights.

SUMMARY OF RESULTS

An analysis was made of the turbojet engine as the power plant for a bomber-type airplane having a design supersonic Mach number of 2. For turbojet engines having currently available components, airplane radius, for an all-supersonic flight plan, is relatively insensitive to afterburner temperature. For a flight plan involving a relatively large amount of subsonic flight, the total radius increases as the zone afterburner temperature is increased. Regardless of the amount of supersonic cruise, in order to ensure sufficient thrust during take-off, climb, and acceleration, the afterburner temperature in level, unaccelerated supersonic flight should probably not exceed 3000° R.

Based on the assumption that engine specific weight is independent of compressor pressure ratio, the airplane radius for currently available components is relatively insensitive to compressor pressure ratio. For an all-supersonic flight plan, compressor pressure ratios up to 9 give nearly maximum performance. If the flight plan has only a 500-mile radius at supersonic speeds with the remainder of the flight at subsonic velocities, the compressor pressure ratio giving maximum radius is approximately 12.

The use of a variable-area convergent nozzle rather than a variable-area convergent-divergent nozzle penalizes the airplane radius as much as 24 percent, depending on the amount of supersonic flight included in the flight plan.

Engine specific weight has a relatively small effect on bomber-type airplane performance, with a 64-percent change in engine specific weight changing the total radius about 18 percent for a flight plan involving 500 miles at a Mach number of 2. This change in specific weight has about the same effect on airplane radius as a change of about 4 points in the efficiencies of both the compressor and turbine.

Increasing the turbine-inlet temperature from 2000° to 2500° R has a beneficial effect on airplane radius, especially at the higher values of compressor pressure ratio. For a flight plan incorporating 500 miles at a Mach number of 2, a compressor pressure ratio of 12 and a turbine-inlet temperature of 2500° R gave a radius 7 percent greater than the maximum obtained for a turbine-inlet temperature of 2000° R. For an all-supersonic flight plan, the radius may be increased as much as 20 percent for the same increase in turbine-inlet temperature and compressor pressure ratio. 2913

An increase in air-handling capacity from 27 to 33 pounds per second per square foot of frontal area increased the total radius 8 percent for the flight plan involving 500 miles at supersonic speeds. For the all-supersonic flight plan, the increase in radius was 11 percent.

For an advanced engine (one embodying simultaneous improvements in turbine-inlet temperatures, component efficiencies, and air-handling capacity), the airplane maximum radius was about 36 percent greater than that obtained for an engine embodying current engine components. This increase in radius was essentially independent of the amount of supersonic cruise incorporated in the flight plan. The combined effect of several improvements in the engine components is greater than the sum of the individual effects.

For the advanced engine, removing the afterburner gave near-maximum performance for both flight plans considered. However, the airplane had the disadvantage of having no maneuverability at the design supersonic flight condition.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, January 28, 1954

APPENDIX - AERODYNAMIC ASSUMPTIONS

For ease in computing the drag of the bomber, it was assumed that the aircraft consisted of a delta-wing design with a low-drag fuselage and with engines mounted in nacelles. Total drag was computed from the superposition of the wing, fuselage, and nacelle drag. no interference

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Fuselage

The fuselage was assumed to be a low-drag body, 8 feet in diameter, with a fineness ratio of 12. The surface to frontal area was taken as 34.5. The pressure drag coefficient was computed from figures 30 and 32 of reference 1. Variations of friction drag and pressure drag coefficient with Mach number were assumed as in reference 1.

Nacelles

The engine nacelles were assumed to have a length-diameter ratio of 9. They were made up of three longitudinally equal sections: the cowl, a truncated cone; the center section, cylindrical; and the aft section, cylindrical with a 7° boattail. The pressure drag coefficients for the cowl and aft sections were computed from figures 30, 33, and 34 of reference 1. The friction drag coefficient for the entire nacelle was taken from figure 31 of reference 1.

Wing

The delta wing assumed for the bomber had an aspect ratio 2.0 and a thickness-chord ratio of 0.05. The zero-lift pressure drag coefficient for the wing was assumed to have a constant value of 0.0058 above a Mach number of 1.0. The variation of the induced-drag factor (change in drag coefficient divided by the square of the lift coefficient) is shown in figure 13. The data were obtained from references 4 and 5. The variation in the friction drag coefficient with Mach number is shown in figure 31 of reference 1. The tail drag was assumed to be 20 percent of the wing zero-lift drag. NG unless wings are cambered due to lam. separation

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3. Schramm, Wilson B., and Nachtigall, Alfred J.: Analysis of Coolant-Flow Requirements for an Improved, Internal-Strut-Supported, Air-Cooled Turbine-Rotor Blade. NACA RM E51L13, 1952.

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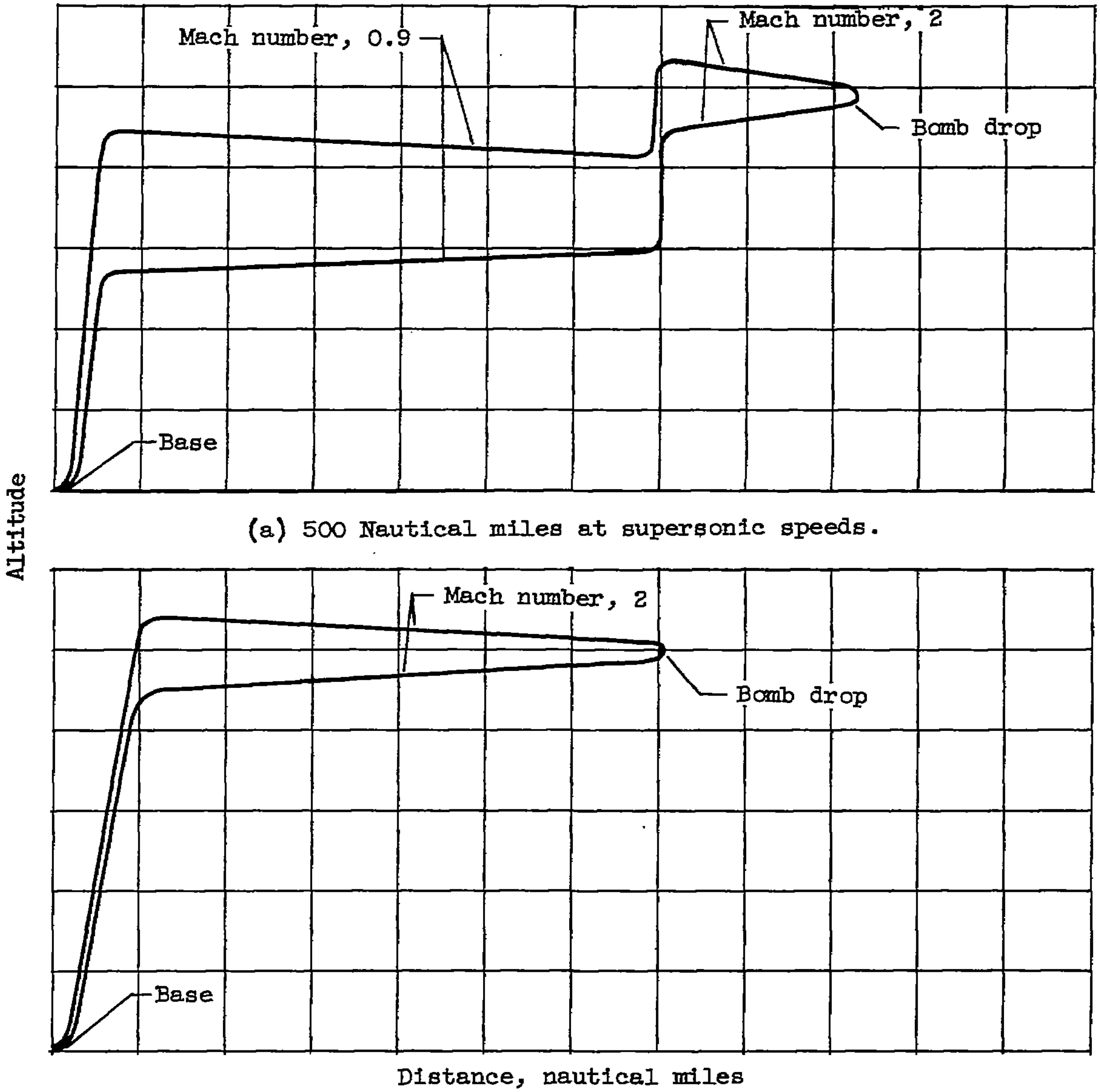
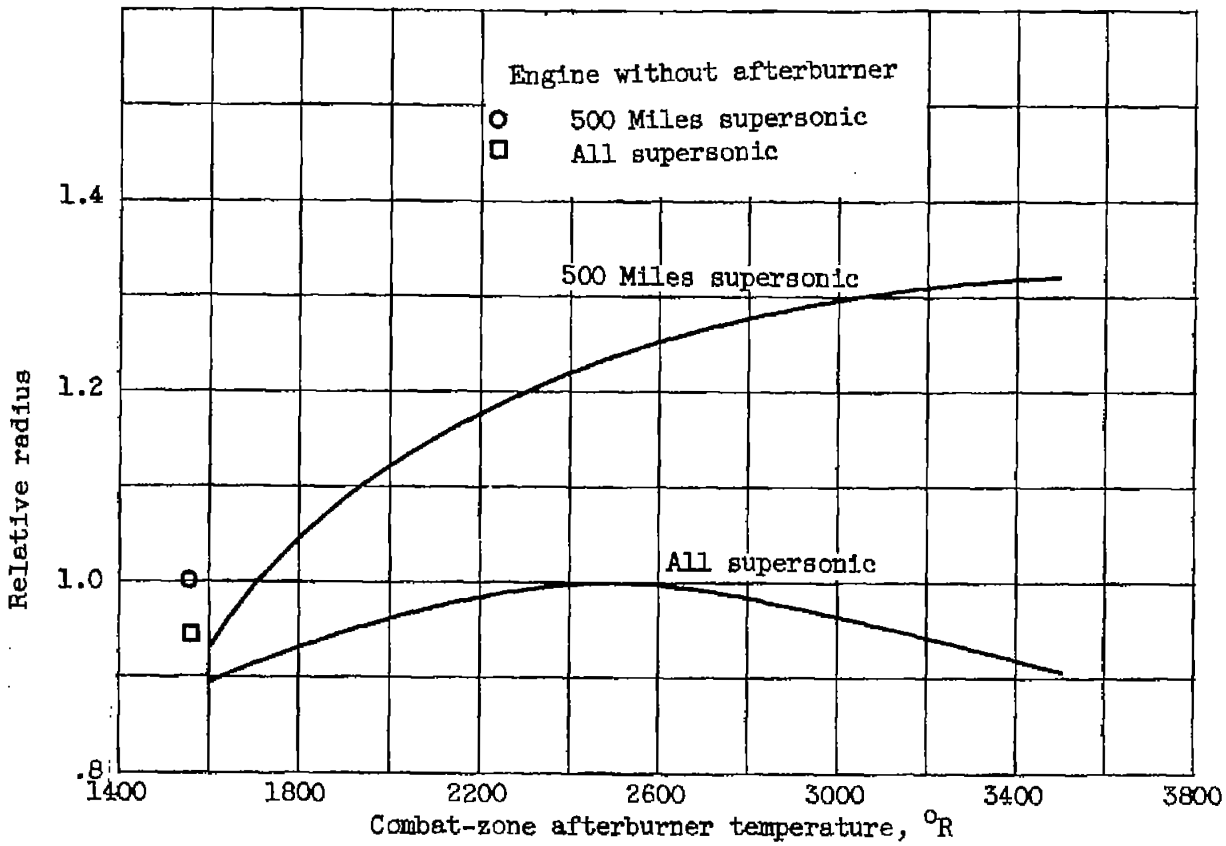


