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RESEARCH MEMORANDUM

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THEORETICAL ANALYSIS OF THE PERFORMANCE OF A SUPERSONIC DUCTED ROCKET

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON
February 13, 1948

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319.98/13

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SUMMARY

Calculated performance characteristics of a ducted rocket that uses gasoline and liquid oxygen as the fuel mixture and has a mass flow of 1 slug per second are presented. Excess fuel is used to maintain the rocket combustion temperature within a limiting value and all this excess fuel is considered to burn in the duct enclosing the rocket. Evaluations of net thrust, frontal area, thrust per unit frontal area, and specific fuel consumption are given for flight conditions under which such a propulsion system might operate. An evaluation is made of the relative thrust increase due to the jet-ejector effect of the rocket enclosed in a duct and that due to the burning of the excess fuel from the rocket exhaust in the duct. The performances of units with nozzles designed to give exhaust-jet expansion to ambient pressure and of units designed to give the maximum thrust per unit frontal area are given.

At flight Mach numbers of 1, 2, and 3 and an altitude of 30,000 feet, thrusts per unit frontal area for the ducted rocket were of the order of twice those available from a ram jet. The corresponding specific fuel consumptions were about two to three times those of a ram jet, but about one-half to one-third that of the rocket alone. The thrust increase was due primarily to the burning of the excess fuel from the rocket inasmuch as only about one-fourth of the total increase in thrust resulting from the ducting was attributable to the jet-ejector effect experienced when the rocket is enclosed in a duct under these flight conditions. For the same flight conditions, the specific fuel consumption of the ducted rocket was lower than that of a combination of equal frontal area consisting of a similar rocket and a ram jet operating independently at stoichiometric fuel-air ratio. The net thrust of the ducted rocket was higher at Mach numbers of 2 and 3, but lower at a Mach number of 1 than the thrust of the combination at comparable flight velocities.

At static conditions, the thrust of the ducted rocket was approximately 9 percent greater than the thrust of the rocket alone.

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The use of reasonable values of the component efficiencies did not alter the characteristics presented enough to affect seriously the performance of the ducted rocket under the flight conditions considered.

INTRODUCTION

A possible method of increasing the output of an aircraft power plant of the jet-propulsion type is to enclose the jet within a duct through which a relatively large air flow exists. The resultant mixing of the exhaust jet and the air flowing through the duct produces a jet-thrust augmentation, which may be of such magnitude that a considerable increase in thrust results. This thrust-augmentation phenomenon has been investigated by Jacobs and Shoemaker (reference 1) and analyses have been made of aircraft-propulsion systems that use this method to increase the thrust available from a rocket (references 2 and 3).

Because an increase in thrust can be obtained without an increased expenditure of fuel, the efficiency of the cycle as compared with the efficiency of the same power plant without augmentation is thereby increased. In order to further increase the thrust, additional fuel may be supplied and burned within the duct, in which case the efficiency may or may not be greater than that of the unit without augmentation, depending on the relative efficiencies of the primary cycle and the ram-jet cycle within the duct.

A preliminary analysis of such a system utilizing a rocket mounted inside a duct was made at the NACA Cleveland laboratory. The rocket on which the analysis was based is a hypothetical unit that burns a mixture of gasoline and liquid oxygen. The characteristics of the rocket were so chosen that they lie within the ranges of values obtainable in the operation of actual rockets.

The maximum allowable temperature in the combustion chamber of a rocket is ordinarily determined by the temperature limits of the structural materials used. In the present case, an amount of gasoline in excess of that required for a stoichiometric mixture was assumed in order that the arbitrarily chosen temperature limits might not be exceeded. In most of the calculations, the excess fuel from the rocket was assumed to burn completely in the duct; these calculations therefore served to evaluate the total thrust increase available by mounting the rocket under consideration inside a duct. In order to estimate the relative thrust increases due to the action of the unit as a jet ejector and to the combustion within the duct of the excess rocket fuel, supplementary calculations were made with the assumption that none of the excess fuel from the rocket burned in the duct.

The performance data are presented as a function of the ratio of the mass of the air flow through the duct to the mass of the rocket-gas flow. Data are presented for several typical flight conditions at which such a propulsion system might be utilized. These conditions include the performance at design flight conditions of ducted rockets designed to operate at Mach numbers of 1, 2, and 3 at an altitude of 30,000 feet, and the performance at a Mach number of 1 at altitudes from sea level to 45,000 feet of a unit designed to operate at a Mach number of 1 at an altitude of 30,000 feet. Data are presented showing the effect on performance of ducting a rocket for static sea-level operation. An evaluation is made of the maximum thrust per unit frontal area obtainable with the ducted rocket considered at an altitude of 30,000 feet at flight Mach numbers of 1, 2, and 3. The decreases in the net thrust and in the thrust per unit frontal area that result from pressure losses in the diffuser and the nozzle are evaluated for a flight Mach number of 2 and an altitude of 30,000 feet. A comparison on the basis of the net thrust produced and of the specific fuel consumption for ducted rockets and for units powered by an independently operating stoichiometric ram jet and a rocket similar to that used in the ducted rocket is given. The two propulsion systems are assumed to have equal frontal areas.

Physical characteristics of the ducted rocket (other than the required frontal area) are not considered, nor is any attempt made to evaluate the advantages or disadvantages of such a system as compared with other available means of propulsion in the same speed range insofar as weight, simplicity of design, aerodynamic characteristics, and so forth are concerned.

CONFIGURATIONS

The basic scheme of the ducted rocket as considered herein is shown in figure 1. Diffusion of the air flow from the flight velocity to a lower velocity occurs between stations 1 and 2. At station 2 the air flow and the rocket gases enter the constant-area mixing section and mixing is assumed to proceed until, at station 3, the two flows are completely mixed and no transverse velocity gradients exist. Combustion of the excess fuel from the rocket occurs between stations 3 and 4 in an extension of the constant-area mixing section and then expansion of the jet occurs in the nozzle from stations 4 to 5.

Basic considerations of flow through nozzles indicate that for a given mass flow and available pressure ratio the greatest thrust occurs with a nozzle designed to give expansion of the jet to ambient pressure. If the pressure ratio is greater than the critical value, however, the greatest thrust per unit of nozzle-exit area is attained with a nozzle that provides for expansion only to sonic velocity, but the thrust is lower than if complete expansion is used. Calculations were therefore made for two types of configuration: The first type employs a nozzle designed to give expansion to ambient pressure; whereas, the second type has a nozzle providing expansion only to sonic velocity, or slightly higher. The first calculations indicate the maximum thrusts obtainable from the arrangement under consideration and the second set of calculations evaluates the maximum thrusts per unit frontal area that might be attained with such a propulsion system.

The total pressure of the air stream flowing through the ducted rocket is changed by two factors: The total pressure is increased by the momentum interchange between the air and the rocket gases, but the momentum changes accompanying the addition of heat in the combustion chamber result in a loss in total pressure. Because the net result for the cases considered was generally an increase in total pressure, a converging-diverging nozzle was generally required where full expansion was desired.

The pressure losses due to combustion vary inversely as the cross-sectional area of the combustion chamber. Inasmuch as a large nozzle-exit area was required for expansion to ambient pressure, an equal combustion-chamber area was therefore used, which maintained the momentum-pressure losses due to burning as low as possible without increasing the frontal area. The second set of calculations evaluates the maximum thrusts per unit frontal area that can be obtained with an exhaust nozzle providing incomplete expansion. For flight Mach numbers of 1 and 2, the combustion-chamber area and the nozzle-exit area were chosen equal and of such a size that thermal choking occurred at the combustion-chamber exit, which thereby permitted no expansion in the nozzle. For a flight Mach number of 3, the combustion-chamber and nozzle-outlet areas were chosen equal to the diffuser-inlet area because the area corresponding to thermal choking at the combustion-chamber exit was smaller than this value. Some expansion through a convergent-divergent nozzle was therefore possible in this case.

In evaluating the effect of altitude on the operation of a ducted rocket, the configuration employed was designed for stoichiometric operation at a flight Mach number of 1 and an altitude of 30,000 feet. The inlet geometry was considered fixed for all altitudes, but the exhaust nozzle was considered variable with a maximum area of the exit equal to the combustion-chamber area.

SYMBOLS

The following symbols are used in this analysis:

A	area, square feet
a	velocity of sound in gas, feet per second
c_p	specific heat of gas at constant pressure, Btu per pound $^{\circ}R$
F	thrust, pounds
g	ratio of absolute to gravitational units of mass, 32.17
I	specific impulse, pounds thrust per second per pound fuel
M	Mach number
m	mass flow, slugs per second
P	total pressure, pounds per square foot absolute
p	static pressure, pounds per square foot absolute
Q	heat content of fuel burned in duct, Btu
R	gas constant, foot-pounds per pound $^{\circ}R$
r	ratio of air mass flow to rocket-gas mass flow
T	total temperature, $^{\circ}R$
t	static temperature, $^{\circ}R$
V	velocity of gas flow, feet per second
γ	ratio of specific heats
ρ	density of gas at static temperature and pressure, slugs per cubic foot

Subscripts:

a	air
j	jet

n net

r rocket

0 free stream

1 duct inlet

2 entrance to mixing region (rocket exit)

3 after mixing and before combustion in duct

4 after completion of combustion in duct

5 nozzle exit

3-4 average between stations 3 and 4

BASIC ASSUMPTIONS

Rocket characteristics. - Calculations were based on a rocket having a mass flow of 1 slug per second using gasoline as the fuel and liquid oxygen as the oxidant. Because of structural considerations, a maximum temperature of 4000° R was arbitrarily assumed as a limit in the rocket. A rocket combustion-chamber pressure of 20 atmospheres was used. According to data obtained at the Guggenheim Aeronautical Laboratory, California Institute of Technology, these values of the combustion pressure and the maximum temperature in the rocket required a fuel-oxidant ratio of 0.606 with a gasoline having a hydrogen-carbon ratio of 0.176 and a lower heating value of 18,500 Btu per pound. All the oxidant in the rocket charge was assumed to be consumed completely within the rocket. Consideration of available experimental data on the thrust of rockets together with preliminary calculations based on values of c_p and γ obtained at California Institute of Technology, indicated that 171 pounds thrust per second per pound fuel would be a reasonable value for the specific impulse of the rocket under consideration at sea-level pressure. This value of specific impulse was therefore assumed for sea-level pressures and used in the calculations for this altitude. At any higher altitude, expansion to ambient pressure could be effected by a change in the rocket-nozzle dimensions, thereby giving a higher jet velocity and the maximum specific impulse for the particular altitude. Calculations indicated that a specific impulse of about 191 pounds thrust per second per pound fuel could be obtained with the rocket under consideration at an altitude of 30,000 feet if the rocket-nozzle dimensions were such that complete expansion to ambient pressure might be accomplished in the rocket exhaust nozzle. This value was therefore used for the comparison of the performance of the ducted rocket with that of the rocket alone at an altitude of 30,000 feet.

Characteristics of flow within duct. - Unless stated otherwise, the excess fuel used to maintain the rocket combustion temperature at 4000° R was assumed to burn completely within the duct. This condition limited the consideration to air flows large enough to give mixtures in the duct that were leaner than stoichiometric. Because dissociation and equilibrium effects would probably prevent attainment of temperatures above 4000° R in the duct, this value was used in all cases where calculations based on the heating value of the fuel and the specific heat of the gas mixture indicated higher combustion temperatures.

