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REVISED AND EXTENDED PERFORMANCE CHARACTERISTICS OF RAM JETS

By

D. F. Collins and Francis M. Gordon

Revised and Extended Performance
Characteristics of Ram Jets

- by -

D.F. Collins and Frances M. Gordon.

SUMMARY

Previous performance estimates of ram jets operating at the design conditions and using oblique shock intakes, are revised and extended.

The range of maximum cycle temperature covers the highest value possible when burning kerosene at any particular altitude and operating at a flight Mach number between the limits of 1.0 and 3.0.

At one value of maximum cycle temperature a comparison is made of the performance of ram jets when designed with either oblique or normal shock intakes.

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1.0 Introduction

A previous report by Singham and Bartlett[1] predicts the performance of ram jets for values of maximum cycle temperature up to 2000°K at sea level and about 1500°K in the stratosphere. Since the issue of that report, maximum cycle temperatures of the order of 2200°K have been recorded on the test bed. The present work, therefore, was primarily aimed at extending the performance calculations to maximum cycle temperatures covering the highest values possible for a ram jet, whether designed for sea level or stratospheric operation and burning kerosene. It was found, however, that it was not sufficient merely to extend the previous work owing to certain simplifying assumptions having been made which, in the previous report, had the effect of over-estimating the thrust at the higher values of combustion chamber inlet Mach number.

This report, therefore, presents a new set of design point performance estimates, revising and extending those of the previous report[1], which it now supersedes.

2.0 Scope of Investigation and Method of Calculation

The investigation refers to a ram jet having a parallel-sided combustion chamber of unit cross-sectional area, and, in dealing with the estimation of the air mass flow, thrust, fuel consumption and corresponding quantities for the design cases only, does not consider the performance of a specific ram jet over varying flight conditions.

The methods of calculation follow along the usual thermodynamic lines and are presented in the Appendix, while the main assumptions made are dealt with in the following section.

3.0 Assumptions

All the calculations are for a ram jet having a combustion chamber of unit cross-sectional area.

3.1 Intake and Diffuser

The overall intake adiabatic efficiency, as defined in the Appendix, includes the diffuser loss and is assumed to vary with design flight Mach number in the manner indicated by the curve in Fig. 1. The efficiency variation used, is that which can be attained by a simple pitot intake up to a design flight Mach number of about 1.3 and by an intake of the Oswatitsh type employing a series of oblique shocks for the higher flight Mach numbers.

For comparison purposes, in Fig. 1, the efficiency curve for a plane shock is also plotted. For design flight Mach numbers above about 1.3, the efficiency curve for a practical pitot intake would tend to fair into that for a plane shock as indicated by the dotted line in Fig. 1.

It can be seen that at the higher supersonic speeds, the efficiency of the simple intake falls so much, that, from the performance point of view and apart from other considerations, a more complex form of intake such as the oblique shock type will probably be necessary. It should be pointed out that values of efficiency obtained from Fig. 1 apply only to the design case, and that the variation of the performance over varying flight conditions of a specific intake is not considered in this report.

Values obtained from the curves in Fig. 1, are used as overall intake and diffuser efficiencies, implying that the subsonic diffusion is carried out without loss. This assumption is made, for a general investigation such as this, since it has been demonstrated that with a suitable design of diffuser, very high efficiencies can be obtained, and the performance estimates of this report can then be regarded as attainable targets.

3.2 Combustion Chamber

The pressure loss caused by the combustion stabilising system is taken as four times the dynamic pressure at entry to the chamber. Combustion is assumed to take place in a parallel-sided duct; the efficiency based on temperature rise, being in general, equal to 90%.

In accordance with the usual practice in ram jet combustion work, the combustion efficiency, at any given fuel/air ratio, is defined as the ratio of the actual temperature rise to the theoretical temperature rise. For any particular fuel, the theoretical temperature rise can be plotted as a function of the air inlet temperature to the combustion system and the fuel/air ratio, and such curves for kerosene as given in reference 2 are used for the present estimates. When the fuel/air ratio reaches the stochiometric value, the theoretical temperature rise attains a maximum. It is clear, therefore, that if a constant value of combustion efficiency of 90% is assumed, the highest maximum cycle temperature to be used at a given flight Mach number, would be restricted to give an actual combustion temperature rise equal to 90% of the theoretical maximum. The results, however, include performance estimates for maximum cycle temperatures higher than those conforming to the above limitation and thus for these temperatures the assumption of combustion efficiencies higher than 90% is made. Curves plotted in Fig. 2, for the standard sea level and stratospheric values of ambient temperature and for an intermediate value of 250°K, show the manner in which the combustion efficiency has to be increased above the nominal value of 90% in order to achieve the highest maximum cycle temperatures.

In order to present the results in non-dimensional form it is necessary to assume a mean value of ambient temperature, since for a given combustion temperature rise, the fuel-air ratio is to some extent, affected by the entry air temperature to the combustion system. For the majority of the thrust calculations, therefore, a mean value of ambient temperature of 250°K is used, though at some of the highest maximum cycle temperatures, the stratospheric value of ambient temperature of 216.5°K is used, in order to apply the assumed combustion efficiency of 90% over the greatest possible range, as explained in the preceding paragraph.

It must be pointed out that the assumption of a mean ambient temperature rather than the true one for any particular altitude, causes a negligible difference in the estimated thrust, though the fuel consumption parameters are more affected. Fuel consumptions are therefore calculated for three values of ambient temperature i.e. 288°K, 250°K, and 216.5°K.

Allowance is made for variations in the value of the gas constant R, at high fuel/air ratios, but the effect is found to be negligible, and in practical ram jet calculations a constant value of R can safely be assumed.

3.3 Expansion

After leaving the combustion chamber, the gases are assumed to expand through a nozzle having an isentropic efficiency (total head to static) of 95%. At low flight Mach numbers, the nozzle is convergent only and the expansion is continued to ambient pressure; while at the higher flight Mach numbers considered (i.e. 1.5 to 3.0), a convergent-divergent nozzle is assumed, with the qualification that the divergent portion is never allowed to extend so far as to cause the exit cross-sectional area to be greater than that of the combus-

4.0 Presentation of Results

The air flow passing through a ram jet having unit combustion chamber cross-sectional area is plotted in non-dimensional form in Fig. 3, against flight Mach number.

All the results of Figs. 4 to 27 are plotted with the Mach number at entry to the combustion chamber as abscissae; while the dependent quantities such as thrust, fuel consumption and their variants (given in the list of figures), are plotted in non-dimensional form for various maximum cycle temperatures and flight Mach numbers.

The range of flight Mach number covered is from 1.0 to 3.0 and the range of the non-dimensional form of maximum cycle temperature is from 1250°K to 3000°K.

5.0 Discussion

5.1 Air Mass Flow and Choking Conditions

Fig. 3 illustrates the fact that the air flow of a ram jet operating at a particular flight Mach number, under fixed conditions of ambient temperature and pressure, is dependent only on the combustion chamber inlet Mach number. Choking of the combustion chamber exit imposes an upper limit on the inlet Mach number, which limit depends on the ratio of the maximum cycle temperature to the combustion chamber entry temperature.

An inspection of the thrust and specific fuel consumption curves shows that a limiting line termed the "choking line with no nozzle" has been imposed. Also, it can be seen that for given values of maximum cycle temperature and flight Mach number, a unique curve is obtained for thrust or specific fuel consumption when plotted against combustion chamber inlet Mach number up to some higher value in the range, after which point two diverging curves are shown. These curves can be explained by considering the operating conditions of a parallel-sided combustion chamber followed by a convergent-divergent nozzle. As the entry Mach number to the combustion chamber is increased, and at given entry and exit temperatures, the exit conditions approach choking and the ratio of nozzle throat area to combustion chamber cross-sectional area tends to unity (this latter tendency is shown in Figs. 23 to 27). When choking actually occurs, the nozzle throat area becomes equal to the combustion chamber area, i.e. in theory. the nozzle becomes a parallel pipe, but in practice, no nozzle would be fitted. For given values of flight Mach number and maximum cycle temperature, therefore, the combustion chamber has one entry Mach number corresponding to exit choking, and at this condition there is no nozzle. At all conditions up to combustion chamber outlet choking, however, the thrust is calculated allowing for a nozzle pressure loss. The true thrust at the choking condition is therefore higher than that computed with this pressure loss taken into account. A more realistic result would be obtained by increasing the nozzle efficiency as the combustion chamber choking condition is approached so that a value of 100% results when no nozzle is required. In order to present this more realistic picture, thrust and specific fuel consumption curves have been smoothed to their respective "no nozzle" values by broken lines.

5.2 Thrust and Fuel Consumption

up to the value corresponding to combustion chamber outlet choking. It will be realised therefore, that when operating at the design condition of a high flight Mach number and a low value of maximum cycle temperature, a ram jet designed with an oblique shock intake should also be provided with a propelling nozzle, in order to obtain the maximum possible range. This combination of a high flight speed and a low maximum cycle temperature may quite possibly be required at the cruising condition, since after the cruising Mach number has been attained, one way of reducing the thrust coefficient from the climb value to that required for steady level flight, is for the fuel metering device to cause a reduction in the maximum cycle temperature. Whether or not a propelling nozzle is, in fact, desirable however, will depend on whether the high thrust coefficient necessary for the climb can be realised for a particular ram jet; and this in turn will depend, amongst other things, on the performance of the oblique shock intake at a flight condition different from that for which the intake is designed.

The curves of specific fuel consumption of Figs. 18 to 22 are calculated for the same values of ambient temperature as the thrust curves. In order to avoid confusion, the additional curves necessary to cover a range of ambient temperature from 216.5°K to 288°K, such as are plotted in the fuel consumption figs., are omitted in those relating to specific fuel consumption. Values of specific fuel consumption, for ambient temperatures other than those for which the curves are plotted, can easily be calculated by use of the fuel consumption curves; since, for given values of flight Mach number, combustion chamber entry Mach number and maximum cycle temperature, the thrust is sensibly constant for all values of ambient temperature and the specific fuel consumption, therefore, varies directly as the fuel consumption itself. The curves in Figs. 18 to 22 show that at a particular flight Mach number and maximum cycle temperature, the specific fuel consumption increases rapidly as the entry Mach number to the combustion chamber approaches the limit imposed by combustion chamber outlet choking.

5.3 Comparison with Ram Jet Designed with a Normal Shock Intake

Figs. 28 and 29 are included in order to compare the performance of a ram jet when designed with an intake of either the oblique shock or the normal shock type. The comparison is made for an equivalent maximum cycle temperature of 2000°K, thrust coefficient and specific fuel consumption, being plotted as a function of flight Mach number for values of combustion chamber inlet Mach number of 0.10 and 0.15. It can be seen that as the design flight Mach number is increased above the value at which the efficiency is approximately the same for both types of intake, the thrust for the normal shock design, falls steadily as compared with the rising curve obtainable with the oblique shock intake. At a design flight Mach number of 3.0 and a combustion chamber entry Mach number of 0.15, the thrust of the ram jet using the normal shock intake has fallen in value to 40% of that attainable using the oblique shock intake, while the specific fuel consumption has increased by about 28%.

It appears therefore, that for ram jet flight at very high Mach numbers, and provided that the problems, associated with operation at conditions other than those of the design, can be overcome, a complex intake of the oblique shock type has a great advantage over the simple pitot intake.

REFERENCES

No.	Author	Title
1	J.R. Singham and W.J. Bartlett.	The Performance of Propulsive Ducts. Power Jets Report No. R.1204. A.R.C.10,735
2	J.E.C. Topps.	Thermal Data for the Combustion Products of Kerosene Air Mixtures. N.G.T.E. Report No. R.14. A.R.C.10,828
3	A.D. Richards and W.G. Lydiard.	Gas Flow Data Sheets for Propulsive Duct Calculations. N.G.T.E. Report No. R.13. A.R.C.11,054

APPENDIX

An outline of the methods used to estimate Ram Jet performance is given below. The planes of reference are shown in Fig. 1A.

Notation

The symbols used are similar to those of N.G.T.E. Report No. R.13, but are set out below for sake of completeness.

T = Static Temperature °K

P = Static Pressure lb/sq.in.

T_t = Total Head Temperature °K

P_t = Total Head Pressure. lb/sq.in.

V = Gas Velocity. ft/sec.

η_i = Intake Isentropic Efficiency = $\dfrac{\text{Isentropic Temperature Rise}}{\text{Actual Temperature Rise}}$

M = Mach Number = $\dfrac{\text{Gas Velocity}}{\text{Local Velocity of Sound}}$

Kp = Specific Heat at Constant Pressure

γ = Ratio of Specific Heats

A = Cross-Sectional Area sq.ft.

Q = Air or Gas Mass Flow. lb/sec.

Q_F = Fuel Mass Flow. lb/sec.

η_C = Combustion Efficiency = $\dfrac{\text{Actual Temp. Rise}}{\text{Theoretical Temp. Rise}}$

R = The Gas Constant. ft.lb/lb.

J = The Mechanical Equivalent of Heat. ft.lb/C.H.U.

g = Acceleration due to gravity. ft/sec/sec

q = Fuel/air ratio.

F = Net Thrust lb.

C_T = Thrust Coefficient = $\dfrac{\text{Net Thrust per unit Combustion Chamber Cross-Sect. Area.}}{\text{Incompressible Velocity Head of the Free Stream.}}$

ϕ = $\dfrac{P_{2t} - P_{2t}^1}{P_{2t} - P_2}$ = Stabilising Baffle Pressure Loss Factor

$$Y = \frac{\sqrt{\gamma}\, M \sqrt{1 + \frac{\gamma - 1}{2} M^2}}{1 + \gamma M^2} \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots *$$

$$Z = \frac{1 + \gamma M^2}{\left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}}} \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots *$$

The significance of the asterisk which follows some of the equations is that an equation so marked can be evaluated using the curves given in N.G.T.E. Report No. R.13.

