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RESEARCH MEMORANDUM

PERFORMANCE POSSIBILITIES OF THE TURBOJET SYSTEM AS A POWER PLANT FOR SUPERSONIC AIRPLANES

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
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SUMMARY

The performance of hypothetical turbojet systems, without thrust augmentation, as power plants for supersonic airplanes has been calculated. The thrust, thrust power, air-fuel ratio, specific fuel consumption, cross-sectional area, and thrust coefficient are shown for free-stream Mach numbers from 1.2 to 3. For comparison, the performance of ram-jet systems over the same Mach number range has also been calculated.

For Mach numbers between 1.2 and 2 the calculated thrust coefficient of the turbojet system was found to be larger than the estimated drag coefficient, and the specific fuel consumption was calculated to be considerably less than the specific fuel consumption of the ram-jet system. The turbojet system therefore appears to merit consideration as a propulsion method for free-stream Mach numbers between approximately 1.2 and 2.

INTRODUCTION

The National Advisory Committee for Aeronautics has issued several papers that present analyses of the performance possibilities of jet engines. In references 1 and 2, for example, calculations of the performance of compressor-turbine jet-propulsion, or turbojet, systems operating at subsonic airplane speeds are given. In reference 3 results of an investigation of the performance of continuous-flow ram-compression jet-propulsion, or ram-jet, systems propelling aircraft at supersonic speeds are presented. The question naturally arises as to the potentialities and the limitations of the turbojet system as a power plant for airplanes at supersonic speeds. The results obtained from analyses of the turbojet system operating at subsonic speeds, however, cannot be expected to give quantitative information about the performance of the turbojet system operating at supersonic speeds. Because of the difference in the compression available from the forward speed, the optimum blower compression ratio for operation at supersonic speeds, for example, may be quite different from the ratio for operation at subsonic speeds.

The purpose of the present paper is to report an analytical investigation of the turbojet system as a means of propelling airplanes and missiles at supersonic speeds.

A comparison of the performances of the turbojet and the ram-jet systems at supersonic free-stream speeds is also given herein. These two systems differ in three inherent characteristics that affect their performance as power plants, namely, maximum fluid temperature, maximum fluid pressure, and maximum cross-sectional area. The maximum fluid temperature that can be used in the turbojet system is limited by the mechanical properties of the turbine blades. The ram-jet system, however, has no turbine, and higher maximum fluid temperatures can therefore be used. The maximum fluid pressure in the ram-jet system is limited to the pressure obtainable from ram. The turbojet system, however, has a mechanical compressor, and higher maximum fluid pressures can therefore be used. The cross-sectional area of the combustion chamber in the ram-jet system is greater than in the turbojet system because of the smaller density resulting from less compression. The turbojet system, however, has a compressor and a turbine, the cross-sectional areas of which may exceed the cross-sectional area of the combustion chamber. In the present paper the effects on performance of the differences in maximum temperature, maximum pressure, and maximum cross-sectional area are shown.

The scope and the limitations of the present paper should be understood. Answers are given to the questions as to whether the turbojet system has a high enough thrust coefficient and a low enough specific fuel consumption to merit any consideration as a means of supersonic-airplane propulsion. The effects of augmentation of the thrust of the turbojet system by means of injection of additional fuel behind the turbine, which is sometimes called afterburning, are not considered. Thrust augmentation might be used for flight conditions in which additional thrust is needed, but an analytical study of thrust augmentation is too extensive for inclusion herein.

The present paper, furthermore, considers only the performance of power plants and does not consider the performance of complete airplanes that are to be propelled by these power plants. As far as the problem of choosing between the turbojet and the ram-jet methods of propulsion is concerned, the present paper therefore gives only a part of the information that is required for making a choice. Additional information of a character different from that given herein is needed for making a choice of power plant for an airplane that would meet a given set of specifications as to speed, range, and cargo capacity. This information should include the rate of air induction (some of the results presented herein are given in terms of rate of flow of inducted air), the altitude of level flight, the estimated weights of the turbojet and the ram-jet systems, and whether the airplane must land at the end of the flight or is to be used as a flying bomb.

The symbols used herein are defined in appendix A and the analysis and method of calculation are given in appendix B.

DESCRIPTION OF PROPULSION SYSTEMS AND ASSUMPTIONS OF COMPONENT PERFORMANCE

The turbojet system considered herein is represented in figure 1(a). Air enters the diffuser at supersonic speed and is delivered to the mechanical compressor at subsonic speed. After the air leaves the compressor, the air and fuel burn in the combustion chamber. The combustion products drive the turbine and then are ejected through the exhaust nozzle.

In the present paper only hypothetical components of the system, as distinguished from production units, were considered. Certain assumptions were made as to the performance of each component. Two types of supersonic diffuser were assumed. One of these types was the fixed-geometry diffuser, which was termed "Diffuser A." The performance of the fixed-geometry diffuser that was used in the present paper is shown by the dashed curve of figure 2. The ratio of total pressures across the diffuser is 0.9 for free-stream Mach numbers from 1.2 to 1.6 and decreases rather rapidly, as Mach number is increased, to a value of 0.33 at a Mach number of 3. The part of the curve between Mach numbers of 1.6 and 3 is the theoretical ratio of total pressures across a normal shock that occurs at free-stream Mach number. This diffuser performance is essentially that obtained in the experimental investigation of fixed-geometry diffusers reported in reference 4.

The other type of diffuser, which was termed "Diffuser B," was assumed to give the total-pressure ratio shown by the solid curve of figure 2. The total-pressure ratio of this diffuser was assumed to be also 0.9 for free-stream Mach numbers between 1.2 and 1.6 and to decrease linearly to a value of 0.75 at a Mach number of 3. The performance shown by the solid curve can perhaps be obtained by variable-geometry diffusers or by diffusers of the type described in references 5 and 6.

The compressor in the turbojet system was assumed to have an efficiency of 85 percent. Various compression ratios were assigned to the compressor. The cross-sectional area of the compressor was obtained by assigning a value of 0.4 to the Mach number at the entrance to the compressor and a value of 0.6 to the hub-tip diameter ratio of the compressor. Burning in the combustion chamber was assumed to be 100 percent complete and to occur in a frictionless channel of constant cross-sectional area. The initial air velocity in the combustion chamber was set at 250 feet per second. Losses of total pressure in the combustion chamber due to the addition of heat to a compressible fluid were taken into account.

The total, or stagnation, temperature at the entrance to the turbine was set at 1500° F for most of the calculations. The over-all efficiency of the turbine was assumed to be 85 percent. The cross-sectional area of the turbine was obtained with the assumptions that the turbine was of the single-stage, axial, impulse type with full admission, that the flow through the blades was frictionless, that the absolute exit velocity from the blades was in the axial direction, that the nozzles were at an angle of 15° to the plane of the wheel, that the turbine developed just the amount of power required to operate the compressor at any given compression ratio and free-stream Mach number, and that the hub-tip diameter ratio was 0.7. These assumptions gave peripheral speeds that exceeded 1350 feet per second in only a few cases.

The efficiency of the exhaust nozzle was assumed to be 100 percent. Thrust was calculated with the assumption that the nozzle was so designed that the exhaust fluid expanded to free-stream static pressure at the nozzle exit.

The ram-jet system considered herein is represented in figure 1(b). Air enters the diffuser at supersonic speed, is delivered to the combustion chamber at subsonic speed, is burned with fuel in the combustion chamber, and then is ejected through the exhaust nozzle.

The two diffuser performance curves (fig. 2) that were used in the calculations for the turbojet system were also used in the calculations for the ram-jet system. The Mach number at the entrance to the combustion chamber was varied upward from 0.2. Burning in the combustion chamber was assumed to be 100 percent complete and to occur in a frictionless channel of constant cross-sectional area. Losses of total pressure in the combustion chamber due to the addition of heat to a compressible fluid were taken into account. The efficiency of the exhaust nozzle was assumed to be 100 percent. Thrust was calculated with the assumption that the exhaust fluid expanded to free-stream static pressure at the nozzle exit.

RESULTS AND DISCUSSION

All the results presented are for operation in the stratosphere, that is, in the isothermal region that begins at an altitude of 35,332 feet. The results for the turbojet system that are discussed in detail are all for a total temperature at the turbine entrance of 1500° F. A few curves for temperatures of 1200° and 1800° are also included.

Thrust and Air-Fuel Ratio

The thrust developed by the turbojet system per pound of air per second is shown in figure 3(a) as a function of the free-stream Mach number. Curves are given for compression ratios of 2, 4, 6, 8, and 10. The curves show that, for compression ratios of 4 and higher, the thrust decreases rapidly as Mach number is increased above 1.2. For a compression ratio of 4 the thrust becomes zero at a Mach number of approximately 3. As the compression ratio is increased above 4, the Mach number at which the thrust vanishes decreases. For a compression ratio of 10 the thrust is zero at a Mach number of about 2.4. At Mach numbers between 1.2 and 1.6, a compression ratio of 2 produces essentially the same thrust as that produced by higher compression ratios. At Mach numbers greater than 1.6 a compression ratio of 2 produces more thrust than that produced by higher compression ratios.

The foregoing results are due principally to the fact that a limit has been placed on the maximum temperature that the fluid can have in the turbojet system and to the value set for that limit. At the smaller Mach numbers shown in figure 3(a) the temperature rise due to ram and to mechanical compression is small enough to allow a large amount of fuel to be burned per pound of air before the limiting temperature of 1500° F is reached. At the larger Mach numbers shown in figure 3(a), however, the temperature rise due to mechanical compression in combination with the temperature rise due to ram brings the air temperature so close to the limit that relatively little fuel can be burned. Air-fuel ratios for the various compression ratios are shown in figure 4. At a Mach number of 1.2 the air-fuel ratio is 56 for a compression ratio of 2. Increasing the compression ratio to 4 increases the air-fuel ratio only to 63. At a Mach number of 3, for a compression ratio of 2 the air-fuel ratio is 126, but for a compression ratio of 4 the air-fuel ratio is 280. At the higher Mach numbers, therefore, little or no thrust is developed by the turbojet system. The thrust produced by a propulsive system is derived from the burning of fuel. If little fuel can be burned per pound of air, then little or no thrust can be developed.

The thrust of the ram-jet system is shown in figure 3(b) for air-fuel ratios of 30, 40, and 60, these air-fuel ratios remaining constant with Mach number. It should be remembered that since there is no turbine there is no limiting maximum temperature. At a Mach number of 1.2 the air-fuel ratio of the turbojet system with a compression ratio of 2 is very close to 60, and the thrust developed by the two systems can therefore be compared at the air-fuel ratio of 60 at a Mach number of 1.2. The thrust of the turbojet system is 42 pounds per pound of air per second and of the ram-jet system is 25 pounds per pound of air per second. As the free-stream Mach number is increased, the thrust of the ram-jet system does not decrease rapidly, as does the thrust of the turbojet system, but, with diffuser B, increases, at least up to a Mach number of 3.

Throughout the present paper it has been assumed that the combustion chambers are of constant cross-sectional area. The cross-sectional area of the combustion chamber in the ram-jet system is a rather important quantity. In the first place, the thrust coefficient of the ram-jet system is based in the present paper on the combustion-chamber area. In the second place, a loss of total pressure occurs when heat is added to a compressible fluid in a constant-area combustion chamber. The loss of pressure is – other factors being constant – a function of the Mach number of the fluid at the entrance to the combustion chamber, which in turn is a function of the combustion-chamber area. In the third place, the Mach number at the combustion-chamber entrance must be below a certain maximum value if choking in the combustion chamber is to be avoided. Thus, the curve for an air-fuel ratio of 30 in figure 3(b) does not extend down to a Mach number of 1.2 but ends at a Mach number of 1.4 because of choking in the combustion chamber. As is well known, when heat is added to a compressible fluid in a constant-area passage, the fluid accelerates and the local Mach number increases along the passage. The increase in Mach number depends upon the initial enthalpy, the initial Mach number, and the amount of heat added (or fuel burned) per pound of air. If the initial enthalpy is low enough, or the initial Mach number is high enough, or enough fuel is burned, a final Mach number of unity is reached at the exit end of the combustion chamber and choking occurs. Figure 3(b) is based on combustion chambers of such cross-sectional areas that the Mach number of the air as it enters the combustion chamber is 0.20. This Mach number is low enough to avoid choking for air-fuel ratios of 40 and 60 but choking does occur for an air-fuel ratio of 30 at a free-stream Mach number of 1.4.

Curves for combustion-chamber entrance Mach numbers of 0.25 and 0.30 are shown in figure 3(c). The fact that some of these curves also end at free-stream Mach numbers greater than 1.2 indicates choking in the combustion chamber. Comparison of figures 3(b) and 3(c) shows that, if choking does not occur, the combustion-chamber entrance Mach number has very little effect on the thrust developed by the ram-jet system.

Thrust Power

The thrust power developed by a jet-propulsion system is, at a given Mach number, proportional to the thrust force developed by that system. The same significant results therefore can be obtained from a plot of thrust power that can be obtained from a plot of thrust. Nevertheless, as a matter of information, the thrust power developed by the systems under consideration has been plotted. The variation with free-stream Mach number of the thrust power per pound of air per second of the turbojet system is shown in figure 5(a). For a constant value of compression ratio, as Mach number is increased above 1.2, the thrust power rises to a maximum value and then decreases. The existence of a maximum is due to the fact that thrust power is proportional to the product of two quantities, of which one, thrust, decreases as Mach number is increased, and the other, free-stream speed, increases as Mach number is increased.

The variation with free-stream Mach number of the thrust power per pound of air per second of the ram-jet system is shown in figure 5(b).

Specific Fuel Consumption

Specific fuel consumption computed as pounds of fuel per hour per pound of thrust is shown in figure 6(a) for the turbojet system. For a compression ratio of 2 the specific fuel consumption remains at values near 1.4 for all Mach numbers between 1.2 and 3.0 if diffuser B is used. With diffuser A the specific fuel consumption increases rapidly at the higher Mach numbers. At Mach numbers of about 1.2 to 1.6 the specific fuel consumption is decreased as the compression ratio is increased, at least as far as a compression ratio of 10. For compression ratios of 6 and higher, however, the specific fuel consumption begins to increase rapidly at a Mach number of about 1.8 and reaches very high values at Mach numbers between 2.0 and 3.0.

The specific fuel consumption of the ram-jet system is shown in figure 6(b). For air-fuel ratios from 30 to 60, the specific fuel consumption of the ram-jet system is considerably greater than that of the turbojet system at Mach numbers between 1.2 and 2.0. At Mach numbers between 2.0 and 3.0 the specific fuel consumption of the ram-jet system either is approximately the same as or is less than that of the turbojet system. On the basis of specific fuel consumption, therefore, the turbojet system shows a decided superiority over the ram-jet system at Mach numbers between 1.2 and 2.0.

Specific fuel consumption computed as pounds of fuel per thrust horsepower-hour is shown for the turbojet system in figure 7(a). At a free-stream Mach number of 3.0 a compression ratio of 2 gives a lower specific fuel consumption than higher compression ratios. As the Mach number is decreased, the compression ratio that gives the lowest specific fuel consumption increases. Thus, at a Mach number of 2.0 a compression ratio of 6 gives a lower specific fuel consumption than any other compression ratio. At a Mach number of 2.0, however, the difference between the specific fuel consumptions of systems with compression ratios from 2 to 10 is very small. At a Mach number of 1.2, however, a compression ratio of 10 gives a much lower specific fuel consumption than a compression ratio of 2 but only a slightly lower specific fuel consumption than a compression ratio of 6.

The curves for the various compression ratios show that quite low values of specific fuel consumption are, at least theoretically, attainable in the supersonic turbojet system. At a Mach number of 1.2 the minimum specific fuel consumption shown is 0.46 pounds per horsepower-hour, at a Mach number of 2.0 it is 0.32, and at a Mach number of 3.0 it is 0.26. These values are much lower than those attainable at present with a conventional engine-propeller system at subsonic airplane speeds.

As has been shown in the section entitled "Thrust and Air-Fuel Ratio," as the free-stream Mach number is increased the thrust of the turbojet system with the higher values of compression ratio decreases rapidly to zero. Correspondingly, as the thrust approaches zero, the specific fuel consumption of the systems with the higher values of compression ratio increases rapidly.

In figure 7(b) the specific fuel consumption of the ram-jet system is shown for comparison with that of the turbojet system. The specific fuel consumption of the ram-jet system is shown to be much greater than that of the turbojet system at a Mach number of 1.2. The difference between the specific fuel consumptions of the two systems decreases as the Mach number is increased. At a Mach number of 3.0 the specific fuel consumptions of the ram-jet system and of the turbojet system with a compression ratio of 2 are about the same.

Cross-Sectional Areas

Cross-sectional areas that are of interest are the free-stream area of the inducted air, the total area of the compressor, the area of the combustion chamber, the total area of the turbine, and the area of the exhaust-nozzle exit. The free-stream cross-sectional area A_0 of the inducted air, which is the same as the air-inlet area of a diffuser of the type shown in figure 1, is given in figure 8 for various altitudes. Although the area A_0 is not important in itself, all plots of other areas are given in terms of A_0 and not in terms of W_a in order that a single curve will apply at all altitudes in the stratosphere and that therefore it will not be necessary to show a different curve for each altitude as is done in figure 8.

The ratio of the compressor total area A_c to A_0 is shown in figure 9. Figure 9 applies to all the turbojet systems considered herein, that is, to all compression ratios, all values of maximum temperature, and all altitudes in the stratosphere. Figure 9 was plotted on the conservative basis of compressor entrance Mach number equal to 0.4 and compressor entrance hub-to-tip diameter ratio equal to 0.6.

The ratio of the combustion-chamber area A_{cc} to A_0 is shown in figure 10. Figure 10(a), for the turbojet system, was prepared on the basis of combustion-chamber entrance velocity equal to 250 feet per second. Figure 10(b), for the ram-jet system, was prepared for combustion-chamber entrance Mach numbers of 0.20, 0.25, and 0.30.

The ratio of the total cross-sectional area of the turbine A_T to A_0 is shown in figure 11. Figure 11 was prepared on the assumption that the turbine was a single-stage impulse type with a hub-to-tip diameter ratio of 0.7. The curves for the higher compression ratios were not extended to the larger values of Mach number, because single-stage turbines would not be sufficient in this region unless their peripheral speeds exceeded 1350 feet per second. The curves for the higher values of compression ratio actually are not needed in the region of large Mach numbers, because the thrust decreases rapidly to zero in that region. The assumptions used herein for calculating turbine areas are conservative and lead to rather large turbine areas for the lower values of the compressor compression ratio. Less conservative assumptions could give considerably smaller areas.

For a compression ratio of 2 the turbine areas are much larger than the compressor areas. For a compression ratio of 4 the turbine areas are somewhat larger than the compressor areas. For a compression ratio of 6 the turbine and compressor areas are very nearly the same. For compression ratios greater than 6 the area of the compressor is greater than the area of the turbine.

The ratio of exhaust-nozzle exit, or maximum, area A_{exit} to A_0 is shown in figure 12. These areas were calculated on the assumption that the exhaust fluid expanded at the nozzle exit to free-stream static pressure.

Thrust Coefficient

One of the most significant quantities for showing the performance of a power plant is the thrust coefficient. The data on thrust and areas given in preceding figures have been used to obtain figure 13, which shows the variation with free-stream Mach number of the thrust coefficients of the turbojet system and, for comparison, of the ram-jet system. The thrust coefficient C_T is defined as the thrust divided by the product of free-stream dynamic pressure and an area. For the turbojet system (fig. 13(a)) the area that was used in calculating the thrust coefficient was the cross-sectional area of the compressor or of the turbine, whichever area was larger.