The values of c_p and γ used in the analysis were weighted averages of the values for the different constituents of the gas at the temperature under consideration. The values of c_p and γ for a process covering a range of temperatures were taken as the arithmetic average of the values corresponding to the end points of the temperature range. In the present case, only a slight discrepancy exists between data calculated by the use of these average values and data based on the integrated specific heat, inasmuch as the variation of c_p and γ with temperature is approximately linear over the ranges of temperature considered.

In order to facilitate the calculations, the mixing and combustion of the rocket gases and the air charge were considered to occur separately. These processes were assumed to occur with conservation of total momentum in a duct of constant cross section.

The temperature of the gas after combustion in the duct was determined from the heating value of the fuel and the heat capacities of the reactants and of the products after complete combustion.

Following the completion of combustion, any expansion was assumed to occur in conformity with the one-dimensional adiabatic-flow relations.

METHODS OF CALCULATION

General method. - In determining the performance of the ducted rocket, two of the variables were considered as independent and values were assigned for these in order to compute the corresponding values of the other variables. Values of the ratio of air mass flow

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through the duct to rocket-gas mass flow r and air Mach number at the rocket exit $M_{a,2}$ were chosen. Inasmuch as the diffusion from the flight velocity was assumed to occur with complete recovery of pressure, fixing the values of these two variables made possible the determination of the pressures, the temperatures, and the stream Mach numbers at stations 2 and 3 and of the cross-sectional area at stations 2, 3, and 4. The amount of unburned fuel discharged from the rocket was considered to be constant for this analysis and all the fuel was assumed to burn in the duct. As a result, the heat released to the air flow was constant and the pressures, the temperatures, the cross-sectional area, and the gas velocities at all stations following combustion could be calculated.

The following general equations were used to determine the state of the gas flow at any particular station by substituting the appropriate values for the variables:

$$\frac{T}{t} = \left(\frac{P}{p}\right)^{\frac{\gamma - 1}{\gamma}} = 1 + \frac{\gamma - 1}{2} M^2 \tag{1}$$

$$V = Ma$$
$$= M\sqrt{\gamma gRT} \tag{2}$$

$$A = \frac{m}{\rho V}$$
$$= \frac{mgRt}{pV} \tag{3}$$

Condition of air and rocket gases before mixing. - After values were chosen for r and $M_{a,2}$, the values of p, V, and A for the air flow at station 2 were obtained from equations (1), (2), and (3), respectively. The values of P and T used in the equations were those corresponding to the desired flight altitude and Mach number. The value 53.4 foot-pounds per pound $^{\circ}R$ was used for R when the flow of air was considered. Corresponding values for the rocket exit were obtained from the same equations using the rocket combustion-chamber total pressure and total temperature, the average value of γ over the expansion in equation (1), and the value of γ corresponding to the rocket-exit temperature in equation (2). Because the value of the specific impulse I of the rocket was assumed, an effective value of R for the rocket gases in agreement with the assumed rocket characteristics was obtained from the relation

$$F_r = m_r V_{r,2} = m_r gI$$

or

$$V_{r,2} = gI$$

$$M_{r,2}\sqrt{\gamma_{r,2}\, g\, R_{r,2}\, t_{r,2}} = gI$$

$$R_{r,2} = \frac{I^2 g}{(M_{r,2})^2\, \gamma_{r,2}\, t_{r,2}} \tag{4}$$

For the mixture of rocket gases and air in the duct, the value 53.4 foot-pounds per pound °R was used for R.

Mixing and combustion. - The total momentum of the gas stream after mixing (station 3) was obtained from the sum of the total momentums of the two individual streams at station 2, or

$$m_3V_3 + p_3A_3 = (m_aV_{a,2} + p_{a,2}\, A_{a,2}) + (m_rV_{r,2} + p_{r,2}\, A_{r,2}) \tag{5}$$

Inasmuch as the mixing and combustion processes were considered to occur separately, the total temperature after mixing was obtained by equating the total enthalpies of the two streams before mixing and the enthalpy of the stream after mixing and solving for T_3 from the relation

$$T_3 = \frac{m_r\, c_{p,r,2}\, T_{r,2} + m_a\, c_{p,a,2}\, T_{a,2}}{(m_r + m_a)\, c_{p,3}} \tag{6}$$

The total temperature after combustion (station 4) was obtained from

$$T_4 = T_3 + \frac{Q}{m_3 g c_{p,3-4}} \tag{7}$$

except that 4000° R was used for T_4 in those cases where equation (7) indicated values higher than this.

The total and static pressures at station 3 were obtained from the total momentum by a method employing dimensionless parameters based on the total temperature. These parameters are given in convenient chart form in reference 4. By substituting these pressures in equation (1), the Mach number before combustion M_3 was determined. The Mach number after combustion M_4 was obtained from a chart based on the equation

$$\frac{T_4}{T_3} = \frac{\dfrac{M_4^2\left(1 + \dfrac{\gamma - 1}{2} M_4^2\right)}{\left(1 + \gamma M_4^2\right)^2}}{\dfrac{M_3^2\left(1 + \dfrac{\gamma - 1}{2} M_3^2\right)}{\left(1 + \gamma M_3^2\right)^2}} \tag{8}$$

The value of M_4 was substituted into the equation

$$\frac{P_4}{P_3} = \frac{\dfrac{\left(1 + \dfrac{\gamma - 1}{2} M_4^2\right)^{\frac{\gamma}{\gamma - 1}}}{\left(1 + \gamma M_4^2\right)}}{\dfrac{\left(1 + \dfrac{\gamma - 1}{2} M_3^2\right)^{\frac{\gamma}{\gamma - 1}}}{\left(1 + \gamma M_3^2\right)}} \tag{9}$$

thus giving the total pressure following combustion P_4. Equations (8) and (9) were developed on the assumption of constant total momentum during the addition of heat to a gas flowing in a duct of constant area.

Expansion and thrust calculations. - Expansion following the completion of combustion was assumed to be in accordance with the one-dimensional adiabatic-flow relations; therefore, the pressure, the temperature, and the velocity of the gas at the nozzle exit and the nozzle-exit area were obtained by applying equations (1), (2), and (3). The jet thrust of the unit was then determined from

$$F_j = (m_a + m_r)\ V_j \tag{10}$$

and the net thrust was obtained by substracting from the jet thrust the initial momentum of the air entering the duct $m_a\ V_0$.

In most cases the duct was so designed that the cross-sectional areas of the mixing and combustion chamber and of the nozzle exit were equal. In order to accomplish this result, it was necessary to follow a trial-and-error procedure inasmuch as the relative areas of the mixing and the combustion chamber and of the nozzle exit could be obtained only by following through the calculations. Different values of $M_{a,2}$ were chosen (and combustion-chamber areas thereby determined) until the desired relative values of the areas were obtained.

DISCUSSION

Rocket thrust. - In comparing the net thrusts available from the rocket alone and from the ducted rocket, the rocket was assumed to have a nozzle of such design that full expansion to the ambient pressure would occur. With this provision, the thrust of the rocket assumed for the analysis would have the values 5500 pounds and 6150 pounds at altitudes of sea level and 30,000 feet, respectively.

Thrust of ducted rocket. - The net thrust available from the ducted rocket under consideration is shown in figure 2 for an altitude of 30,000 feet. These data cover a range of the mass-flow ratio r from 3 (approximately stoichiometric) to 10 and flight Mach numbers of 1, 2, and 3. In the calculation of these data, it was assumed that the exhaust nozzle was designed to give expansion to ambient pressure; therefore, the values given may be considered as the maximum values obtainable from the ducted rocket on which this analysis was based.

The net thrust of the ducted rocket at a flight Mach number of 1 is just under 10,000 pounds for a mass-flow ratio r of 3 and increases to almost 14,000 pounds as r increases to 10. The corresponding range for a flight Mach number of 2 is from 13,500 to 22,000 pounds, and for a flight Mach number of 3 is from 14,500 to 23,500 pounds. The rapid rate of decrease of the thrust as r decreases below a value of about 5 is due partly to the use of a maximum-temperature limitation in the duct.

The ratio of these net-thrust values to the thrust of the rocket alone is shown in figure 3. At a flight Mach number of 1, ducting the rocket results in net thrusts from 1.6 to 2.25 times the thrust of the rocket alone over the range of mass-flow ratios considered. Comparable net thrusts of the ducted rocket for flight Mach numbers of 2 and 3 are from 2.2 to 3.55 and from 2.4 to 3.8 times the rocket thrust, respectively.

These thrust values were obtained assuming complete combustion in the duct of all excess fuel from the rocket exhaust.

Fuel consumption. - The specific fuel consumption of the ducted rocket is given in figure 4. The consumption varies from 12 to 8.5 pounds per hour per pound of thrust for values of r varying from 3 to 10 at a flight Mach number of 1. At a flight Mach number of 2, the fuel consumption varies from 8.5 to 5.5 over the range of r considered; whereas over the same range of the mass-flow ratio the value for a Mach number of 3 is approximately 0.5 pound per hour per pound of thrust below the corresponding value at a Mach number of 2. The specific fuel consumption for a ram jet operating at a stoichiometric fuel-air ratio at an altitude of 30,000 feet is 4.7 pounds per hour per pound thrust at a Mach number of 1, 2.8 at a Mach number of 2, and 3.6 at a Mach number of 3. The first two of these values are given in reference 5 and the third was obtained by the method used in reference 5. If a ram jet were operated at a fuel-air ratio leaner than stoichiometric, a lower specific fuel consumption would result and consequently, there would be a greater difference in the fuel consumptions of these units than is given here. The thrust of the ram jet at lean operation, however, would also decrease.

Frontal areas. - In order to realize the thrust increases resulting from ducting a rocket, a unit with a much greater frontal area than the rocket alone is required. Because this increase in area gives an increase in the external drag of the body, the external resistances would have to be taken into account in order to determine whether ducting a rocket would give a resultant net gain. Figure 5 shows the frontal areas associated with the configurations required to realize the thrust characteristics previously shown. The exit-nozzle area for the fully expanding rocket at an altitude of 30,000 feet is 0.7 square foot. At a flight Mach number of 1, the ducted rocket has a frontal area of 11.5 square feet at a mass-flow ratio of 3 and an area of 29 square feet when r equals 10. The frontal areas required at higher flight velocities are smaller because of the higher total density associated with these velocities. These areas vary from 5.5 to 12 square feet for a flight Mach number of 2 and from 3.5 to 7 square feet for a Mach number of 3 for the range of r considered.

Specific thrust. - The data of figures 2 and 5 can be presented in a more significant manner as the ratio of the net thrust to the frontal area required to produce this thrust. Values of this specific thrust are given in figure 6 for the data already presented. The specific thrust at a Mach number of 1 decreases from a value of 850 to 500 pounds per square foot of frontal area as r increases from 3 to 10. For Mach numbers of 2 and 3, corresponding specific thrusts are from 2450 to 1800 and from 4350 to 3300 pounds per square foot of frontal area, respectively. Specific thrusts for a ram jet operating at a stoichiometric fuel-air ratio under similar flight conditions are 407, 1860, and 4390 pounds per square foot of frontal area at flight Mach numbers of 1, 2, and 3, respectively. The values for the ram jet at Mach numbers of 1 and 2 are given in reference 5 (based on nacelle rather than combustion-chamber diameter) and the value for a Mach number of 3 was computed by the method of reference 5. The specific thrusts of the ducted rocket at flight Mach numbers of 1 and 2 are generally greater than those of a stoichiometric ram jet operating at the same Mach numbers. At a flight Mach number of 3, the specific thrust of the ram jet is larger. This fact results from the occurrence of the low combustion-chamber velocities in the ducted rocket at high flight velocities when the combustion-chamber area is equal to the nozzle-exit area required for complete expansion.