(b) All supersonic.

Figure 1. - Flight plans.



(a) Bomber relative radius.

Figure 2. - Effects of combat-zone afterburner temperature. Rated compressor pressure ratio, 7; maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; convergent-divergent nozzle; air-handling capacity, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; combat-zone Mach number, 2.

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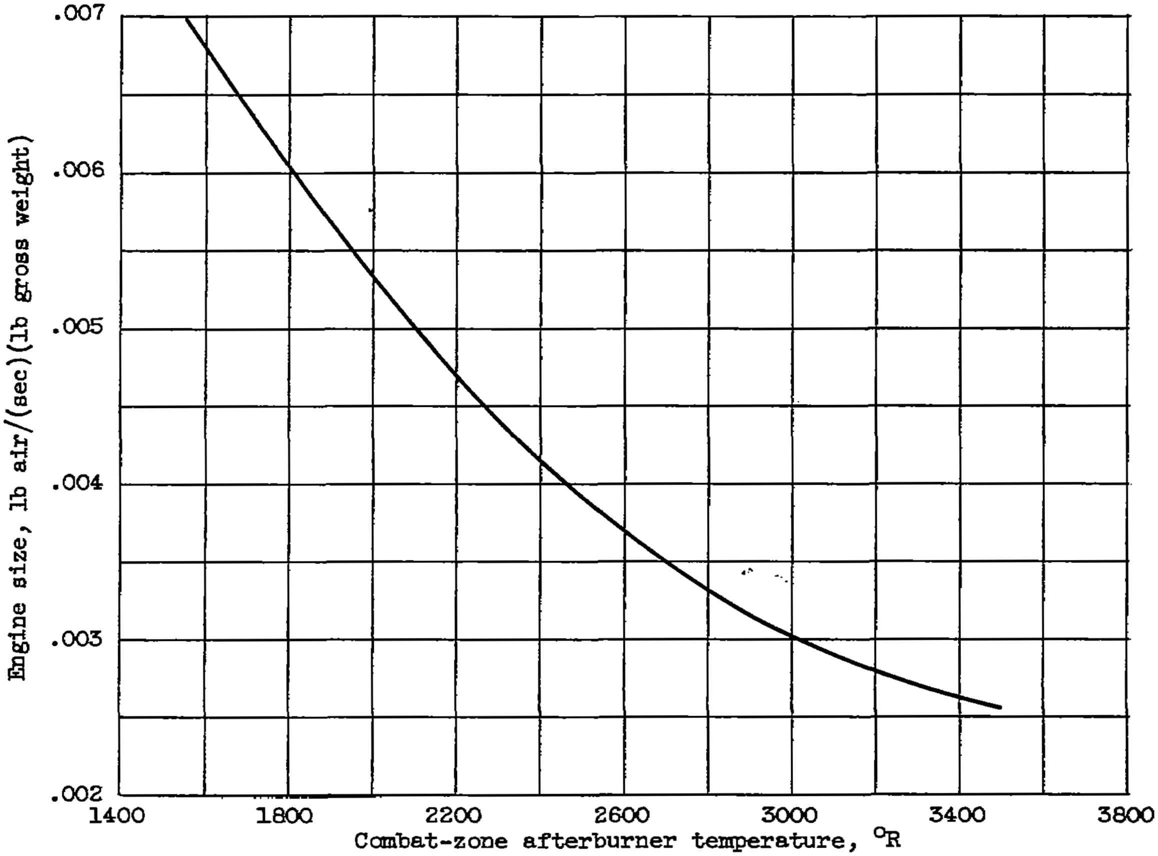

(b) Engine size.

Figure 2. - Continued. Effects of combat-zone afterburner temperature. Rated compressor pressure ratio, 7; maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; convergent-divergent nozzle; air-handling capacity, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; combat-zone Mach number, 2.

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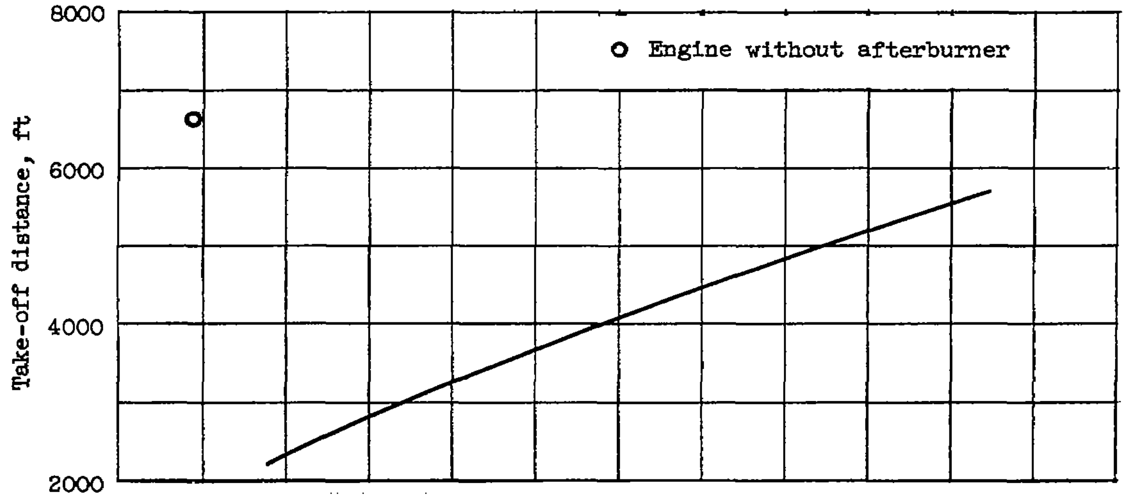
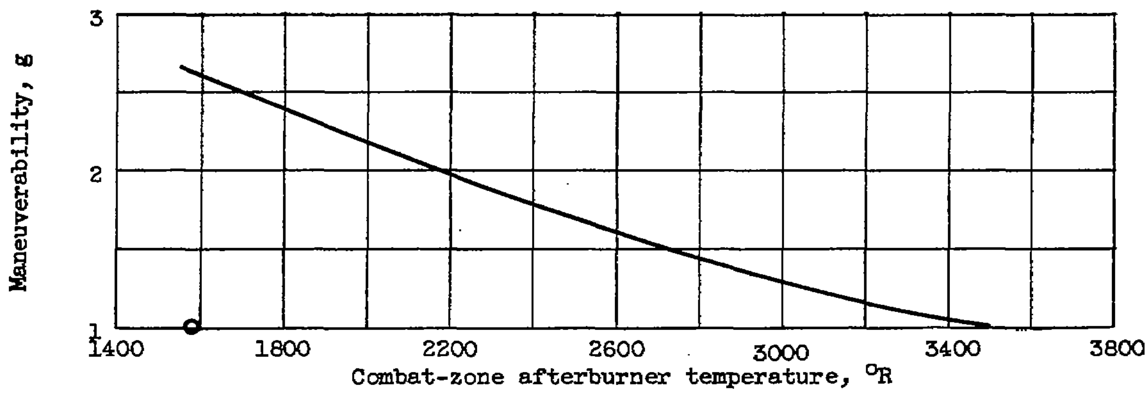


(c) Bomber take-off distance.