Comparison of figures 9 and 10(a) shows that the cross-sectional area of the compressor is greater than that of the combustion chamber. Comparison of figures 9 and 11 shows that for compression ratios of 2 and 4 the cross-sectional area of the turbine is greater than that of the compressor, for a compression ratio of 6 the areas of the turbine and the compressor are very nearly the same, and for compression ratios of 8 and 10 the area of the compressor is greater than that of the turbine. Comparison of figures 9 and 12(a) shows that at Mach numbers less than 2.2 the area of the compressor is greater than that of the exhaust-nozzle exit and that at Mach numbers of 2.2 and higher the area of the nozzle exit is either approximately equal to or greater than the area of the compressor. Nozzle-exit areas, however, were not used in calculating C_T. If, for a given set of conditions, the thrust of a jet system were calculated for various nozzle-exit areas, the thrust would have a maximum value at the nozzle-exit area that expanded the fluid to the ambient pressure. There is a large variation in the nozzle-exit area that gives almost maximum values of thrust, however, and not much thrust is sacrificed by using exit areas considerably smaller than the exit area that gives maximum thrust. (See reference 3.) For the purpose of reduction of external drag, supersonic jet power plants would undoubtedly use exhaust nozzles the exit areas of which were no greater than the maximum area of the rest of the power plant. The thrust coefficients shown in figure 13(a) can therefore be considered to be based on maximum cross-sectional areas.

The thrust coefficients of the turbojet system with high compression ratios are large at the smaller values of Mach number shown in figure 13(a). The thrust coefficients decrease rapidly to zero as the Mach number is increased because the thrust decreases rapidly to zero. At the smaller values of Mach number the thrust coefficients for the lower compression ratios are much less than for the higher compression ratios because of the large turbine cross sections that are required with low compression ratios at the smaller Mach numbers.

No details are given in the present paper of the external design of the housing, or fuselage, of the turbojet system, such as the wedge angle of the leading edge of the air inlet, the length of the fuselage, or the smoothness of the fuselage surface; accordingly, no precise estimate of the drag of the housing is attempted. At a given Mach number the drag due to shock depends on the wedge angle, and the drag due to friction depends on the smoothness of the surface and on the Reynolds number behind the shock. This Reynolds number in turn depends on the wedge angle, the fuselage length, and the altitude. The drag coefficient of the system based on maximum cross-sectional area can, however, be roughly estimated to be 0.15. This value of drag coefficient can be compared with the thrust coefficients shown in figure 13(a). The thrust coefficients are seen to be adequate with low or high compression ratios at the smaller Mach numbers and with low compression ratios at the larger Mach numbers.

The thrust coefficients of the ram-jet system are shown in figures 13(b) and 13(c). These coefficients are based on the cross-sectional area of the combustion chamber. (See fig. 10(b).) The thrust coefficients generally far exceed the estimated drag coefficient of 0.15. The thrust coefficients of the ram-jet system are generally greater than those of the turbojet system at the larger Mach numbers. The thrust coefficients of the ram-jet system with diffuser A were calculated for the same kind of diffuser that was assumed in part of the analytical studies of reference 3. Diffuser B is much more efficient at the higher Mach numbers than diffuser A and results in much larger values of the thrust coefficient at the higher Mach numbers. The large difference in the thrust coefficients with the two diffusers is due principally to differences in the area of the combustion chamber. (See fig. 10(b).) Comparison of figures 13(b) and 13(c) shows also that an appreciable gain in thrust coefficient is obtained by making the Mach number at the entrance to the combustion chamber as large as possible. The Mach number at the entrance to the combustion chamber is made larger by decreasing the combustion-chamber area, and the resulting increase in thrust coefficient is due to the decrease in combustion-chamber area.

Maximum Temperature

The curves showing the performance of the turbojet system were calculated with the use of a maximum fluid total temperature of 1500° F at the entrance to the turbine. The fact that a maximum limiting temperature is necessary in the turbojet system has a large effect on the performance of the system, particularly in that a maximum temperature causes the thrust of the system to become zero at some value of free-stream Mach number in the neighborhood of 3. In subsonic turbojet systems some variation of maximum temperature from 1500° F does not have a very large effect on the performance of the system. Figures 14 and 15 are presented for the supersonic turbojet system with diffuser A, with a compression ratio of 4, and with maximum temperatures of 1200°, 1500°, and 1800° F. Figure 14 shows the variation in the free-stream Mach number at which the system develops no thrust, and figure 15 shows the variation in the free-stream Mach number at which the specific fuel consumption reaches a minimum and begins to increase rapidly.

CONCLUDING REMARKS

The performance of hypothetical turbojet systems of propulsion of supersonic airplanes has been calculated for free-stream Mach numbers from 1.2 to 3. For comparison, the performance of the ram-jet system has also been calculated for the same range of Mach number.

At Mach numbers between 1.2 and 2 the calculated thrust coefficient of the turbojet system was found to be greater than the estimated drag coefficient. As Mach number is increased the thrust coefficient of the system with a compression ratio of 6 or more rapidly decreases and reaches zero at Mach numbers between 2 and 3. At Mach numbers between 2 and 3, therefore, only compression ratios less than 6 should be considered.

At a Mach number of 1.2 the specific fuel consumption of the turbojet system is much less than that of the ram-jet system with comparable thrust or thrust coefficient. As the Mach number is increased toward 2, the difference between the specific fuel consumptions of the two systems decreases.

The fact that at free-stream Mach numbers between 1.2 and 2 the turbojet system has adequate thrust coefficient and low specific fuel consumption (compared with the ram-jet system) means that the turbojet system can logically be considered as a possible power plant for supersonic airplanes operating at those Mach numbers. The choice between use of the two systems should, of course, not be based solely on the fact that the specific fuel consumption of the turbojet system is lower than that of the ram-jet system. The turbojet system is inherently heavier than the ram-jet system. For flights of great enough time duration, however, the combined weight of the power plant and the fuel will be less for the turbojet system. Other factors than flight duration - such as the advantage that the turbojet system may have at take-off, in accelerating to flight speed, and in landing - should also be considered.

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APPENDIX A

SYMBOLS

A	cross-sectional area, sq ft
A_c	total cross-sectional area of compressor, sq ft
A_{cc}	cross-sectional area of combustion chamber, sq ft
A_{exit}	cross-sectional area of exhaust-nozzle exit, sq ft
A_T	total cross-sectional area of turbine, sq ft
c_p	specific heat at constant pressure, Btu/lb/°F
C_T	thrust coefficient $\left(\frac{2T}{\rho_0 V_0{}^2 A}\right)$
g	acceleration due to gravity, ft/sec^2 (32.2)
H	static enthalpy, Btu/lb
H_t	total enthalpy, Btu/lb
J	mechanical equivalent of heat, ft-lb/Btu (778)
K	total-pressure ratio across diffuser
M	Mach number
p	static pressure, lb/sq ft
p_t	total pressure, lb/sq ft
P	thrust power, hp
P_T	turbine power, ft-lb/sec
r	compression ratio
R	gas constant, ft-lb/lb/°F abs. (53.5)
t_{max}	maximum total temperature, °F

T	static temperature, °F abs.
T_t	total temperature, °F abs.
T	thrust, lb
u	peripheral speed of turbine, ft/sec
V	velocity, ft/sec
W_a	weight rate of flow of air, lb/sec
W_f	weight rate of flow of fuel, lb/sec
α	nozzle angle (angle between V_5 and u)
γ	ratio of specific heats
η_c	efficiency of compressor
η_T	efficiency of turbine
ρ	static mass density, slug/cu ft
ρ_t	total mass density, slug/cu ft

Subscripts:

0,... 7 stations, shown in figure 16

APPENDIX B

ANALYSIS AND METHOD OF CALCULATION

Turbojet System

The stations that were used in the analysis of the turbojet system are shown in figure 16(a). The performance of the system was obtained for each set of values assigned to the parameters by calculating the state of the working fluid at each station.

At station 0, in the free stream,

$$p_{t_0} = p_0\left(1 + \frac{\gamma - 1}{2} M_0^2\right)^{\frac{\gamma}{\gamma - 1}} \tag{1}$$

$$T_{t_0} = T_0\left(1 + \frac{\gamma - 1}{2} M_0^2\right) \tag{2}$$

$$\frac{A_0}{W_a} = \frac{1}{g\rho_0 V_0} \tag{3}$$

$$V_0^2 = \gamma g R T_0 M_0^2 \tag{4}$$

The diffuser is between stations 0 and 1. The function of the diffuser is to receive air of Mach number greater than unity from the free stream and to deliver air of Mach number less than unity to the compressor with a minimum loss of total pressure. The computations were made for two types of diffuser. One type is the fixed-geometry diffuser, a theoretical and experimental investigation of which was reported in reference 4. The theory of reference 4 takes into account the fact that the configuration of the diffuser must be such that establishment of supersonic flow inside the diffuser can be effected. The establishment of supersonic flow in the converging part of the diffuser is generally preceded, before design speed is reached, by a regime in which there is a bow wave in front of the diffuser and subsonic flow in the converging section, as shown in figure 17. The shock first moves up to the entrance of the diffuser, provided that the throat, or minimum section, of the diffuser is large enough to accommodate the passage, at a Mach number not greater than unity, of all the fluid contained between the dashed lines of figure 17. The shock then enters the diffuser, passes through the throat, and is established in the diverging section.

The net result of these considerations is, however, that the throat area cannot be much smaller than the entrance area, and therefore little diffusion of the supersonic flow is obtainable. The minimum Mach number that can be obtained at the throat, according to the results of reference 4, is shown in figure 18 as a function of free-stream Mach number. It is evident that the Mach number at the throat is only slightly smaller than the free-stream Mach number. For stability, furthermore, the shock should occur not at the throat but in the diverging section behind the throat. In the diverging section the Mach number increases downstream, and the shock that occurs in the diverging section must occur at a Mach number that is greater than the throat Mach number and that approaches free-stream Mach number. Losses in the subsonic part of the diffuser contribute to the total loss between stations 0 and 1. In figure 19 the upper curve is the theoretical ratio of total pressure across a normal shock that occurs at the minimum throat Mach number shown in figure 18. The test-point symbols in figure 19 are the experimentally obtained ratios of total pressure across the diffuser, as given in reference 4. The lower curve is the theoretical ratio of total pressures across a normal shock that occurs at free-stream Mach number. The experimental points lie close to the lower curve. This curve, which is also shown in figure 2, was used herein as the performance curve of the fixed-geometry diffusers (of the kind described in reference 4) for free-stream Mach numbers between 1.6 and 3.

The other type of diffuser for which computations were made was assumed to give the total pressure ratio shown by the solid curve of figure 2. The diffuser performance shown by this curve can perhaps be obtained with variable-geometry diffusers or with diffusers of the type described in references 5 and 6. Both types of diffusers were assumed to give a pressure ratio of 0.9 at Mach numbers from 1.2 to 1.6.

Conditions at the entrance to the compressor (station 1, fig. 16(a)) are given by equations (5) to (11) as follows:

$$p_{t_1} = Kp_{t_0} \tag{5}$$

$$\rho_{t_1} = K\rho_{t_0} \tag{6}$$

$$T_{t_1} = T_{t_0} \tag{7}$$

$$\rho_1 = \frac{\rho_{t_1}}{\left(1 + \frac{\gamma - 1}{2} M_1{}^2\right)^{\frac{1}{\gamma - 1}}} \tag{8}$$

$$T_1 = \frac{T_{t_1}}{1 + \frac{\gamma - 1}{2} M_1{}^2} \tag{9}$$

$$V_1{}^2 = \gamma g R T_1 M_1{}^2 \tag{10}$$

$$\frac{A_1}{W_a} = \frac{1}{g \rho_1 V_1} \tag{11}$$

In the present paper, in order that the cross-sectional areas of the compressors could all be calculated on the same basis, the compressor-entrance Mach number M_1 was set at the same value, 0.4, for all compressors, that is, for all values of the compression ratio. The ratio of the hub diameter to the total diameter of the axial-flow compressor was set at 0.6 for all compression ratios. The total cross-sectional area of the compressor A_c is then $1.56A_1$.

Conditions at the exit from the compressor (station 2) are given by equations (12) to (14) as follows:

$$p_{t_2} = r p_{t_1} \tag{12}$$

$$H_{t_2} = \frac{H_{t_1}}{\eta_c}\left(r^{\frac{\gamma - 1}{\gamma}} + \eta_c - 1\right) \tag{13}$$

$$\rho_{t_2} = \frac{p_{t_2}}{gRT_{t_2}} \tag{14}$$

A compressor efficiency of 85 percent was used in all the calculations.

The temperature corresponding to a given value of the enthalpy, or the enthalpy corresponding to a given value of the temperature, was found from the relation

$$H = \int_0^T c_p \, dT = 0.239202T + \left(1.634 \times 10^{-6}\right)T^2 + \left(1.104 \times 10^{-9}\right)T^3 \tag{15}$$

which was adapted from equation (3) of reference 7. Equation (15) written with H_t and T_t in place of H and T gives the relation between total enthalpy and total temperature. It should be noted that the enthalpy as defined by equation (15) and as used in the present paper is based on a value of zero absolute enthalpy at zero absolute temperature. Ratios of enthalpy frequently occur in the equations given herein. In these equations, therefore, values of absolute enthalpy must be used and not values of relative enthalpy based on a zero of enthalpy at some other temperature than absolute zero.

The initial velocity in the combustion chamber V_3 was set at 250 feet per second. An isentropic change in flow area between stations 2 and 3 was assumed. Then

$$p_{t_3} = p_{t_2} \tag{16}$$

$$H_{t_3} = H_{t_2} \tag{17}$$

$$\rho_{t_3} = \rho_{t_2} \tag{18}$$

The quantity T_{t_3} is found by means of equation (15). Then

$$M_3^2 = \frac{1}{\dfrac{\gamma g R T_{t_3}}{V_3^2} - \dfrac{\gamma - 1}{2}} \tag{19}$$

inasmuch as

$$T_{t_3} = T_3 \left(1 + \frac{\gamma - 1}{2} M_3^2\right) \tag{20}$$

and

$$M_3^2 = \frac{V_3^2}{\gamma g R T_3} \tag{21}$$

Then

$$\rho_3 = \frac{\rho_{t_3}}{\left(1 + \frac{\gamma - 1}{2} M_3^2\right)^{\frac{1}{\gamma - 1}}} \tag{22}$$

and

$$\frac{A_3}{W_a} = \frac{1}{g \rho_3 V_3} \tag{23}$$

Between stations 3 and 4 combustion occurs. It is assumed that enough fuel is burned to raise the total temperature at station 4 to a maximum permissible value. Three values of maximum total temperature were used, 1200°, 1500°, and 1800° F. The corresponding values of H_{t_4} are 406, 483, and 561 Btu per pound, respectively. In order to simplify the calculations, the effect on the specific heats of the change in the chemical composition of the air when fuel is added is not taken into account herein in the calculations for the turbojet system. Some degree of approximation is therefore introduced. The ratio of air to fuel is, however, generally large in the turbojet system, and the error introduced by the simplification is therefore small.

In the combustion chambers in current use, rather large losses of total pressure occur. Much of the loss is doubtless due to changes in the shape and in the area of the flow passages. Data for satisfactorily calculating these losses are not available. In the present paper the only loss in total pressure in the combustion chamber is assumed to be the loss due to the addition of heat to a compressible fluid. For constant-area combustion the ratio of total pressures at entrance and exit is a function only of the entrance Mach number M_3 and the ratio of total enthalpies at entrance and exit H_{t_3}/H_{t_4}. The total pressure at the combustion-chamber exit p_{t_4} is given by equations (24) and (25), in which there are only two unknowns, M_4 and p_{t_4}.

$$\frac{p_{t_4}}{p_{t_3}} = \frac{\left(1 + \gamma M_3^2\right)\left(1 + \frac{\gamma - 1}{2} M_4^2\right)^{\frac{\gamma}{\gamma-1}}}{\left(1 + \gamma M_4^2\right)\left(1 + \frac{\gamma - 1}{2} M_3^2\right)^{\frac{\gamma}{\gamma-1}}} \tag{24}$$

$$\frac{H_{t_4}}{H_{t_3}} = \frac{M_4^2\left(1 + \frac{\gamma - 1}{2} M_4^2\right)\left(1 + \gamma M_3^2\right)^2}{M_3^2\left(1 + \frac{\gamma - 1}{2} M_3^2\right)\left(1 + \gamma M_4^2\right)^2} \tag{25}$$

The air-fuel ratio is

$$\frac{W_a}{W_f} = \frac{19{,}000 - H_{t_4} + H_{t_3}}{H_{t_4} - H_{t_3}} \tag{26}$$

In equation (26) the value of the heat of combustion of the fuel has been taken to be 19,000 Btu per pound.

For the purpose of calculating the cross-sectional area of the turbine, a number of assumptions were made regarding the turbine. These assumptions permit at least an approximate calculation of the area to be made. The turbine was assumed to be of the single-stage, axial, impulse, full-admission type. Friction in the blades was neglected. The velocity of the fluid with respect to the casing was assumed to be in an axial direction at the exit from the blades. (For frictionless impulse blades, this condition gives maximum efficiency.) The power developed by the turbine is then

$$\begin{aligned} P_T &= W_a J\left(H_{t_5} - H_{t_6}\right) \\ &= \frac{W_a}{g} u\left(V_5 \cos \alpha - 0\right) \end{aligned} \tag{27}$$

In equation (27) V_5 is the absolute velocity of the fluid after the fluid has passed through the turbine nozzles and is entering the turbine blades. For frictionless impulse blades the velocity of the fluid relative to the blades is the same at entrance to and exit from the blades, and the absolute value of the angle between the relative velocity of the fluid and the turbine axis is the same at entrance and exit. Under the additional condition that the absolute exit velocity from the wheel is in an axial direction, the peripheral velocity of the wheel is then

$$u = \frac{1}{2} V_5 \cos \alpha \tag{28}$$

Equation (27) can then be rewritten as

$$\begin{aligned} P_T &= W_a J\left(H_{t_5} - H_{t_6}\right) \\ &= \frac{W_a}{2g} V_5{}^2 \cos^2\alpha \end{aligned} \tag{29}$$

The work done by the turbine, however, is assumed to be equal to the work done on the compressor

$$H_{t_5} - H_{t_6} = H_{t_2} - H_{t_1} \tag{30}$$

Therefore

$$\begin{aligned} P_T &= W_a J\left(H_{t_2} - H_{t_1}\right) \\ &= \frac{W_a}{2g} V_5{}^2 \cos^2\alpha \end{aligned} \tag{31}$$

and

$$V_5{}^2 = \frac{2gJ\left(H_{t_2} - H_{t_1}\right)}{\cos^2\alpha} \tag{32}$$

The nozzle angle α was assumed herein to be 15°. Equation (32) was used to find V_5, and then equation (28) was used to find the peripheral speed of the turbine u. So long as the value of u did not exceed approximately 1350 feet per second, the present method of calculating turbine areas could be used. When the value of u exceeded 1350 feet per second for the higher values of compression ratio at the higher values of Mach number, calculations of turbine area were not made.

The Mach number at the entrance to the blades is given by the following equation:

$$M_5{}^2 = \frac{1}{\dfrac{\gamma g R T_5}{V_5{}^2} - \dfrac{\gamma - 1}{2}} \tag{33}$$

The density is

$$\rho_5 = \frac{\rho_{t_5}}{\left(1 + \dfrac{\gamma - 1}{2} M_5{}^2\right)^{\frac{1}{\gamma-1}}} \tag{34}$$

The open area at the entrance to the blades in a radial plane is given by the following equation:

$$\frac{A_5}{W_a} = \frac{1}{g\rho_5 V_5 \sin \alpha} \tag{35}$$

Under the assumption that the ratio of the hub diameter to the total diameter of the turbine wheel is 0.7, the total cross-sectional area of the turbine A_T is $1.96A_5$.

At the outlet from the turbine (station 6) the total pressure is

$$p_{t_6} = p_{t_5}\left(1 - \frac{H_{t_5} - H_{t_6}}{\eta_T H_{t_5}}\right)^{\frac{\gamma}{\gamma-1}} \tag{36}$$

A value of 85 percent was used for the turbine efficiency η_T.