Intake

$$T_{1t}/T_a = 1 + \frac{(\gamma - 1)}{2} M_a^2 = T_{2t}/T_a \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (1)*$$

$$P_{1t}/P_a = \left(1 + \eta_i \frac{(\gamma - 1)}{2} M_a^2\right)^{\frac{\gamma}{\gamma-1}} = P_{2t}/P_a \quad \ldots\ldots\ldots\ldots (2)*$$

where γ is taken as being equal to 1.4, and the intake efficiency η_i is given, for an assumed value of flight Mach No. M_a, by the relationship shown in Fig. 1. If a value is assumed for the Mach No. at entry to the combustion chamber, (M_2), then T_{2t}/T_2 is known from

$$T_{2t}/T_2 = 1 + \frac{(\gamma - 1)}{2} M_2^2 \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (3)*$$

Hence $T_2/T_a = \dfrac{T_{2t}/T_a}{T_{2t}/T_2}$

Also $P_{2t}/P_2 = \left(1 + \frac{\gamma - 1}{2} M_2^2\right)^{\frac{\gamma}{\gamma - 1}} \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (4)*$

and $P_2/P_a = \dfrac{P_{2t}/P_a}{P_{2t}/P_2}$

Air flow

The air flow parameter is given by

$$\frac{Q_2\sqrt{T_a}}{A_2\, P_a} \frac{14.7}{\sqrt{288}} = \frac{Q_2\sqrt{T_2}}{A_2\, P_2} \times \frac{P_2}{P_a} \times \sqrt{\frac{T_a}{T_2}} \times \frac{14.7}{\sqrt{288}} \quad \ldots\ldots\ldots\ldots\ldots\ldots (5)$$

where P_2 and P_a are in lb/sq.in., and $\dfrac{Q\sqrt{T_2}}{A_2 P_2} = M_2 \sqrt{\dfrac{\gamma g}{R}} \times 144$

Combustion Chamber

(a) Flow passed the Baffle

Using Richards curves[3] and taking the loss factor $\phi = 4$, the total head pressure loss and the increase in Mach No. in passing the baffle can be found.

i.e. $M_2{}^1$ and $P_{2t}{}^1/P_{2t}$ are determined

$$\therefore \quad P_{2t}{}^1/P_a = P_{2t}{}^1/P_{2t} \times P_{2t}/P_a.$$

(b) Heat Addition

Assuming a value of $T_{3t}\left(\frac{288}{T_a}\right)$, and in general that $T_a = 250°K$, the actual combustion temperature rise $T_{3t} - T_{2t}$ is given, and from Ref. 2 is found the theoretical fuel/air ratio q_{TH}. The theoretical temperature rise is given by $\frac{1}{\eta_C}(T_{3t} - T_{2t})$ and the curves of Ref. 2 give the corresponding fuel/air ratio q_A.

Then $Q_3/Q_2 = 1 + q_A$

Also, from Topps' curves[2], is determined the value of the gas constant R, which increases with increase of fuel/air ratio. In actual fact, R only deviates appreciably from the value for air at the highest values of fuel/air ratios. The curves give also γ_3 using T_{3t} and q_{AV}, where q_{AV} is an average value between q_A and q_{TH}.

The fuel consumption parameter is

$$\frac{Q_F\ 14.7}{A_2\ P_a}\sqrt{\frac{288}{T_a}} = q_A\left(\frac{Q\sqrt{T_a}}{A_2\ P_a}\ \frac{14.7}{\sqrt{288}}\right) \times \frac{288}{250} \quad \text{..................(6)}$$

(c) The Combustion Chamber Exit Conditions

The method used depends upon whether or not R deviates appreciably from the value for air. If it is assumed constant at 96 ft.lb/lb., Richards' work[3] gives

$$\frac{Q\sqrt{T_t}}{Y} \text{ is constant}$$

$$\text{i.e.} \quad Y_3 = \frac{Q_3}{Q_2}\sqrt{\frac{T_{3t}}{T_{2t}}}\ Y_2{}^1 \quad \text{.................................(7)}$$

and after evaluating Y_3, M_3 is obtained from the curves of R.13, knowing γ_3. If M_3 is greater than about 0.6, however, T_3 is determined, and hence the corresponding values of γ_3 and M_3, which are slightly different from the values obtained using the total head temperature T_{3t}.

When R is found to have increased considerably above the value for air, the method is as follows:-

Considerations of continuity and momentum give

$$\sqrt{\frac{T_2{}^1}{T_a}}\ \frac{(1 + \gamma_2 (M_2{}^1)^2)}{M_2{}^1} = \sqrt{\frac{T_3}{T_a}}\ \frac{(1 + \gamma_3 M_3{}^2)}{M_3}\ \frac{Q_3}{Q_2}\sqrt{\frac{R_3}{R_2}}\sqrt{\frac{\gamma_2}{\gamma_3}} \quad \ldots.(8)$$

With the aid of curves giving $\frac{1 + \gamma M^2}{M}$ as a function of M for a range of values of γ, M_3 is found by a trial and error process, since, for a given value of T_{3t}/T_a, T_3/T_a is also a function of M_3. It will be observed that the term in R consists of the square root of the ratio of the values of R at exit and entry to the chamber, and it is found in fact, that no appreciable difference is made in the calculations by allowing for the variations in R.

Now R.13 gives

Z_3 as a function of M_3 and γ_3

$Z_2{}^1$ as a function of $M_2{}^1$ and γ_2

Therefore, having determined M_3,

$$P_{2t}{}^1/P_{3t} = Z_3/Z_2{}^1$$

and $$P_{3t}/P_a = \frac{P_{2t}{}^1/P_a}{P_{2t}{}^1/P_{3t}}$$

<u>Propelling Nozzle</u>

Using a nozzle adiabatic efficiency of 95%

$$\frac{\Delta T_{35}}{T_a} = \frac{(P_{3t}/P_5)^{\frac{\gamma - 1}{\gamma}} - 1}{(P_{3t}/P_5)^{\frac{\gamma - 1}{\gamma}}} \times 0.95 \times \frac{T_{3t}}{T_a} \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots(9)$$

where $\Delta T_{35} = T_{3t} - T_5$

$(\gamma - 1)/\gamma$ in the latter equation is a mean value over the expansion and is obtained from curves plotted for mean $\frac{\gamma - 1}{\gamma}$ as a function of pressure ratio P_{3t}/P_a and T_{3t}

$$\therefore \quad V_5 / \sqrt{T_a} = \sqrt{2gJKp\,\frac{\Delta T_{35}}{T_a}}$$

therefore so long as $\frac{Q_3\sqrt{T_a}}{A_5\ P_a}$ is greater than $\frac{Q_3}{Q_2} \times \frac{Q_2\sqrt{T_a}}{A_2\ P_a}$, thrust is calculated as,

$$\frac{F \times 14.7}{A_2\ P_a} = \frac{Q_3}{Q_2} \times \frac{Q_2\sqrt{T_a}}{A_2\ P_a}\ \frac{14.7}{32.2}\frac{V_5}{\sqrt{T_a}} - \frac{Q_2\sqrt{T_a}}{A_2\ P_a}\ \frac{14.7}{32.2}\ \frac{V_a}{\sqrt{T_a}} \qquad \ldots\ldots\ldots\ldots(11)$$

If $\frac{Q_3\sqrt{T_a}}{A_5\ P_a}$ is less than $\frac{Q_3}{Q_2} \times \frac{Q_2\sqrt{T_a}}{A_2\ P_a}$, it is necessary to assume that the divergent portion of the nozzle is shortened, which involves finding a value of M_5 such that P_{3t}/P_5 satisfies an area ratio A_5/A_2 of unity. Using this value of P_{3t}/P_5, thrust is calculated as

$$\frac{F \times 14.7}{A_2 P_a} = \frac{Q_3}{Q_2} \times \frac{Q_2\sqrt{T_a}}{A_2\ P_a}\ \frac{14.7}{32.2}\ \frac{V_5}{\sqrt{T_a}} - \frac{Q_2\sqrt{T_a}}{A_2\ P_a}\ \frac{14.7}{32.2}\ \frac{V_a}{\sqrt{T_a}} + 14.7\left(\frac{P_5}{P_a} - 1\right)144 \qquad \ldots(12)$$

The parameter for specific fuel consumption is derived by dividing that for fuel consumption by thrust i.e.

$$\frac{Q_F}{F}\sqrt{\frac{288}{T_a}} = \frac{14.7\ Q_F}{A_2\ P_a}\sqrt{\frac{288}{T_a}} \times \frac{A_2\ P_a}{F \times 14.7} \qquad \ldots\ldots\ldots\ldots\ldots\ldots\ldots(13)$$

Thrust Coefficient C_T is defined as the net thrust per unit combustion chamber cross-sectional area divided by the incompressible dynamic head of the free stream.

$$\text{i.e.}\quad C_T = \frac{F}{\frac{1}{2}\rho V_a^{\ 2}\ A_2}$$

where ρ is the ambient density in slugs/cu.ft. and V_a is the flight velocity in ft./sec.

$$\text{i.e.}\quad C_T = \left(\frac{F \times 14.7}{A_2\ P_a}\right) \times \frac{2}{144 \times 14.7 \times \gamma \times M_a^{\ 2}} \qquad \ldots\ldots\ldots\ldots\ldots(14)$$

Ratio of Nozzle Throat Area to Combustion Chamber Area

(i) When $M_a = 1.0$ (P_{3t}/P_a less than critical)

$$\frac{A_4}{A_2} = \frac{\frac{Q_1\sqrt{T_a}}{A_2\ P_a} \times \frac{Q_3}{Q_1}}{\frac{Q_3\sqrt{T_a}}{A_5\ P_a}} \qquad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots(15)$$

X_3 and X_4 can be found from Report R.13 since M_3 and M_4 are known. (M_4 = 1.0)

and $$\frac{P_{4t}}{P_{3t}} = \frac{P_4}{P_{3t}} \times \frac{P_{4t}}{P_4}$$

$$= \left(1 - \frac{\frac{\gamma - 1}{2} M_4^{\,2}}{\eta \left(1 + \frac{\gamma - 1}{2} M_4^{\,2}\right)}\right)^{\frac{\gamma}{\gamma - 1}} \frac{P_{4t}}{P_4} \qquad \text{.........................(16)}$$

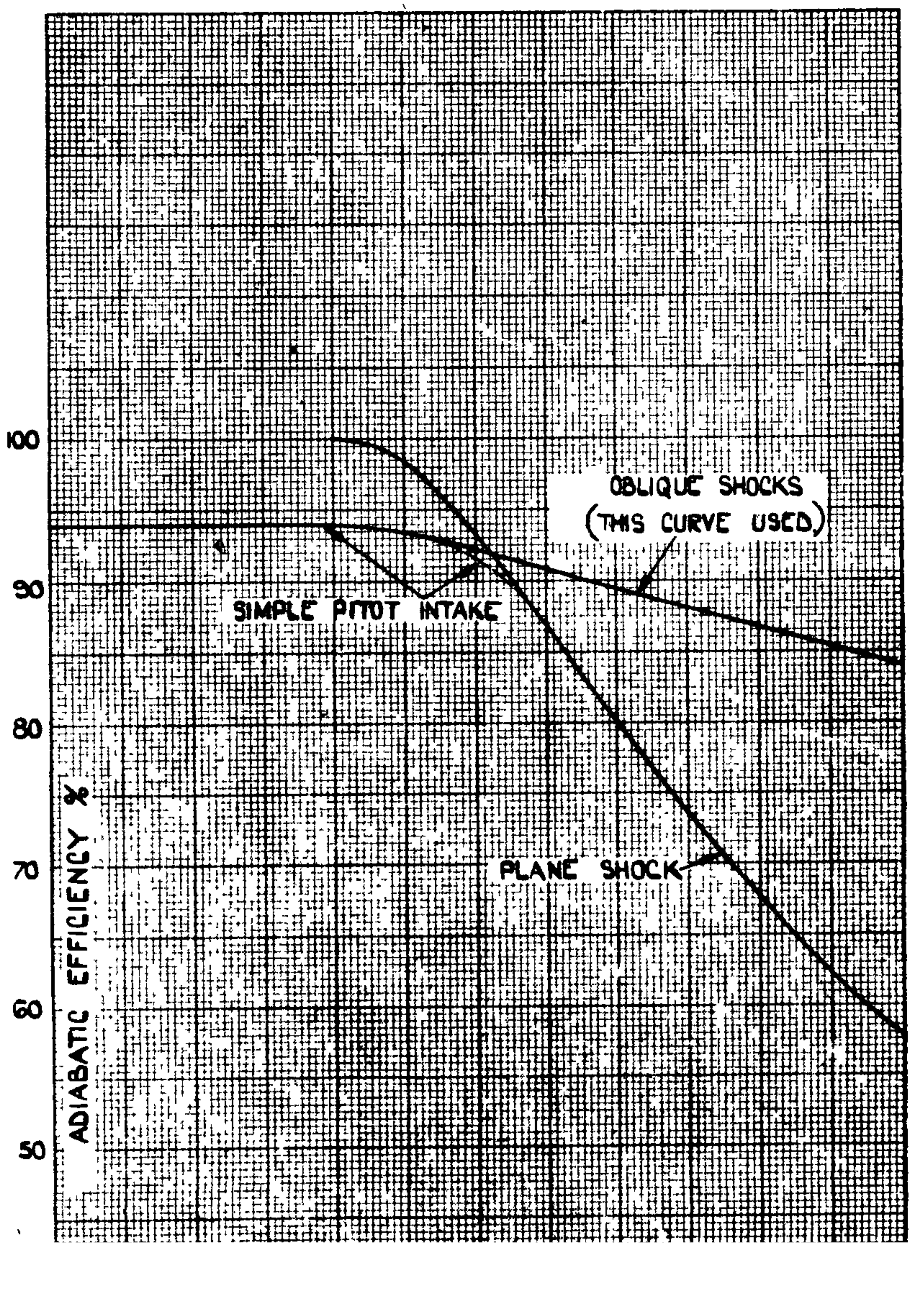
$\frac{\gamma}{(\gamma - 1)}$ is the mean value over the expansion, η is the nozzle efficiency