(d) Bomber maneuverability.

Figure 2. - Concluded. Effects of combat-zone afterburner temperature. Rated compressor pressure ratio, 7; maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; convergent-divergent nozzle; air-handling capacity, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; combat-zone Mach number, 2.

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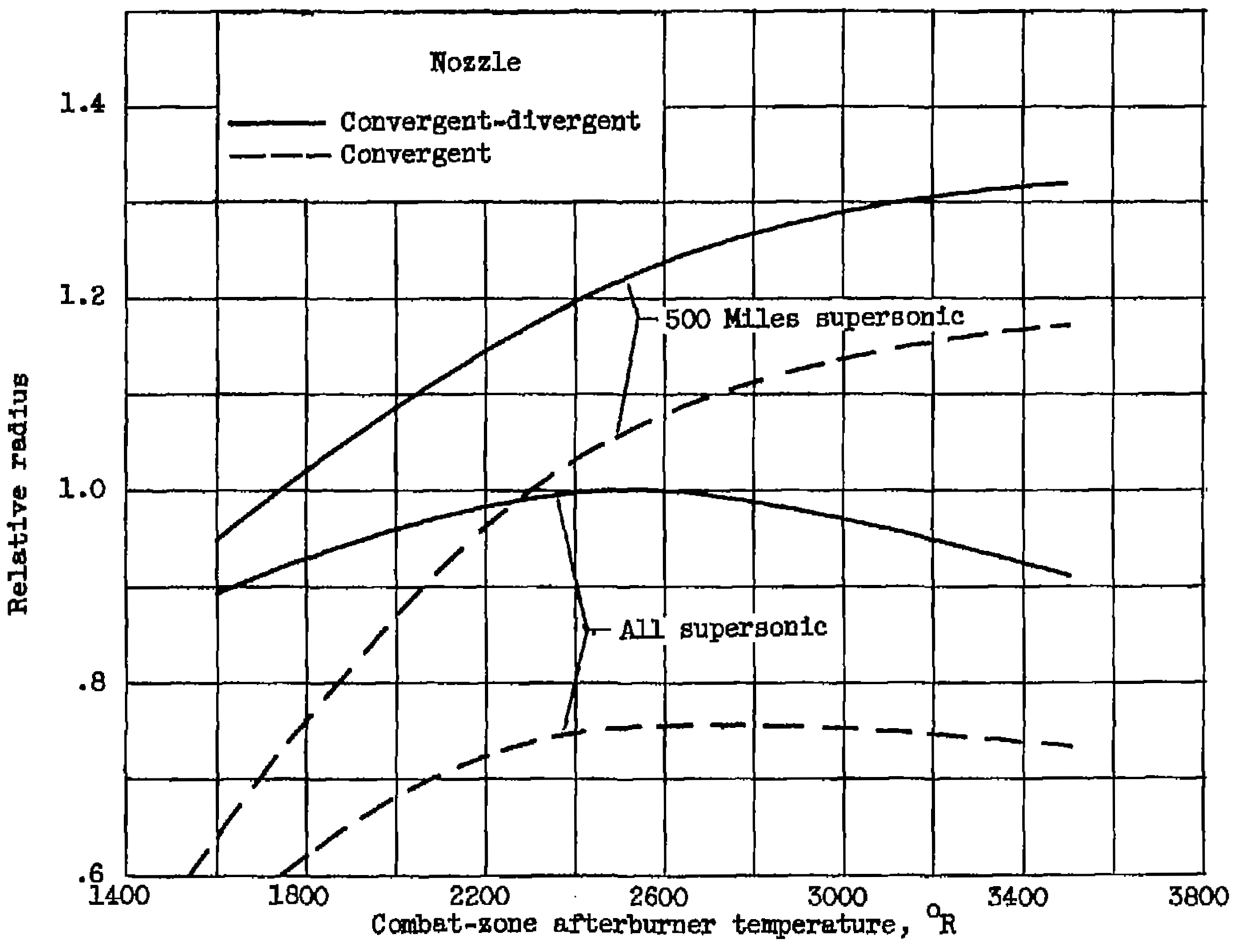

Figure 3. - Comparative radius of bomber with convergent and convergent-divergent exhaust nozzles. Rated compressor pressure ratio, 7; maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; air-handling capacity, 27 pounds per second per square foot; engine specific weight, 650 pounds per square foot; combat-zone Mach number, 2.

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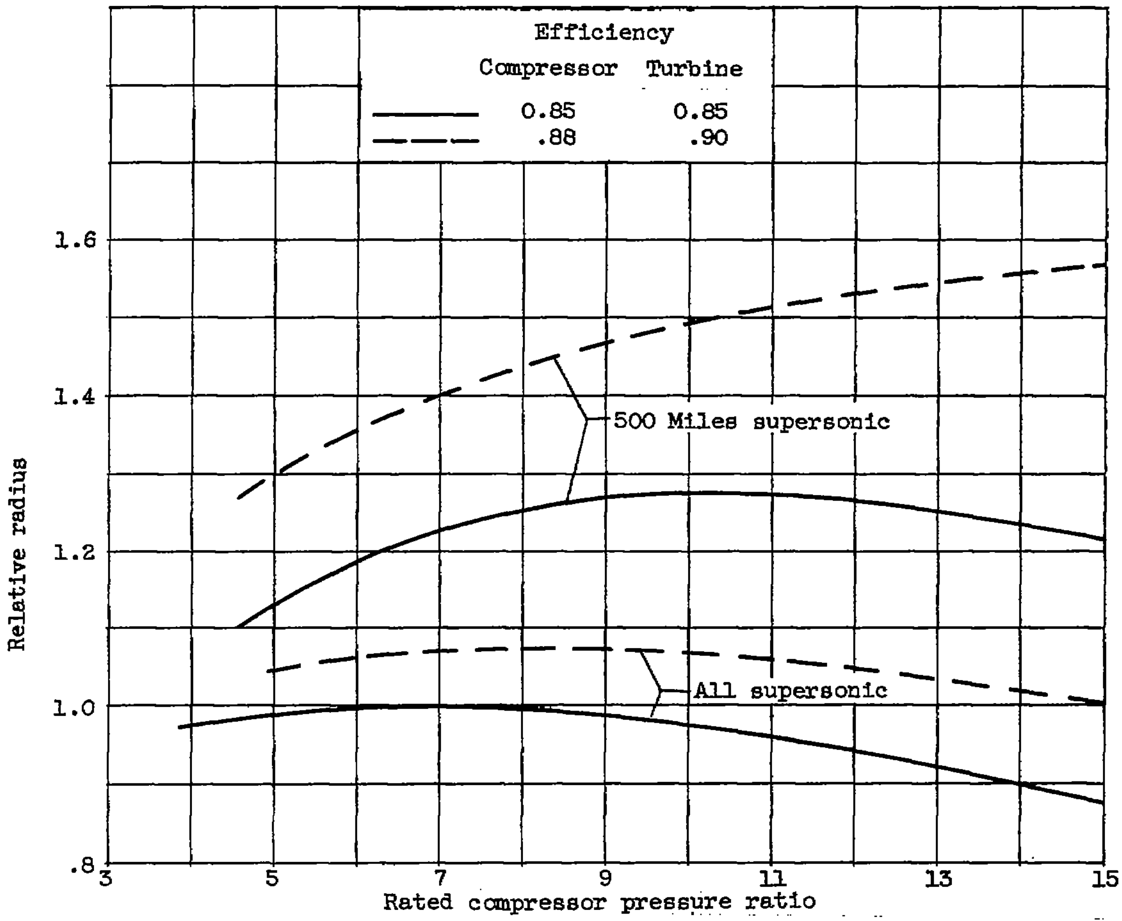
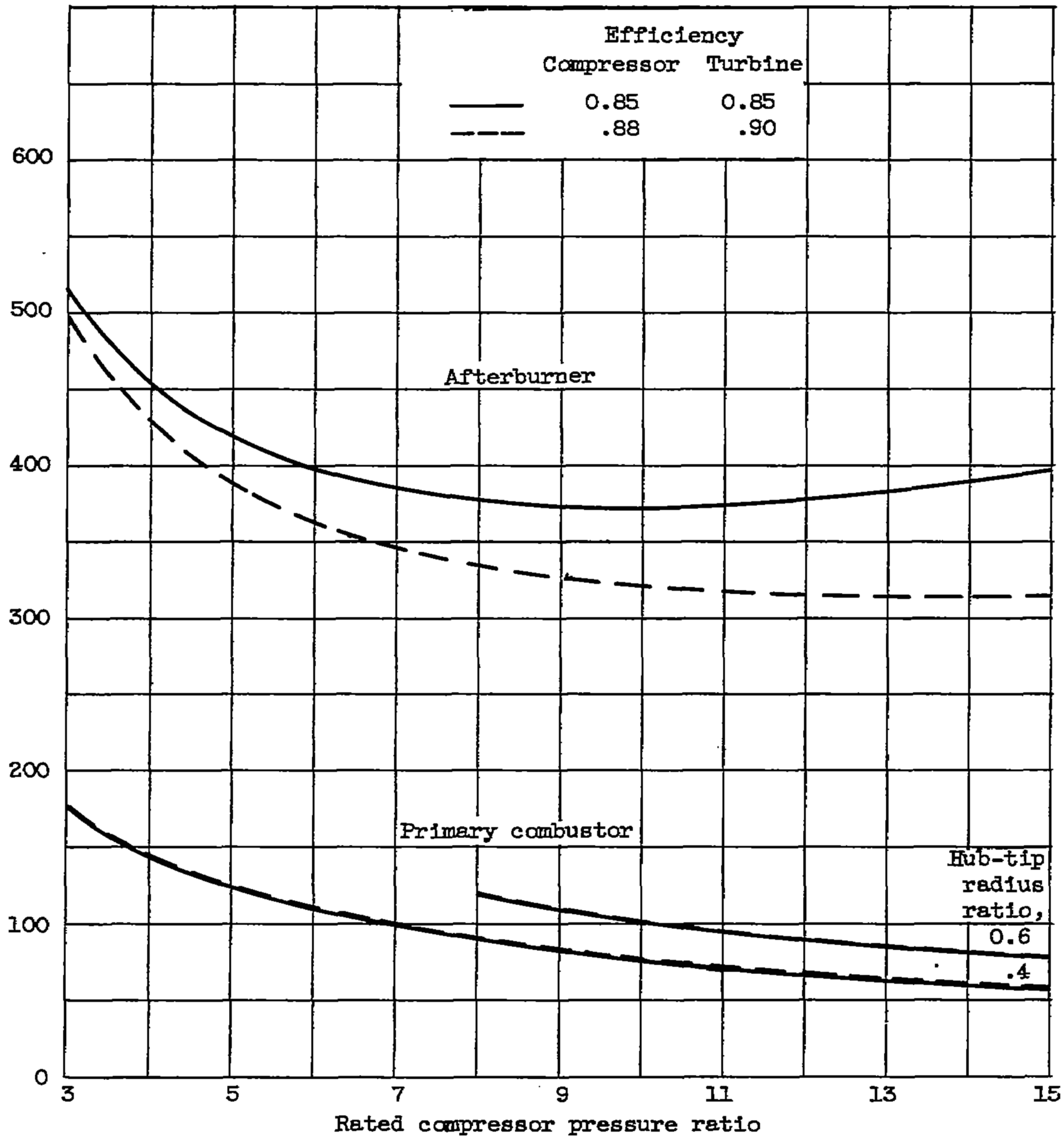