The static enthalpy H_7 of the fluid after expansion through a 100-percent-efficient exhaust nozzle to free-stream static pressure is given by the relation

$$\frac{H_7}{H_{t_6}} = \left(\frac{p_0}{p_{t_6}}\right)^{\frac{\gamma-1}{\gamma}} \tag{37}$$

There is a considerable difference between H_{t_6} and H_7 and consequently also between γ_6 and γ_7. For the present calculations the values of H_7 were obtained from the tables of reference 7. These tables give the enthalpy changes associated with isentropic pressure changes. Use of the tables eliminates much tedious calculation or interpolation on a Mollier chart and automatically takes into account the variation of γ between stations 6 and 7. The tables were computed for air, but in the turbojet system the air-fuel ratio is so high that neglect of the change in γ with air-fuel ratio is justified.

The velocity of exit from the airplane is given by the relation

$$V_7^2 = 2gJ\left(H_{t_6} - H_7\right) \tag{38}$$

The thrust per pound of air per second is

$$\frac{T}{W_a} = \frac{V_7 - V_0}{g} + \frac{W_f V_7}{W_a g} \tag{39}$$

The thrust power per pound of air per second is

$$\frac{P}{W_a} = \frac{V_0 T}{550 W_a} \tag{40}$$

From H_7 and equation (15), T_7 is found. The exhaust-nozzle-exit area per pound of air per second is then

$$\frac{A_7}{W_a} = \frac{1}{g\rho_7 V_7} = \frac{RT_7}{p_0 V_7} \tag{41}$$

Ram-Jet System

The stations that are used in the analysis of the ram-jet system are shown in figure 16(b). Conditions at station 0 in the free stream are given by equations (1) to (4). The supersonic diffuser is between stations 0 and 1. The diffuser performance of figure 2 was used for the ram-jet as well as for the turbojet system. Conditions at the entrance to the combustion chamber (station 1) are given by equations (5) to (11). The Mach number at station 1 was set at 0.20, 0.25, and 0.30.

Conditions at station 6 following combustion are given by equations (24) to (26) with subscripts 1 and 6 substituted for subscripts 3 and 4, respectively. Air-fuel ratios of 30, 40, and 60 were used in equation (26). Figures 20 and 21 were plotted by means of equations (25) and (24), respectively. The ratio H_{t_6}/H_{t_1} is found from equation (26), then M_6 is found from equation (25) or figure 20, then p_{t_6}/p_{t_1} is found from equation (24) or figure 21. The conditions under which choking occurs can be readily seen from figures 20 and 21. These figures also show that even under choking conditions the loss of total pressure due to the addition of heat is not very large.

The enthalpy change in the exhaust nozzle was calculated by equation (37). The quantity γ in equation (37) was adjusted to take into account the fact that the air-fuel ratios for the ram-jet system were small enough for the fuel to have an appreciable effect on the value of γ. The effect on γ of the temperature change between stations 6 and 7 was taken into account by estimating the temperature at station 7, obtaining the corresponding values of γ_7, and then using the average of γ_6 and γ_7 in equation (37). Effects of dissociation and heat-capacity lag were neglected.

Exhaust velocity is given by equation (38), thrust by equation (39), thrust power by equation (40), and exhaust-nozzle exit area by equation (41).

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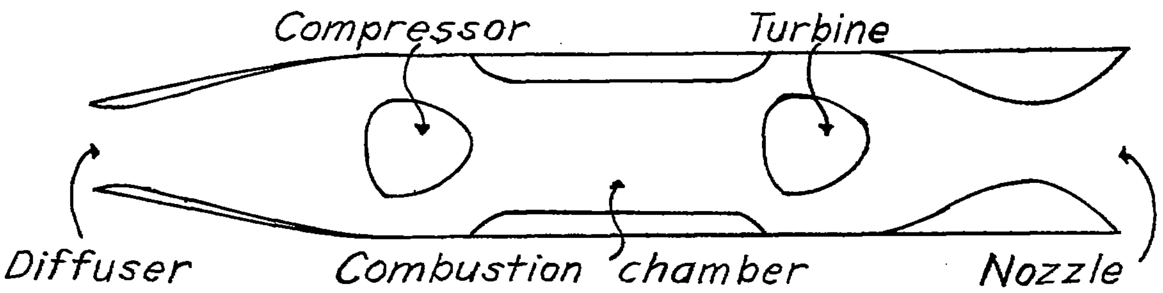
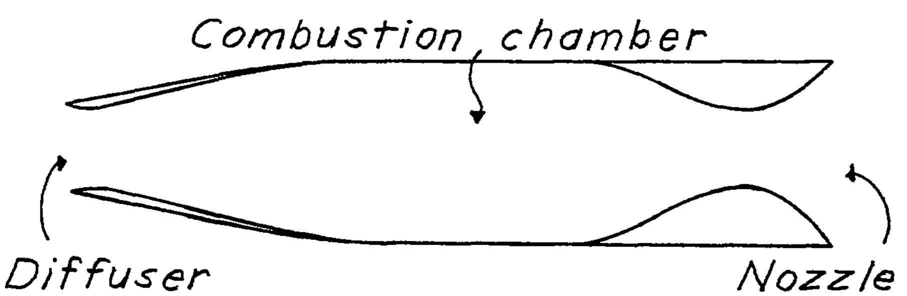


(a) Turbojet system.



(b) Ram-jet system.

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Figure 1.- Jet-propulsion systems.

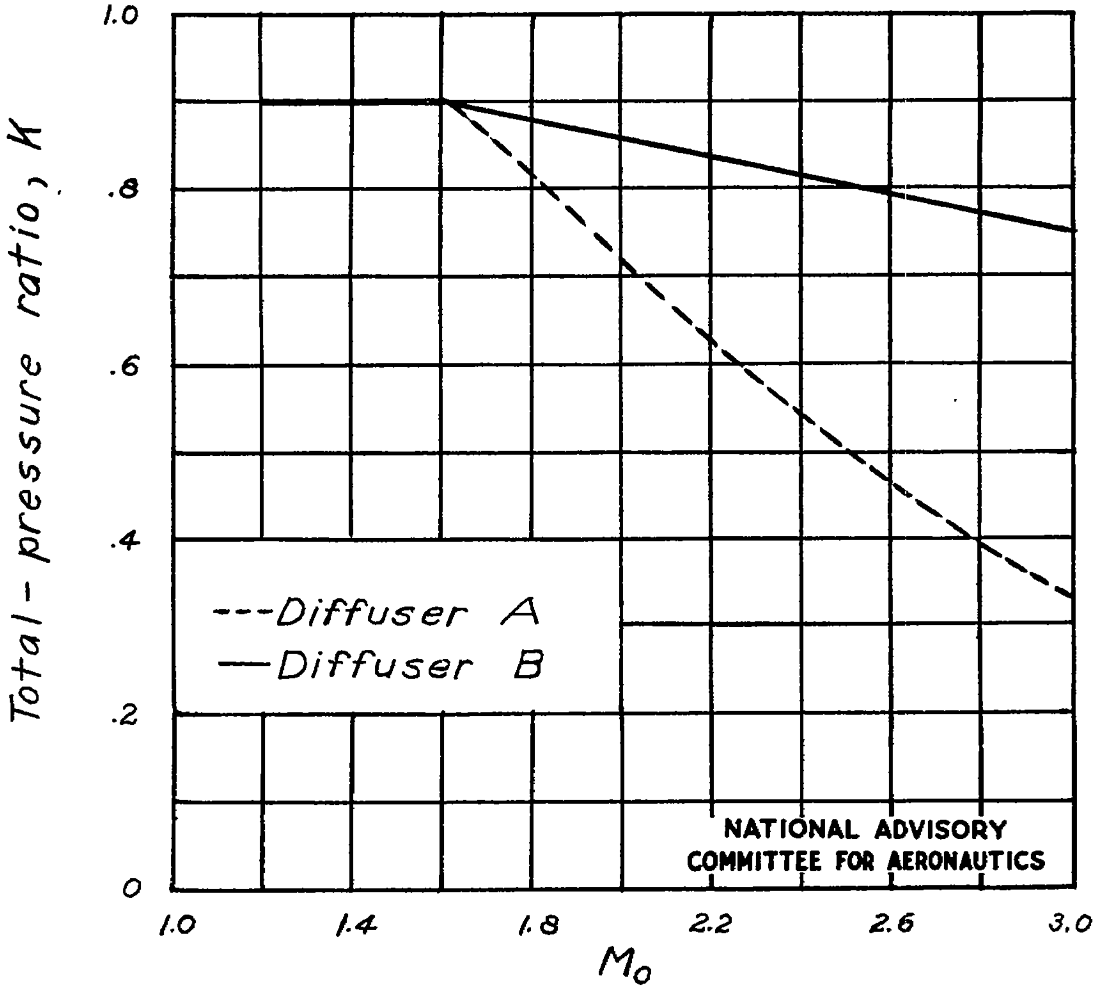
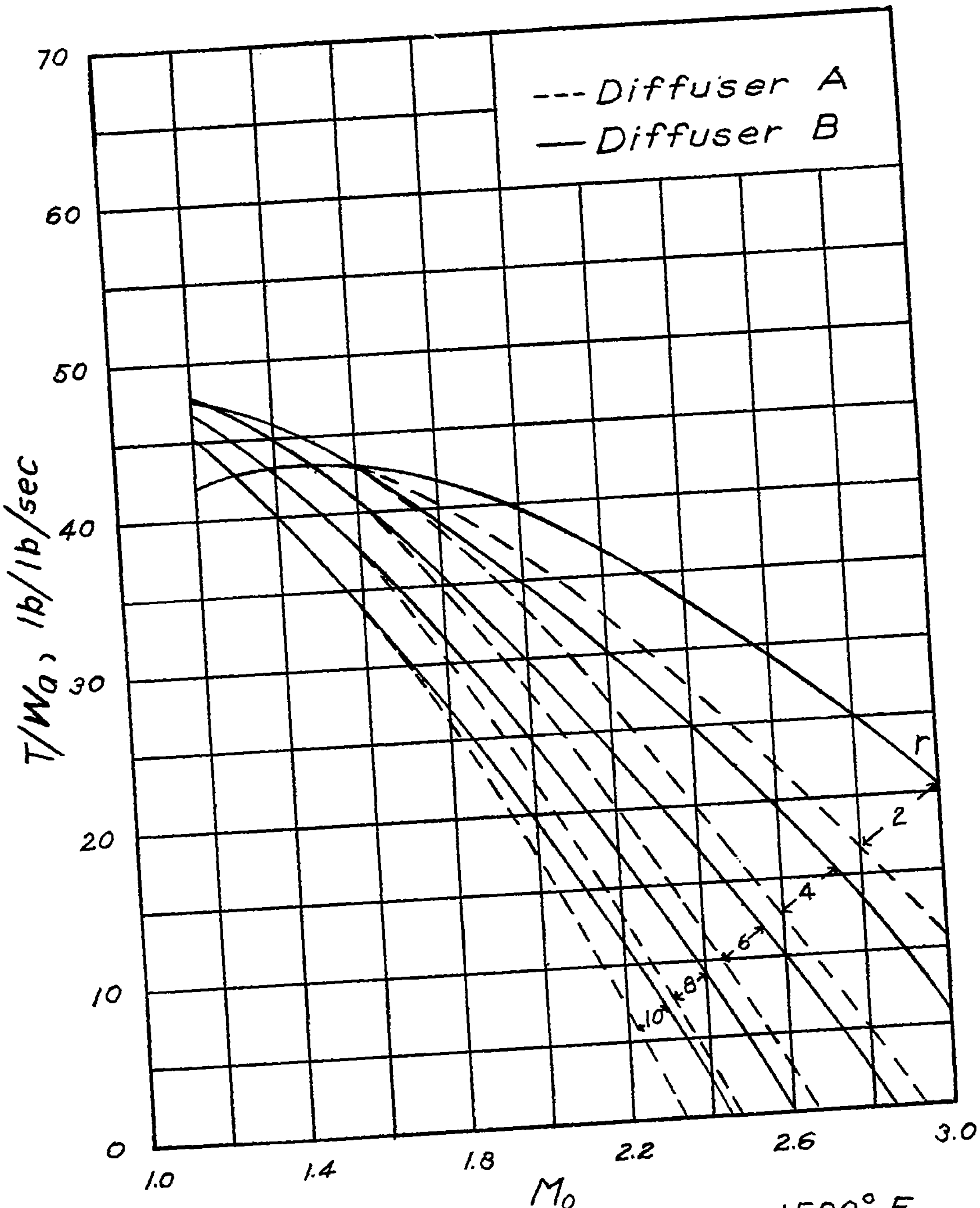


Figure 2.- Total-pressure ratio across diffuser as a function of free-stream Mach number. Turbojet and ram-jet systems.



(a) Turbojet system. t_{max} = 1500° F.

Figure 3.- Thrust per pound of air per second as a function of free-stream Mach number. Altitude, stratosphere.

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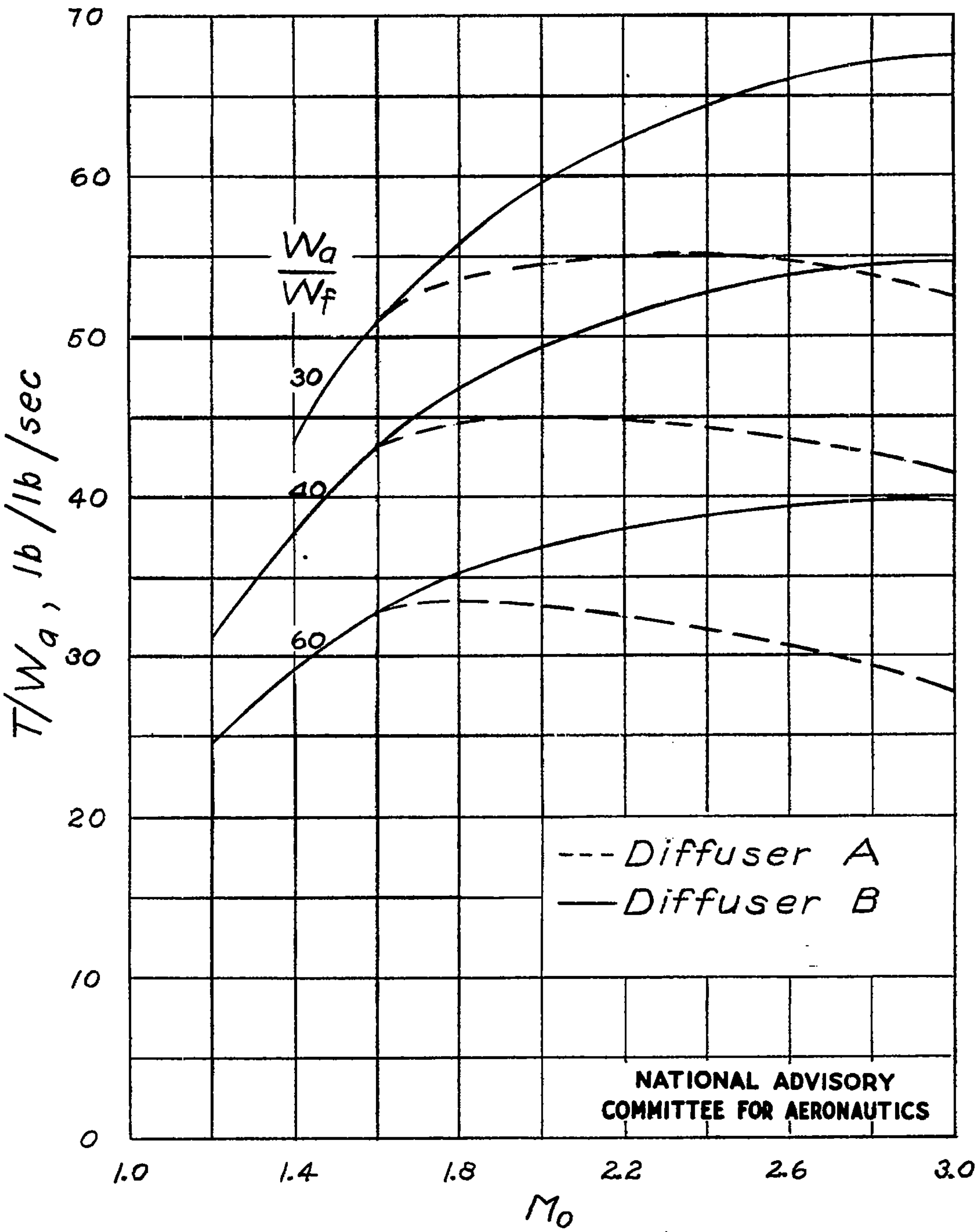
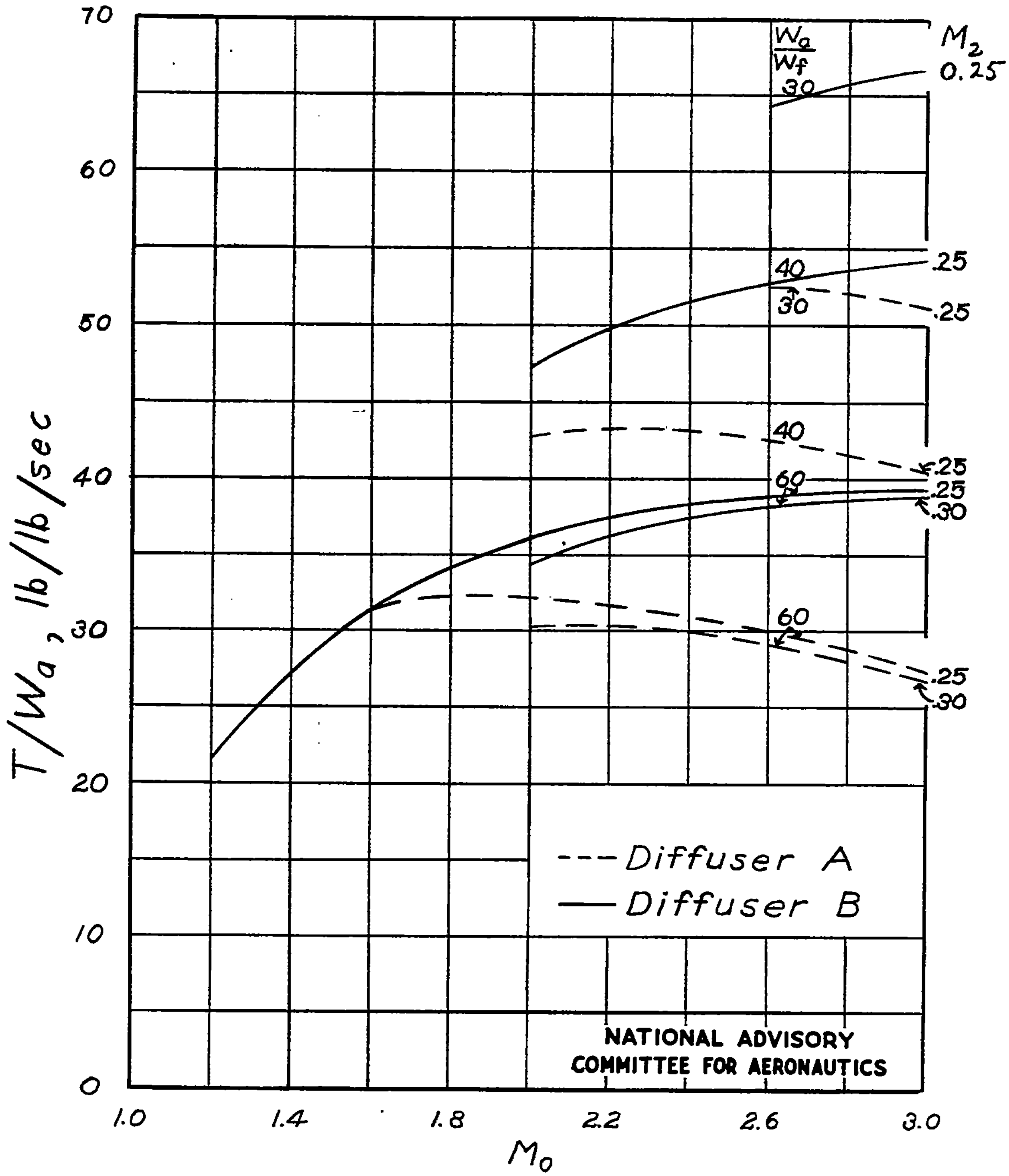


(b) Ram-jet system. $M_2 = 0.2$.