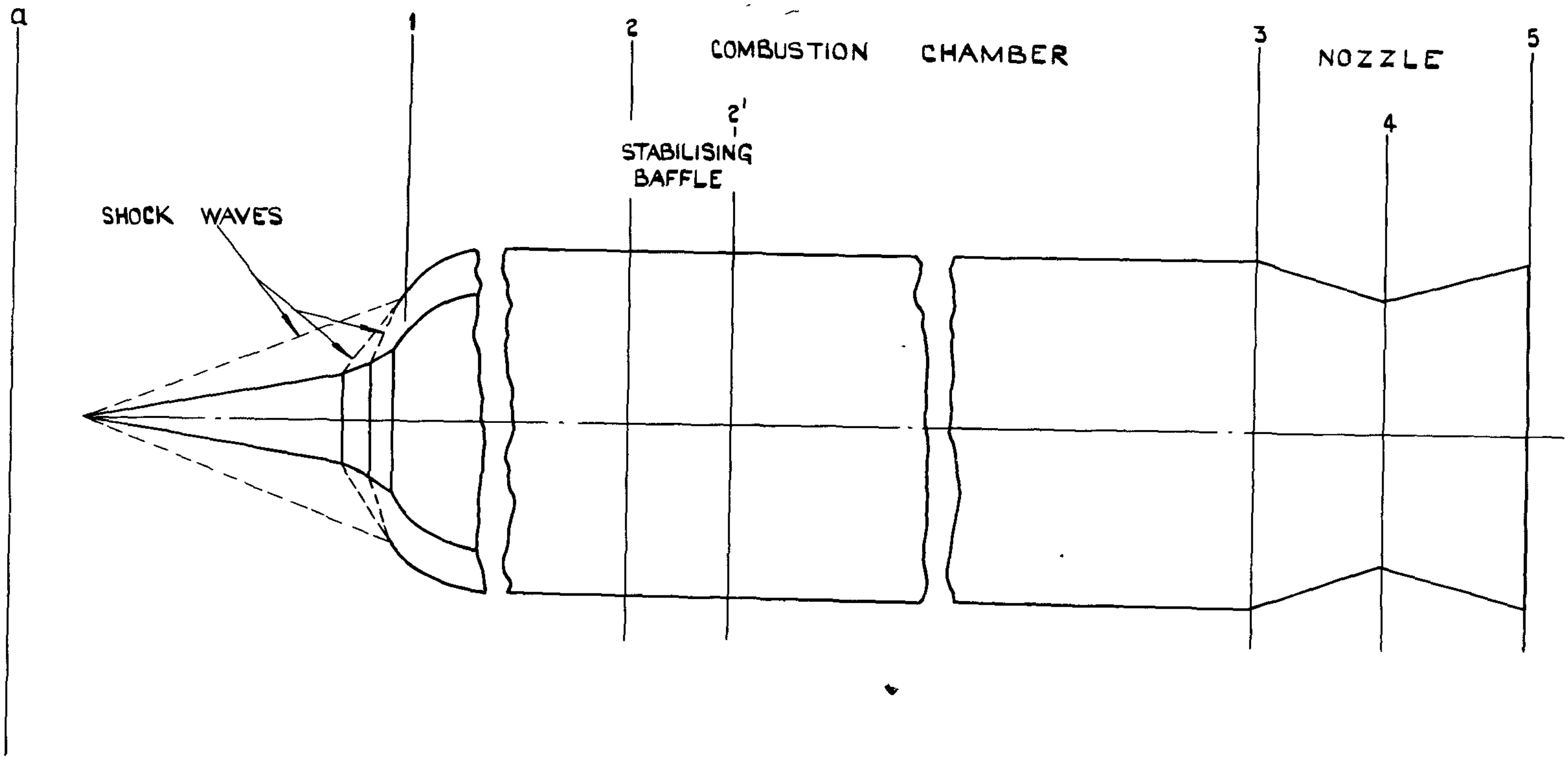
and $\frac{P_{4t}}{P_4}$ can be read off the curves in R.13.

FIG. I.

DESIGN PERFORMANCE OF RAM JETS.

INTAKE EFFICIENCY





RAM JET.

SKETCH SHOWING PLANES OF REFERENCE.

FIG. 1A.

FIG. 2.

DESIGN PERFORMANCE OF RAM JETS.

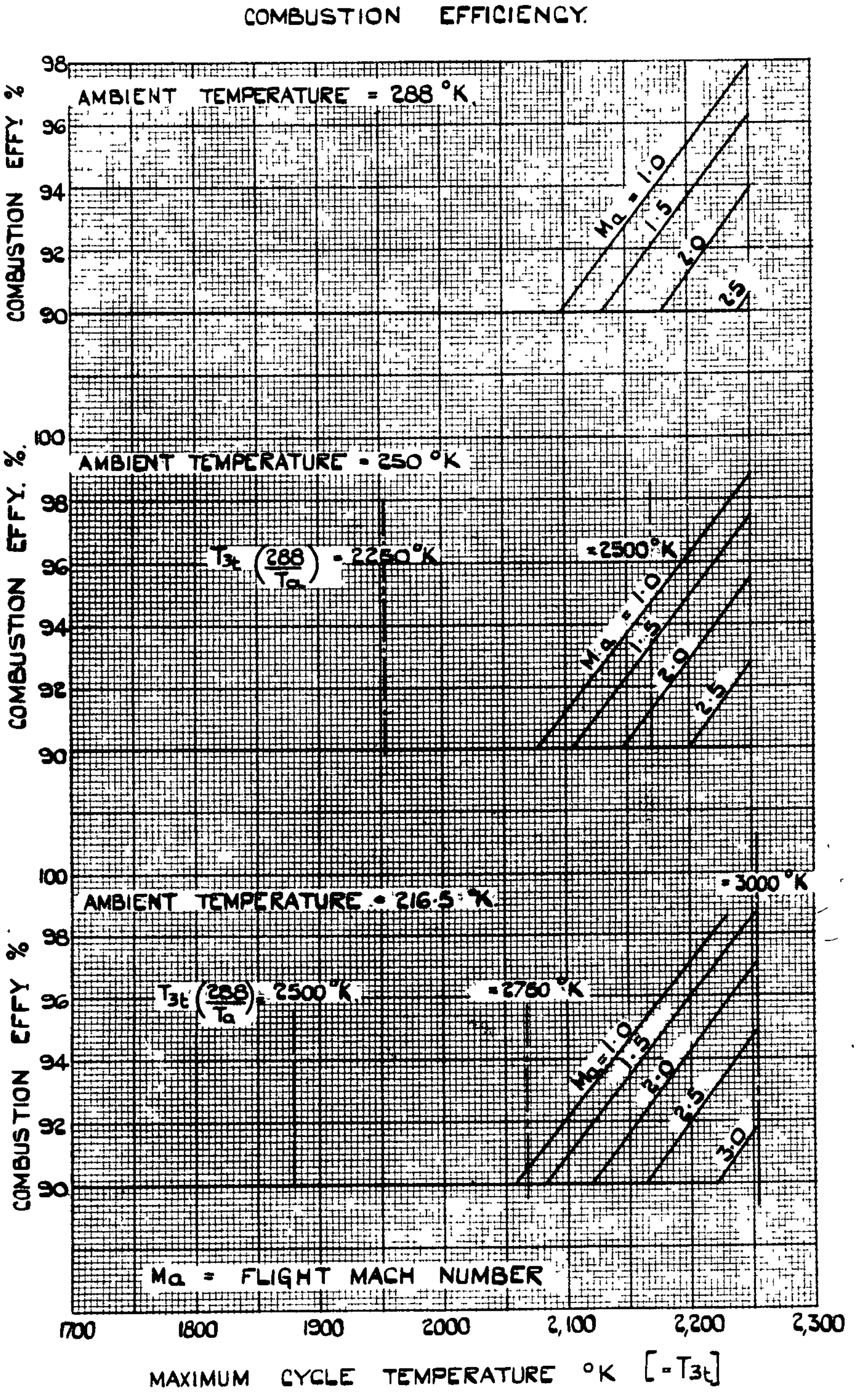
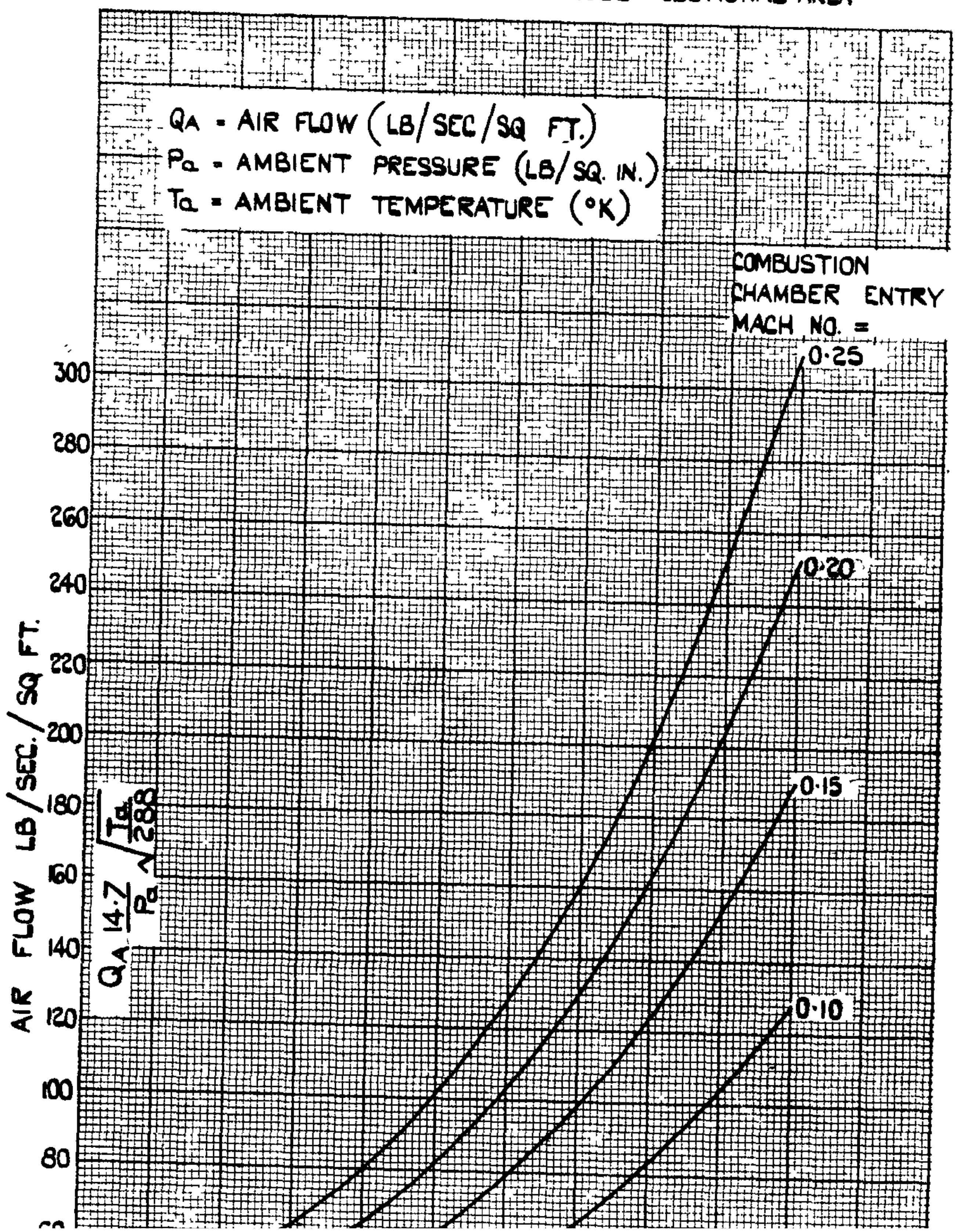


FIG. 3.

DESIGN PERFORMANCE OF RAM JETS.

AIR FLOW

PER UNIT MAXIMUM CROSS-SECTIONAL AREA



DESIGN PERFORMANCE OF RAM JETS. FIG. 4.

THRUST.

PER. SQ FT. OF MAX CROSS - SECTIONAL AREA.