(a) Bomber relative radius. Convergent-divergent nozzle; engine specific weight, 650 pounds per square foot; engine size, 0.004 pound per second per pound gross weight.

Figure 4. - Effects of compressor pressure ratio and compressor and turbine efficiencies. Turbine-inlet temperature, 2000° R; air-handling capacity, 27 pounds per second per square foot; combat-zone Mach number, 2.



(b) Combustion-chamber velocities.

Figure 4. - Concluded. Effects of compressor pressure ratio and compressor and turbine efficiencies. Turbine-inlet temperature, 2000° R; air-handling capacity, 27 pounds per second per square foot; combat-zone Mach number, 2.

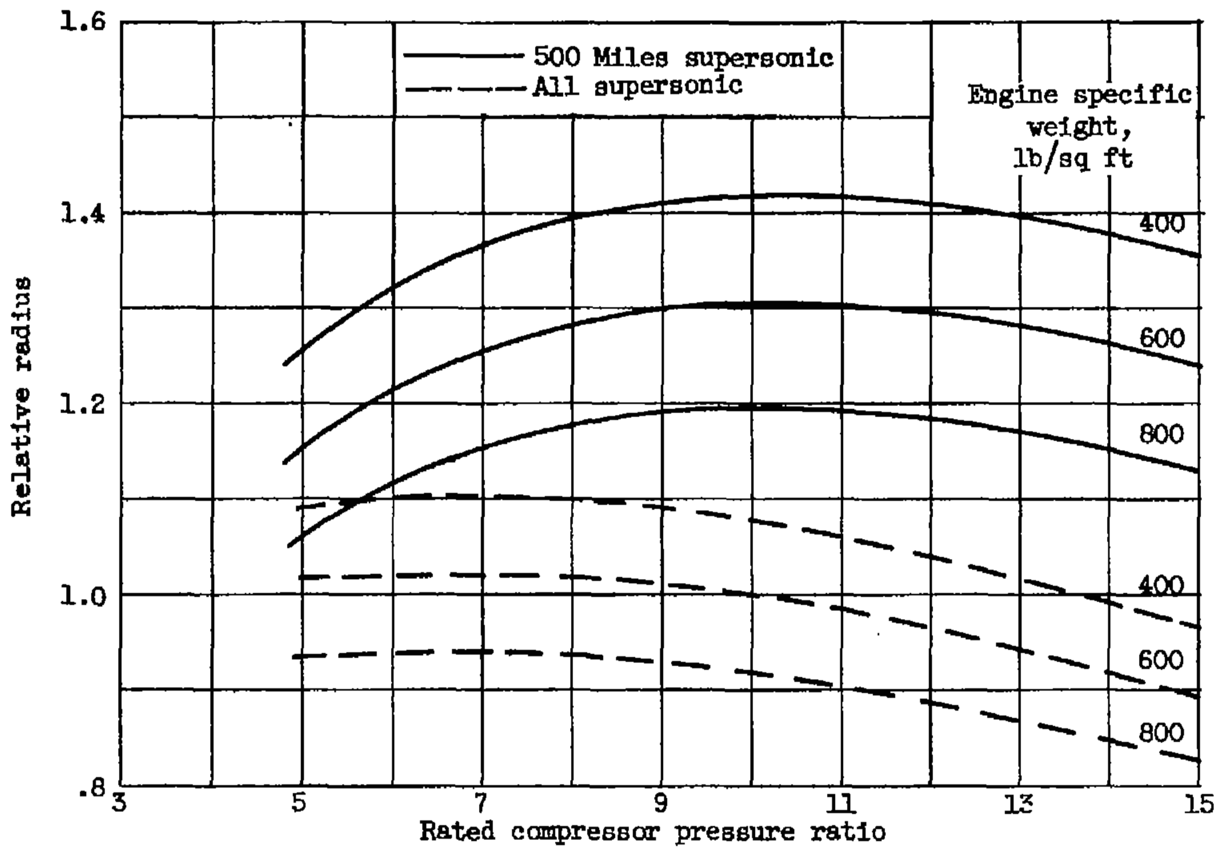
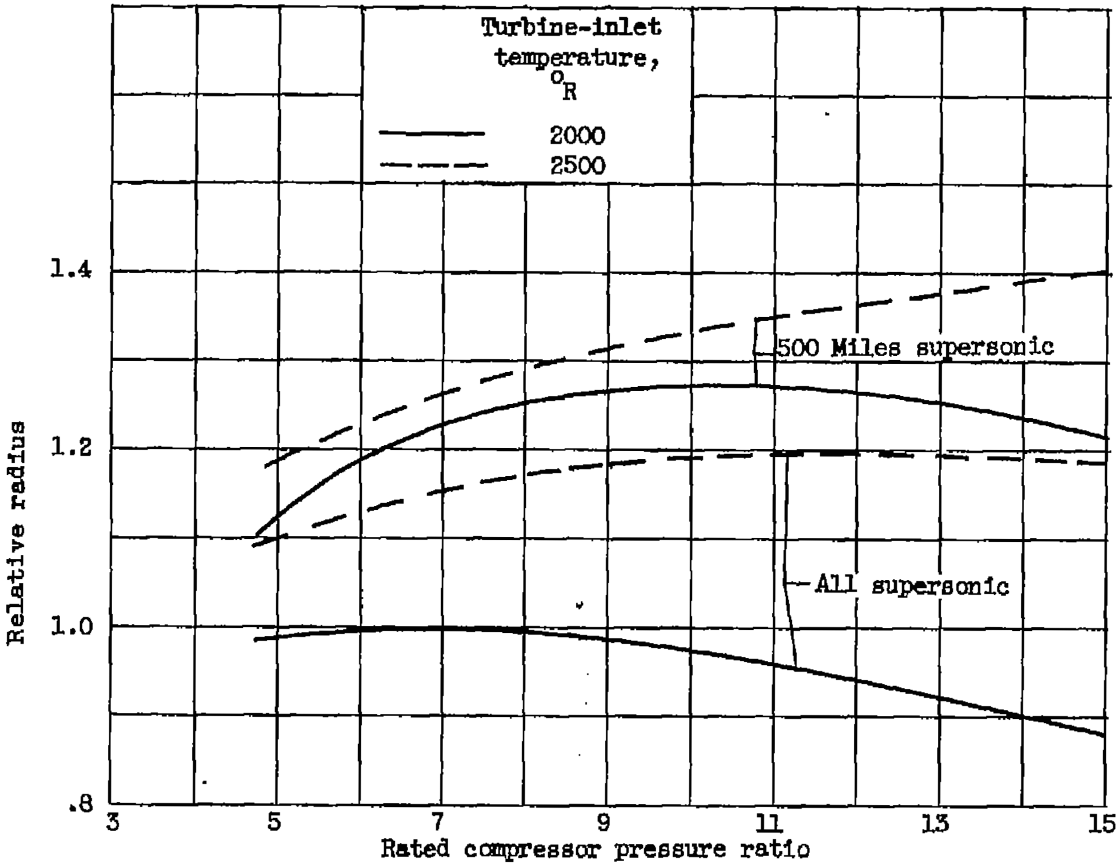
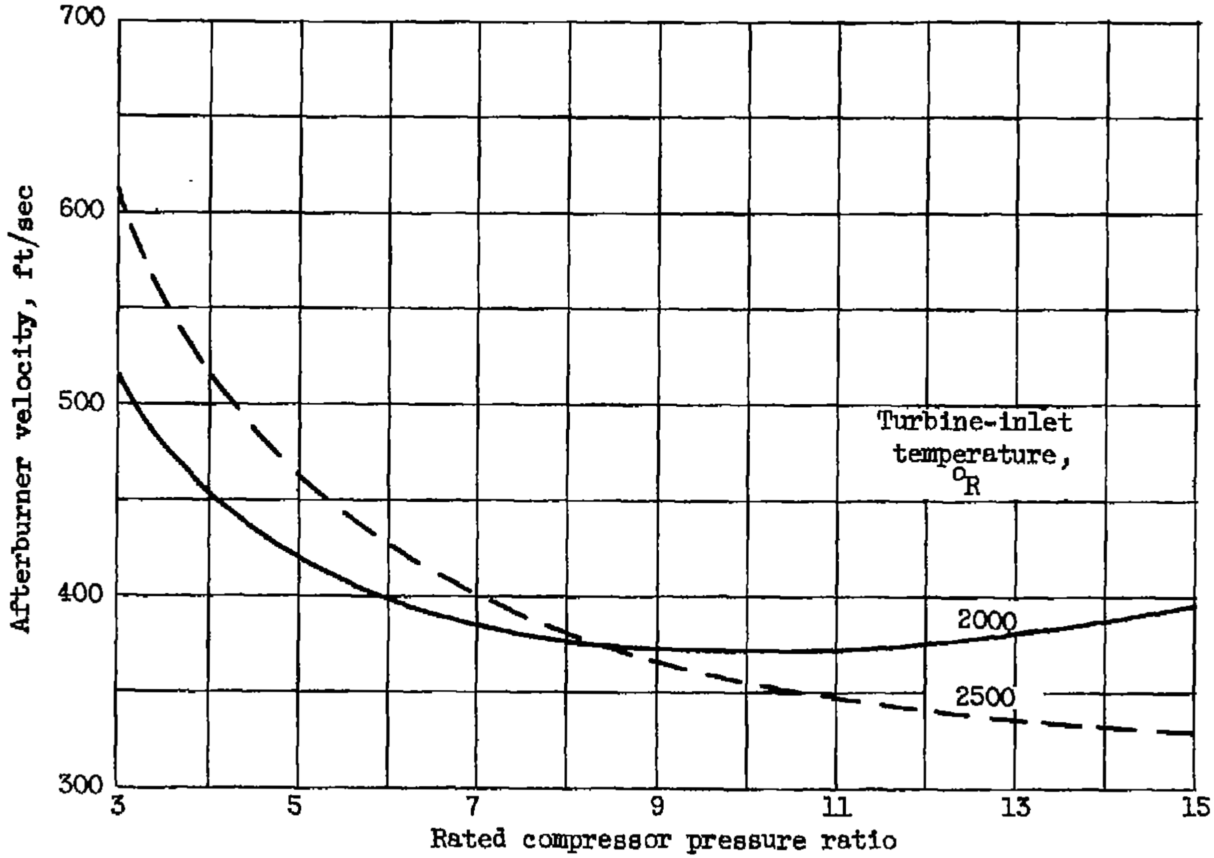