Figure 3.- Continued.



(c) Ram-jet system. M_2 = 0.25 and 0.30.

Figure 3.- Concluded.

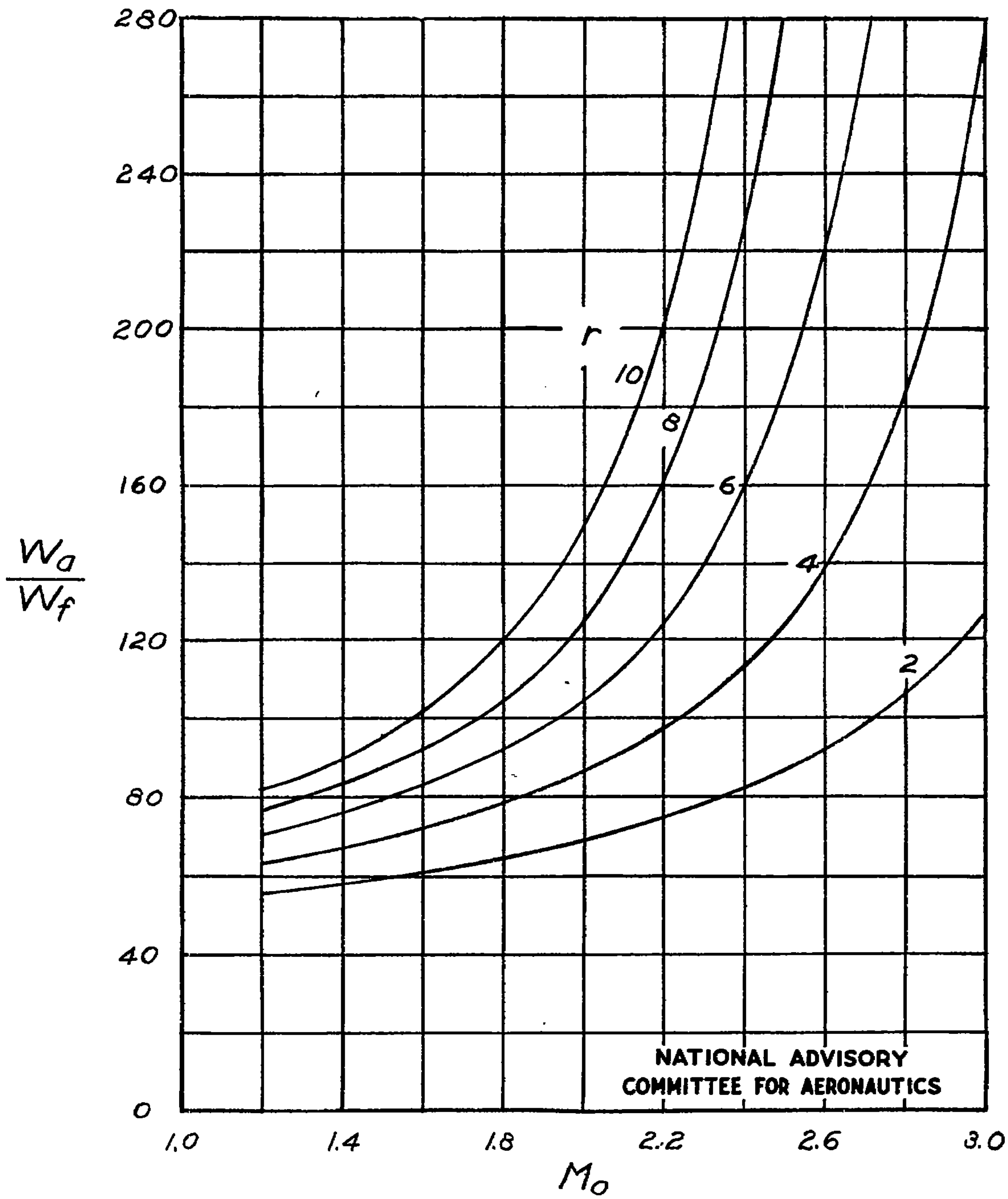
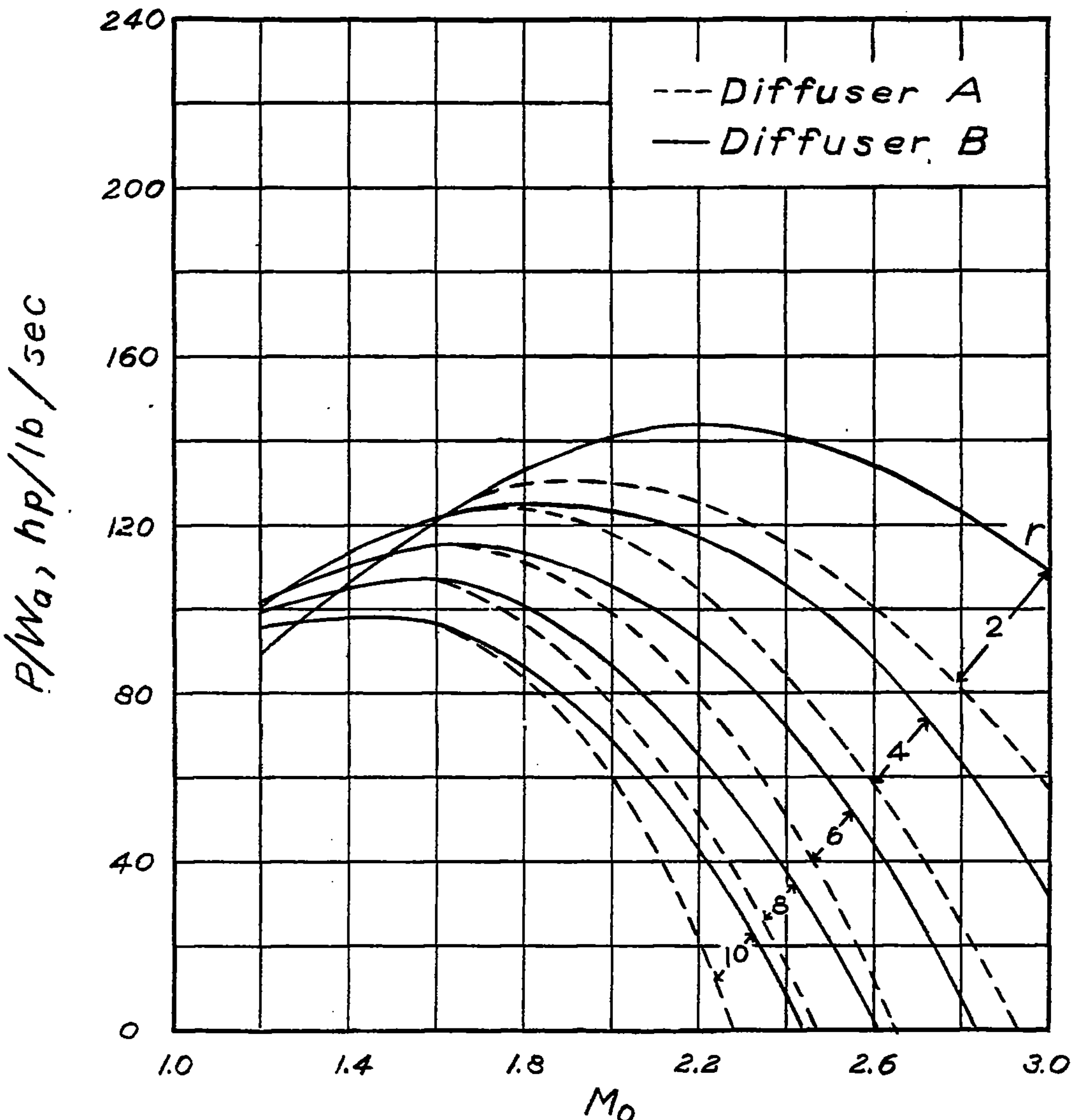
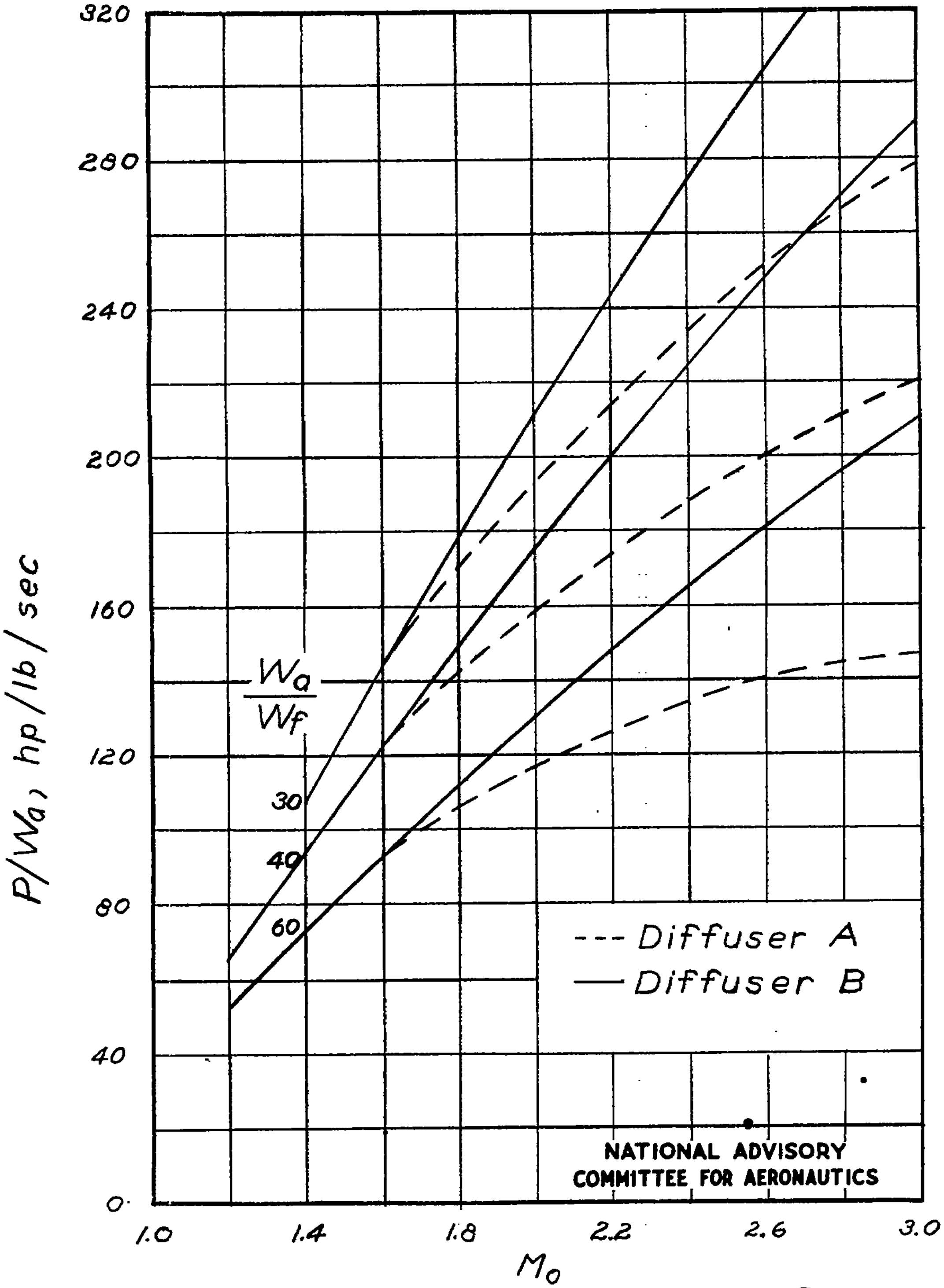


Figure 4.- Air-fuel ratio of turbojet system as a function of free-stream Mach number. Altitude, stratosphere; maximum temperature, 1500°F.



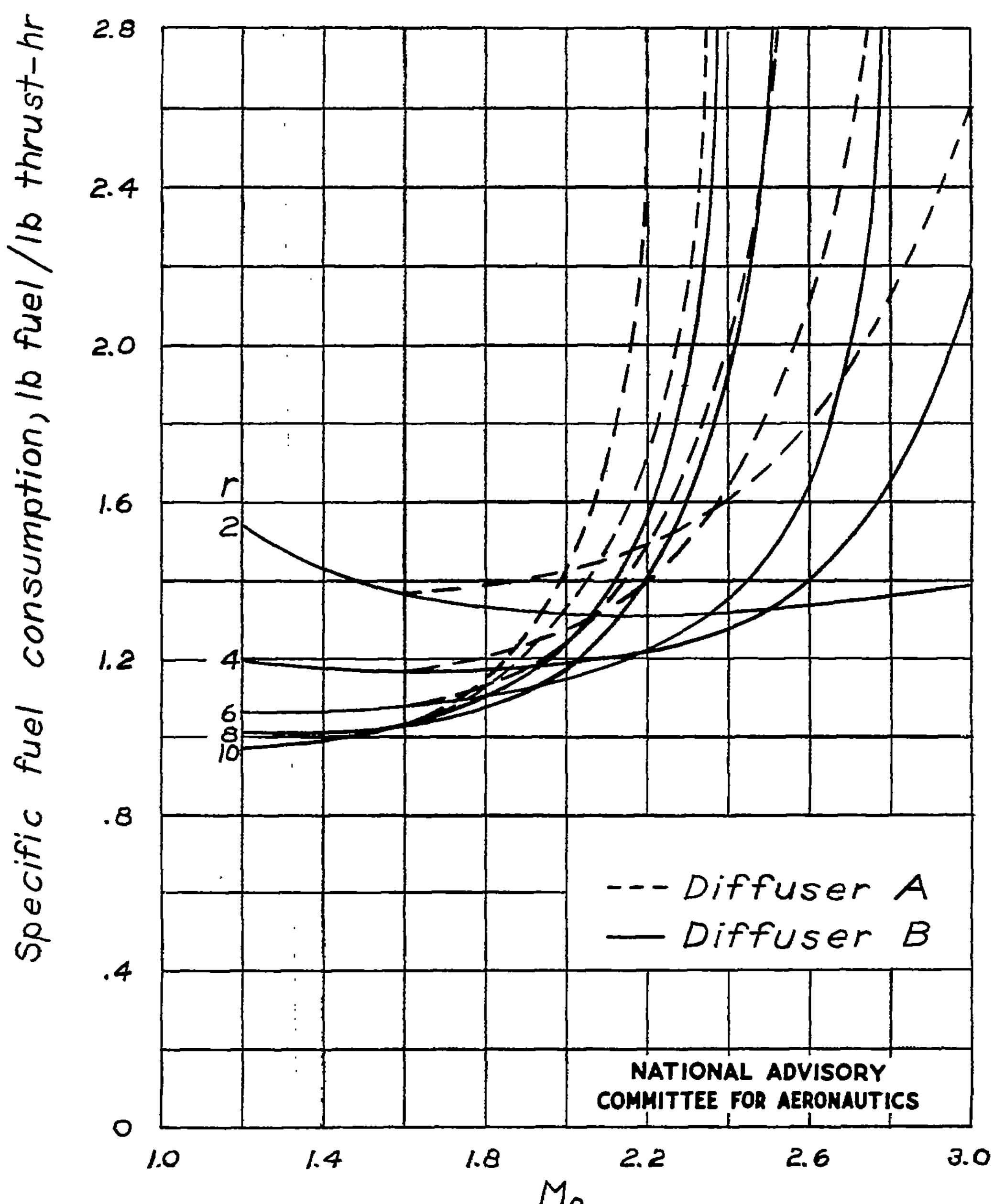
(a) Turbojet system. t_{max} = 1500°F.

Figure 5.- Thrust power per pound of air per second as a function of free-stream Mach number. Altitude, stratosphere.



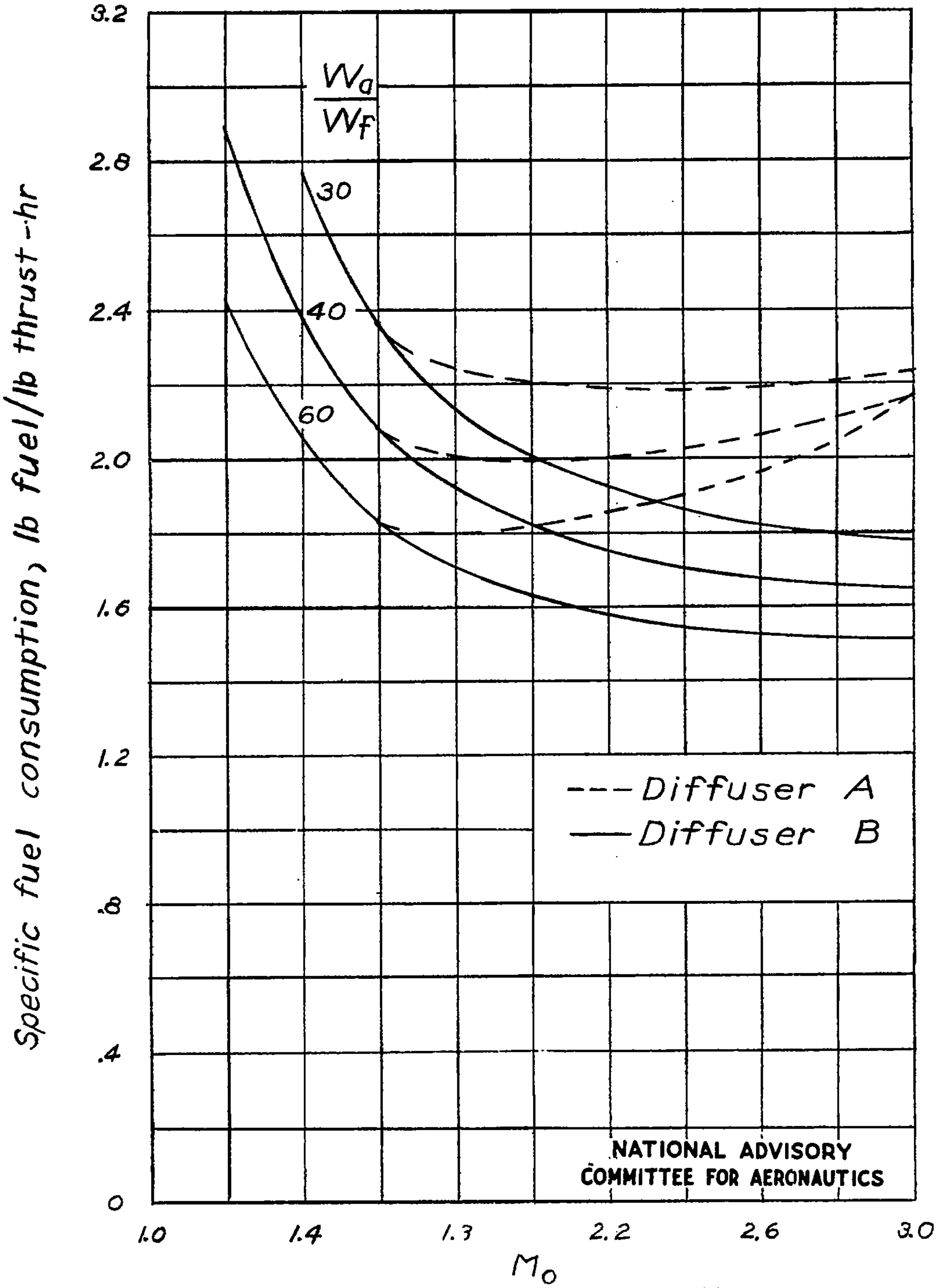
(b) Ram-jet system. $M_2 = 0.2$.