FLIGHT MACH No = 1·0

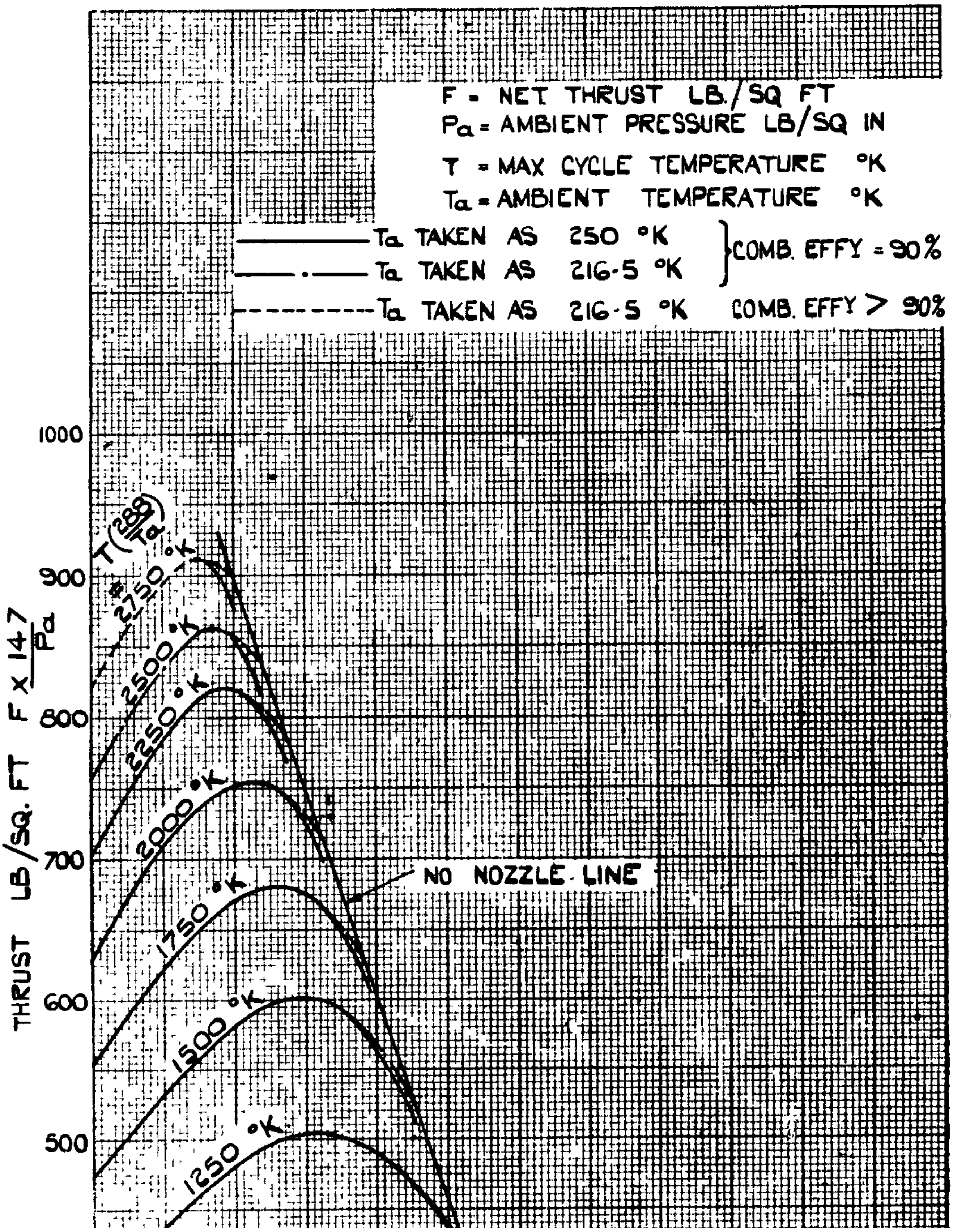


FIG. :

DESIGN PERFORMANCE OF RAM JETS.

THRUST

PER SQ FT OF MAX CROSS - SECTIONAL AREA.

FLIGHT MACH No = 1·5

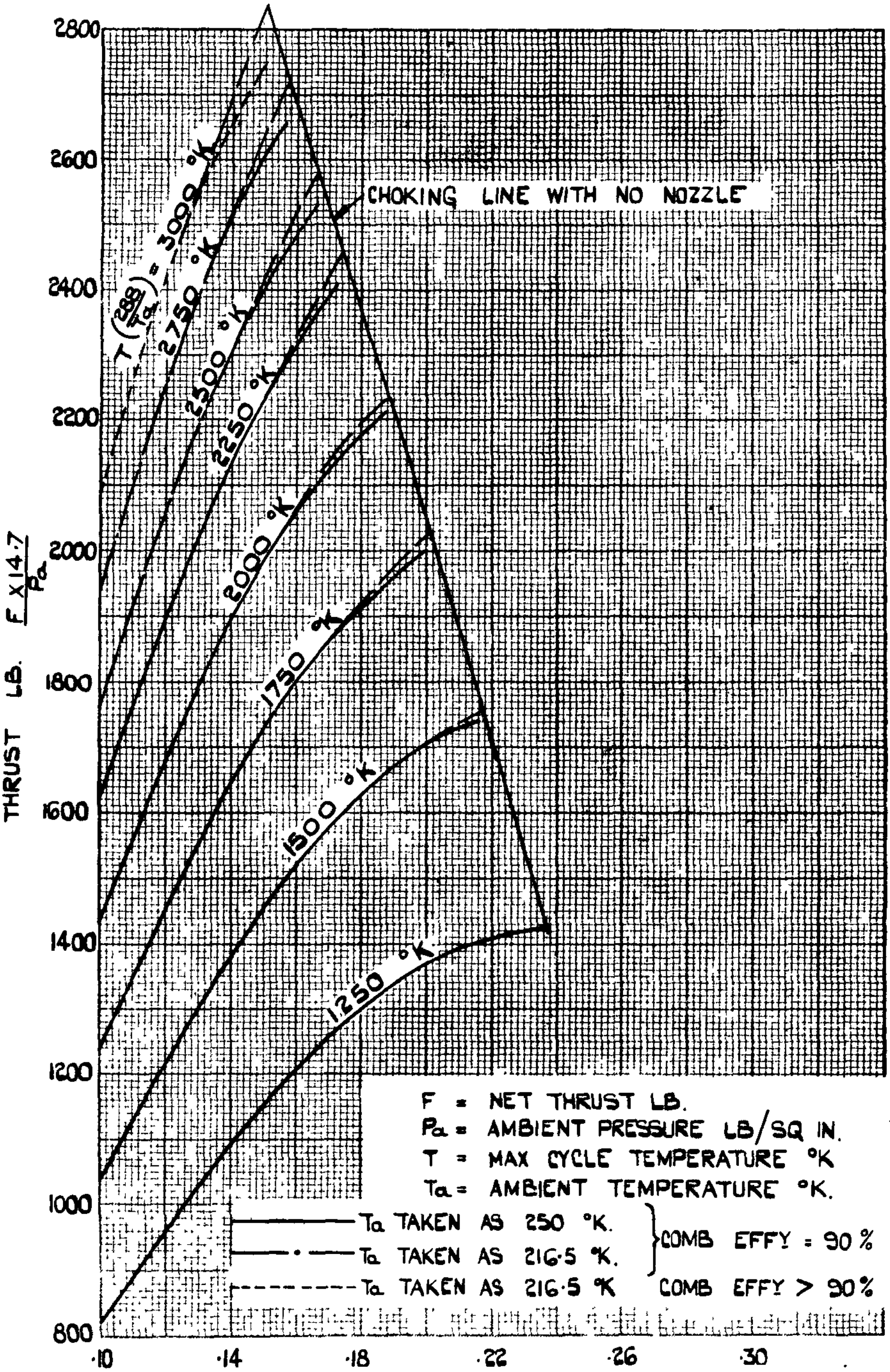
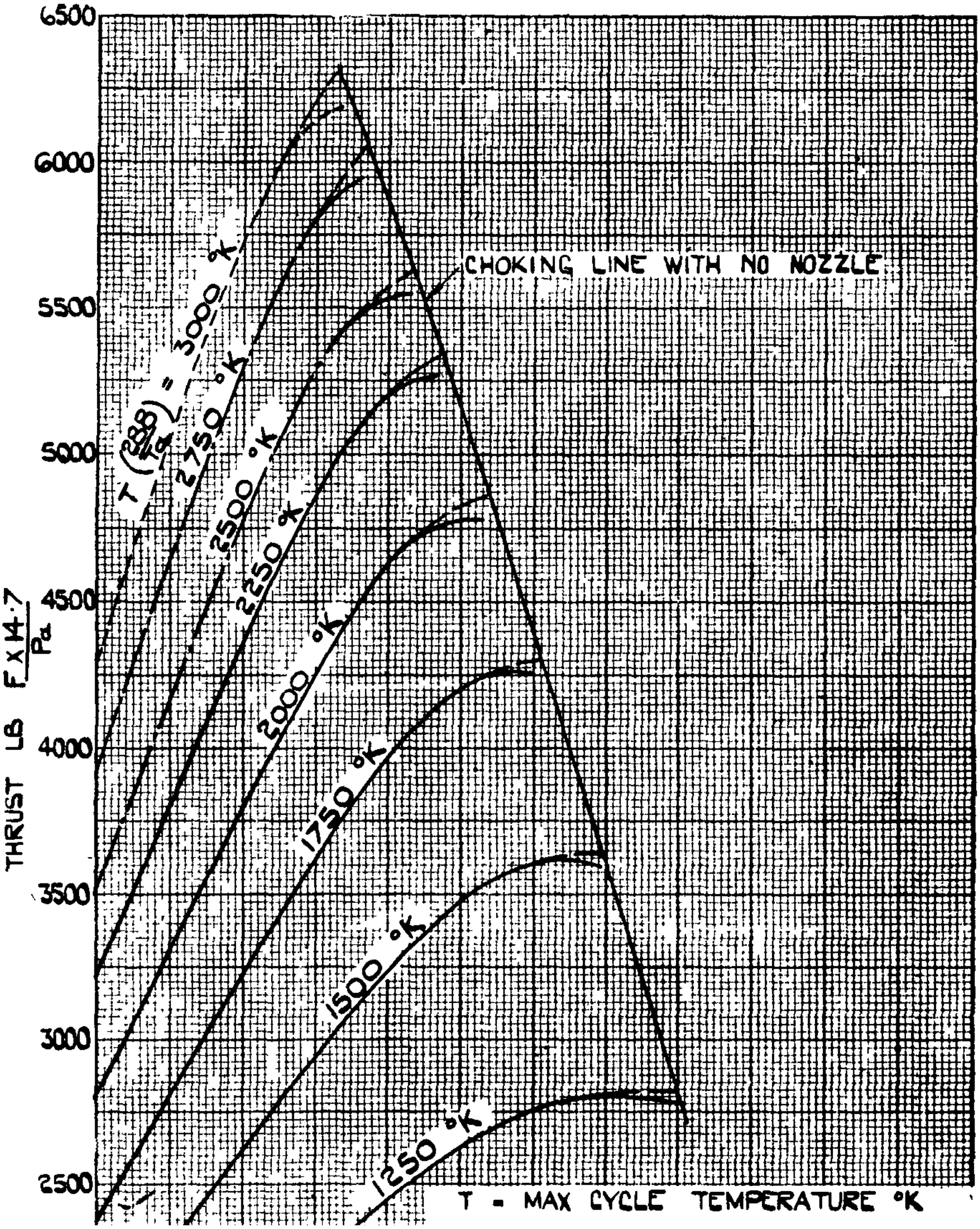


FIG. 6.

DESIGN PERFORMANCE OF RAM JETS.

THRUST.
PER SQ FT. OF MAX CROSS - SECTIONAL AREA
FLIGHT MACH. NO = 2·0



DESIGN PERFORMANCE OF RAM JETS.

THRUST.