Figure 5. - Effect of engine specific weight and compressor pressure ratio on bomber relative radius. Maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; air-handling capacity, 27 pounds per second per square foot; convergent-divergent nozzle; combat-zone Mach number, 2; engine size, 0.004 pound per second per pound gross weight.



(a) Bomber relative radius. Convergent-divergent nozzle; engine specific weight, 650 pounds per square foot; engine size, 0.004 pound per second per pound gross weight.

Figure 6. - Effects of turbine-inlet temperature and compressor pressure ratio. Maximum compressor efficiency, 0.85; turbine efficiency, 0.85; air-handling capacity, 27 pounds per second per square foot; combat-zone Mach number, 2.



(b) Afterburner velocity.

Figure 6. - Concluded. Effects of turbine-inlet temperature and compressor pressure ratio. Maximum compressor efficiency, 0.85; turbine efficiency, 0.85; air-handling capacity, 27 pounds per second per square foot; combat-zone Mach number, 2.

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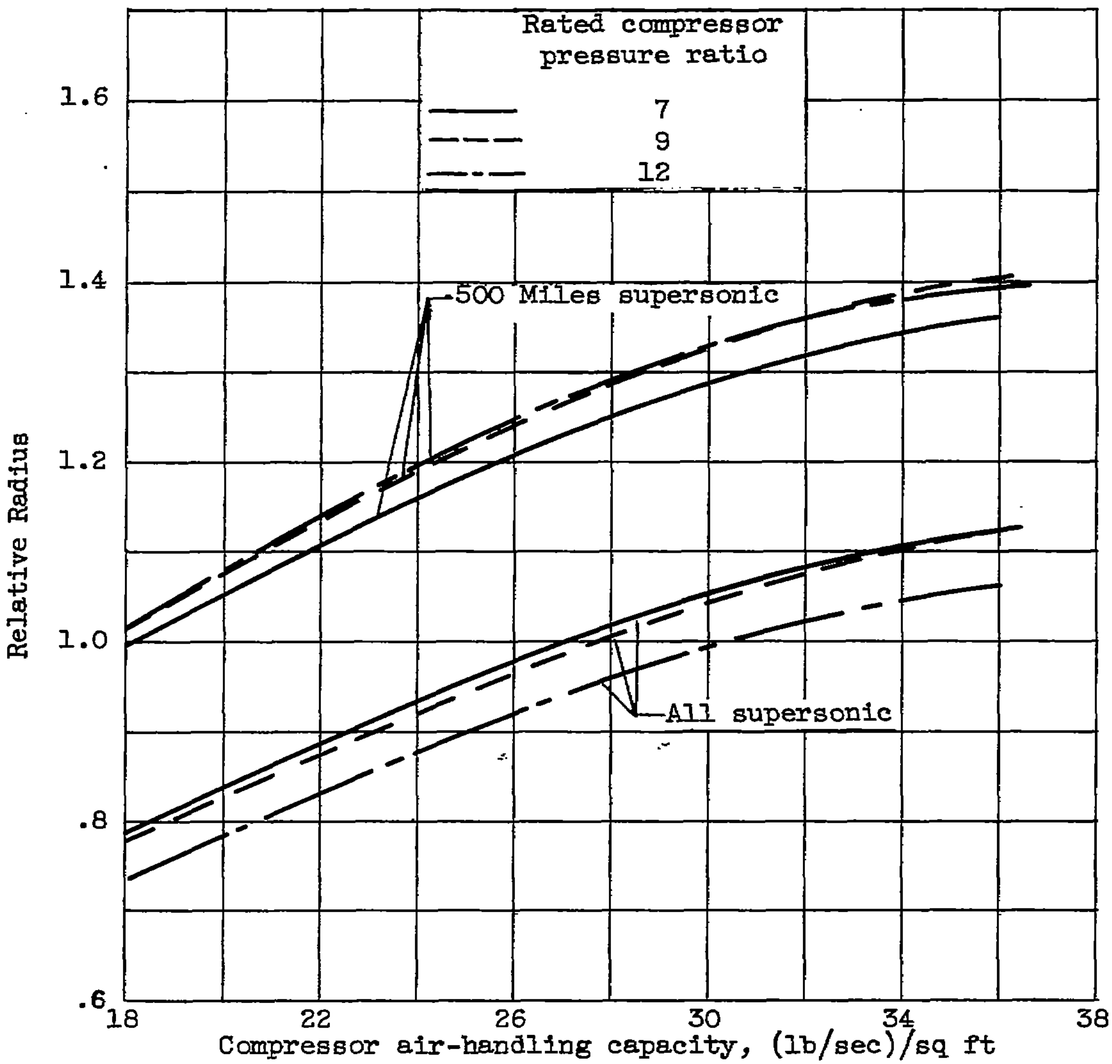

(a) Bomber relative radius. Convergent-divergent nozzle; engine specific weight, 650 pounds per square foot; engine size, 0.004 pound per second per pound gross weight.