Figure 5.- Concluded.



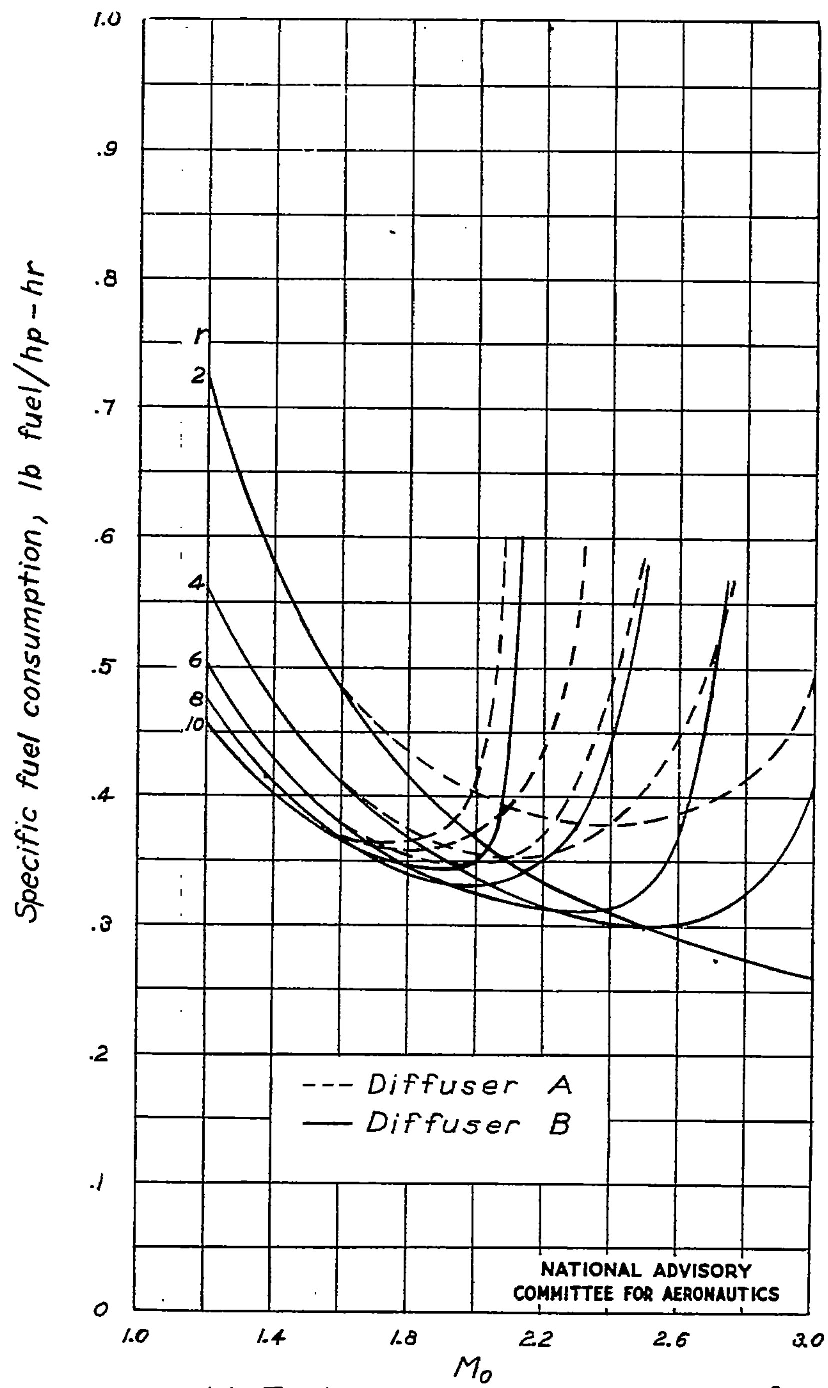
(a) Turbojet system. t_{max} = 1500°F.

Figure 6.- Specific fuel consumption, pounds fuel per hour per pound thrust, as a function of free-stream Mach number. Altitude, stratosphere.



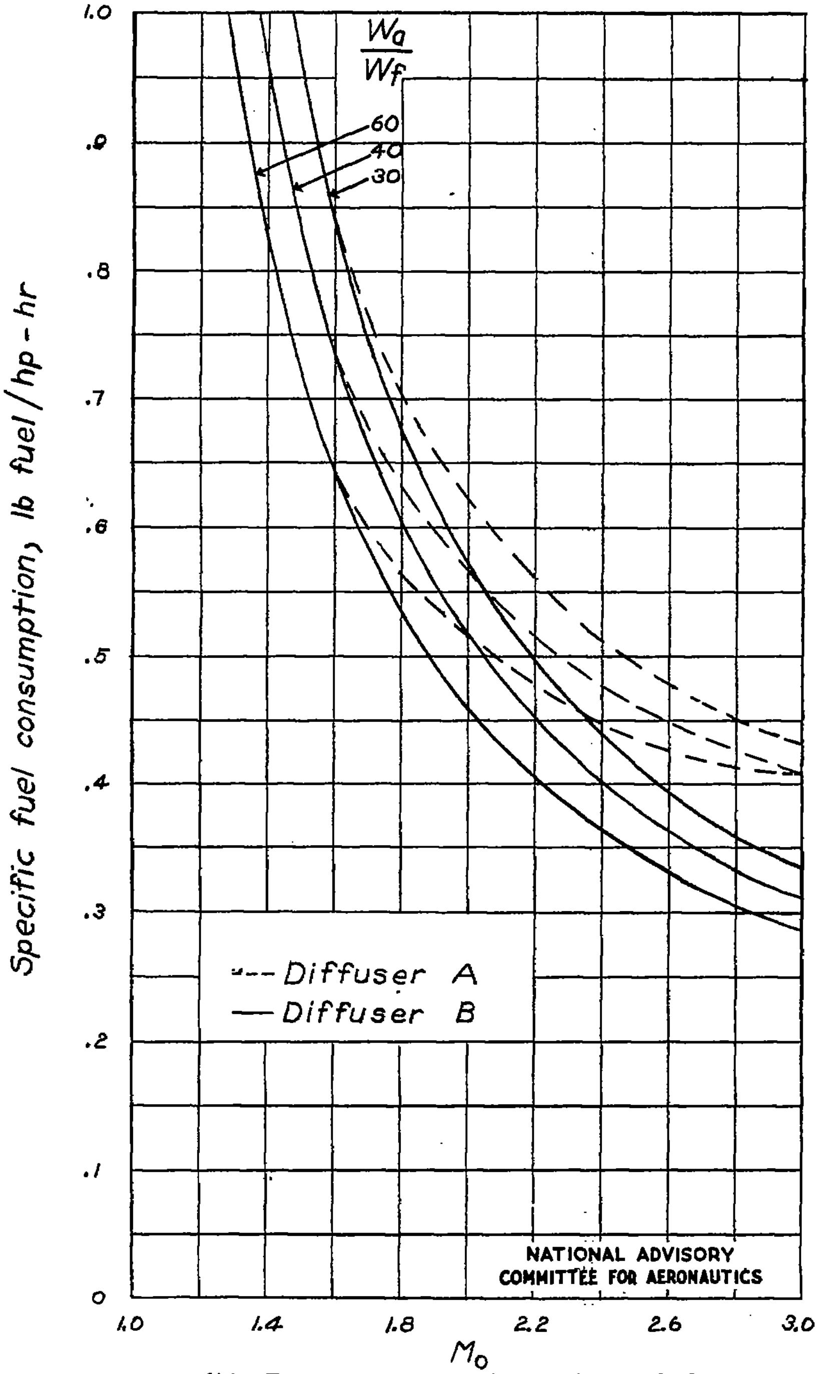
(b) Ram-jet system. $M_2 = 0.2$.

Figure 6.- Concluded.



(a) Turbojet system. $t_{max} = 1500°$ F.

Figure 7.- Specific fuel consumption, pounds fuel per horsepower-hour, as a function of free-stream Mach number. Altitude, stratosphere.



(b) Ram-jet system. $M_2 = 0.2$.

Figure 7.- Concluded.

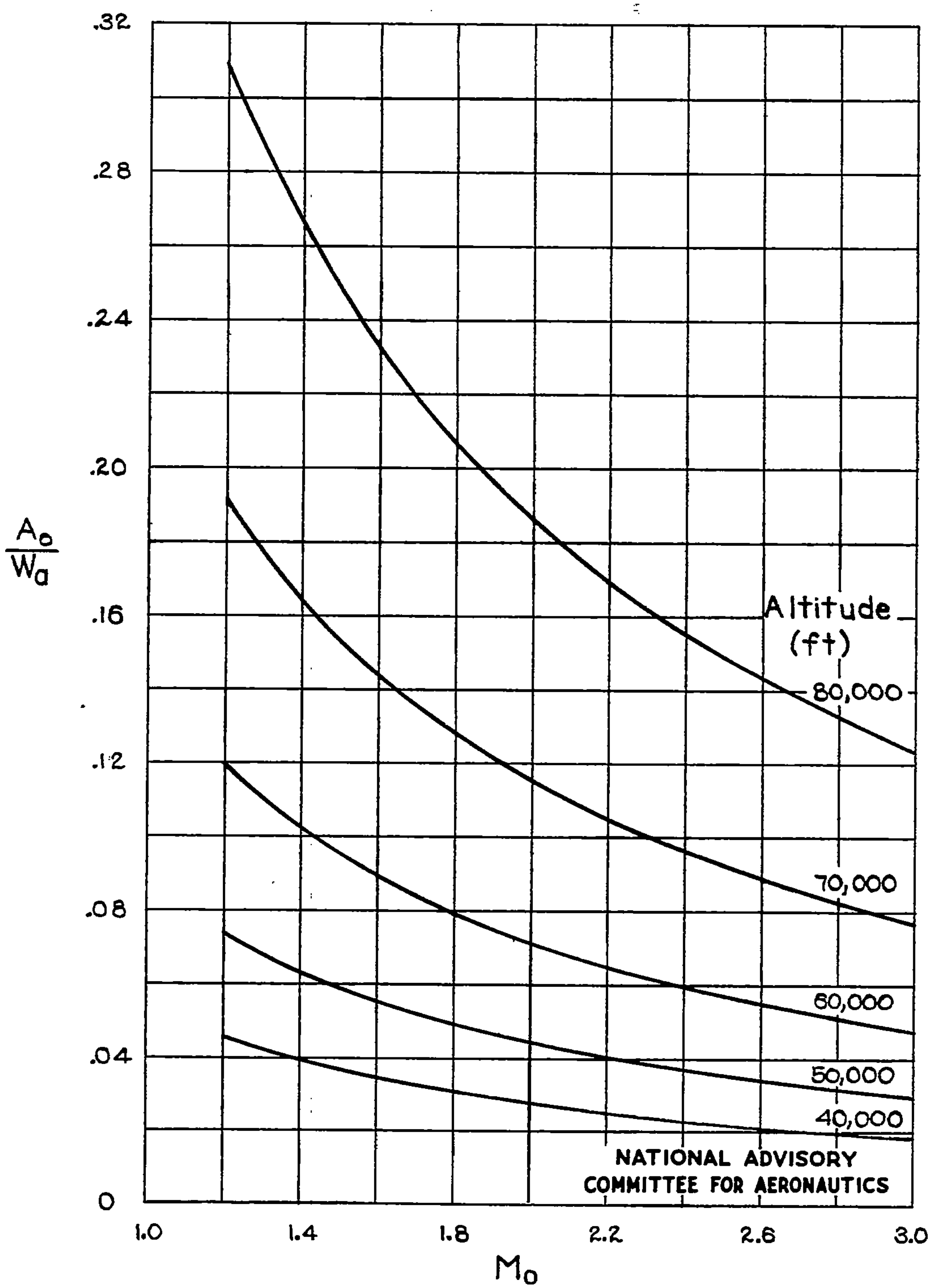


Figure 8.- Air-inlet area per pound of air per second as a function of free-stream Mach number.

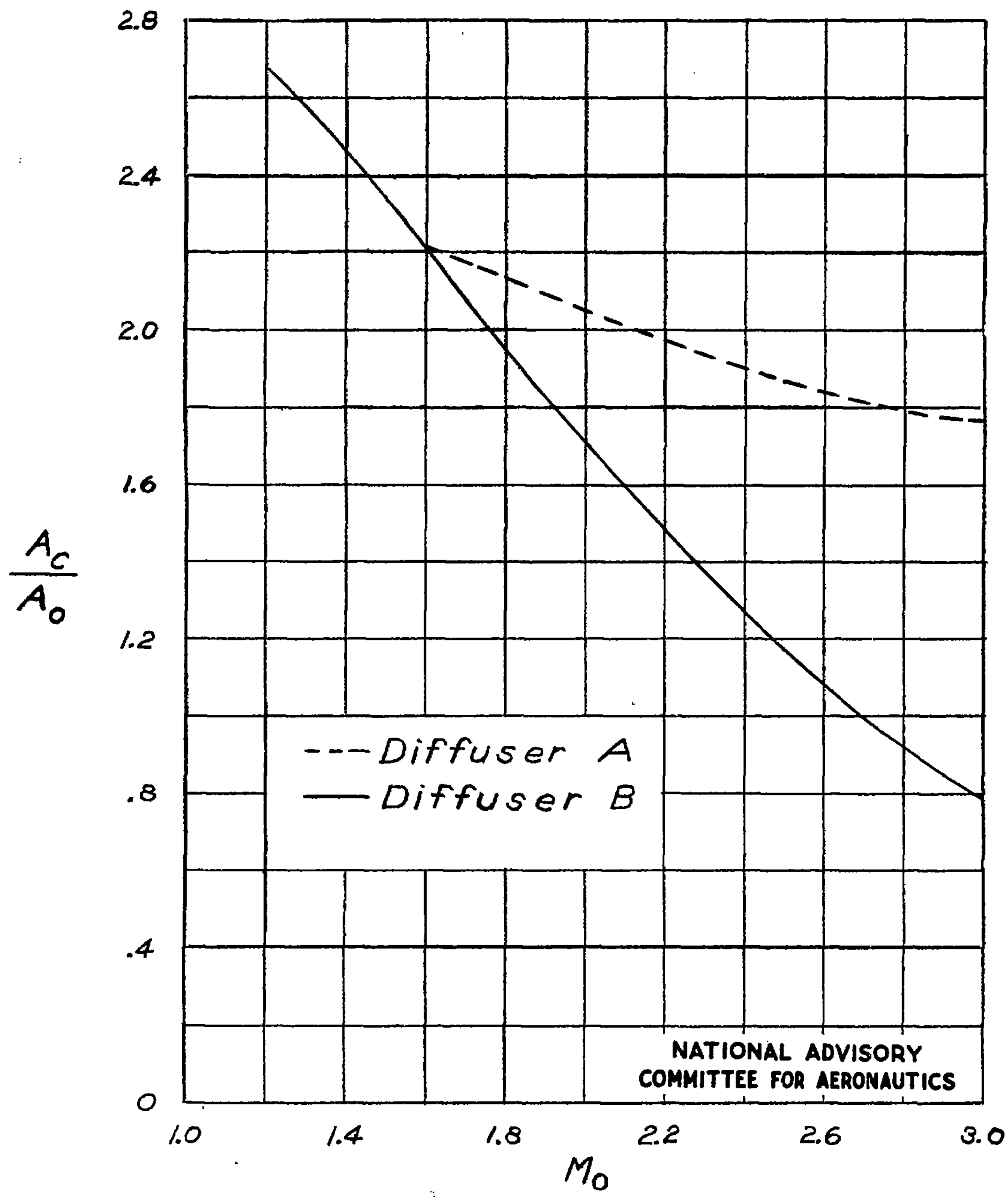
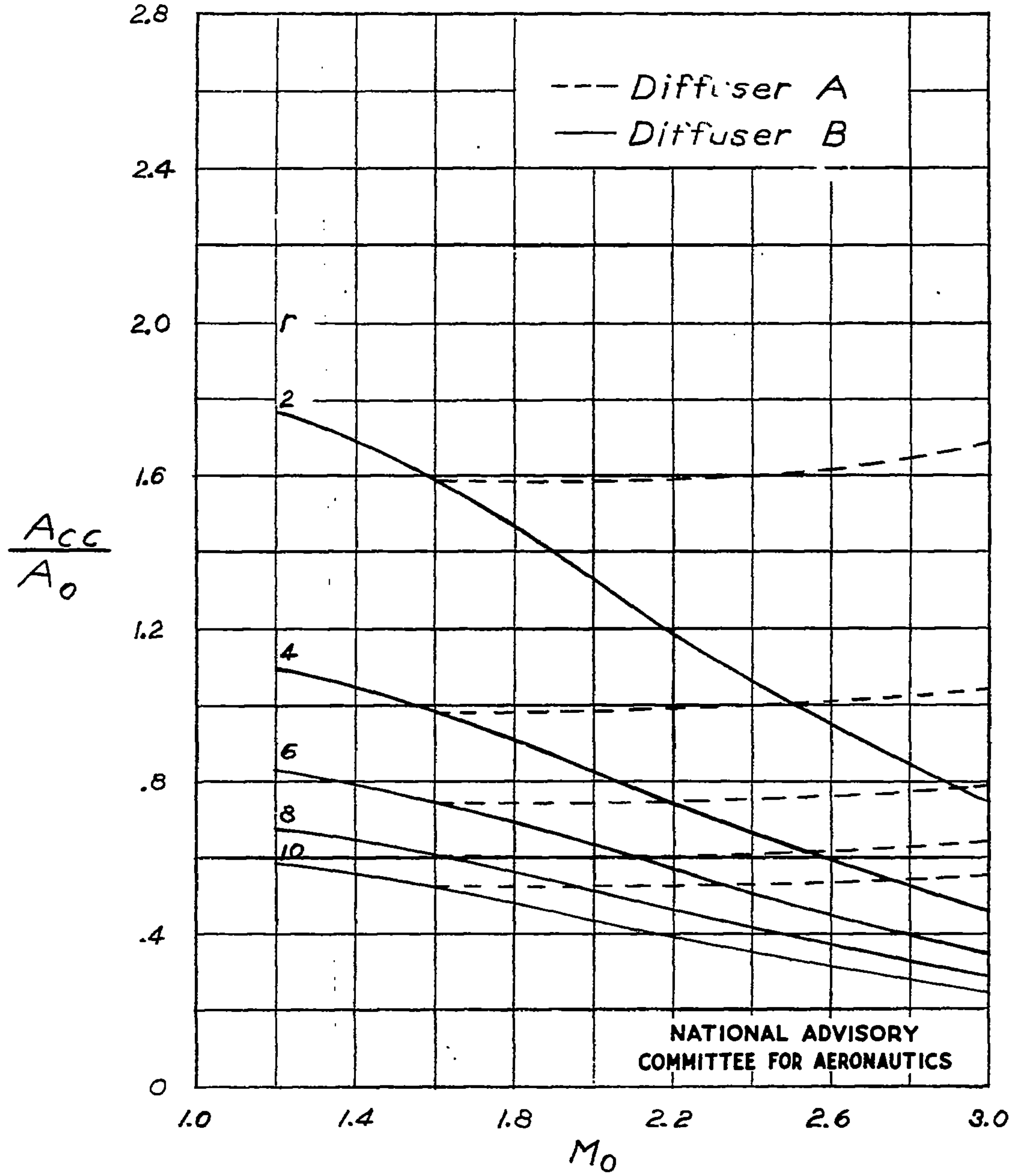
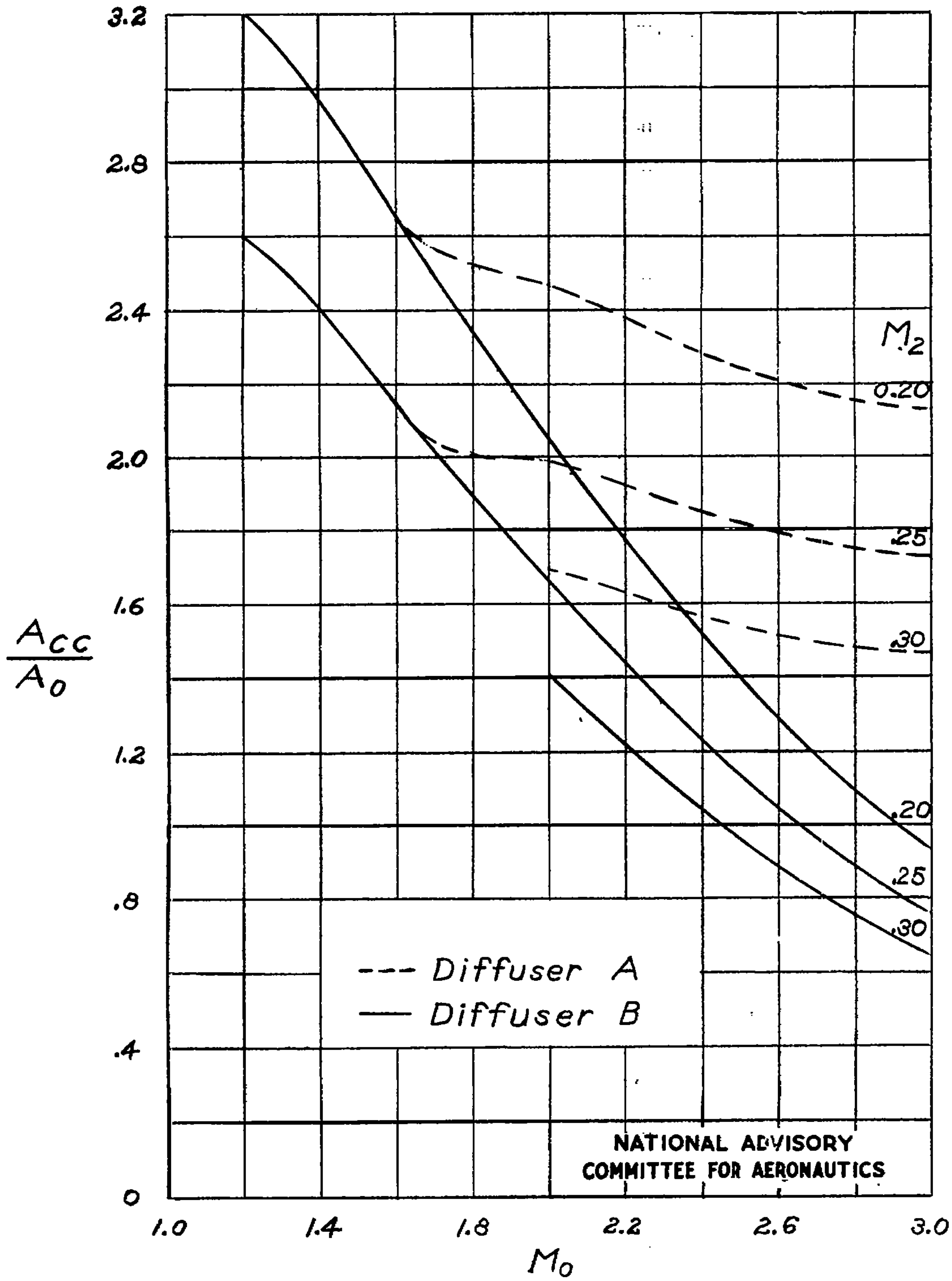