PER SQ FT. OF MAX CROSS - SECTIONAL AREA

FLIGHT MACH. NO = 2·5

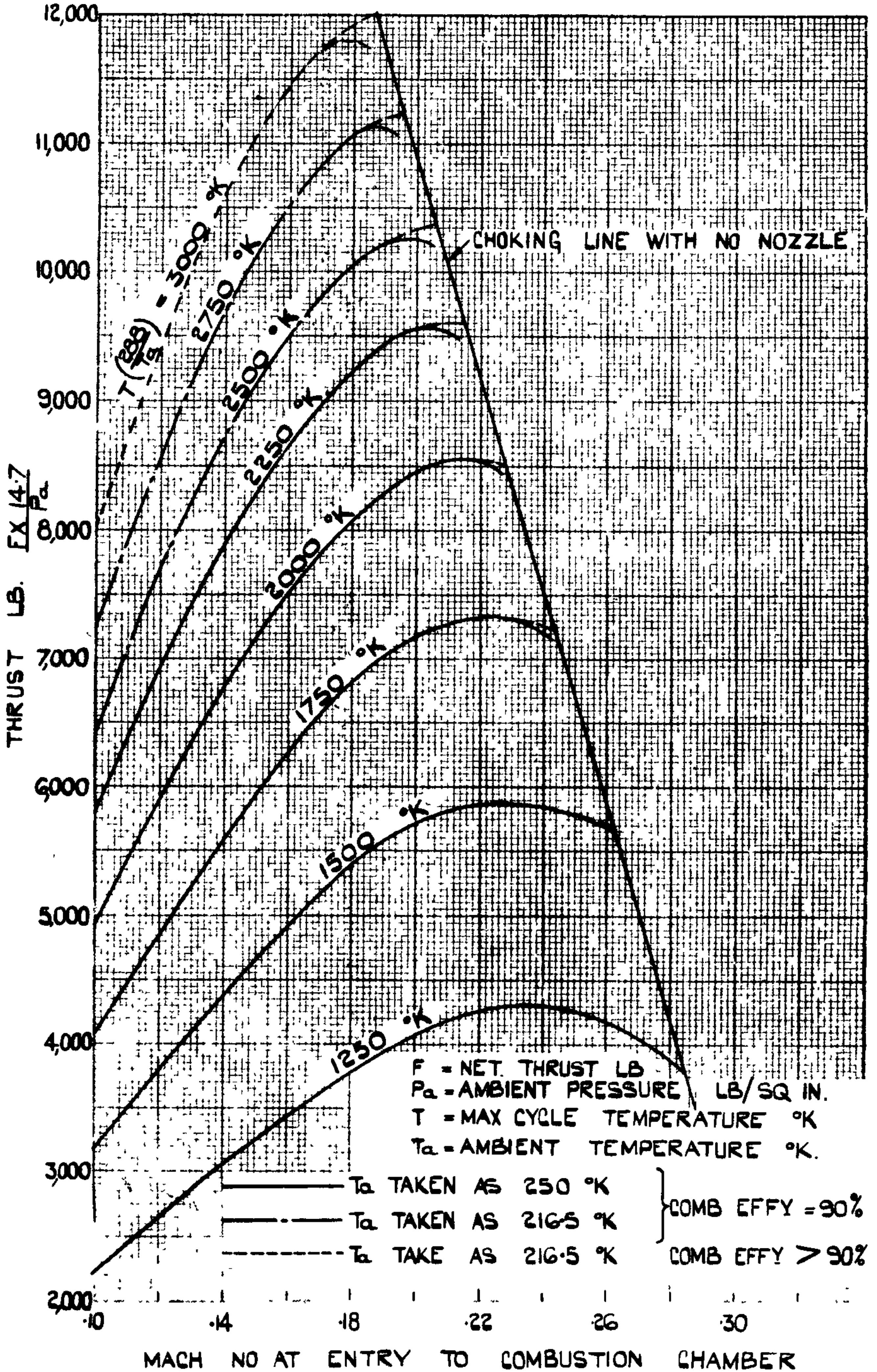


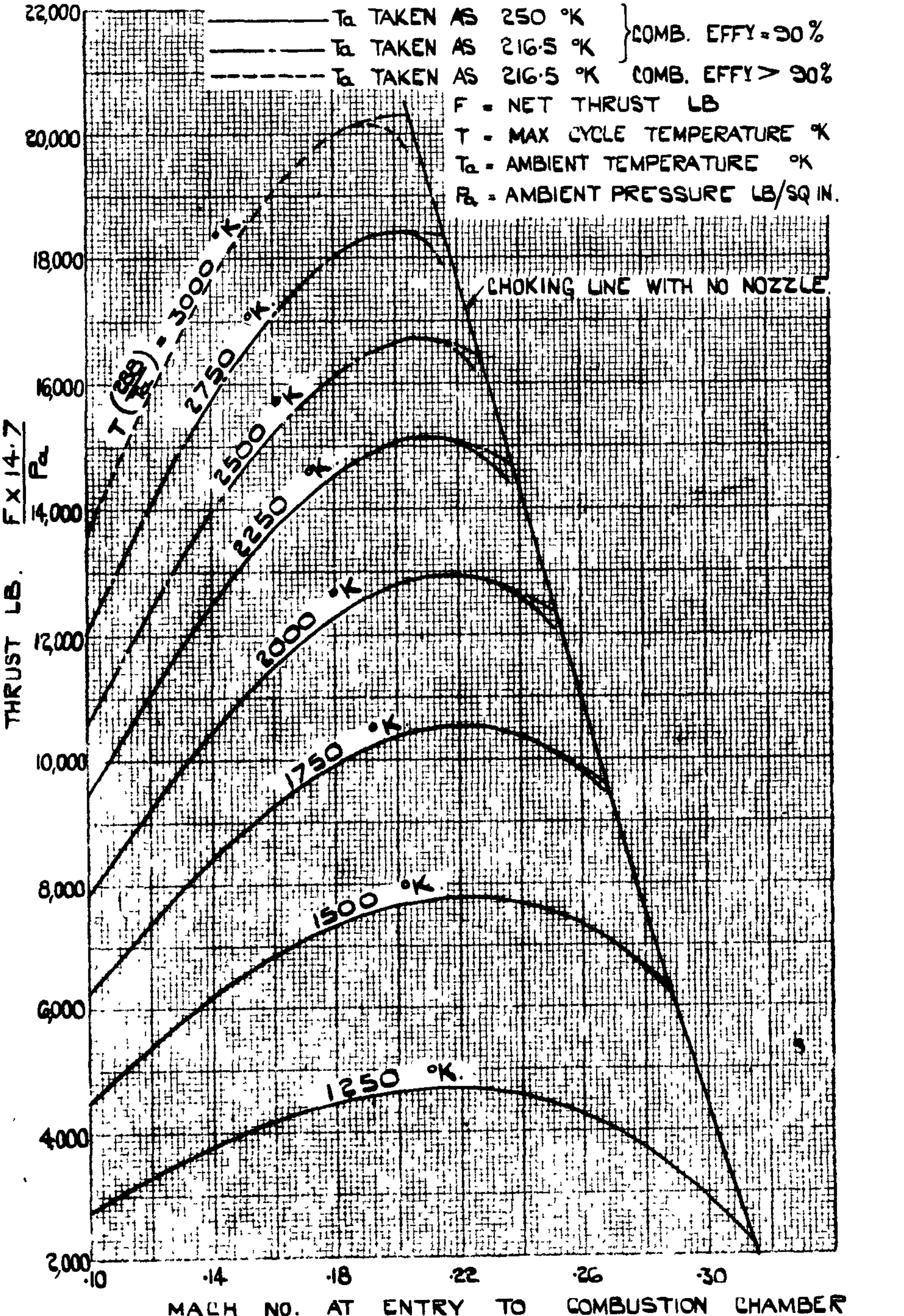
FIG. 8.

DESIGN PERFORMANCE OF RAM JETS.

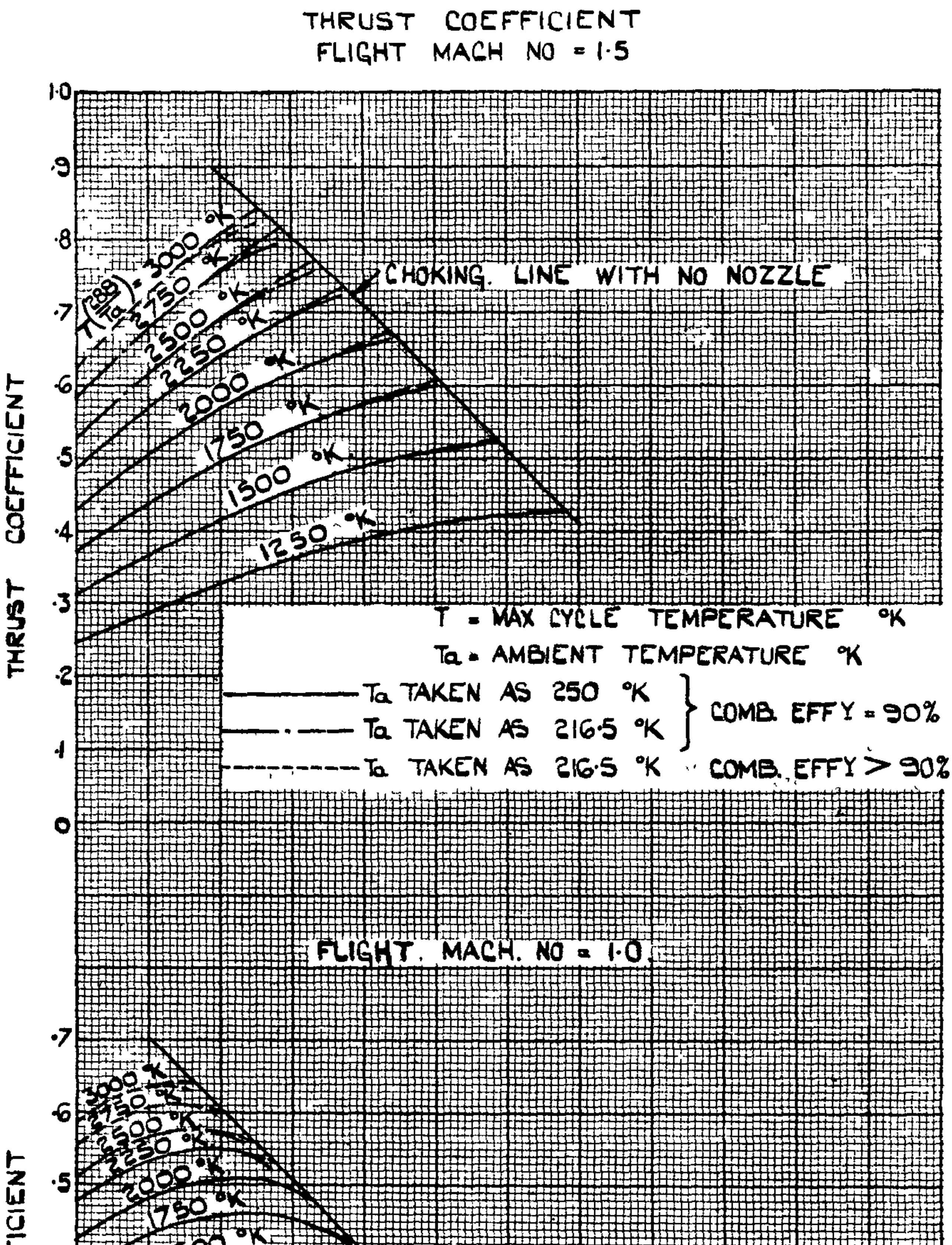
THRUST.

PER SQ FT OF MAX CROSS-SECTIONAL AREA

FLIGHT MACH No = 3·0



DESIGN PERFORMANCE OF RAM JETS.



DESIGN PERFORMANCE OF RAM JETS.

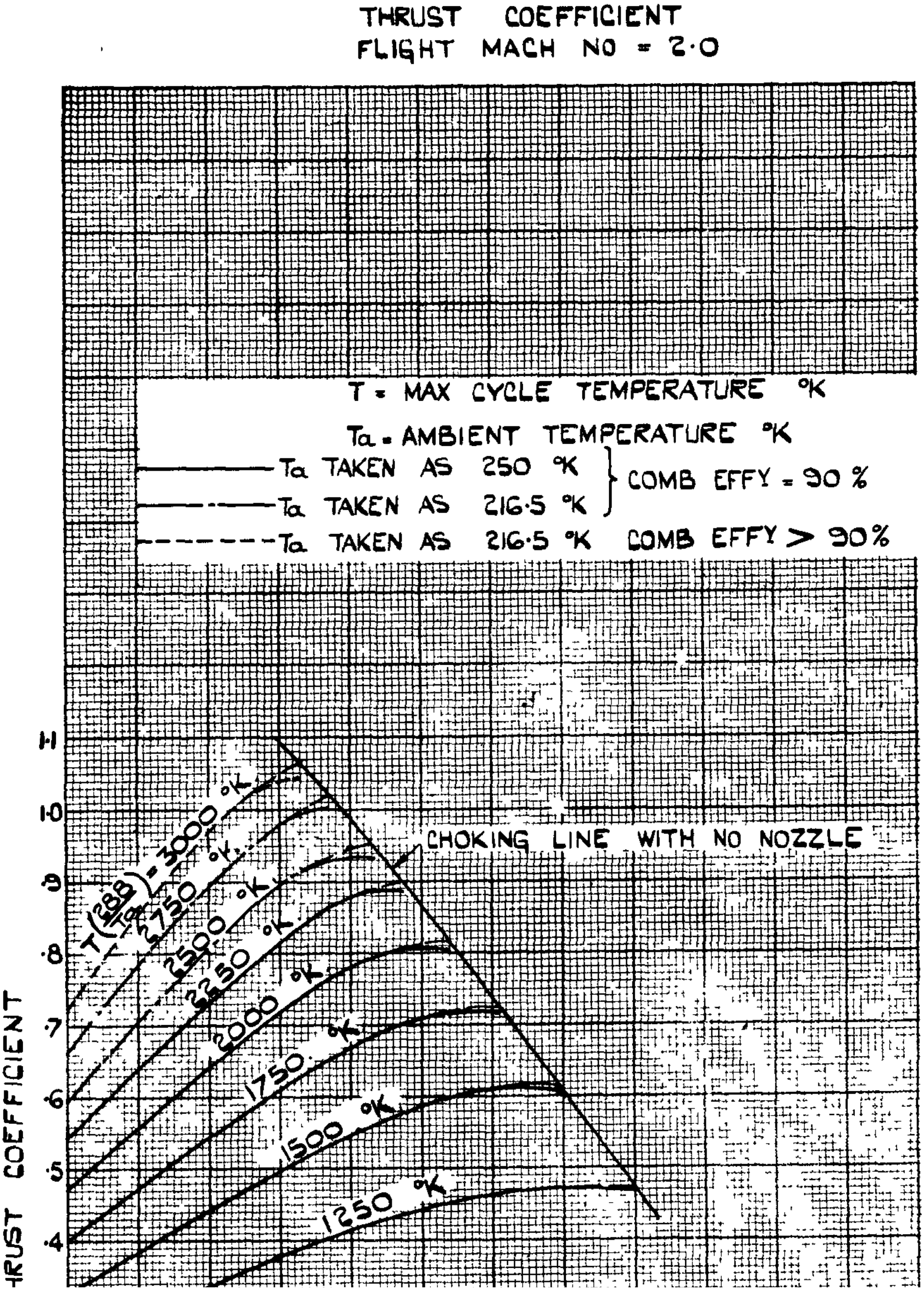


FIG. 11.

DESIGN PERFORMANCE OF RAM JETS.

THRUST COEFFICIENT
FLIGHT MACH. NO = 2·5

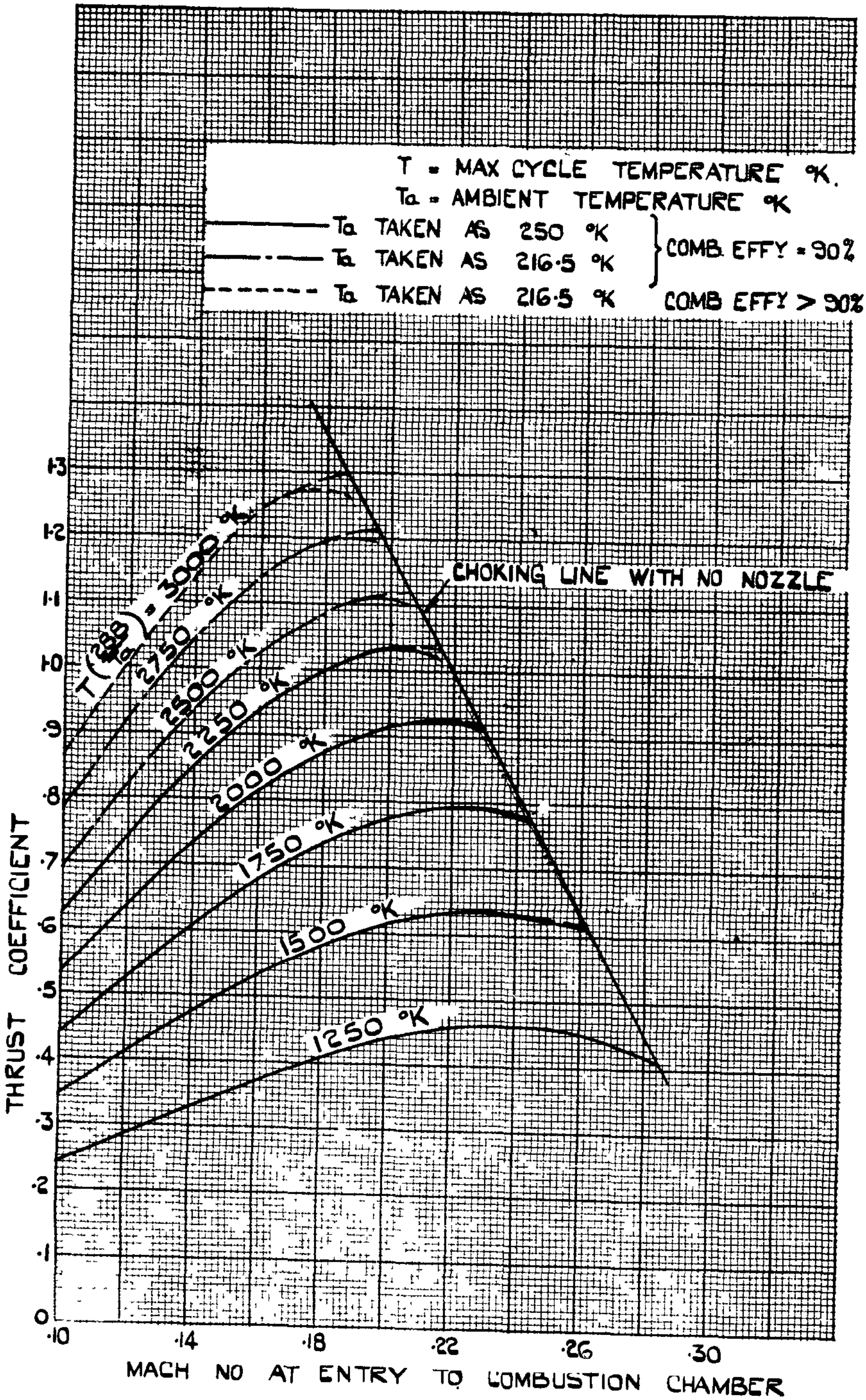


FIG.12.