Configurations for best specific thrusts. - For flight conditions under which a ducted-rocket propulsion system might be used, the specific thrust would probably be a more important criterion than the net thrust. As previously explained, maximum thrusts for a given frontal area and air flow can be obtained by decreasing the combustion-chamber cross-sectional area until thermal choking occurs, or until this area is equal to the duct-inlet area required for the desired air flow, and then using a nozzle having an exit area of the same size. Calculated data for the conditions giving these maximum specific thrusts are presented in figures 7 to 9. The net thrusts are somewhat lower than in the previous case, especially at flight Mach numbers of 2 and 3, as can be seen from a comparison of figures 2 and 7. The lower net thrusts are caused by the greater momentum-pressure losses associated with the smaller cross-sectional areas and by the loss in thrust due to incomplete expansion. Because of the lower thrusts, the specific fuel consumptions would be a little higher than for the cases discussed earlier. The frontal areas required are also smaller for the present configuration (figs. 5 and 8) with the result that the specific thrusts are greater than for the conditions previously considered (figs. 6 and 9). Large increases in specific thrust over the values previously presented are available at Mach numbers of 2 and 3; maximum specific thrusts ranging from 4300 to 2500 pounds per square foot of frontal area as compared with a range from 2450 to 1800 pounds in the case of full expansion of the jet for a Mach number of 2 and from 11,000 to 5800 pounds per square foot of frontal area as compared with a range of 4350 to 3300 pounds for the previous case for a Mach number of 3 are indicated for the range of r considered. These specific thrusts are of the order of twice those previously presented for a stoichiometric ram jet operating at comparable flight Mach numbers. Inasmuch as the specific thrusts for these configurations are much greater than when expansion to ambient pressure occurs in the nozzle, these configurations would be preferred for most applications.

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From figures 6 and 9 it appears that the most logical operating conditions for the ducted rocket considered would be such that the mass-flow ratio would be in the range of 3 to 5. Because this range would give operation at mixtures leaner than stoichiometric, the entire heat content of the rocket-fuel charge could be utilized in the cycle. If leaner operation were desired, greater thrusts could be obtained (figs. 2 and 7) but the specific thrust would be lower.

Performance of ducted rocket with varying altitude. - The performance characteristics of a ducted rocket with fixed inlet geometry when operating at a Mach number of 1 at altitudes from sea level to 45,000 feet are shown in figure 10. Design conditions were stoichiometric operation with expansion to ambient pressure at an altitude of 30,000 feet and a flight Mach number of 1. In the calculations it was assumed that an adjustable exhaust nozzle would allow expansion to ambient pressure at altitudes below 30,000 feet and that at higher altitudes incomplete expansion would occur with the nozzle-exit area equal to the combustion-chamber cross-sectional area. At sea-level operating conditions, the net thrust is about 13,000 pounds; the thrust decreases with increasing altitude and at 45,000 feet a thrust of about 8000 pounds is obtained. The specific thrust and the ratio of net thrust to rocket thrust would vary similarly with altitude. The decrease in thrust with altitude is a consequence of the decreasing air-handling capacity of a unit of fixed-inlet geometry with increasing altitude. This effect can be seen in figure 10: at sea level, three times the design air flow is induced, whereas at an altitude of 45,000 feet the air flow drops to one-half the design value. The specific fuel consumption increases from about 9 pounds per hour per pound thrust at sea level through the design value of about 12 to a value of 14.5 pounds per hour per pound thrust at an altitude of 45,000 feet. At altitudes above 30,000 feet, the specific fuel consumption would be expected to increase rapidly because the unit is then operating at a mixture ratio richer than stoichiometric; therefore, part of the heat content of the excess fuel from the rocket cannot be utilized in the duct.

Static performance. - The performance of the ducted rocket under static conditions is shown in figure 11. The thrust of the rocket alone operating at sea level with expansion to ambient pressure is about 5500 pounds. At static conditions, the net-thrust increase due to ducting the rocket is slight; the net thrust of the rocket was increased about 9 percent by ducting over a range of r from 3 to 5.5. This range of air flows would require a frontal area of 8 to 20 square feet. This small thrust increase under these conditions should be expected, inasmuch as the unit is then effectively a rocket plus a ram jet operating at low ram. When the increase in weight necessary for ducting is considered, it is probable that the take-off performance of a rocket would be adversely affected by ducting it in the manner under consideration.

<u>Relative gains from jet-ejector effect and from burning excess rocket fuel.</u> - In order to determine the relative magnitudes of the thrust increases attributable to the jet-thrust augmentation effect and to the burning of the excess fuel from the rocket in a ram-jet cycle in the duct, calculations were made assuming no combustion of the rocket-exhaust products within the duct. The results of these calculations are given in figure 12. The net thrusts for the flight conditions at an altitude of 30,000 feet lie within the range of 6800 to 10,000 pounds as compared with the range of about 10,000 to 23,500 pounds (fig. 2) with combustion of the excess fuel from the rocket. The jet-thrust augmentation thus contributes only about one-fourth of the thrust increase in the cases presented. From these results, it is apparent that the chief source of the increase in thrust resulting from ducting a rocket is the burning of the excess fuel from the rocket-exhaust gases rather than from the interchange of momentum between the jet from the rocket and the air flowing through the duct.

<u>Effect of lowered component efficiencies on performance of ducted rocket.</u> - In order to evaluate the effects of pressure losses in the diffuser and exhaust nozzle and the effect of a lower combustion efficiency in the duct on the performance of a ducted rocket, calculations were made to determine the magnitude of the losses that would result from lower component efficiencies. These data, for a flight Mach number of 2 and a configuration designed to give maximum net thrust, are presented in figure 13. A diffuser pressure ratio of 0.95 was selected on the basis of data presented in reference 6 and a pressure ratio across the nozzle of 0.96 was used to cover pressure losses in the nozzle. The total decreases in net thrust corresponding to those pressure losses is about 3 percent and the decrease in net thrust is about 6 to 7 percent. The diffuser pressure loss accounts for slightly more than one-half of the decrease in each of these factors.

The decrease in net thrust accompanying a 10-percent decrease in combustion efficiency is about 4 or 5 percent for mass-flow ratios of 5 to 10 percent (fig. 3). For lower air flows, smaller percentage decreases are indicated in the figure because the arbitrary assumption of a 4000° R limiting temperature in the earlier calculations was equivalent to employing a reduced combustion efficiency at the low air flows. With a decrease in combustion efficiency, a smaller nozzle-exit area can be used for the same mass flow; consequently, decreases in specific thrust due to lowered combustion efficiency are of smaller magnitude (about 1 or 2 percent) than are the changes in net thrust.

The losses in net thrust due to the use of expected values of component efficiencies rather than the assumed 100-percent efficiencies are not large enough to affect the general results presented.

Comparison of ducted rocket with combination of rocket and ram jet. - A comparison of the performance of the ducted rocket with that of a combination consisting of an independently operating rocket and a stoichiometric ram jet is shown in figure 14. These data represent operation at an altitude of 30,000 feet. The two propulsion systems should have comparable characteristics: Each would have the ability to take off under its own power and, because the thrust and fuel-consumption characteristics of the two systems should be of the same order of magnitude, they might be considered suitable for the same applications. Similar rockets were assumed for both applications and the frontal areas of the ducted rocket and of the combination were assumed equal. The ram-jet data were calculated by the method of reference 5; the ducted-rocket data are those presented herein for the unit designed to give maximum specific thrusts. For values of r from 3 to 6, the thrust of the ducted rocket is about 30 percent greater than that of the combination at a Mach number of 3 and about 10 percent greater at a Mach number of 2. For a Mach number of 1 and the same range of r, the thrust of the ducted rocket is about 10 percent lower than that of the combination.

In all cases the specific fuel consumption of the ducted rocket is considerably lower than that of the combination. This result would be expected because the increase in thrust of the ducted unit above the rocket thrust is obtained with no additional expenditure of fuel; whereas in the combination extra fuel is necessary for the ram jet in order to develop any thrust above that developed by the rocket alone. If the ram jet in the combination were operated at a fuel-air ratio lower than stoichiometric, the comparison would be more favorable for the ducted rocket because the thrust of the combination would decrease to that of the rocket alone as the fuel consumption decreased to that of the ducted rocket.

SUMMARY OF RESULTS

The following results were derived from a theoretical analysis of the performance of a rocket using a fuel mixture of gasoline and liquid oxygen when the rocket is enclosed in a duct through which air is flowing with all excess fuel from the rocket being burned in the duct:

1. At flight Mach numbers of 1, 2, and 3 and an altitude of 30,000 feet, the ducted rocket provided thrusts per unit frontal area of the order of twice those available from a ram jet; whereas the specific fuel consumptions were about two to three times those of a ram jet or about one-half to one-third that of the rocket alone.

2. The greater part of the thrust increase was due to the burning of the excess fuel from the rocket; only about one-fourth of the total thrust increase at flight Mach numbers of 1, 2, and 3 at an altitude of 30,000 feet was due to the jet-ejector effect experienced when the rocket is enclosed in a duct.

3. At an altitude of 30,000 feet and flight Mach numbers of 1, 2, and 3, the specific fuel consumption of a ducted rocket was lower than that of a combination having the same frontal area and consisting of a similar rocket and a ram jet operating at a stoichiometric fuel-air ratio. At flight Mach numbers of 2 and 3, the thrust of the ducted rocket was greater than that of the combination, but at a Mach number of 1 the thrust of the combination was the greater of the two.

4. At static conditions, the thrust of the ducted rocket was approximately 9 percent greater than the thrust of the rocket alone.

5. Consideration of the pressure losses in the diffuser and the exhaust nozzle would reduce the values given for the net thrust of the ducted rocket by about 3 percent and would decrease the specific-thrust values by about 6 or 7 percent at a flight Mach number of 2. At the same Mach number, if a combustion efficiency of 90 percent were used rather than the assumed value of 100 percent, the net thrusts would be decreased as much as 5 percent and the specific thrusts about 1 or 2 percent.

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REFERENCES

1. Jacobs, Eastman N., and Shoemaker, James M.: Tests on Thrust Augmentors for Jet Propulsion. NACA TN No. 431, 1932.

2. Rasof, Bernard: Theoretical Subsonic Performance of a Ducted Rocket. Rep. No. 3-3, P. P. Lab. Proj. No. MX527, Jet Propulsion Lab., GALCIT, Aug. 14, 1945.

3. Lorell, Jack: Additional Theoretical Results on the Performance of the Subsonic Ducted Rocket. Rep. No. 3-11, P. P. Lab. Proj. No. MX527, Jet Propulsion Lab., GALCIT, Oct. 21, 1946.