Figure 7. - Effects of air-handling capacity and compressor pressure ratio. Maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; combat-zone Mach number, 2.

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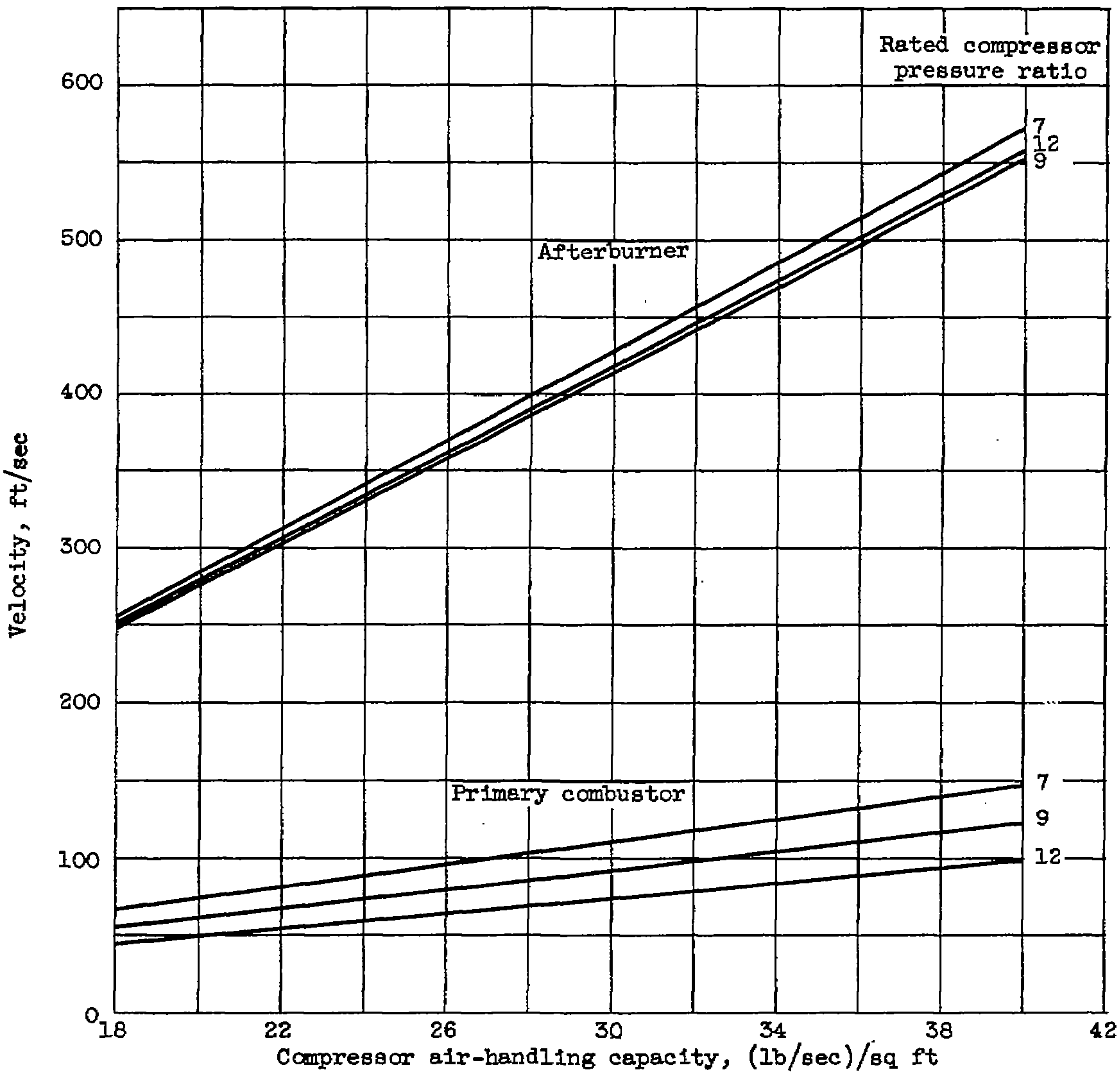

(b) Afterburner velocity.

Figure 7. - Concluded. Effects of air-handling capacity and compressor pressure ratio. Maximum compressor efficiency, 0.85; turbine efficiency, 0.85; turbine-inlet temperature, 2000° R; combat-zone Mach number, 2.

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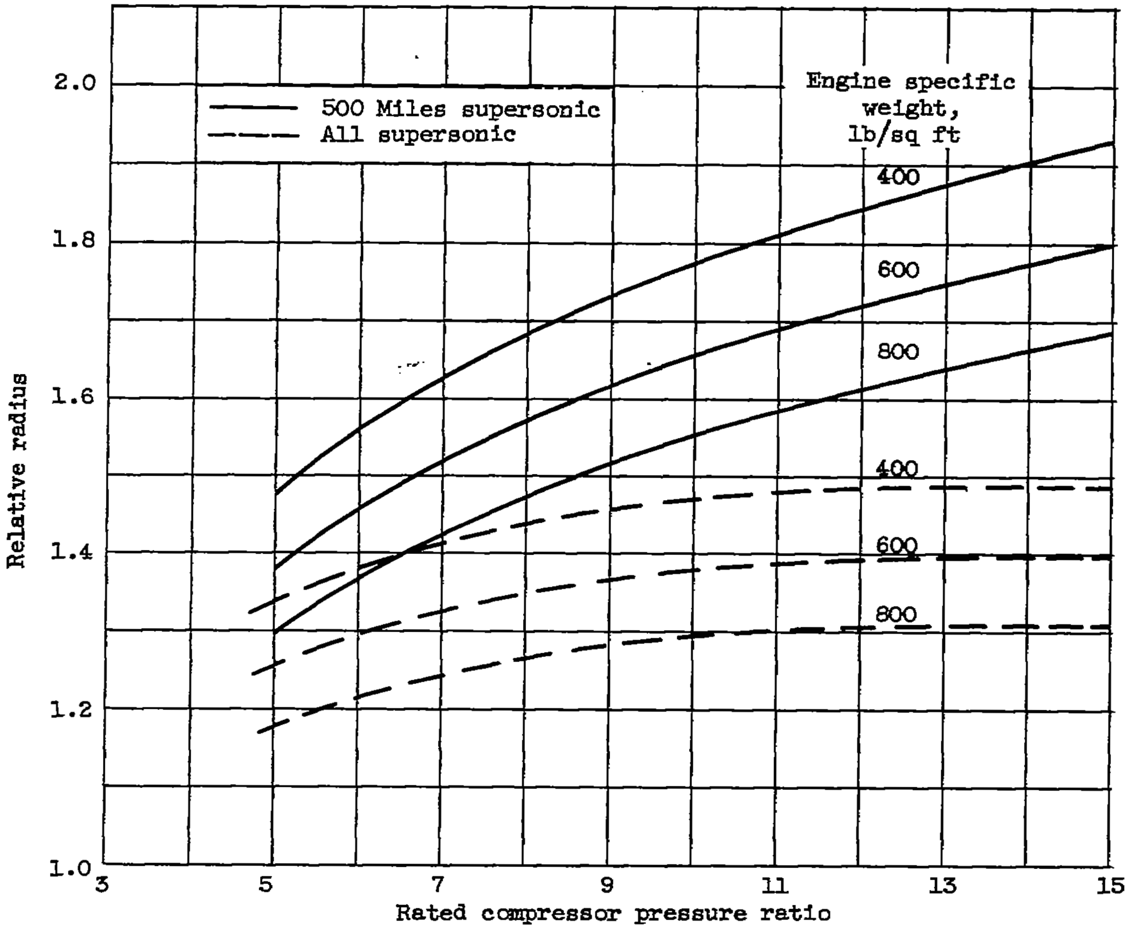

Figure 8. - Effect of rated compressor pressure ratio and engine specific weight on bomber radius. Engine size, 0.004 pound of air per second per pound gross weight; maximum compressor efficiency, 0.88; turbine efficiency, 0.90; turbine-inlet temperature, 2500° R; convergent-divergent nozzle; air-handling capacity, 33 pounds per second per square foot; combat-zone Mach number, 2.