DESIGN PERFORMANCE OF RAM JETS.

THRUST COEFFICIENT.
FLIGHT MACH NO = 3·0

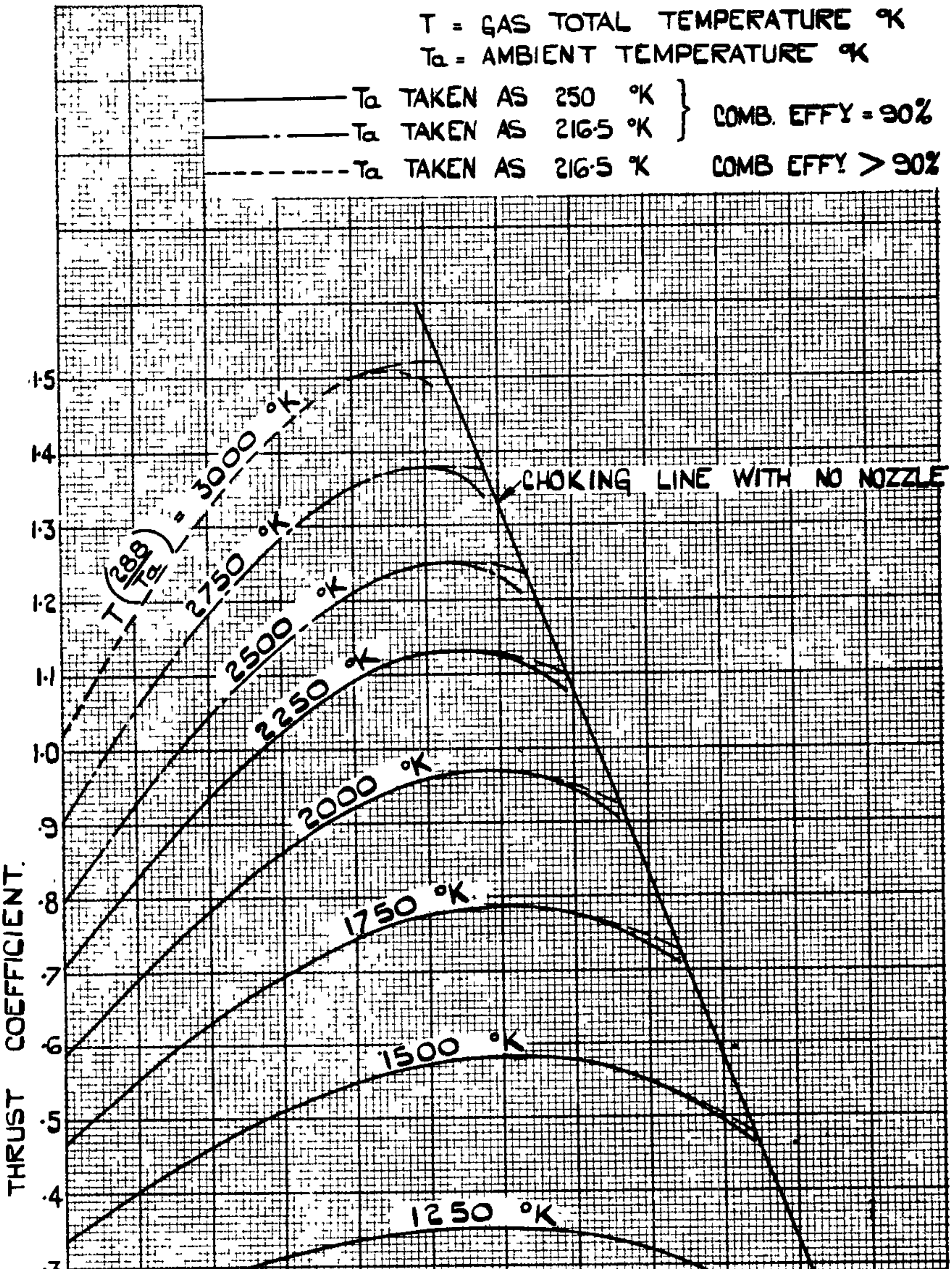


FIG.13.

DESIGN PERFORMANCE OF RAM JETS.

FUEL CONSUMPTION

PER SQ FT OF MAX CROSS - SECTIONAL AREA.

FLIGHT MACH NO = 1·0

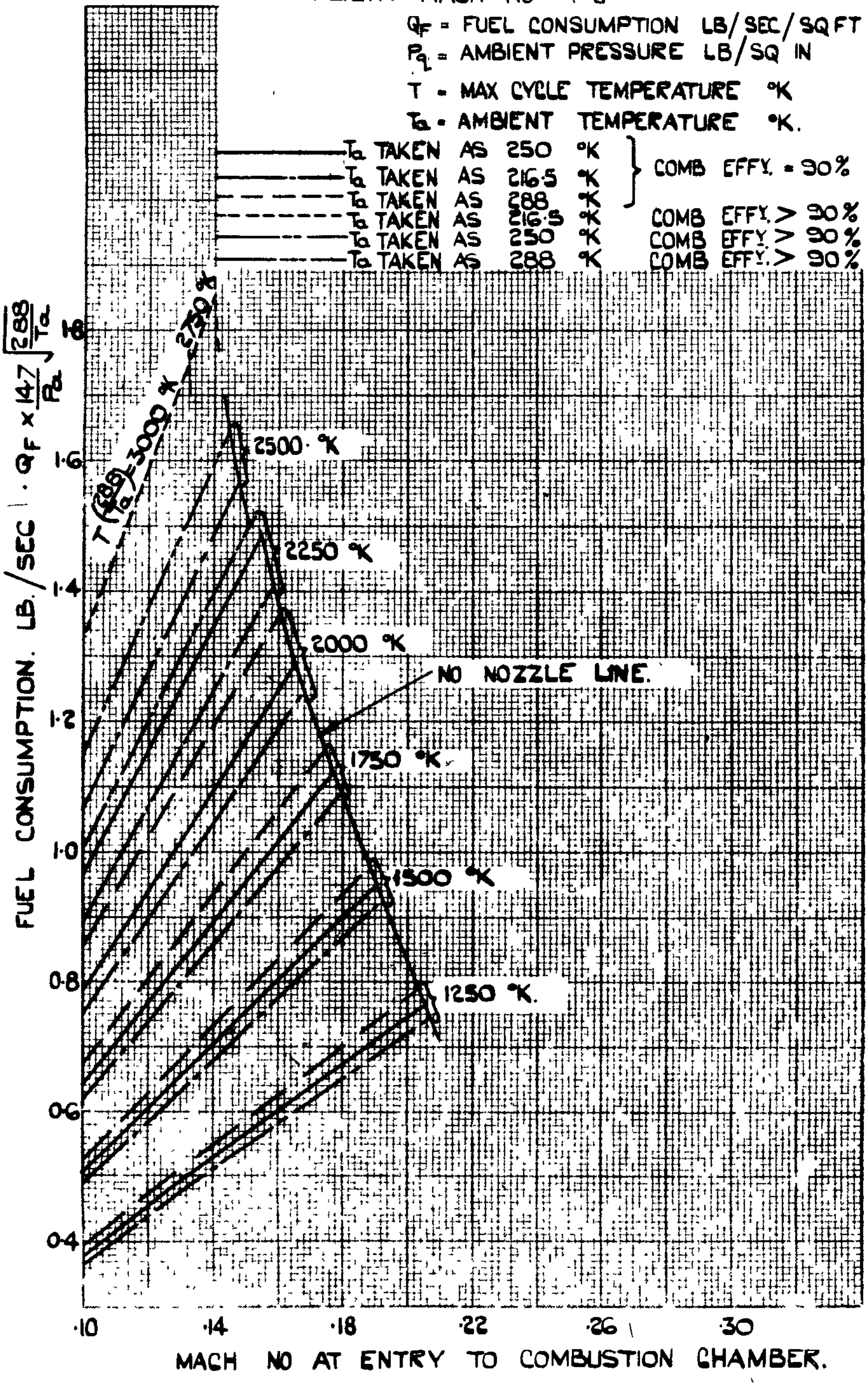


FIG.14.

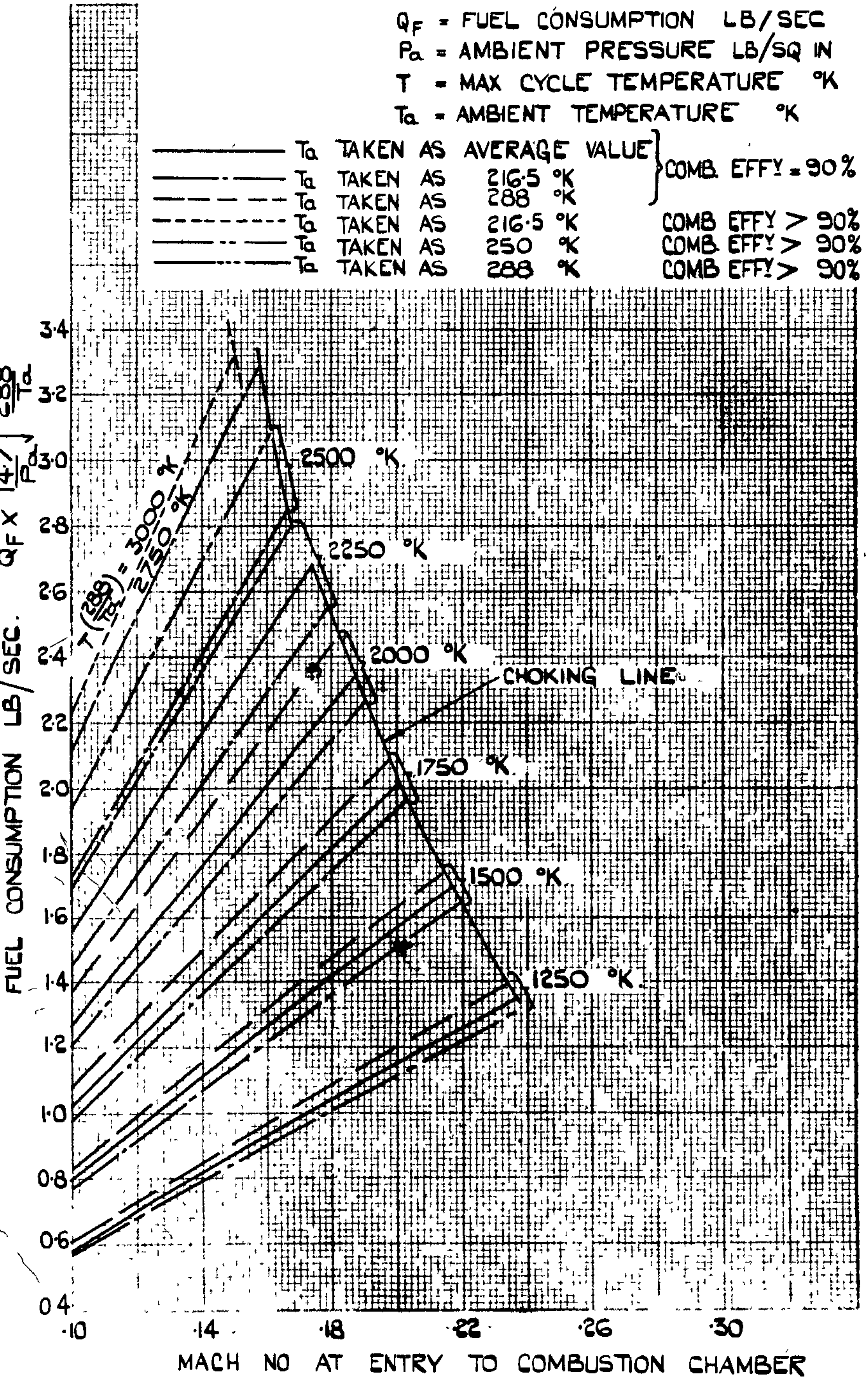
DESIGN PERFORMANCE OF RAM JETS.

FUEL CONSUMPTION

PER SQ FT OF MAX CROSS-SECTIONAL AREA.