4. Turner, L. Richard, Addie, A. N., and Zimmerman, R. H.: Charts for the Analysis of One-Dimensional Compressible Flow. NACA TN No. 1419, 1947.

5. Krebs, Richard P., and Palasics, John: Analytical Comparison of a Standard Turbojet Engine, a Turbojet Engine with a Tail-Pipe Burner, and a Ram-Jet Engine. NACA RM No. E6L11, 1947.

6. Moeckel, W. E., and Connors, J. F.: Investigation of Shock Diffusers at Mach Number 1.85. III - Multiple-Shock and Curved-Contour Cones. NACA RM No. E7F13, 1947.

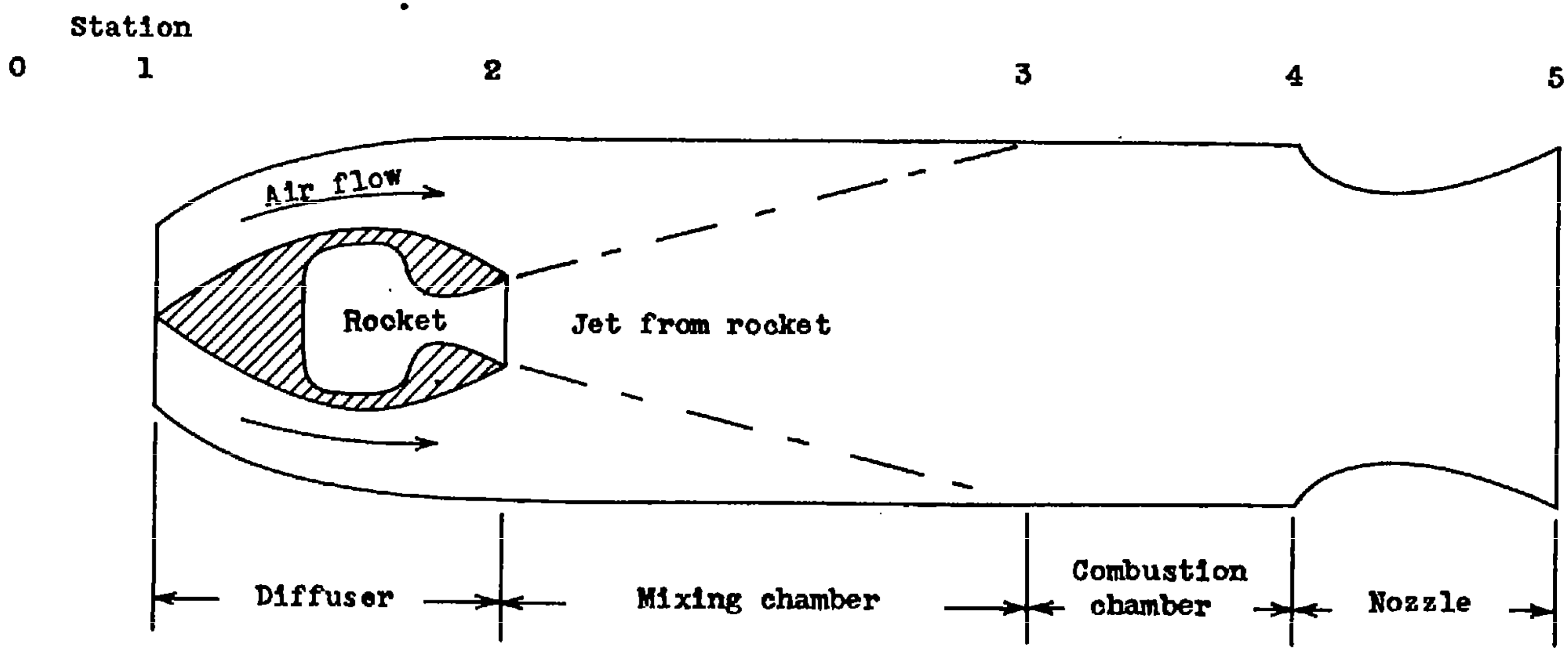

Figure 1. - Schematic diagram of ducted-rocket propulsion system.

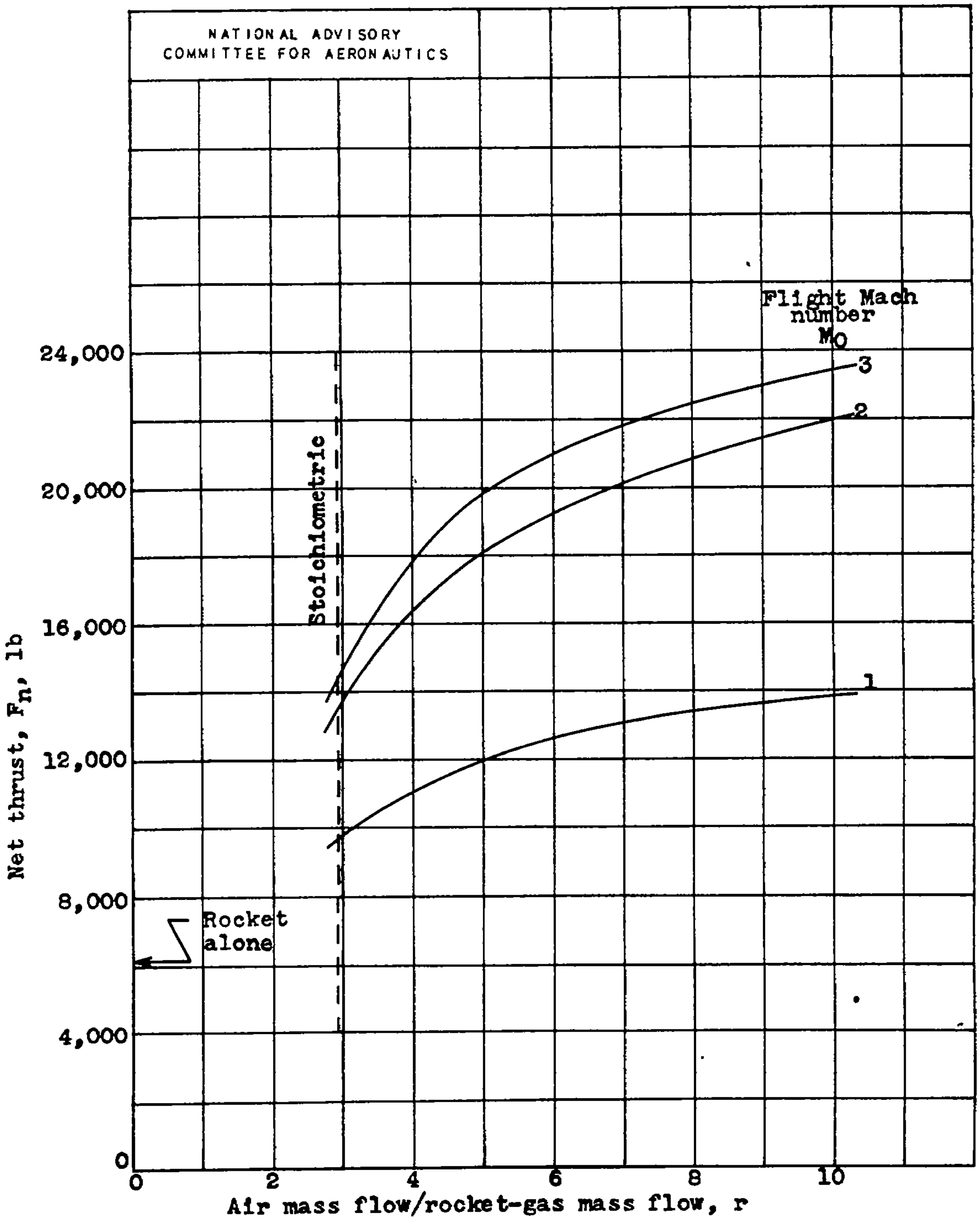

Figure 2. - Variation of net thrust of ducted rocket with flight Mach number and mass-flow ratio. Exhaust nozzle to give expansion of jet to ambient pressure. Rocket-gas mass flow, 1 slug per second; altitude, 30,000 feet.

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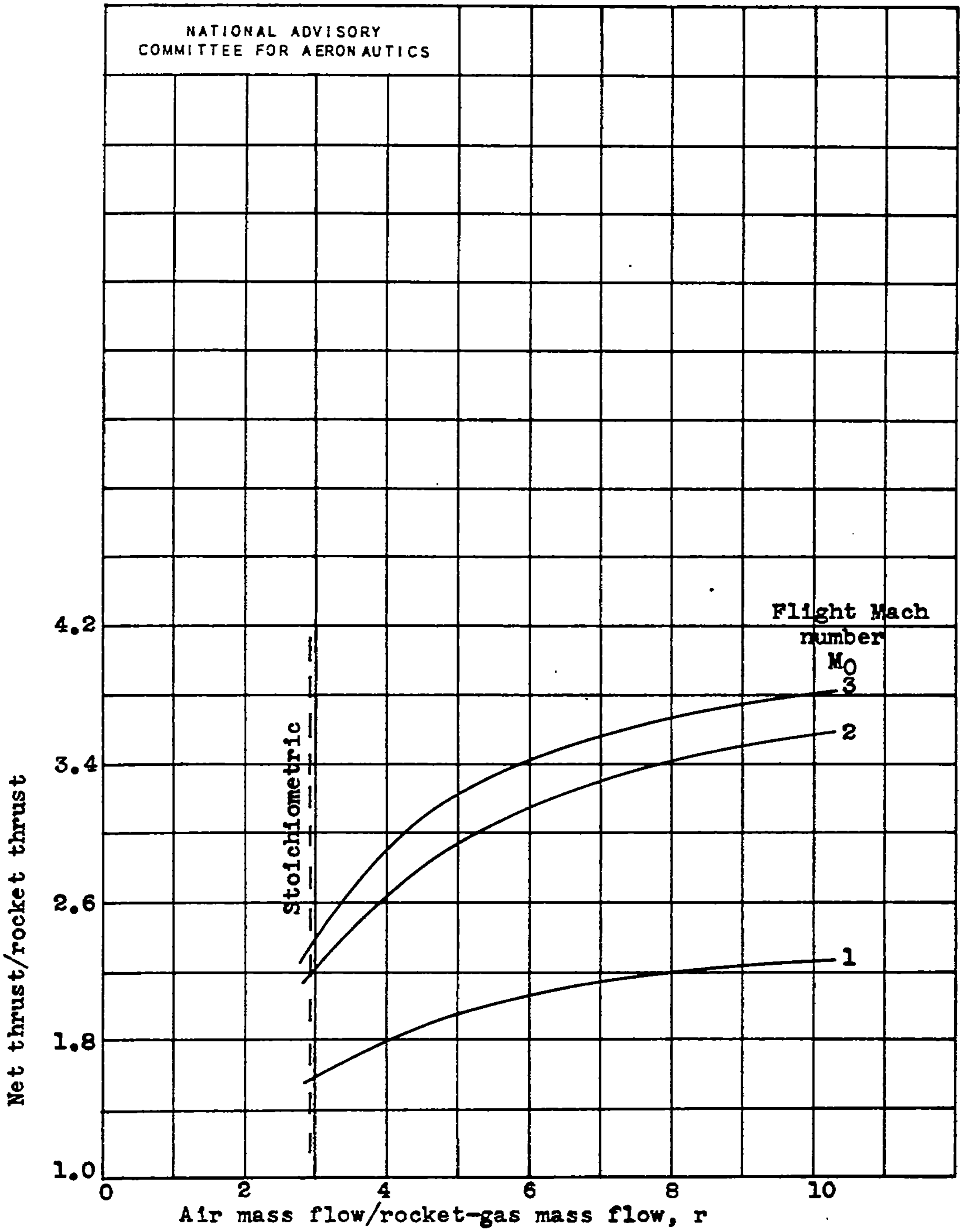

Figure 3. - Comparison of net thrust of ducted rocket with thrust of rocket alone. Exhaust nozzle to give expansion of jet to ambient pressure. Specific impulse of rocket, 191 pounds thrust per second per pound fuel; altitude, 30,000 feet.