Figure 9.- Ratio of total cross-sectional area of compressor of turbojet system to air-inlet area as a function of free-stream Mach number. Altitude, stratosphere; compressor entrance Mach number, 0.4; compressor hub-diameter ratio, 0.6.



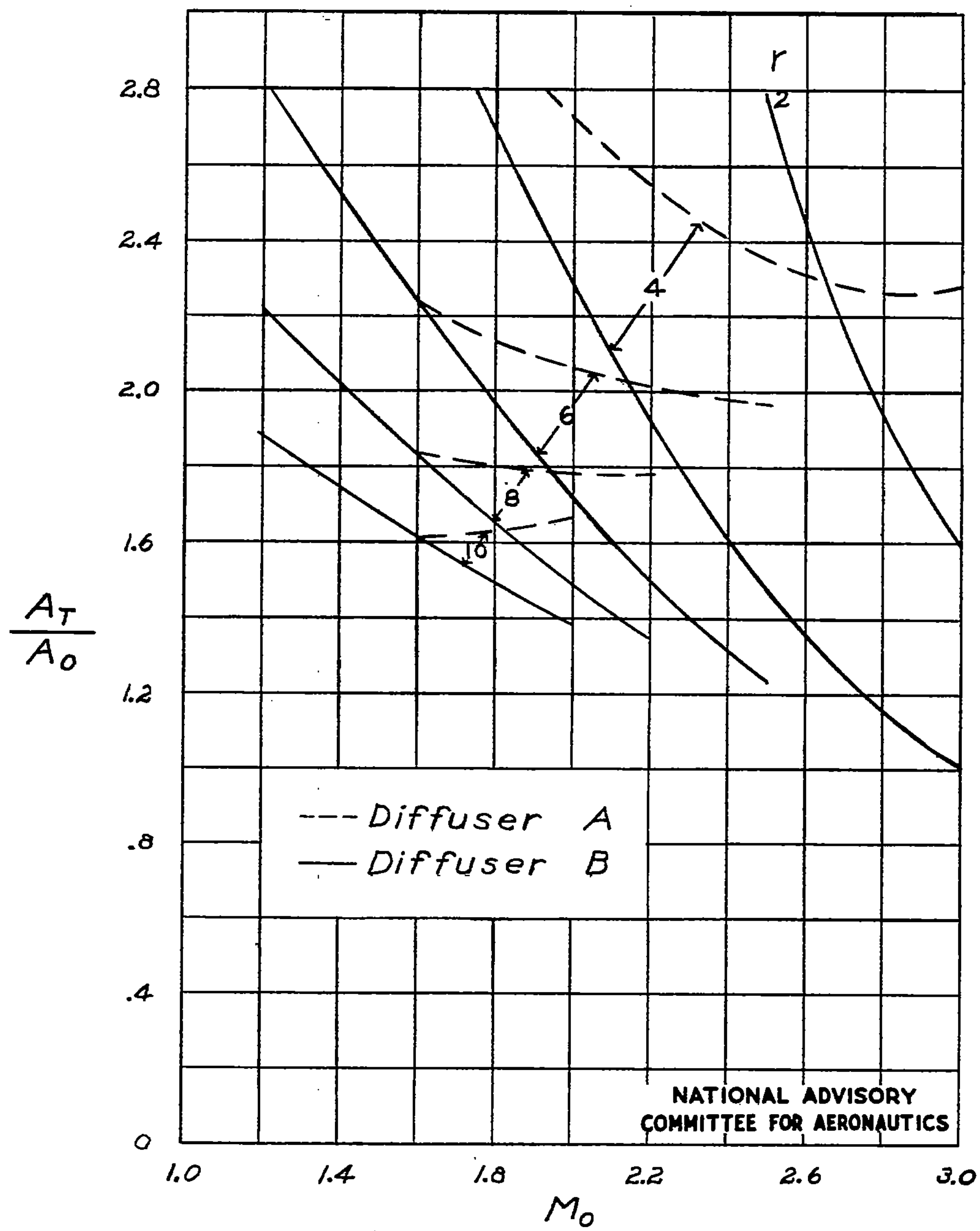
(a) Turbojet system. Combustion-chamber entrance velocity, 250 feet per second.

Figure 10.- Ratio of cross-sectional area of combustion chamber to air-inlet area as a function of free-stream Mach number. Altitude, stratosphere.



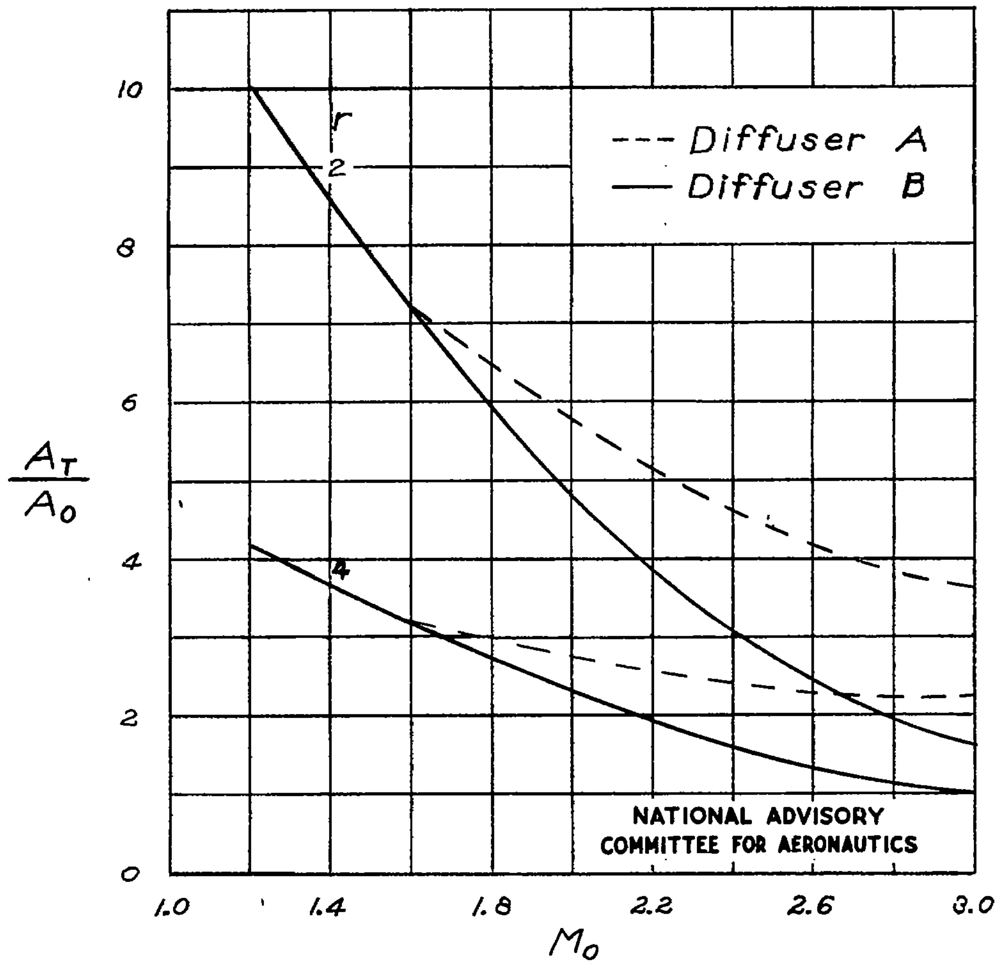
(b) Ram-jet system.

Figure 10.- Concluded.



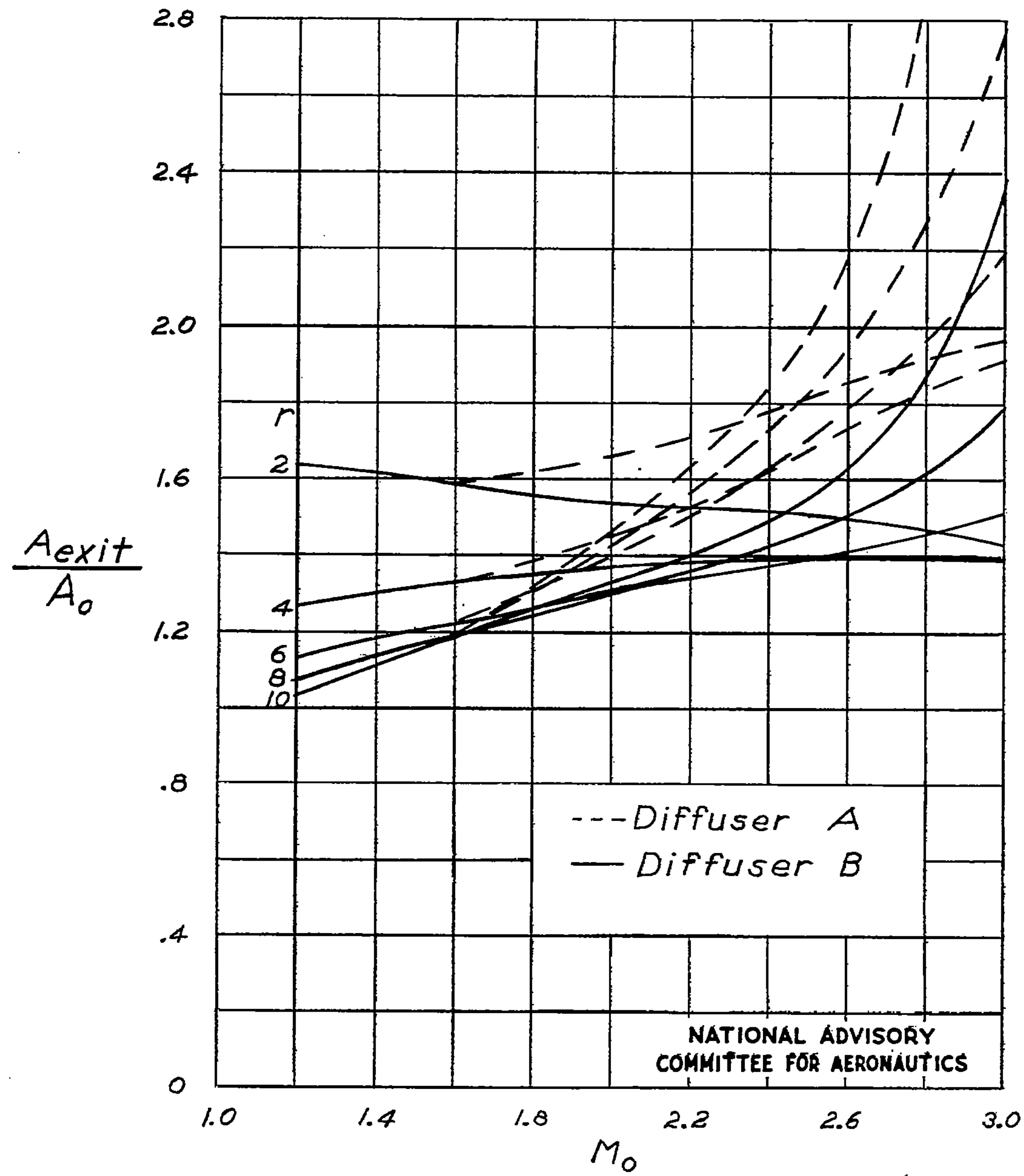
(a) r = 2 to 10.

Figure 11.- Ratio of total cross-sectional area of turbine of turbojet system to air-inlet area as a function of free-stream Mach number. Altitude, stratosphere; turbine hub-tip ratio, 0.7; maximum temperature, 1500°F.



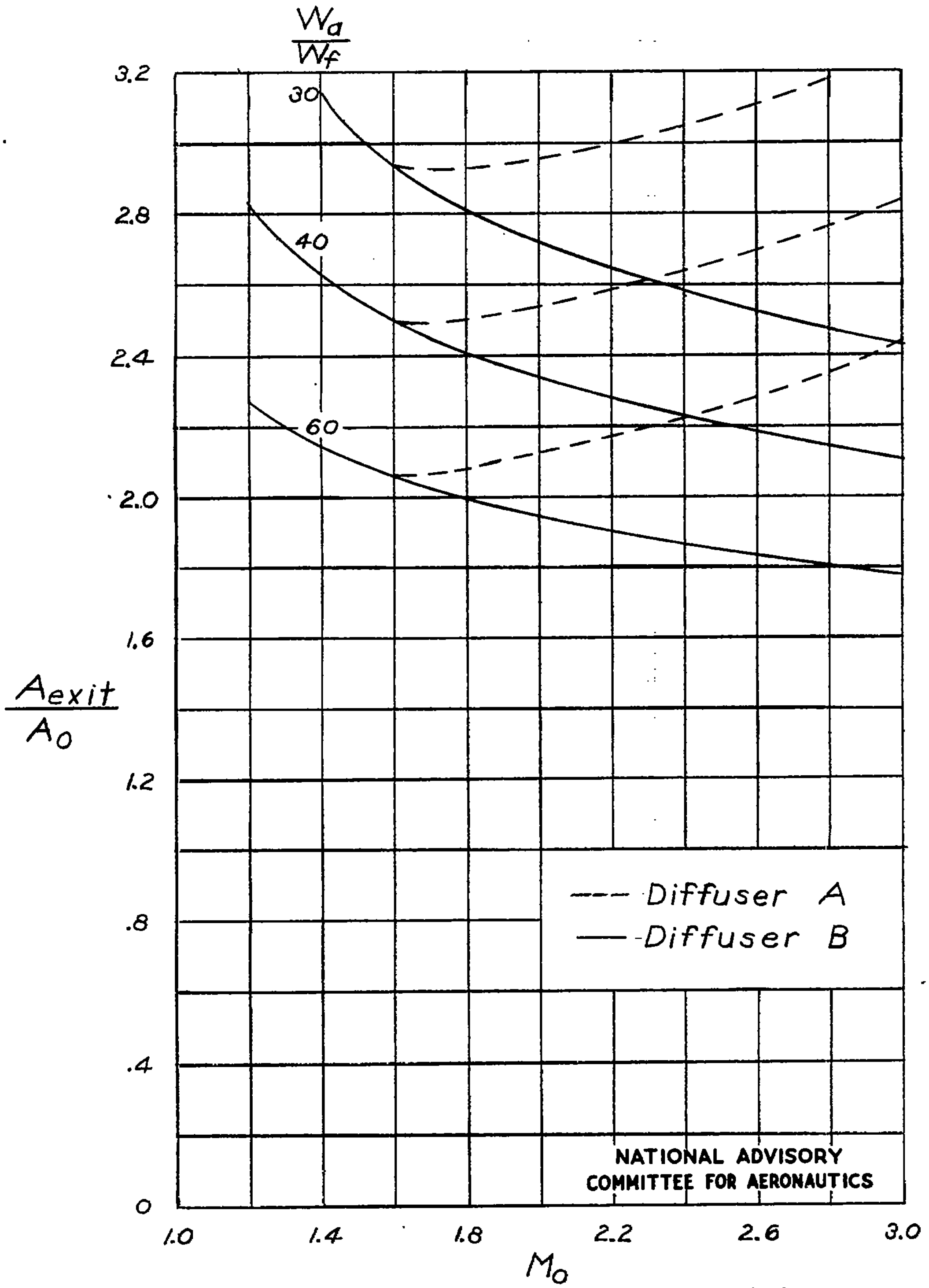
(b) r = 2 and 4.

Figure 11.- Concluded.