FLIGHT MACH NO = 1·5

Q_F = FUEL CONSUMPTION LB/SEC
P_a = AMBIENT PRESSURE LB/SQ IN
T = MAX CYCLE TEMPERATURE °K
T_a = AMBIENT TEMPERATURE °K



DESIGN PERFORMANCE OF RAM JETS.

FIG. 15.

FUEL CONSUMPTION.

PER. SQ FT OF MAX CROSS-SECTIONAL AREA

FLIGHT MACH. NO = 2·0.

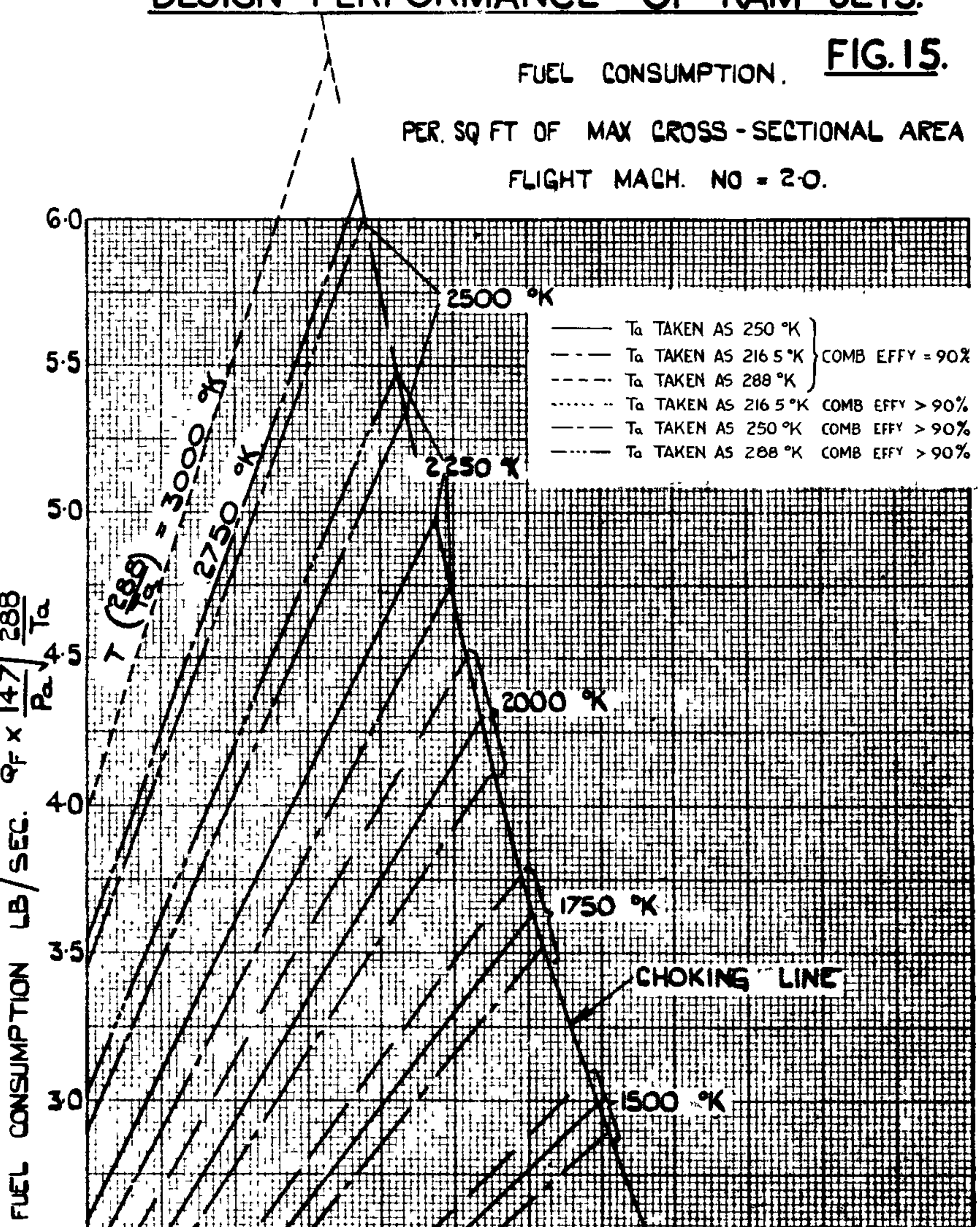


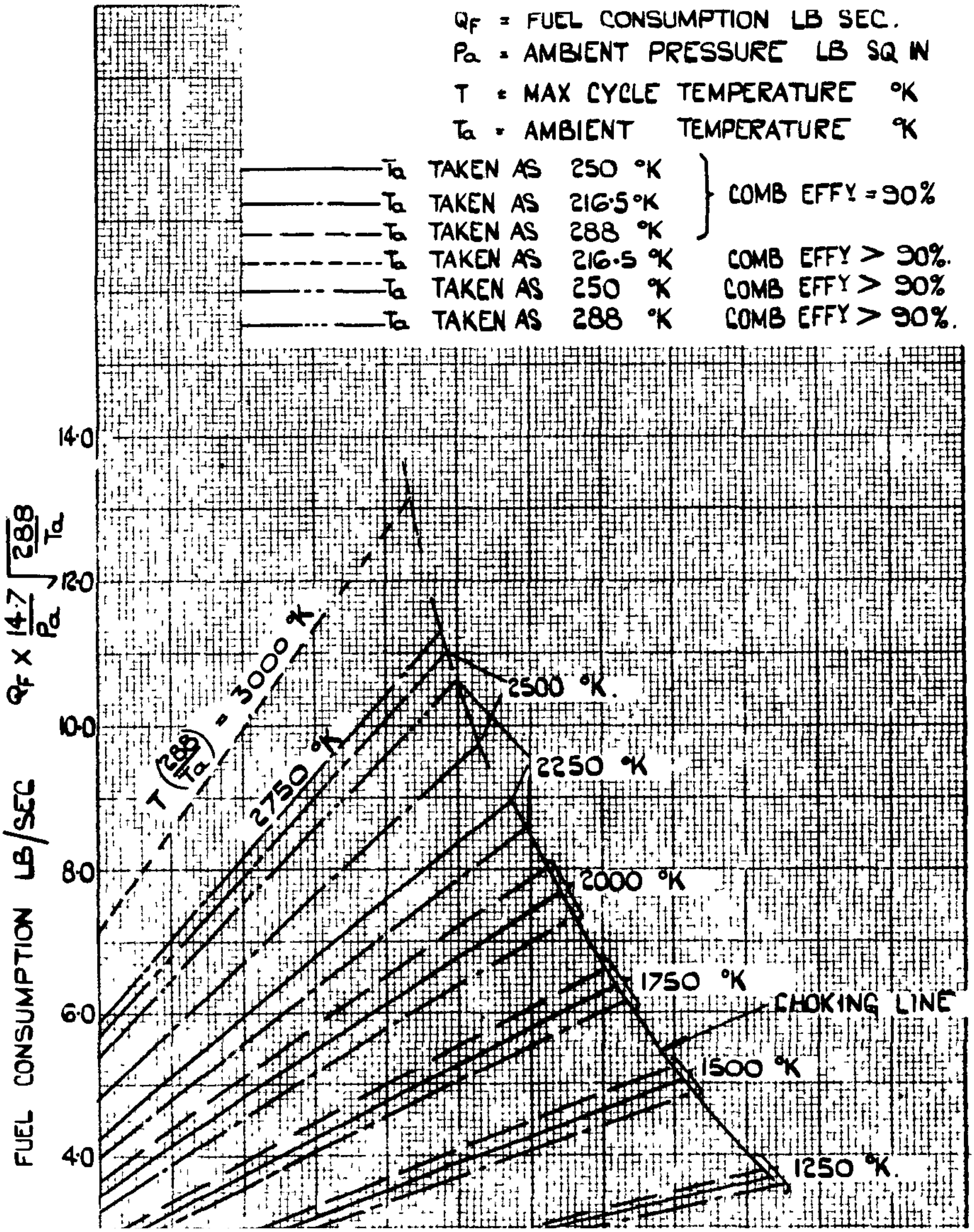
FIG. 16.

DESIGN PERFORMANCE OF RAM JETS.

FUEL CONSUMPTION.

PER. SQ FT OF MAX CROSS-SECTIONAL AREA

FLIGHT MACH NO = 2·5



DESIGN PERFORMANCE OF RAM JETS.

FUEL CONSUMPTION.

PER SQ FT OF MAX CROSS-SECTIONAL AREA

FLIGHT MACH NO = 3·0

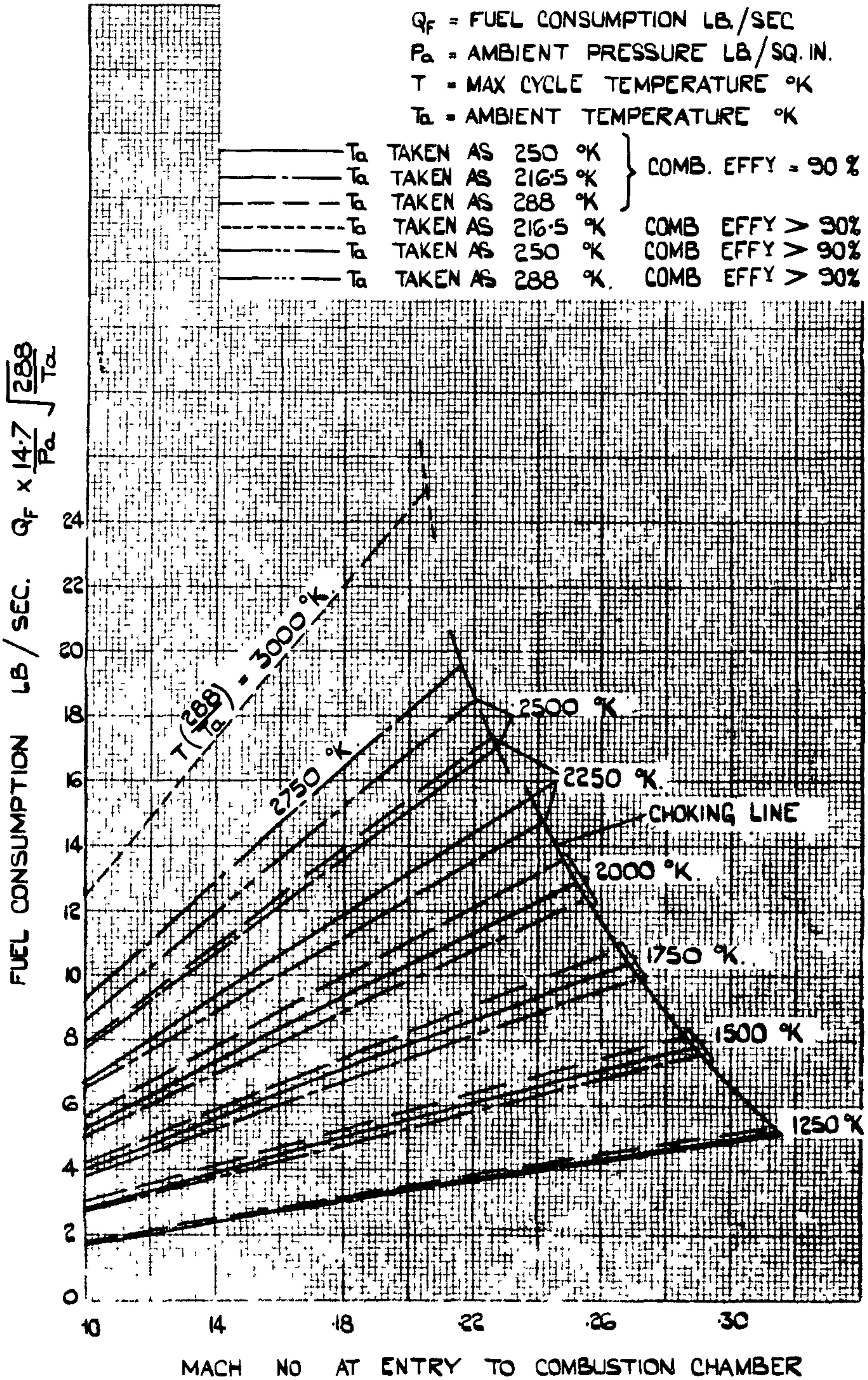


FIG. 18.

DESIGN PERFORMANCE OF RAM JETS.

SPECIFIC FUEL CONSUMPTION.