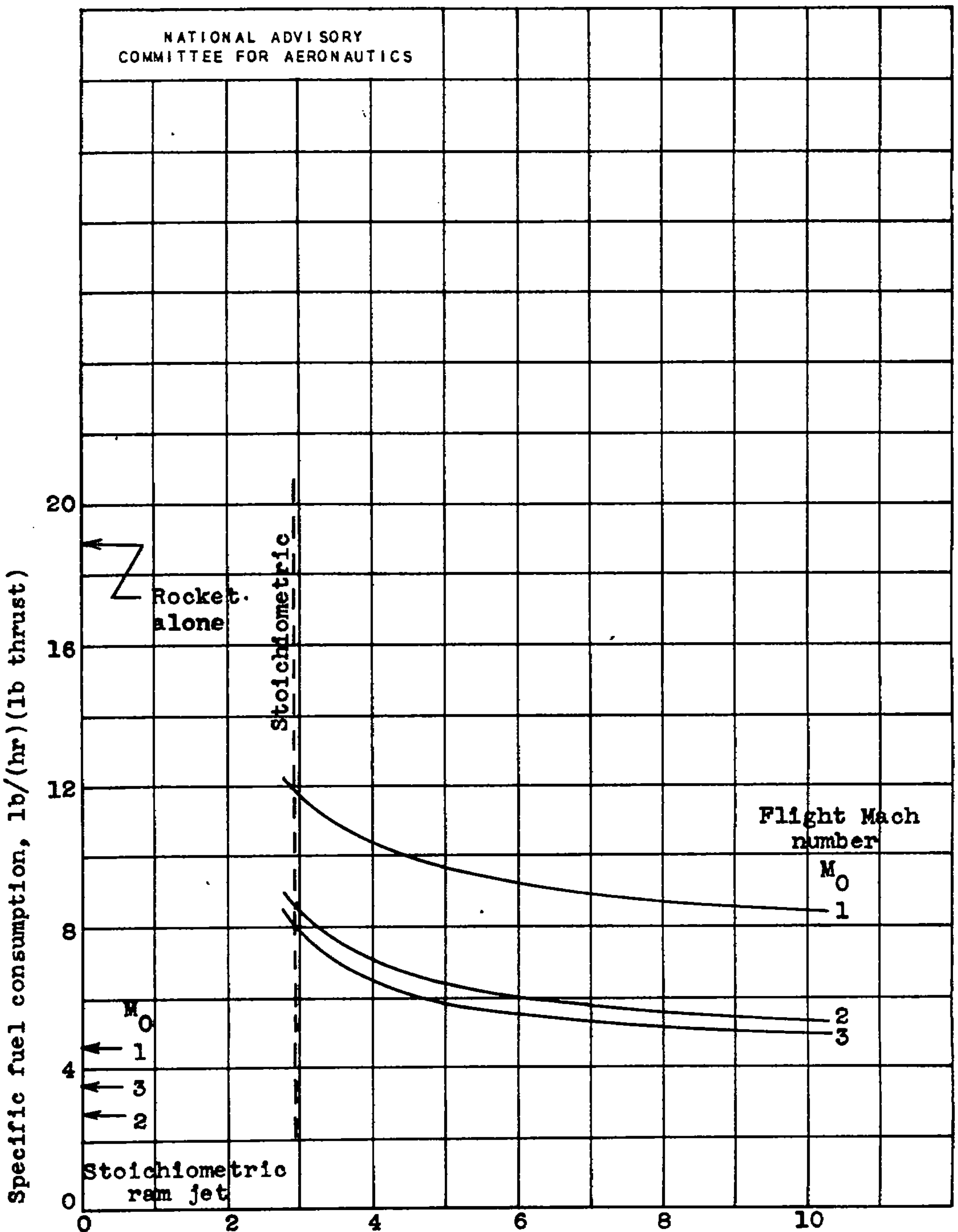

Figure 4. - Variation of specific fuel consumption of ducted rocket with flight Mach number and mass-flow ratio. Exhaust nozzle to give expansion of jet to ambient pressure. Altitude, 30,000 feet.

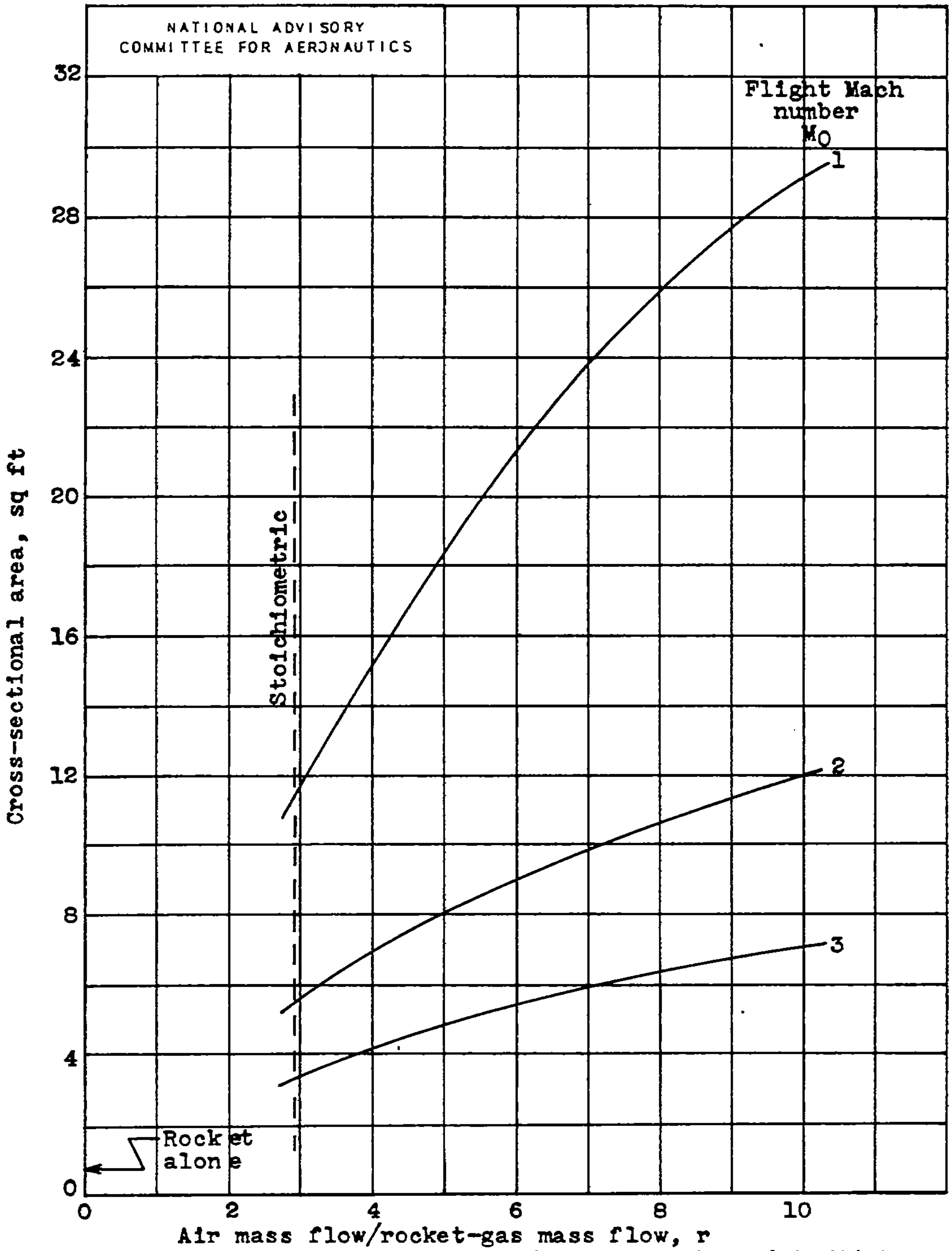

Figure 5. - Variation of frontal area of ducted rocket with flight Mach number and mass-flow ratio. Exhaust nozzle to give expansion of jet to ambient pressure. Rocket-gas mass flow, 1 slug per second; altitude, 30,000 feet.

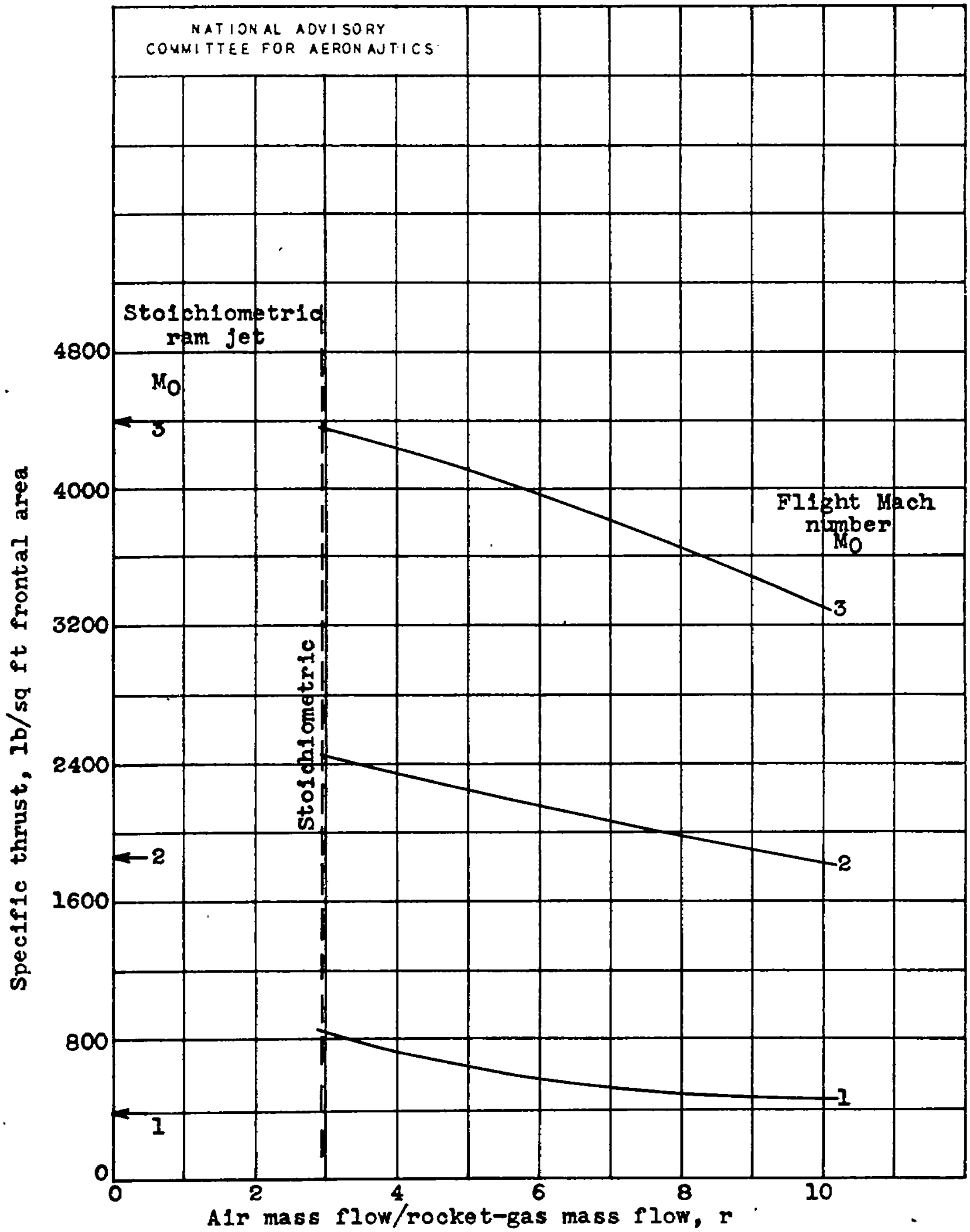

Figure 6. - Variation of specific thrust of ducted rocket with flight Mach number and mass-flow ratio. Exhaust nozzle to give expansion of jet to ambient pressure. Altitude, 30,000 feet.