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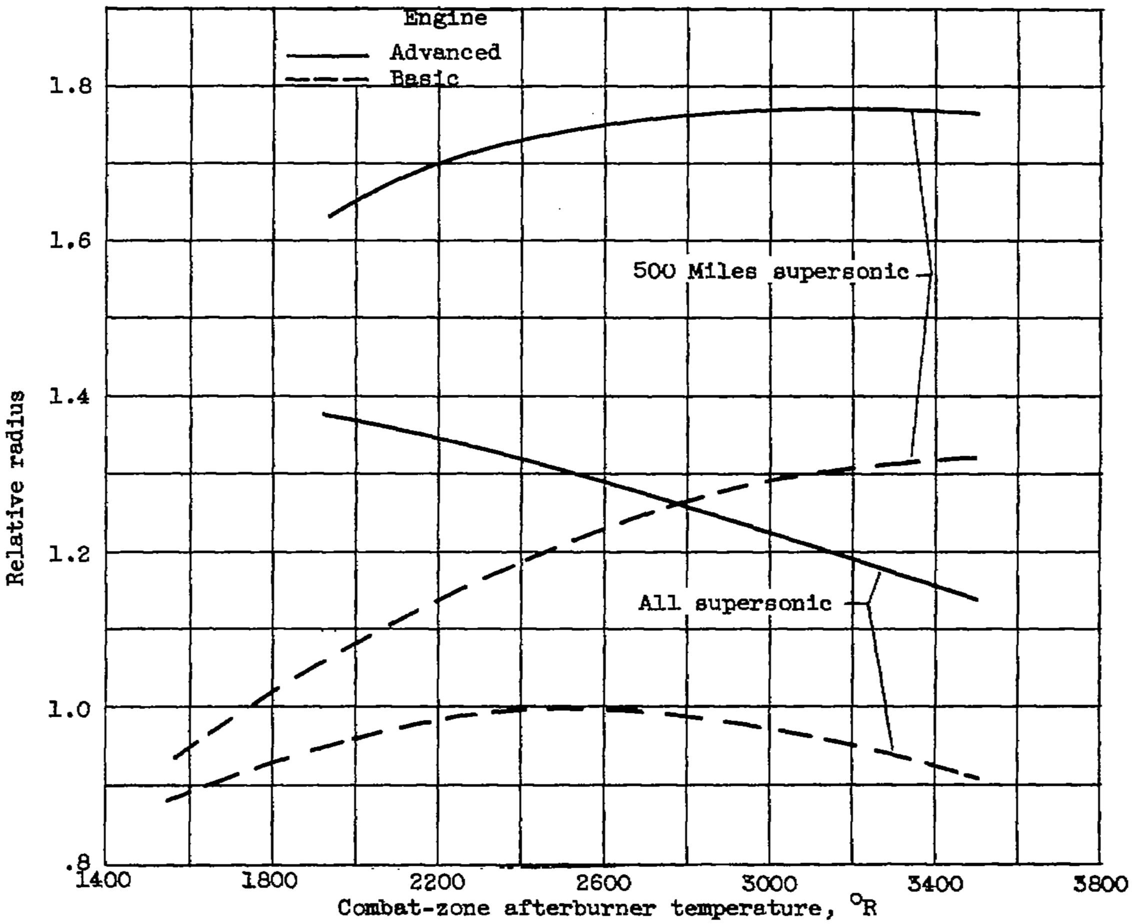

Figure 9. - Comparison of effect of combat-zone afterburner temperature on bomber relative radius with basic and advanced engines. Rated compressor pressure ratios, 7 and 12; maximum compressor efficiencies, 0.85 and 0.88; turbine efficiencies, 0.85 and 0.90; turbine-inlet temperatures, 2000° and 2500° R; convergent-divergent nozzles; air-handling capacities, 27 and 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; combat-zone Mach number, 2.

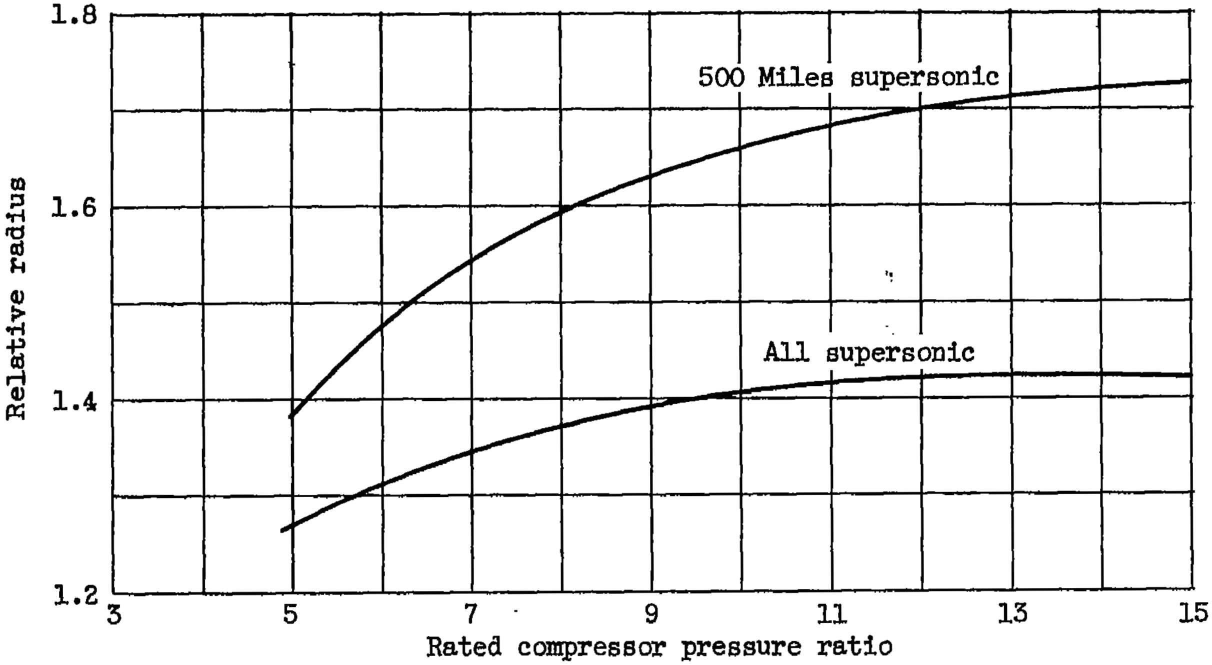

Figure 10. - Effect of rated compressor pressure ratio in advanced engine without afterburner on bomber relative radius. Maximum compressor efficiency, 0.88; turbine efficiency, 0.90; turbine-inlet temperature, 2500° R; convergent-divergent nozzle; air-handling capacity, 33 pounds per second per square foot; engine specific weight, 560 pounds per square foot; combat-zone Mach number, 2.

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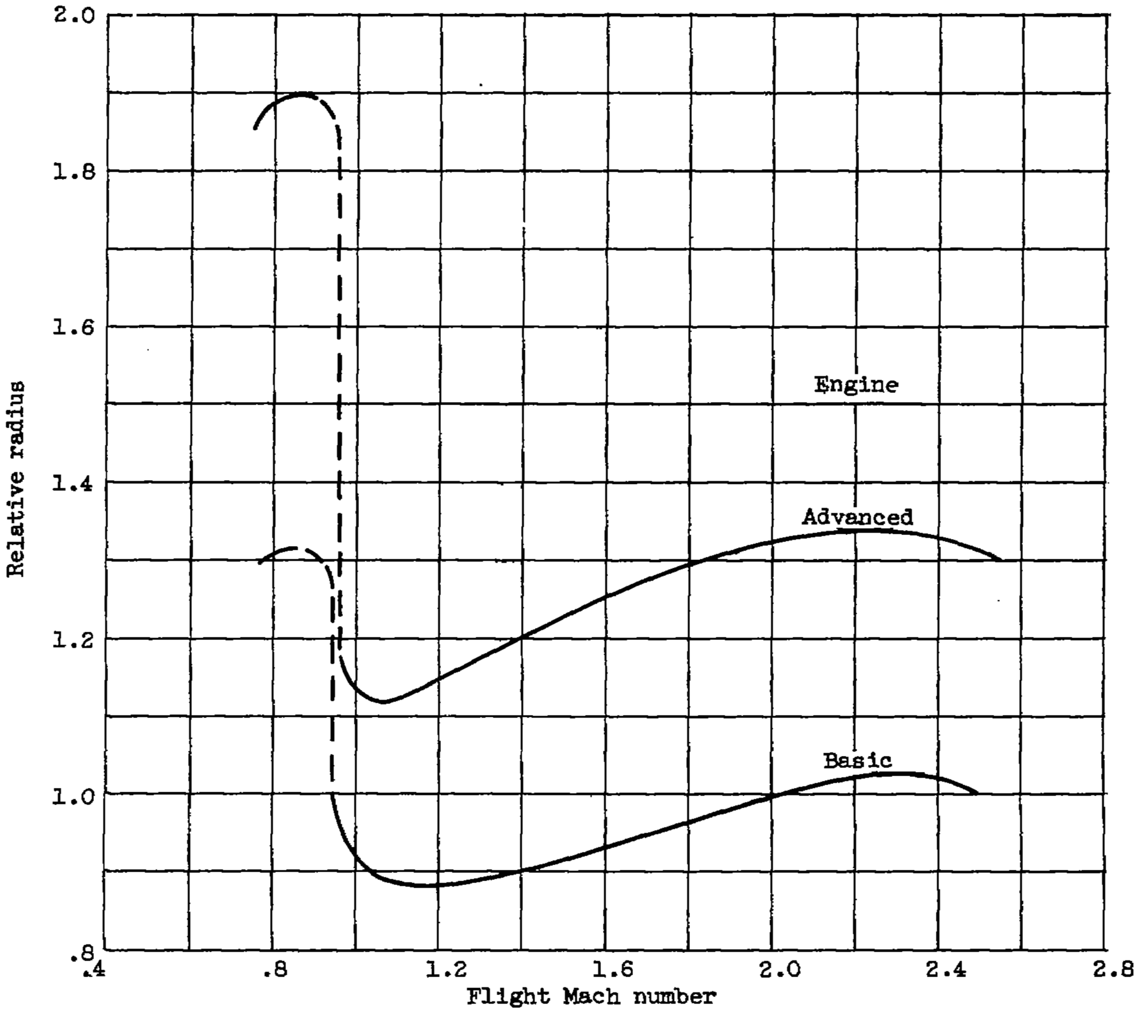

Figure 11. - Effect of flight Mach number on relative radius of supersonic bomber powered by basic and by advanced engines. Rated compressor pressure ratios, 7 and 12; maximum compressor efficiencies, 0.85 and 0.88; turbine efficiencies, 0.85 and 0.90; turbine-inlet temperatures, 2000° and 2500° R; convergent-divergent nozzles; air-handling capacities, 27 and 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; engine sizes, 0.004 and 0.003 pound per second per pound of take-off gross weight.