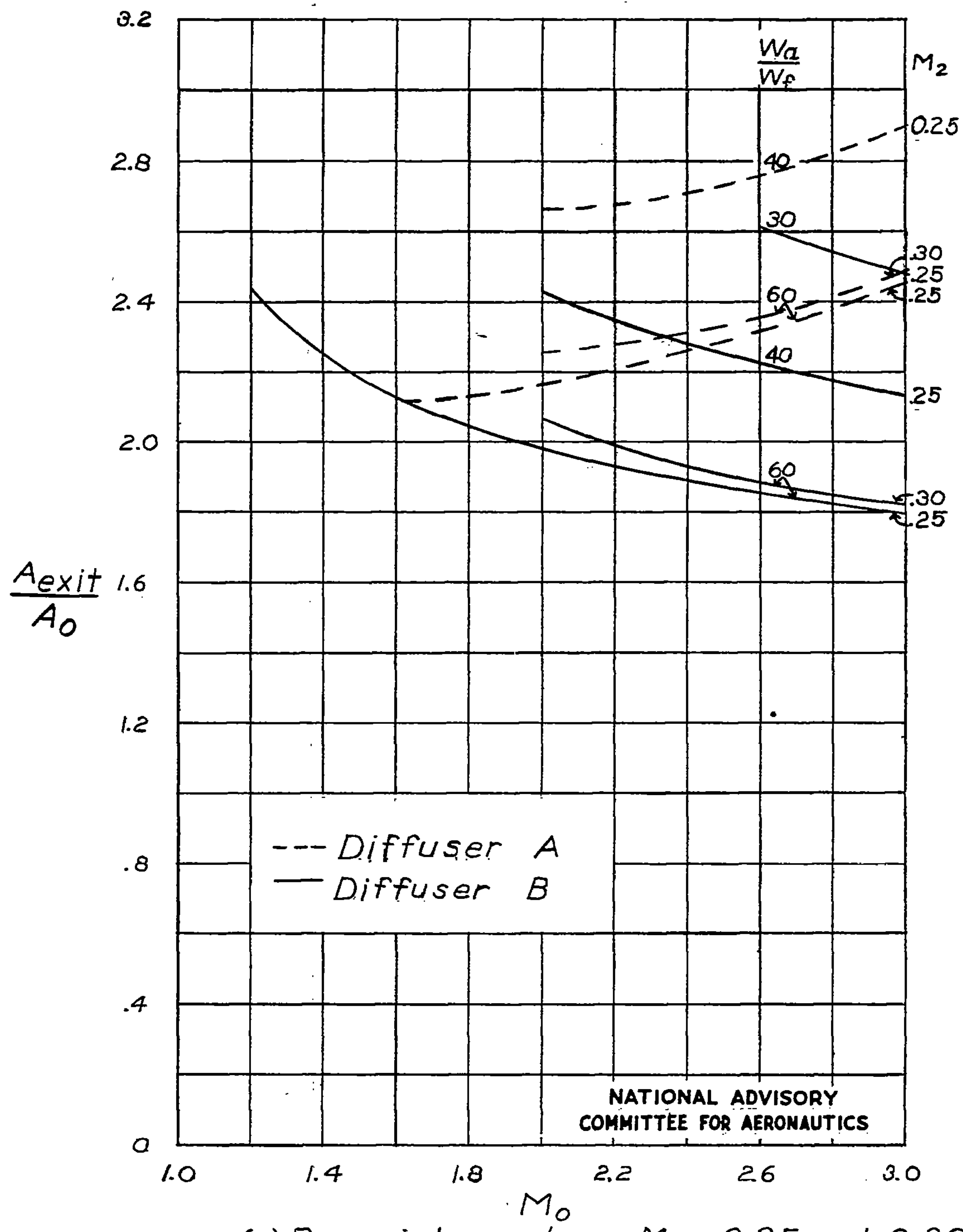
(a) Turbojet system. $t_{max} = 1500°$ F.

Figure 12.- Ratio of exhaust-nozzle exit area to air-inlet area as a function of free-stream Mach number. Altitude, stratosphere.



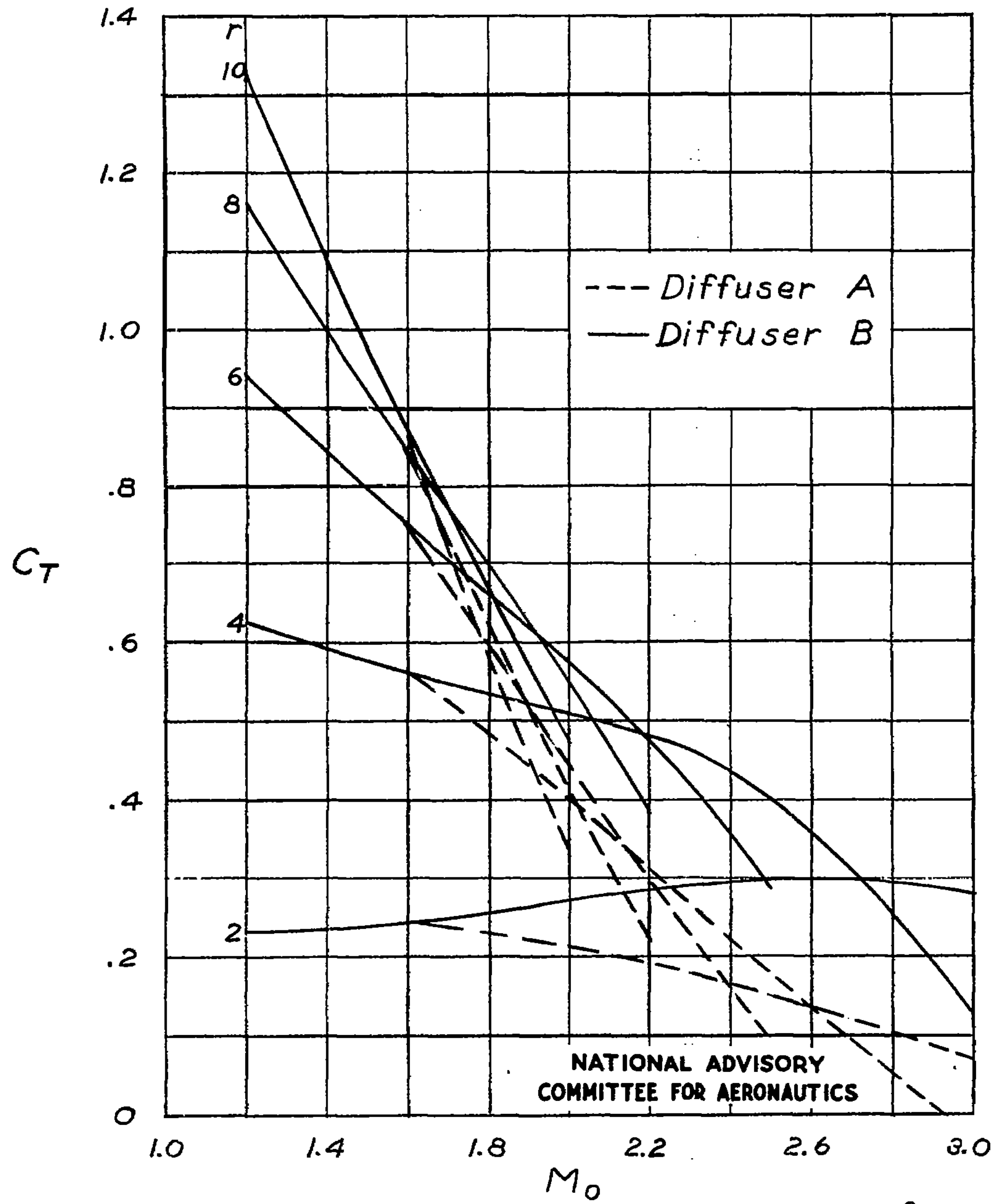
(b) Ram-jet system. $M_2 = 0.20$.

Figure 12.- Continued.



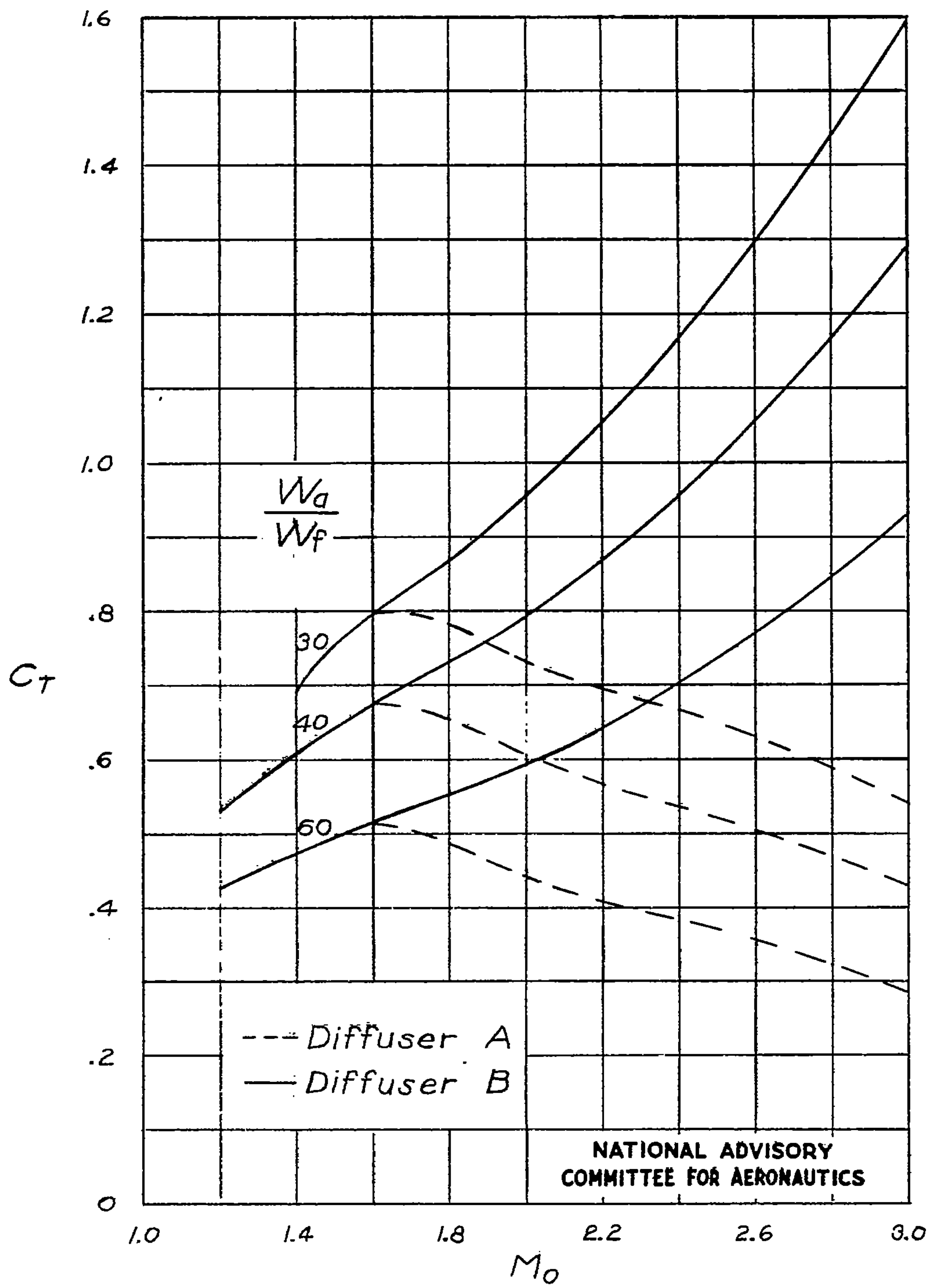
(c) Ram-jet system. $M_2 = 0.25$ and 0.30.

Figure 12.- Concluded.



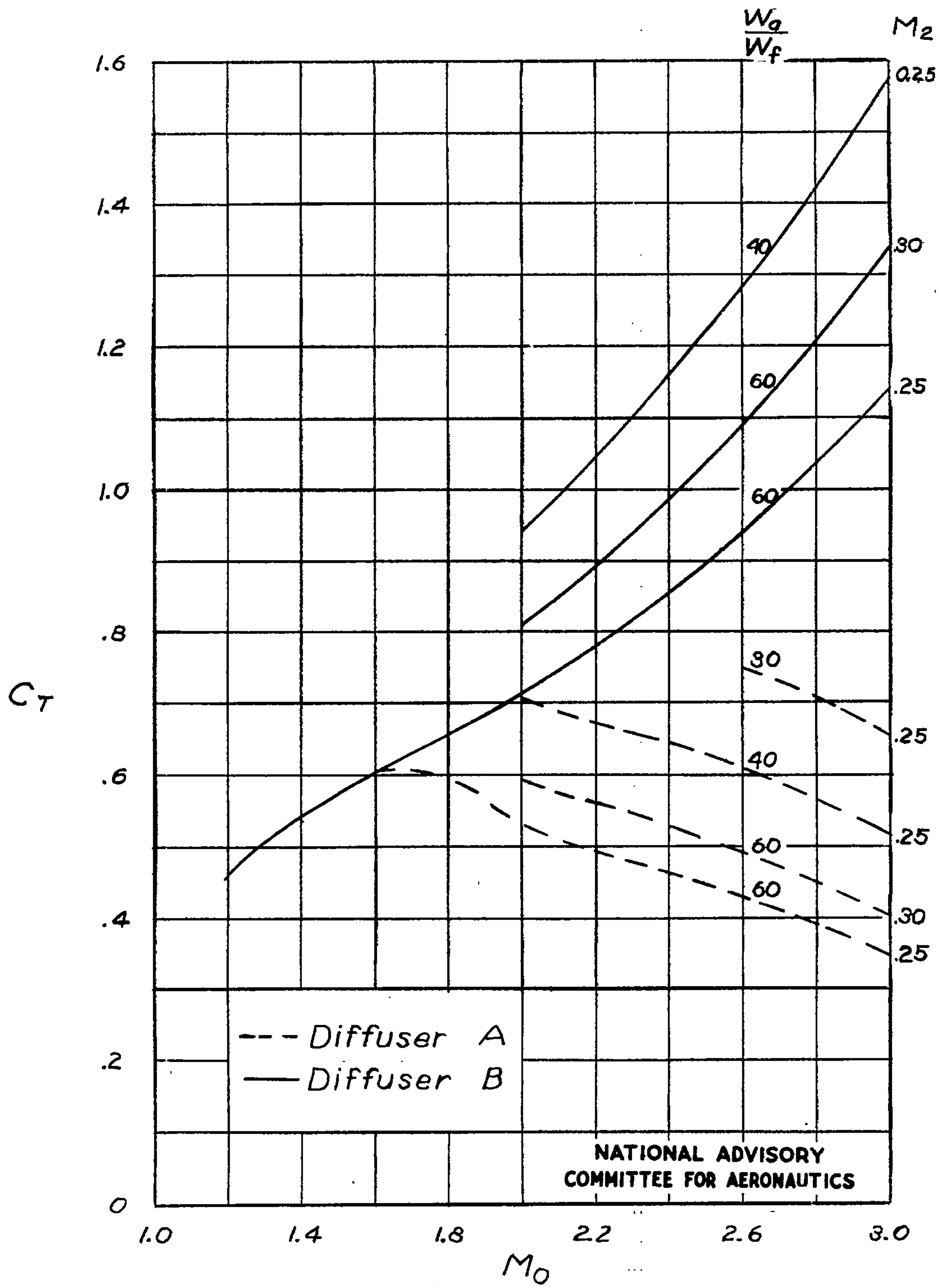
(a) Turbojet system. $t_{max} = 1500°$ F.

Figure 13.- Thrust coefficient as a function of free-stream Mach number. Altitude, stratosphere.



(b) Ram-jet system. $M_2 = 0.20$.

Figure 13.- Continued.



(c) Ram-jet system. $M_2 = 0.25$ and 0.30.

Figure 13.- Concluded.

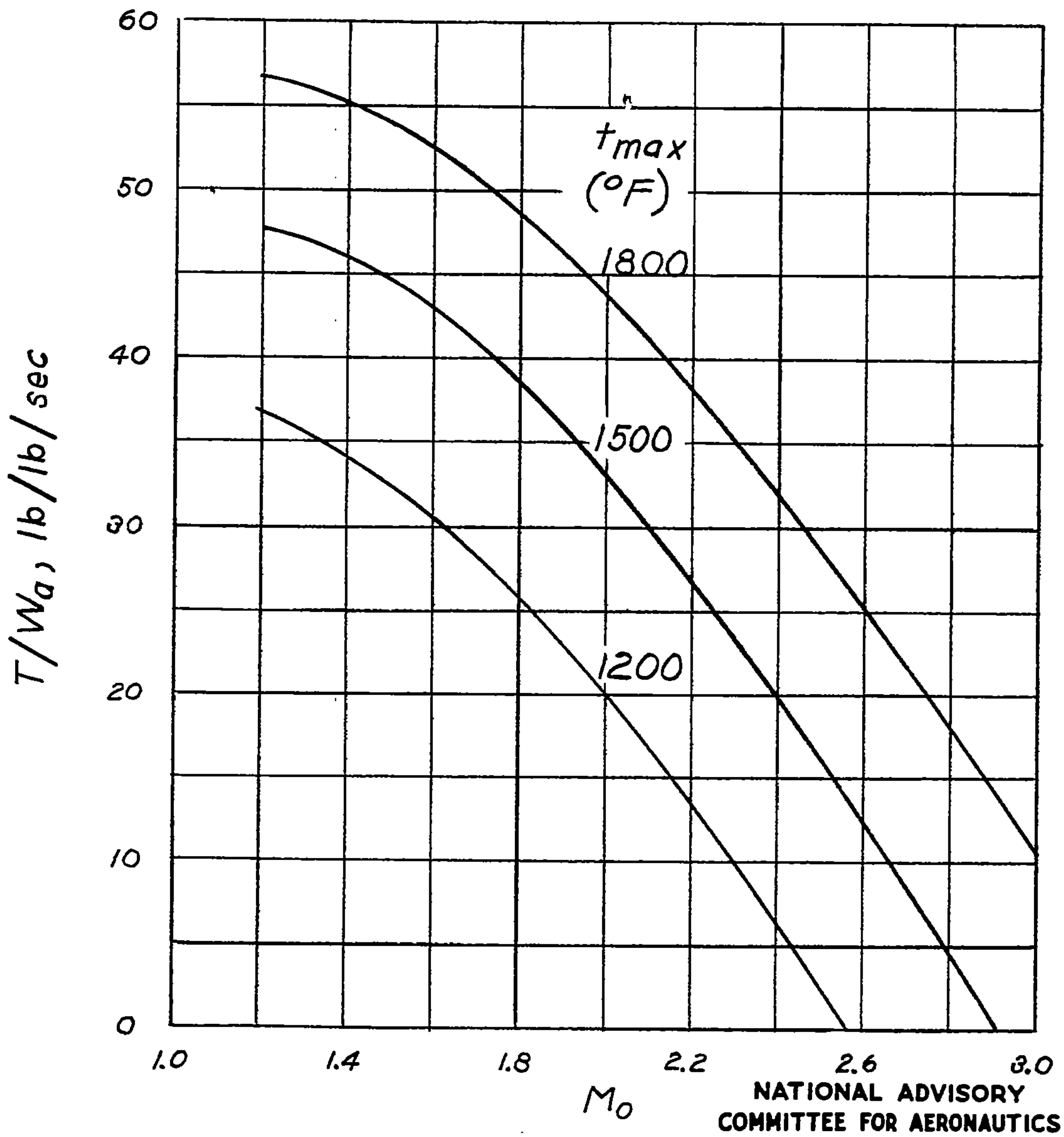


Figure 14.- Effect of maximum temperature on thrust of turbojet system. Compression ratio, 4; diffuser A.