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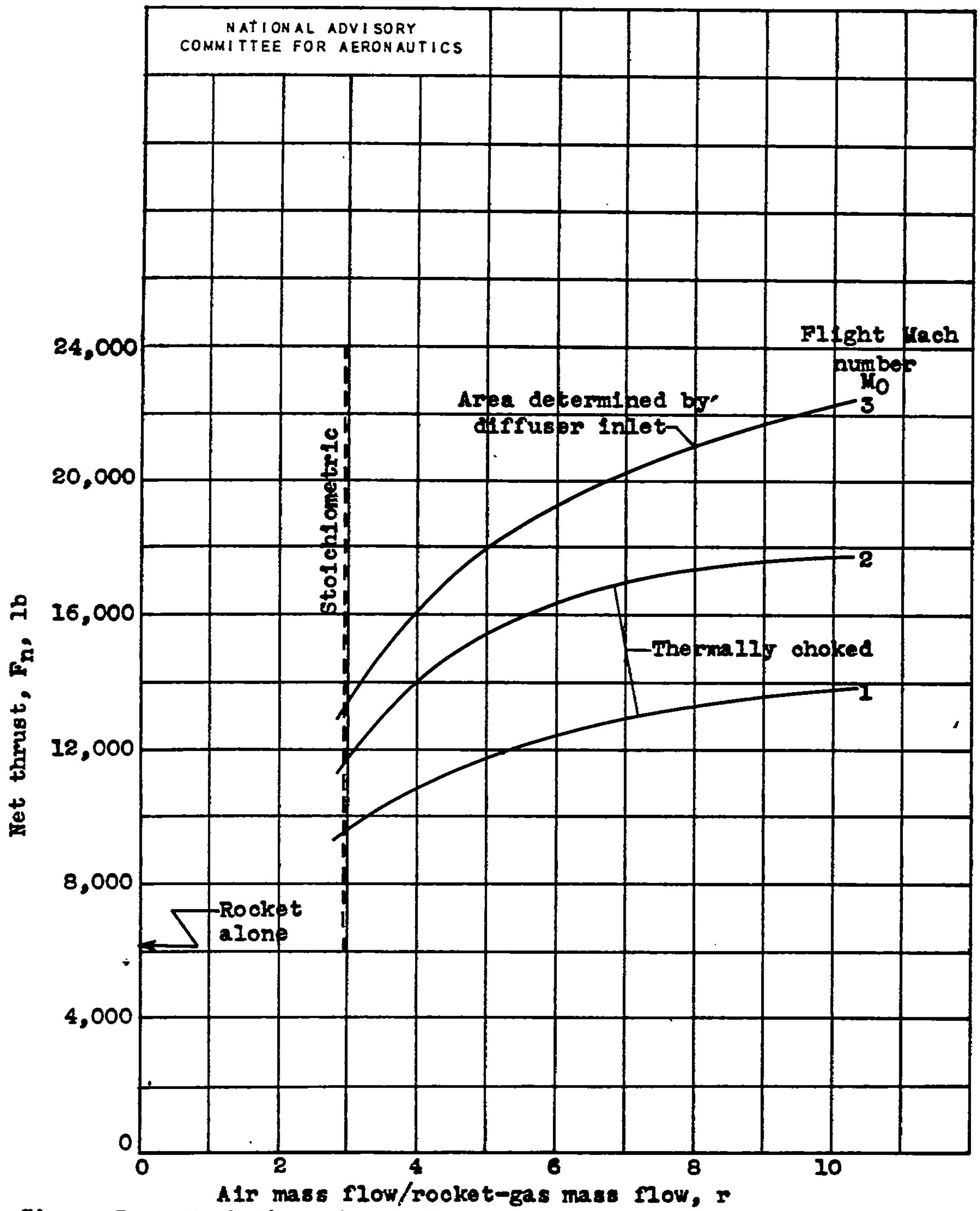

Figure 7. - Variation of net thrust of ducted rocket with flight Mach number and mass-flow ratio when combustion-chamber and nozzle dimensions are chosen to give maximum specific thrusts. Rocket-gas mass flow, 1 slug per second; altitude, 30,000 feet.

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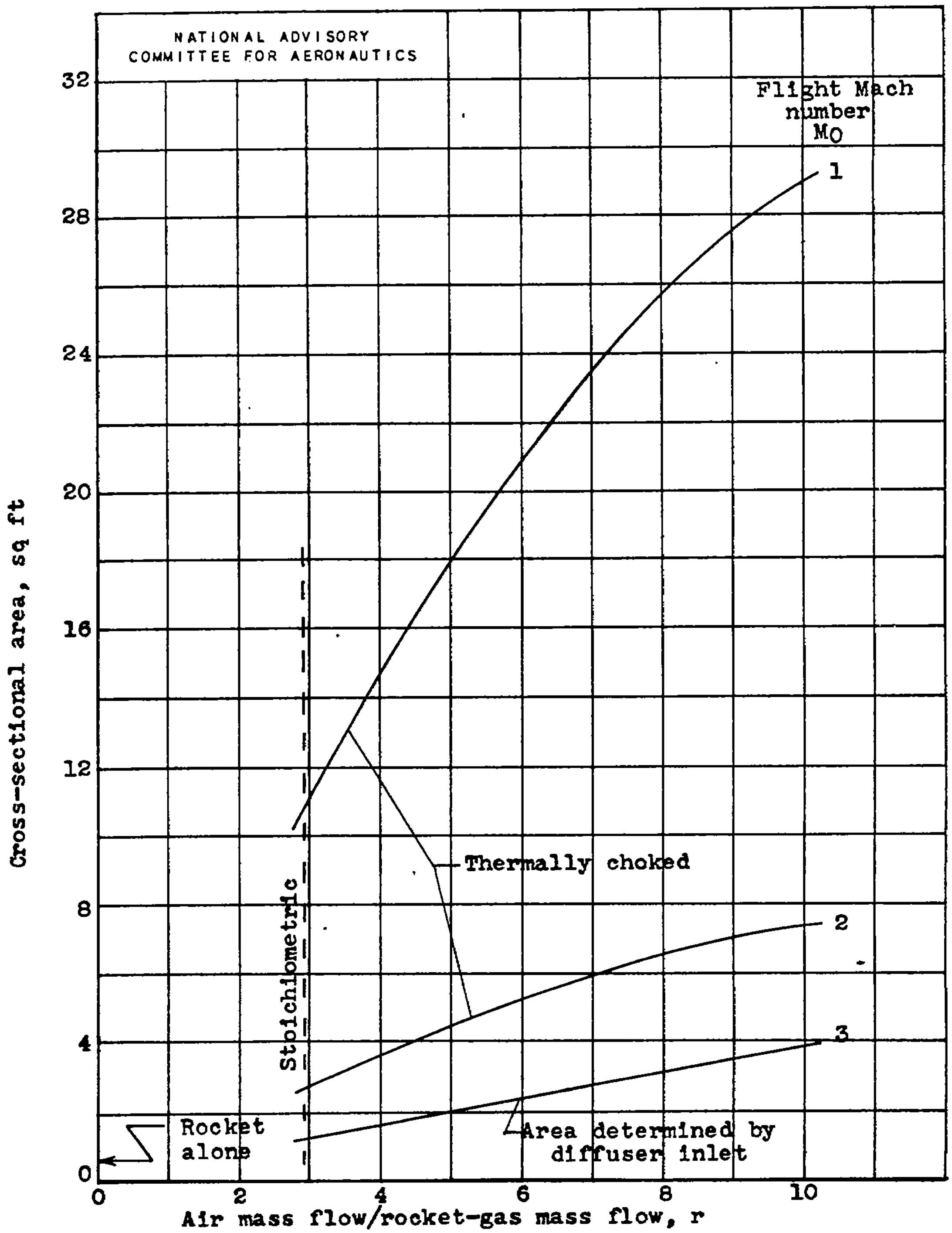

Figure 8. - Variation of cross-sectional area of ducted rocket with flight Mach number and mass-flow ratio when combustion-chamber and nozzle dimensions are chosen to give maximum specific thrusts. Rocket-gas mass flow, I slug per second; altitude, 30,000 feet.