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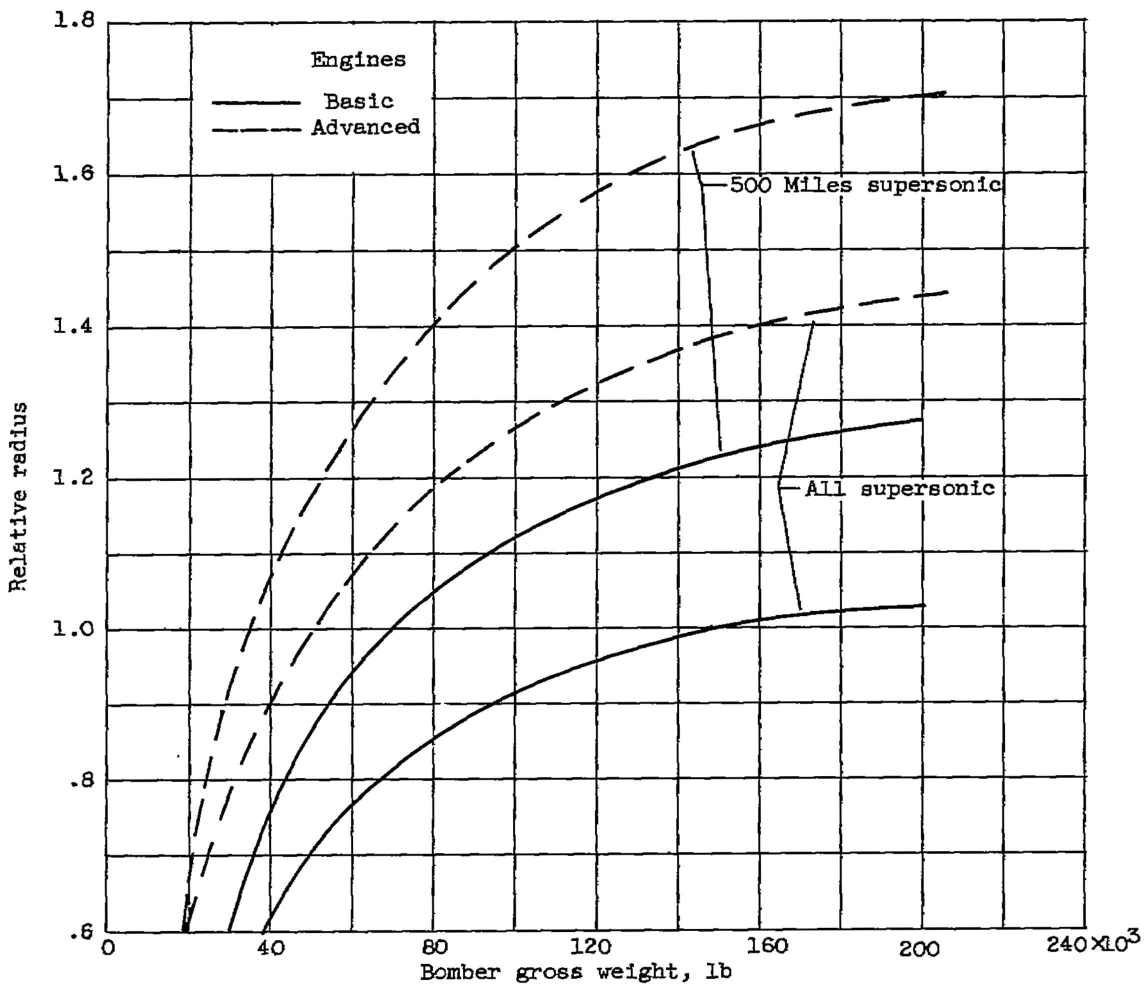

Figure 12. - Effect of gross weight on bomber relative radius when equipped with basic and advanced engines. Rated compressor pressure ratios, 7 and 12; maximum compressor efficiencies, 0.85 and 0.88; turbine efficiencies, 0.85 and 0.90; turbine-inlet temperatures, 2000° and 2500° R; convergent-divergent nozzles; air-handling capacities, 27 and 33 pounds per second per square foot; engine specific weight, 650 pounds per square foot; combat-zone Mach number, 2.

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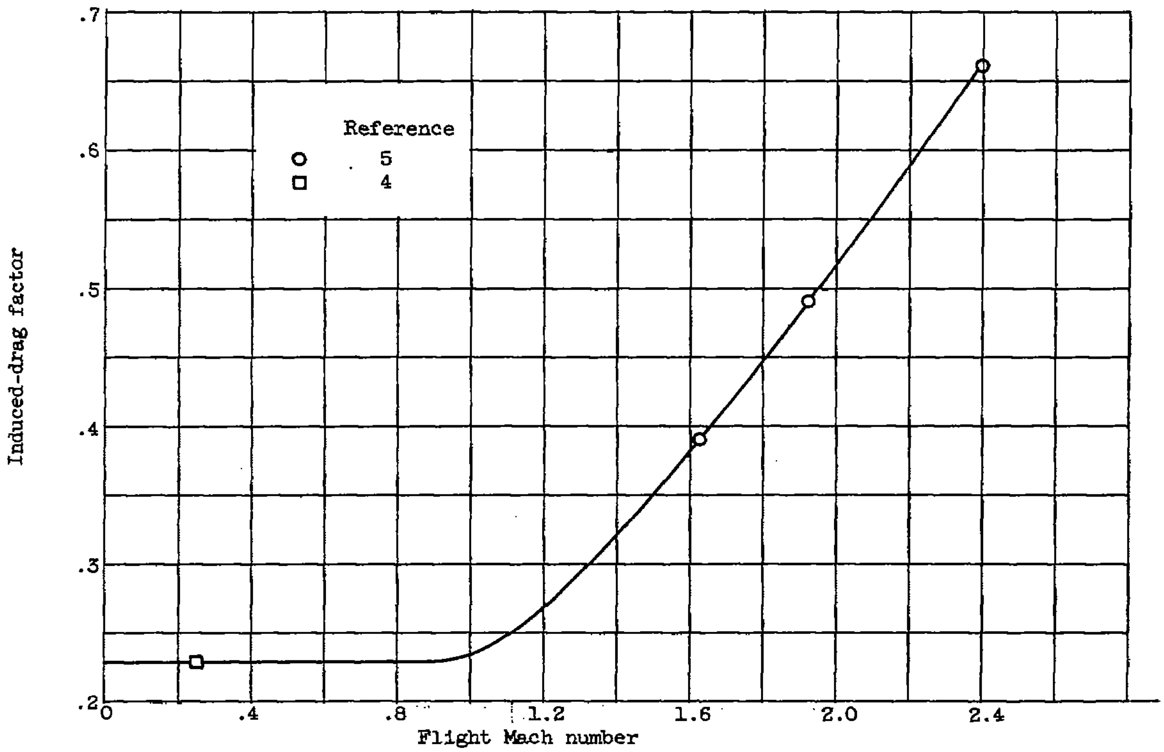

Figure 13. - Assumed relation between induced-drag factor and flight Mach number for delta wing.

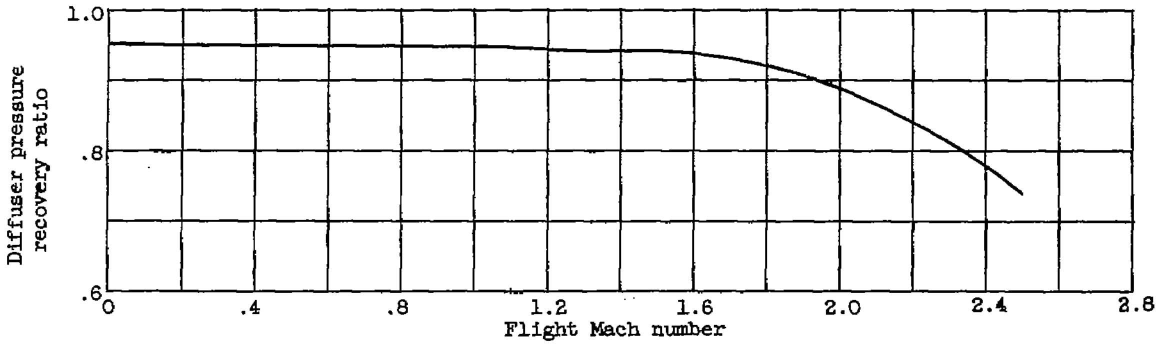

Figure 14. - Assumed relation between diffuser pressure recovery ratio and flight Mach number.

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