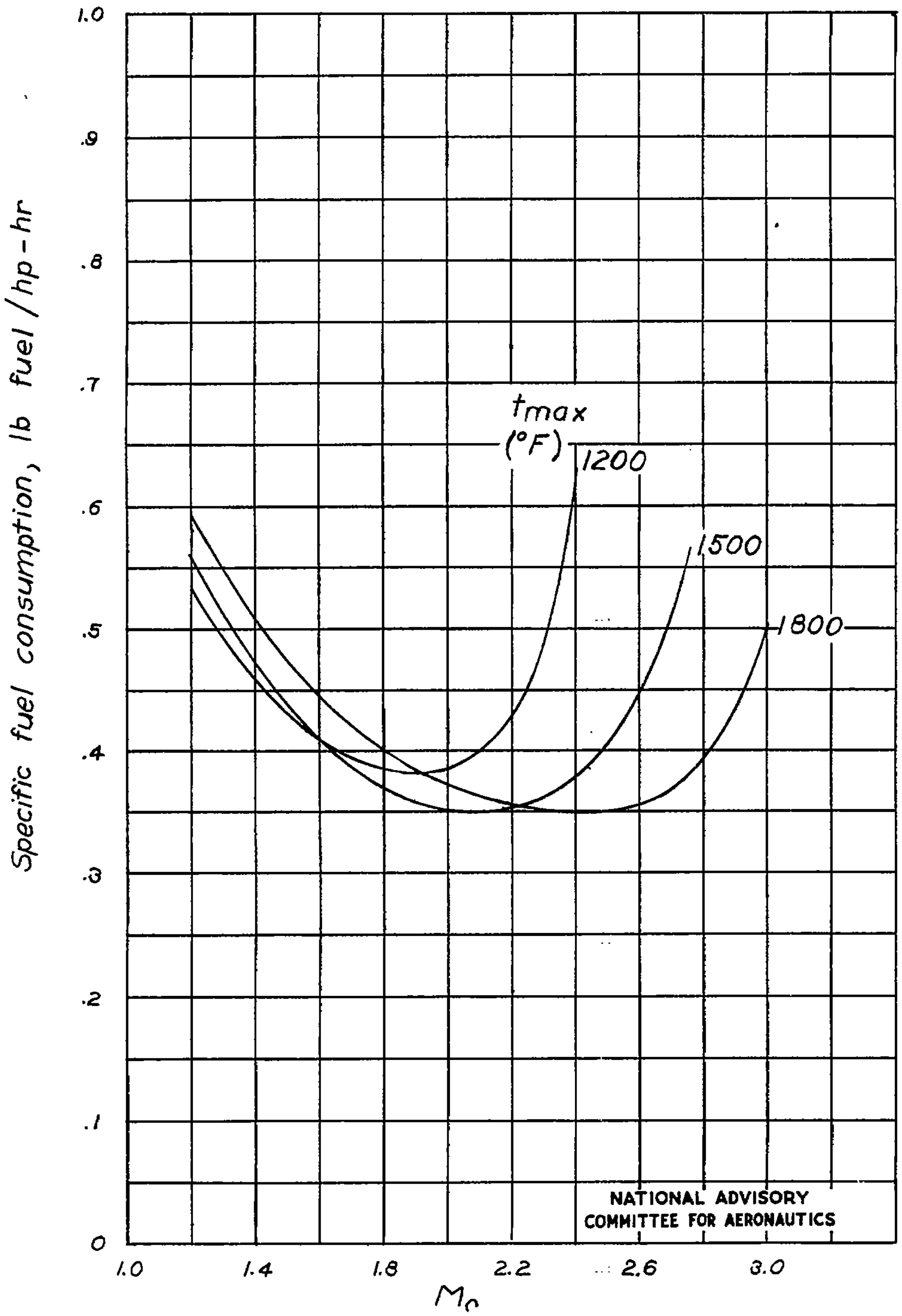
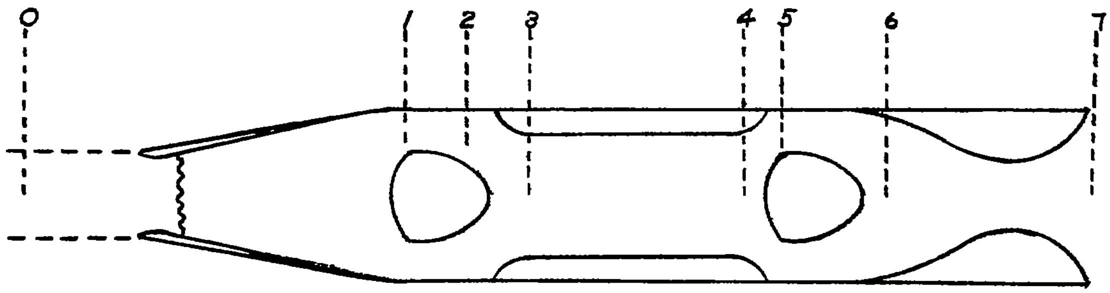
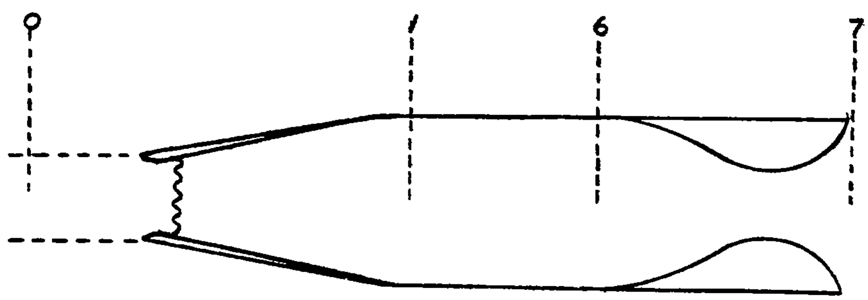


Figure 15.- Effect of maximum temperature on specific fuel consumption of turbojet system. Compression ratio, 4; diffuser A.



(a) Turbojet system.



(b) Ram-jet system.

Figure 16.- Stations used in analysis.

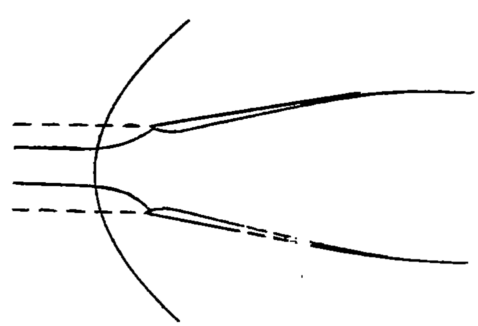


Figure 17.-Air flow at supersonic-diffuser entrance.

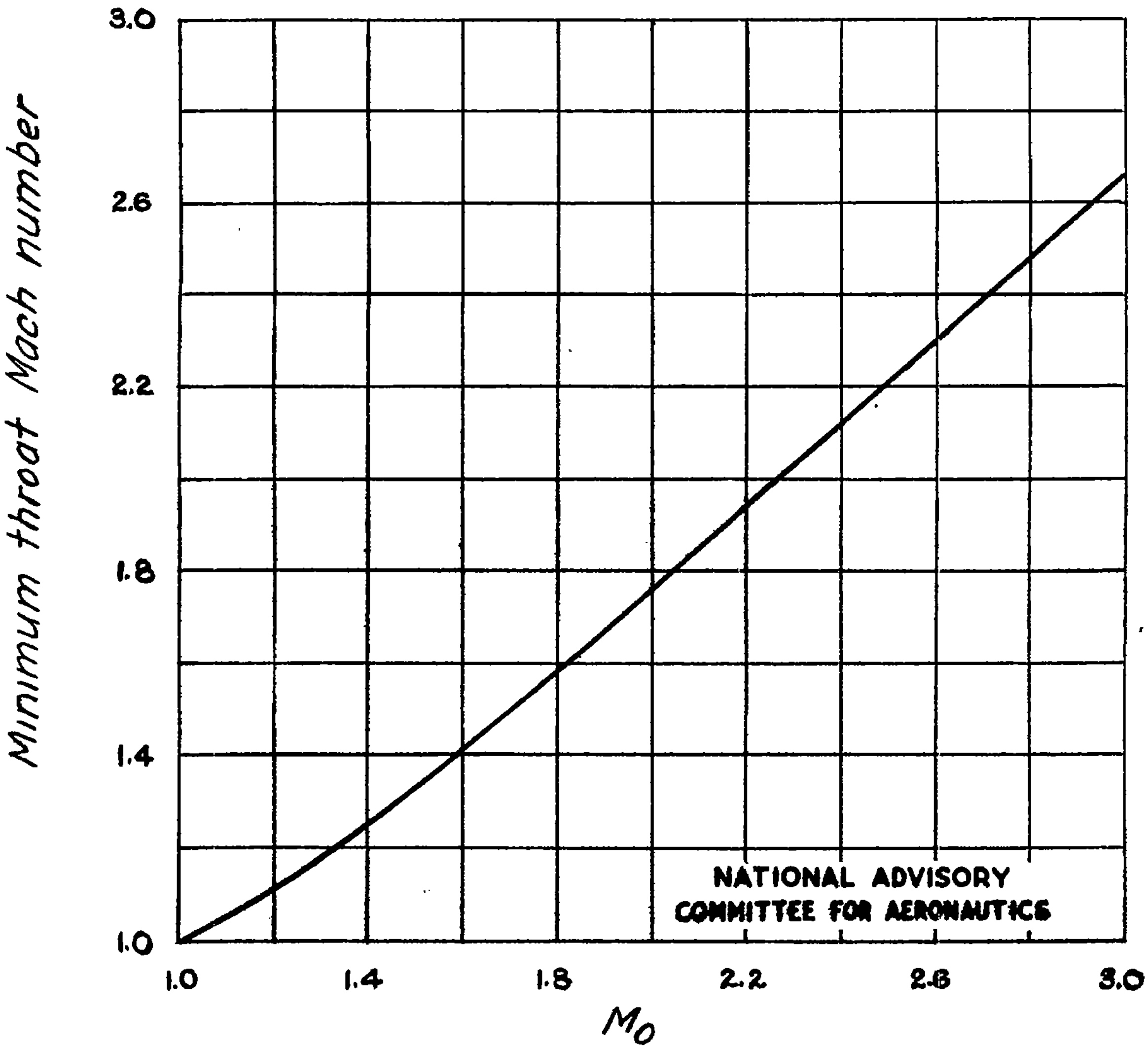


Figure 18.- Minimum diffuser-throat Mach number as a function of free-stream Mach number. (Theory of reference 4.)

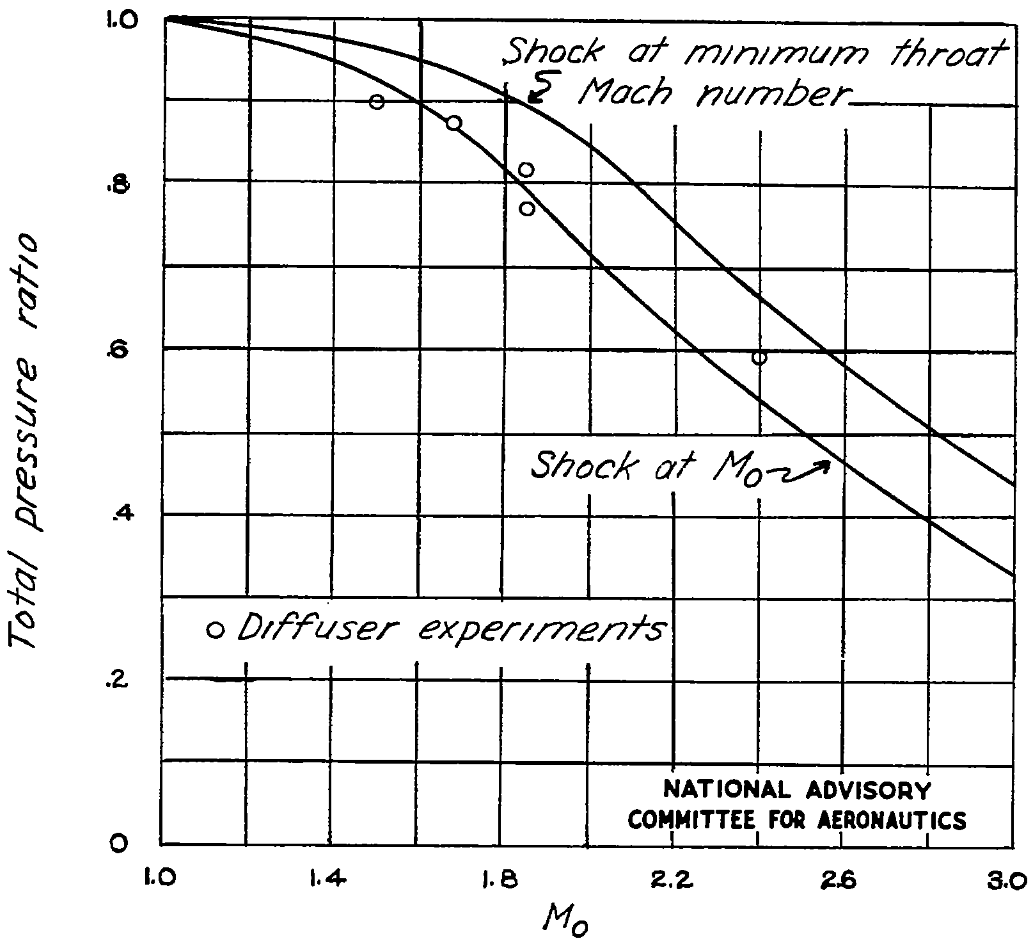


Figure 19.- Theoretical ratio of total pressure across normal shocks and experimental diffuser data of reference 4.

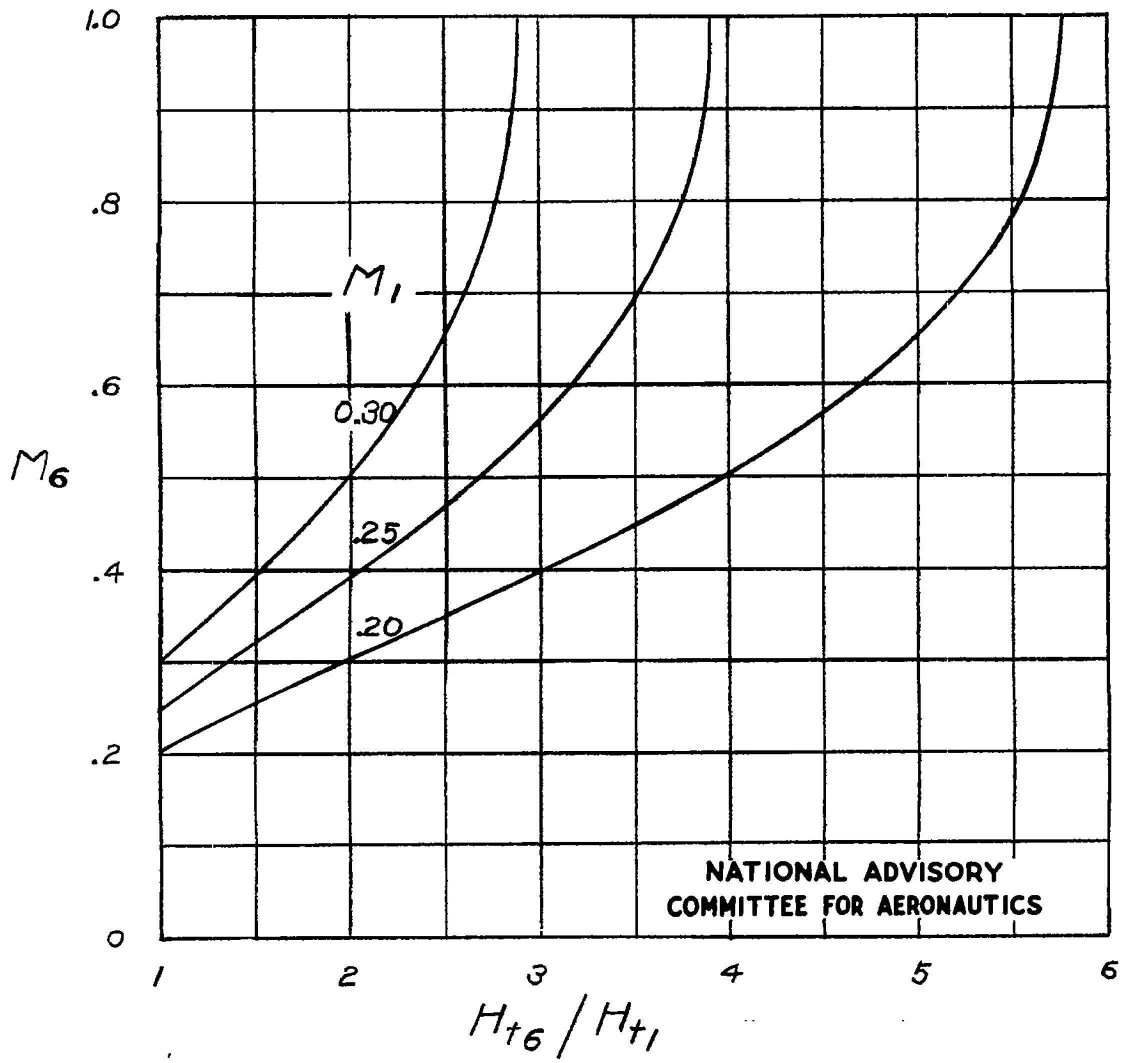


Figure 20.- Mach number following combustion as a function of the total-enthalpy ratio across the combustion chamber.

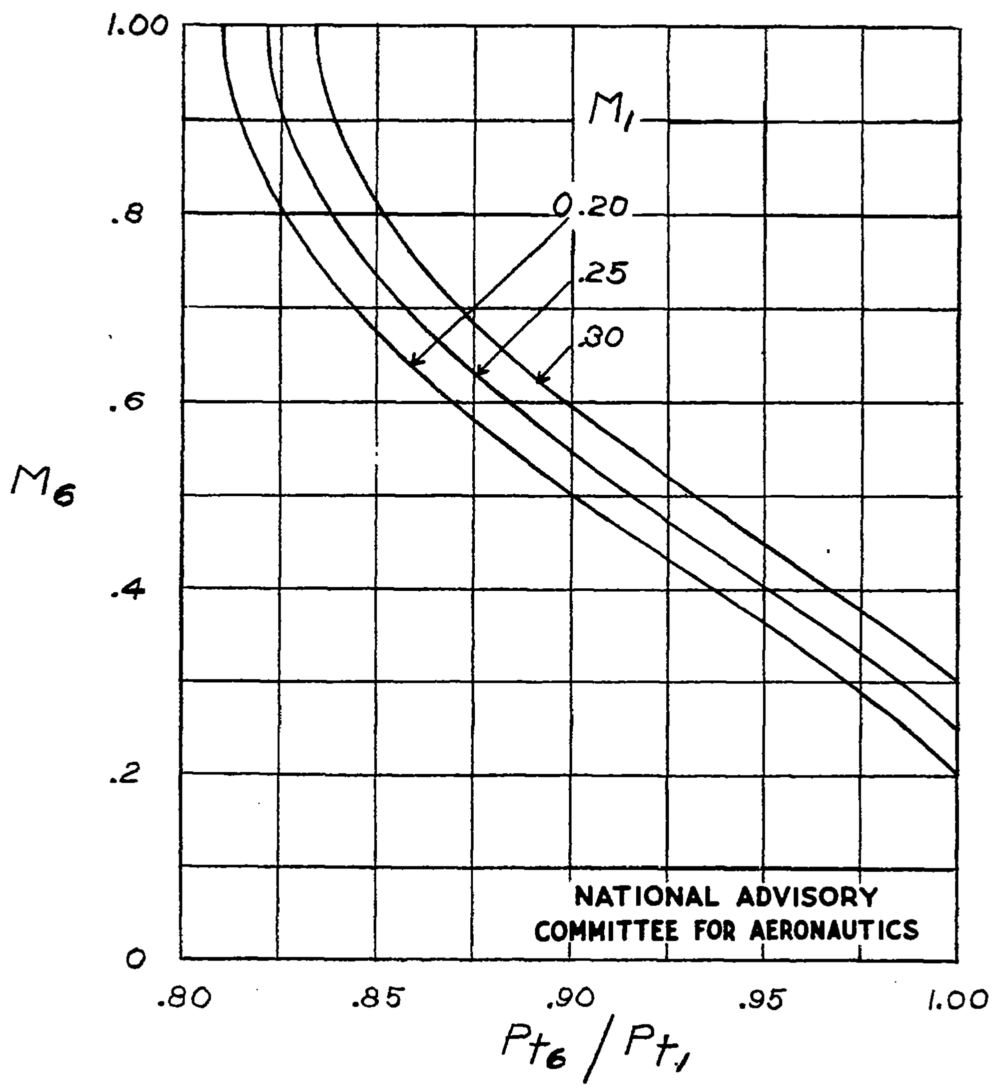


Figure 21.– Mach number following combustion as a function of the total-pressure ratio across the combustion chamber.