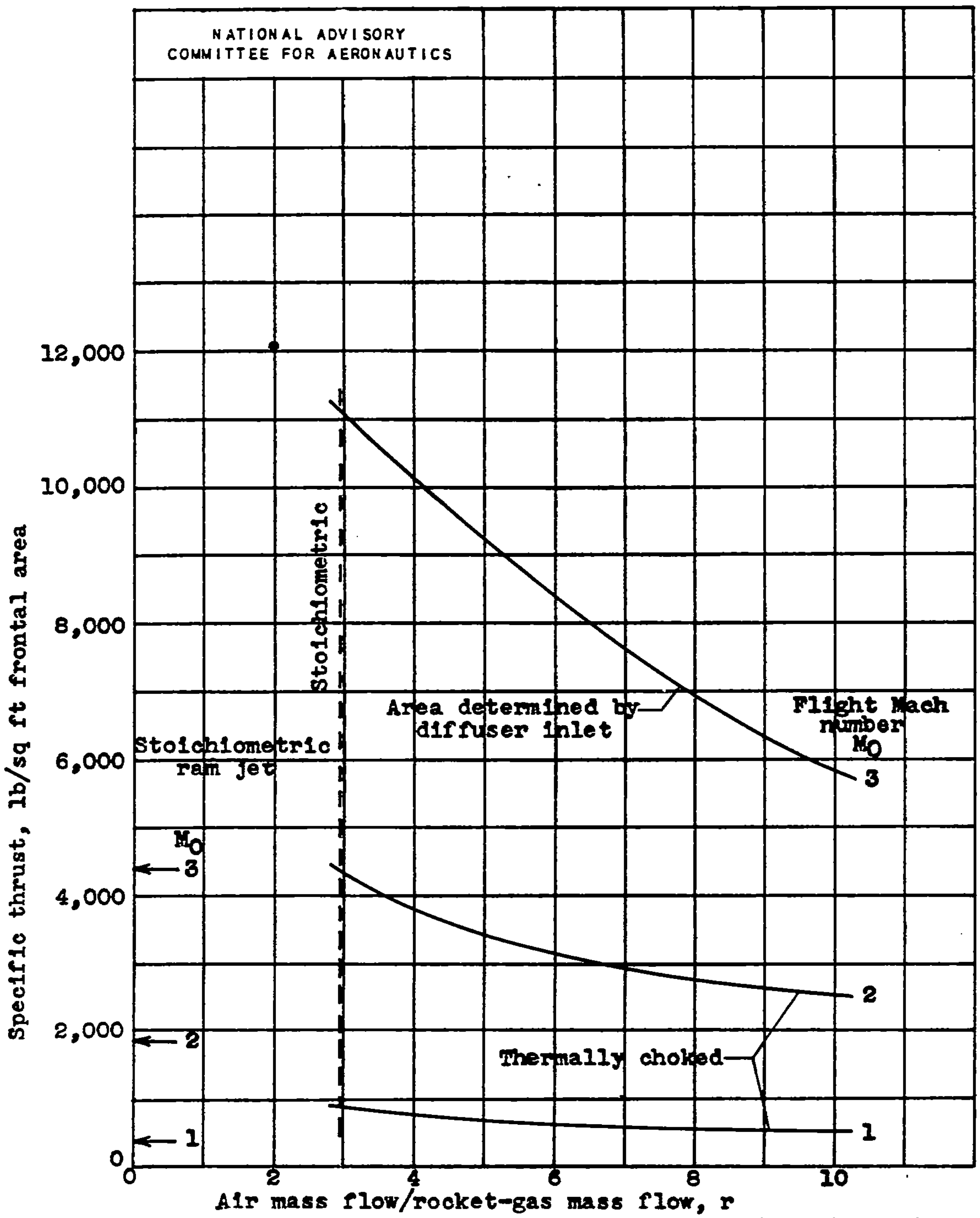

Figure 9. - Variation of specific thrust of ducted rocket with flight Mach number and mass-flow ratio when combustion-chamber and nozzle dimensions are chosen to give maximum specific thrusts. Altitude, 30,000 feet.

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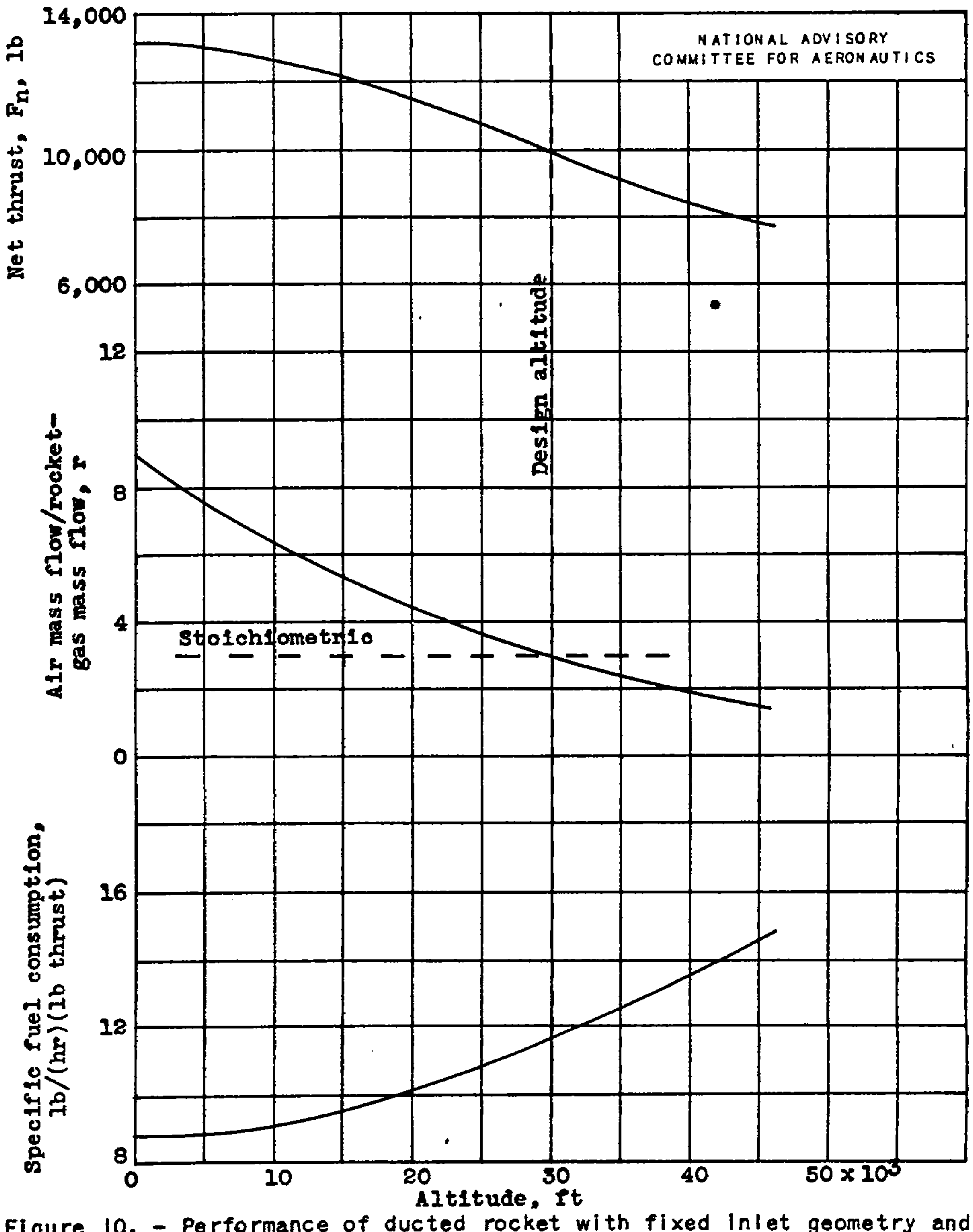

Figure 10. - Performance of ducted rocket with fixed inlet geometry and variable nozzle-exit area at altitudes from sea level to 45,000 feet at flight Mach number of 1. Rocket-gas mass flow, 1 slug per second.

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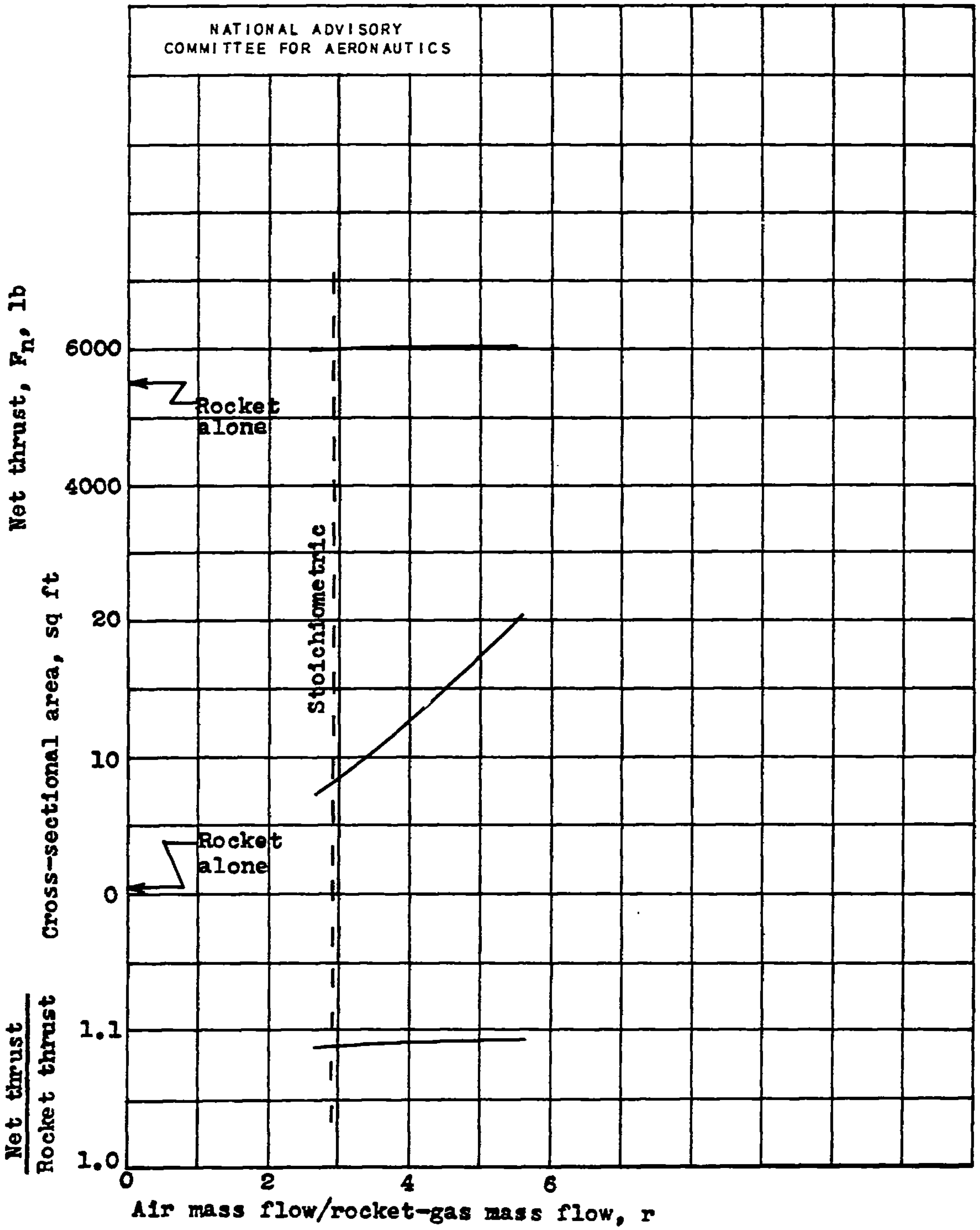

Figure 11. - Performance characteristics of ducted rocket under static conditions. Exhaust nozzle to give expansion of jet to ambient pressure. Rocket-gas mass flow, 1 slug per second; flight Mach number, 0; altitude, sea level.

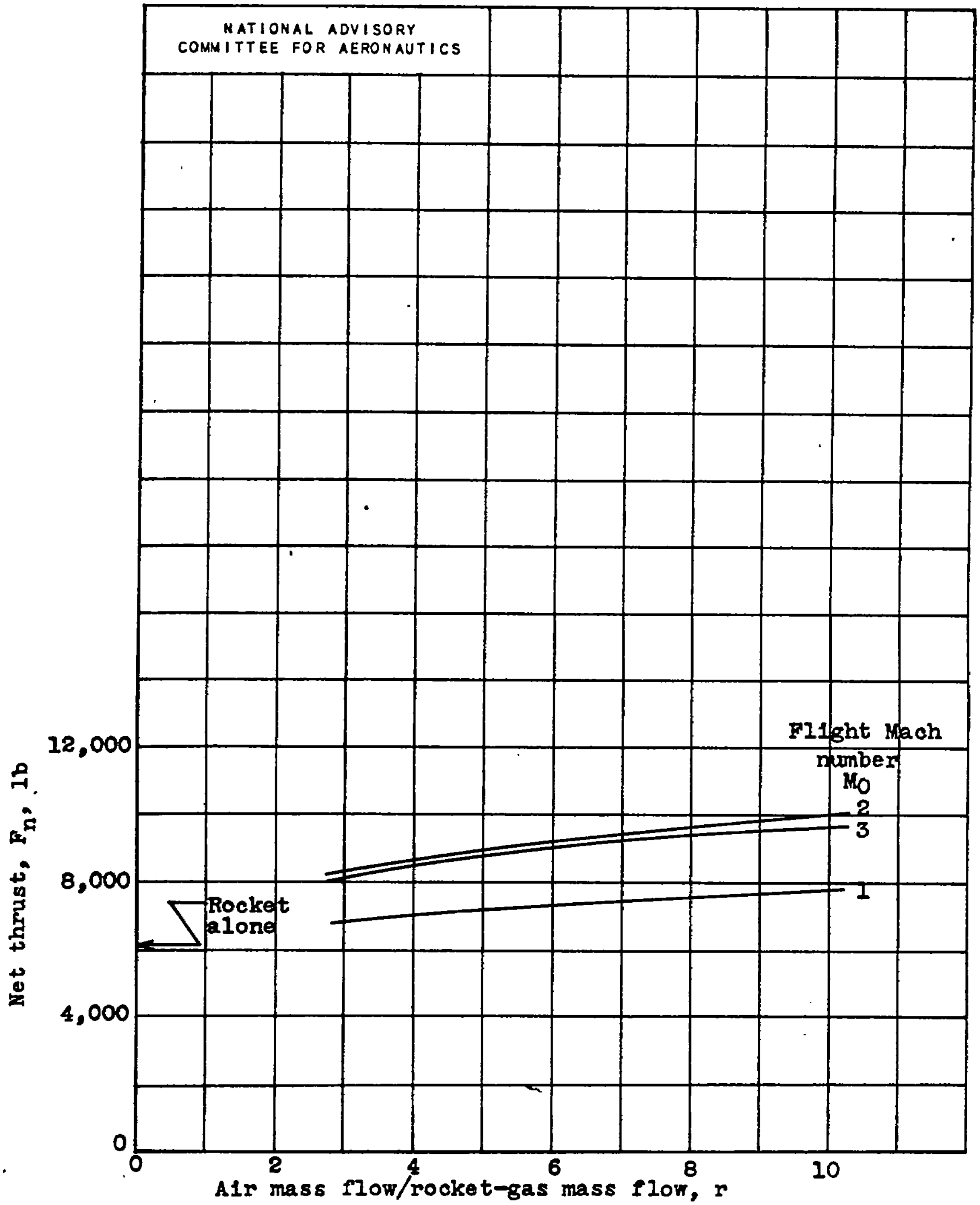

Figure 12. - Variation of net thrust of ducted rocket with flight Mach number and mass-flow ratio when no combustion occurs in duct. Exhaust nozzle to give expansion to ambient pressure. Rocket-gas mass flow, 1 slug per second; altitude, 30,000 feet.

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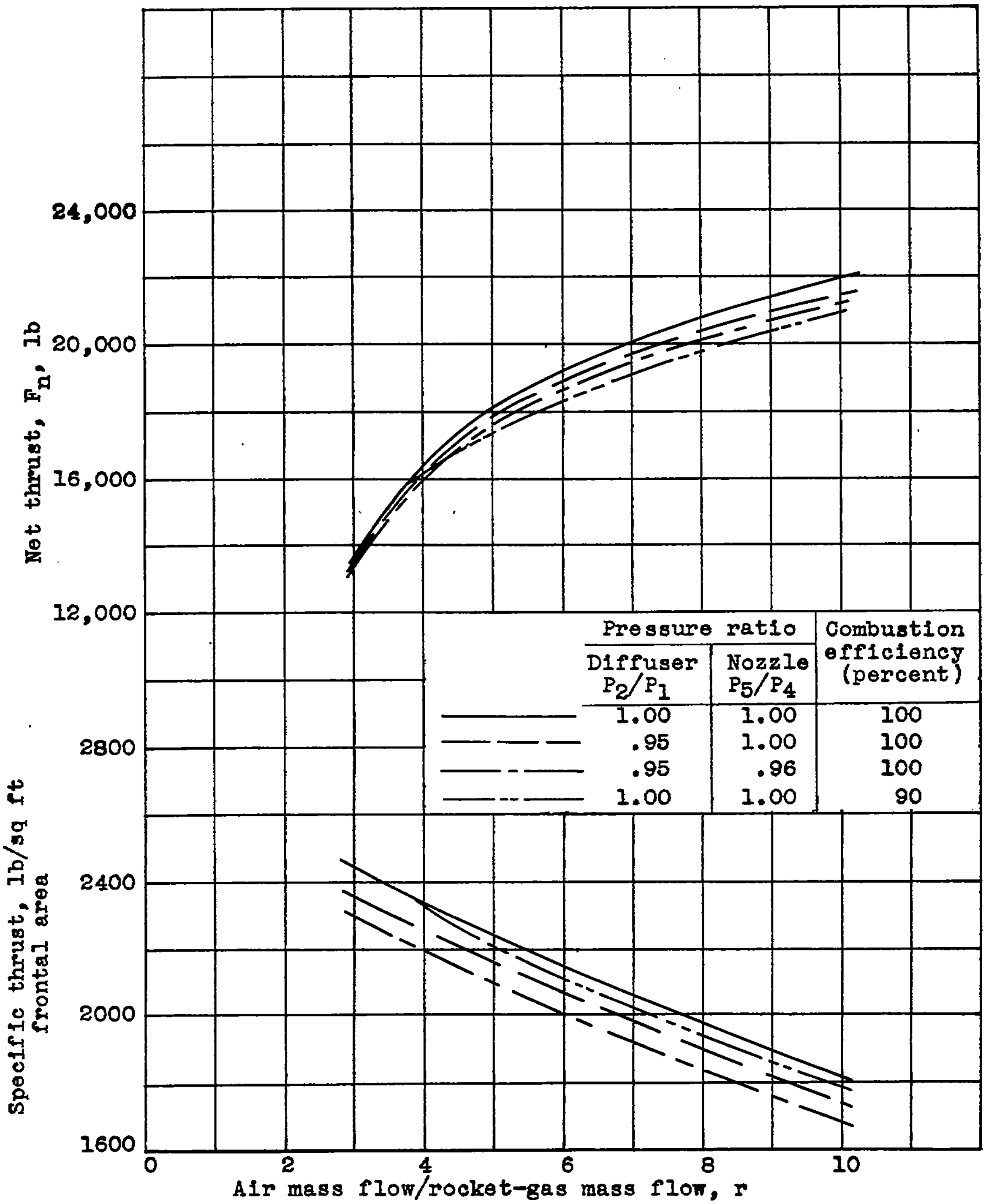

Figure 13. - Effect of lowered component efficiencies on net thrust and specific thrust of ducted rocket. Exhaust nozzle to give expansion of jet to ambient pressure. Rocket-gas mass flow, 1 slug per second; flight Mach number, 2; altitude, 30,000 feet.

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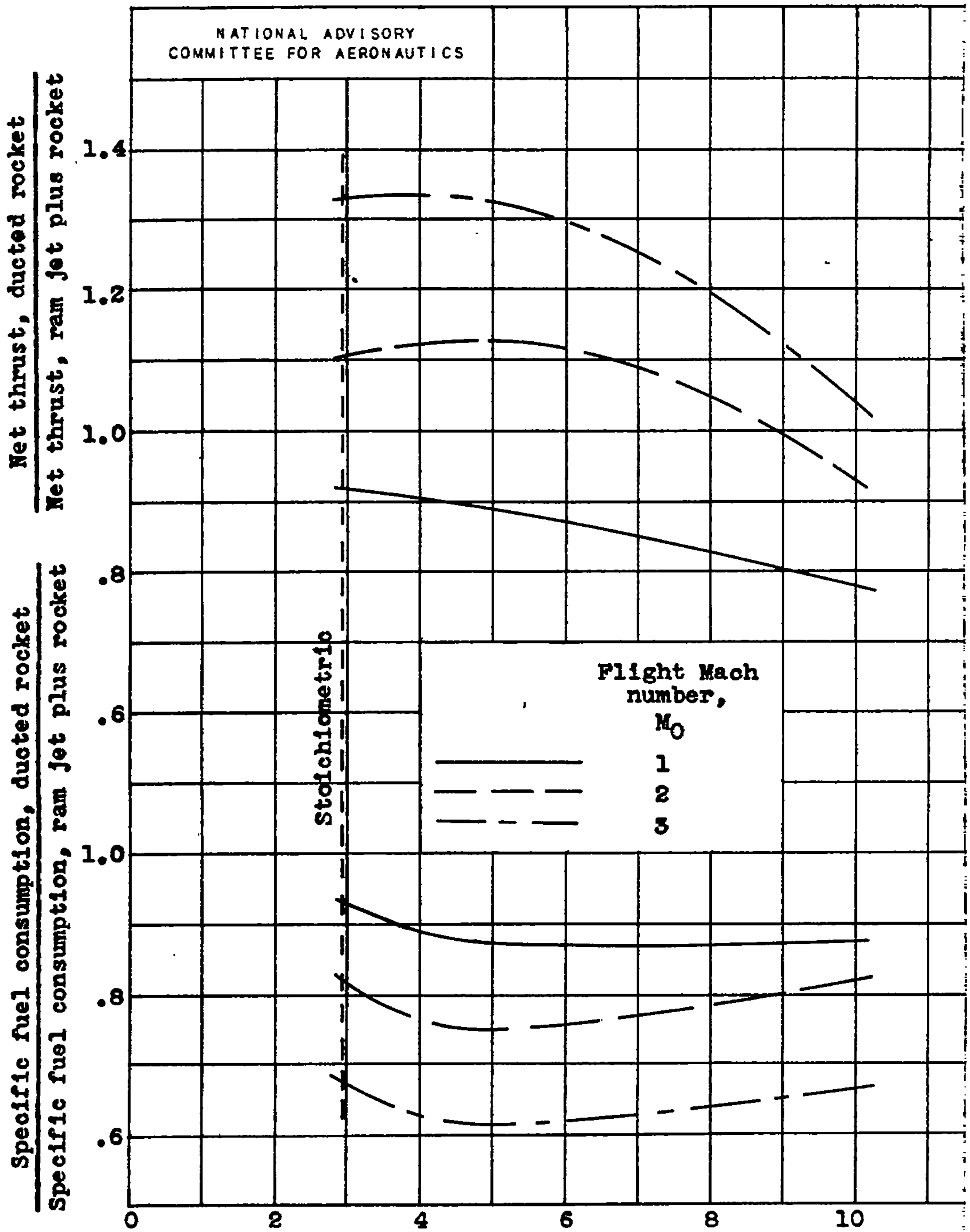

Figure 14. - Comparison of performance of ducted rocket with that of equal-frontal-area combination of similar rocket and ram jet opera-ting at stoichiometric-mixture ratio. Ducted rocket designed to g maximum specific thrusts. Altitude, 30,000 feet.