FLIGHT MACH NO = 1·0

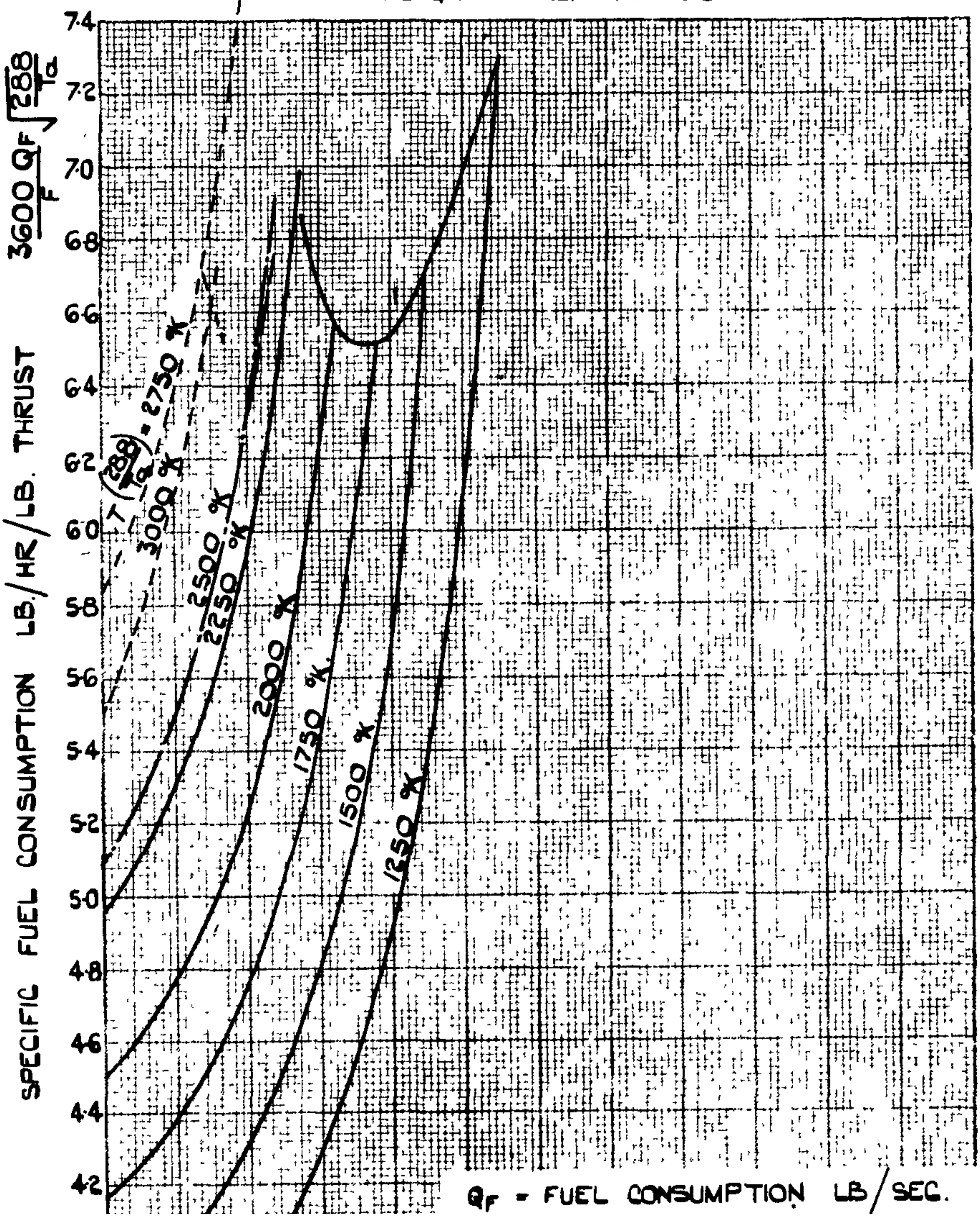


FIG. 19.

DESIGN PERFORMANCE OF RAM JETS.

SPECIFIC FUEL CONSUMPTION

FLIGHT MACH. NO = 1·5

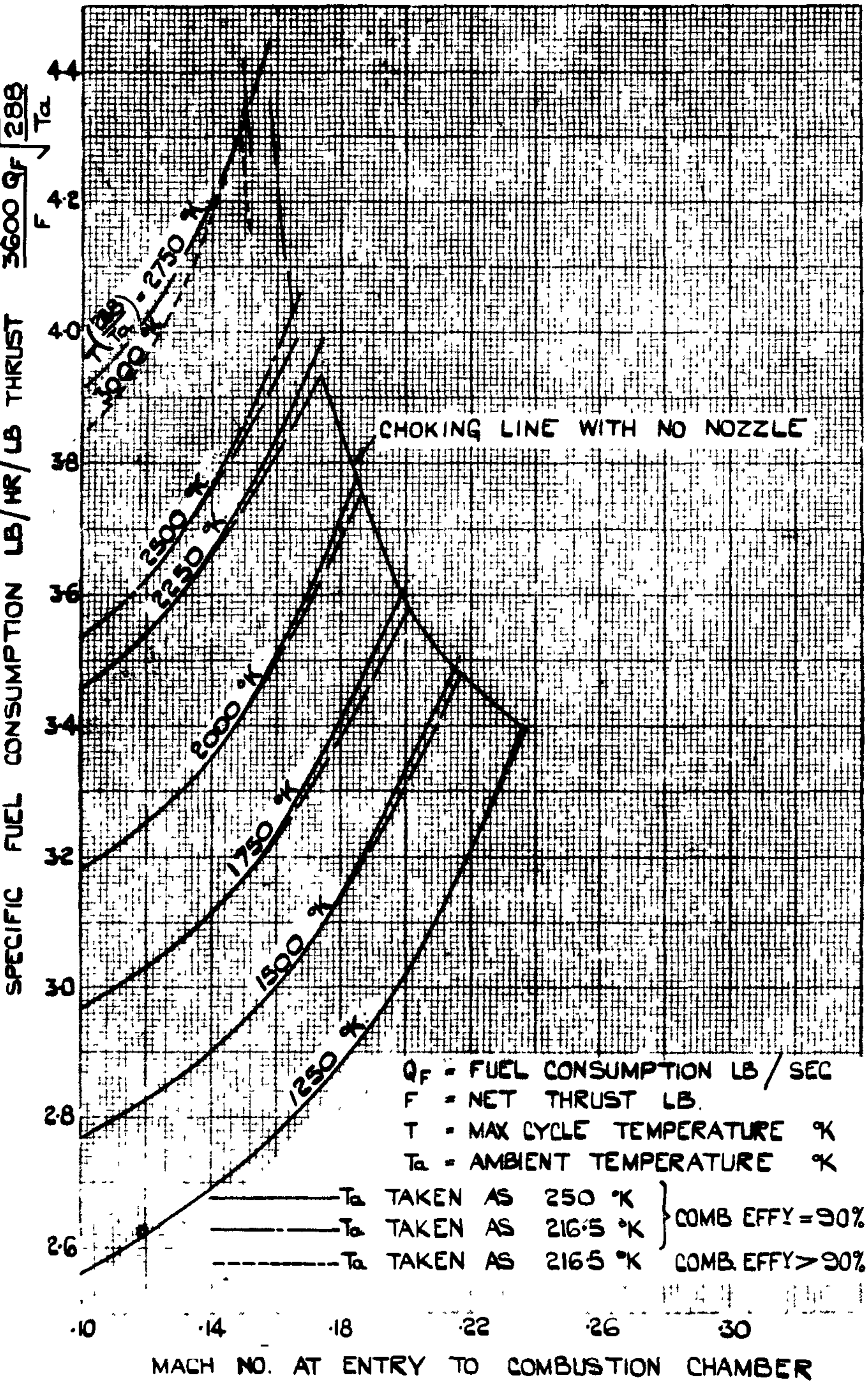


FIG. 20

DESIGN PERFORMANCE OF RAM JETS.

SPECIFIC FUEL CONSUMPTION.

FLIGHT MACH NO = 2·0

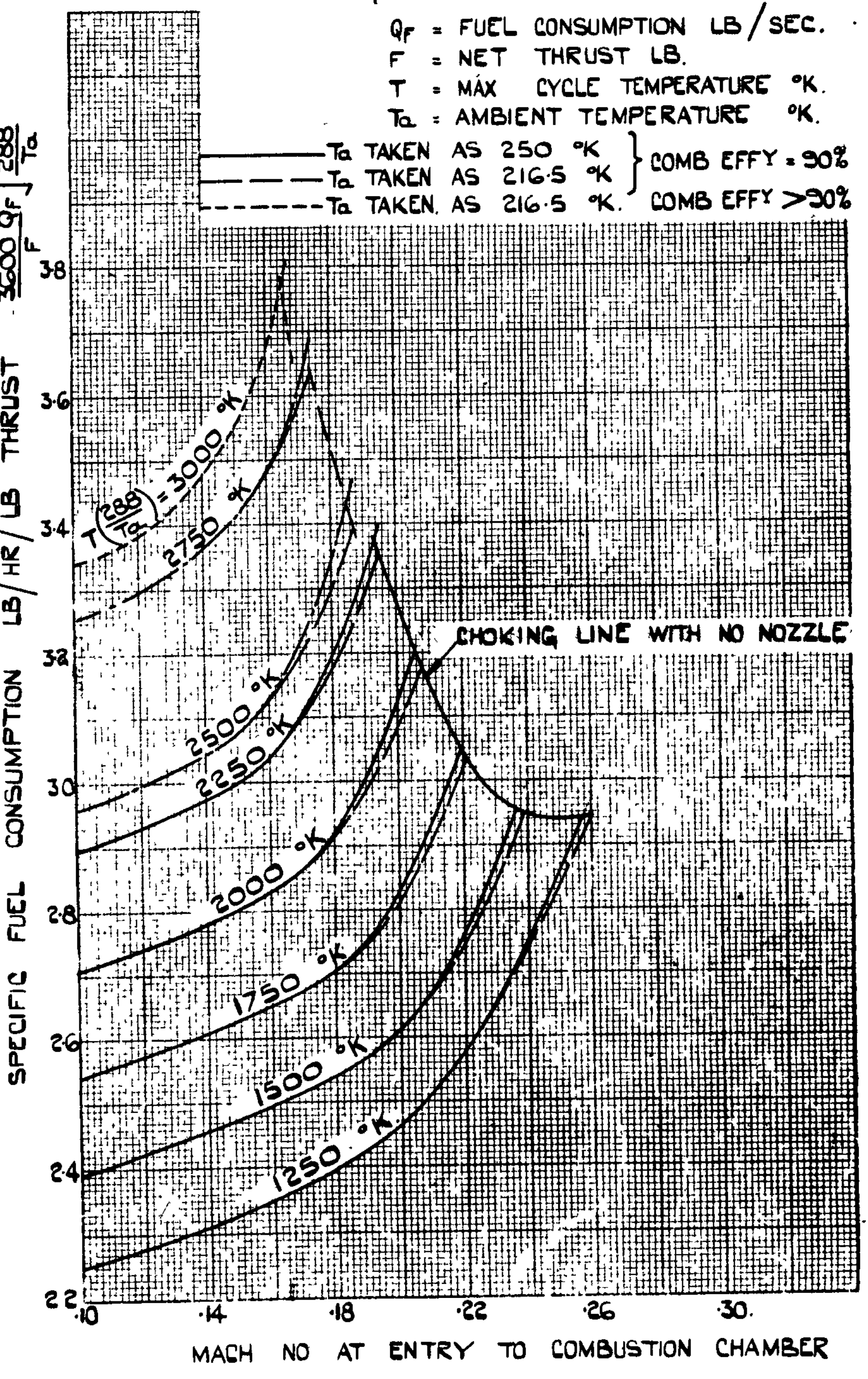
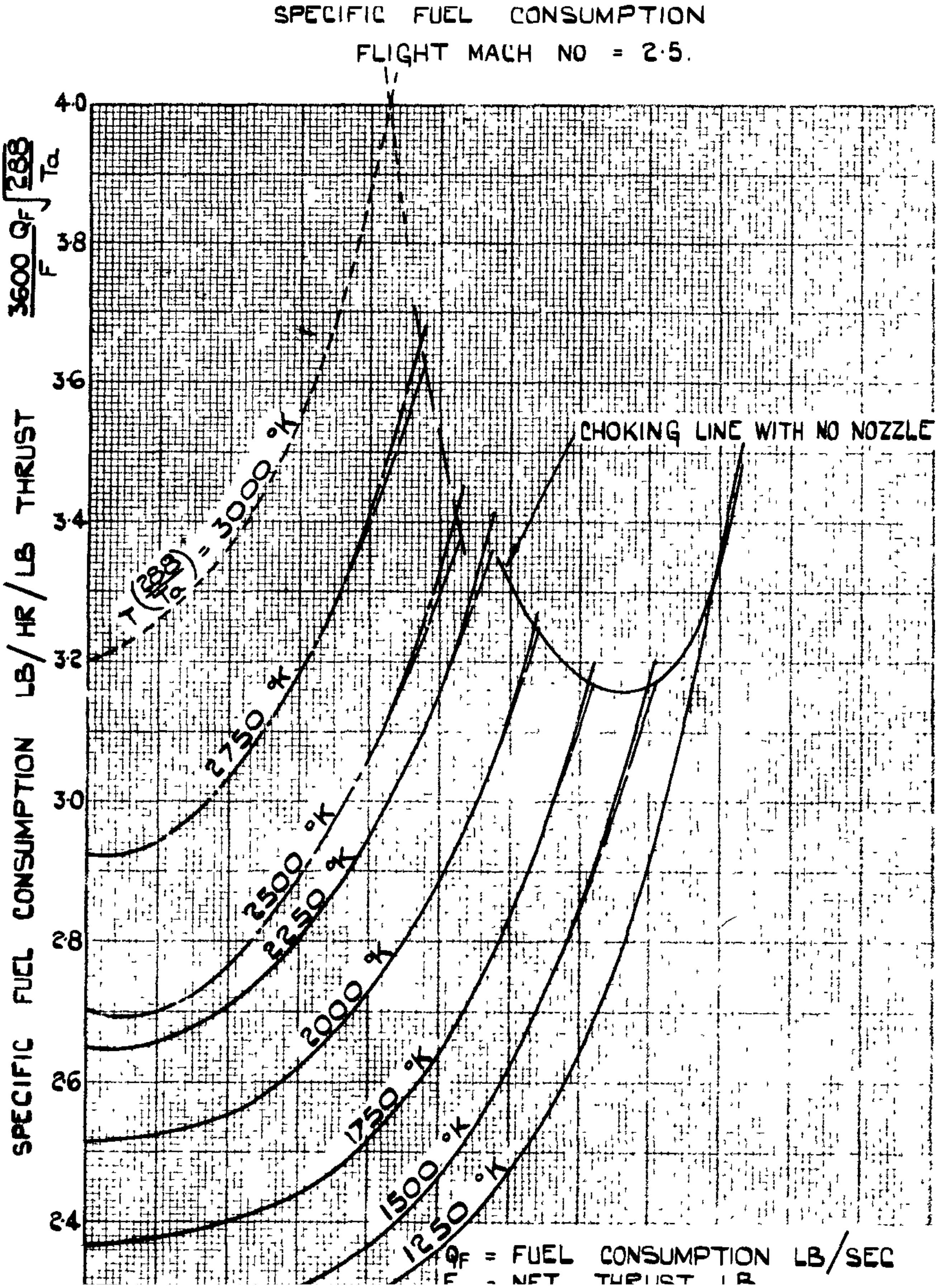


FIG. 21.

DESIGN PERFORMANCE OF RAM JETS.



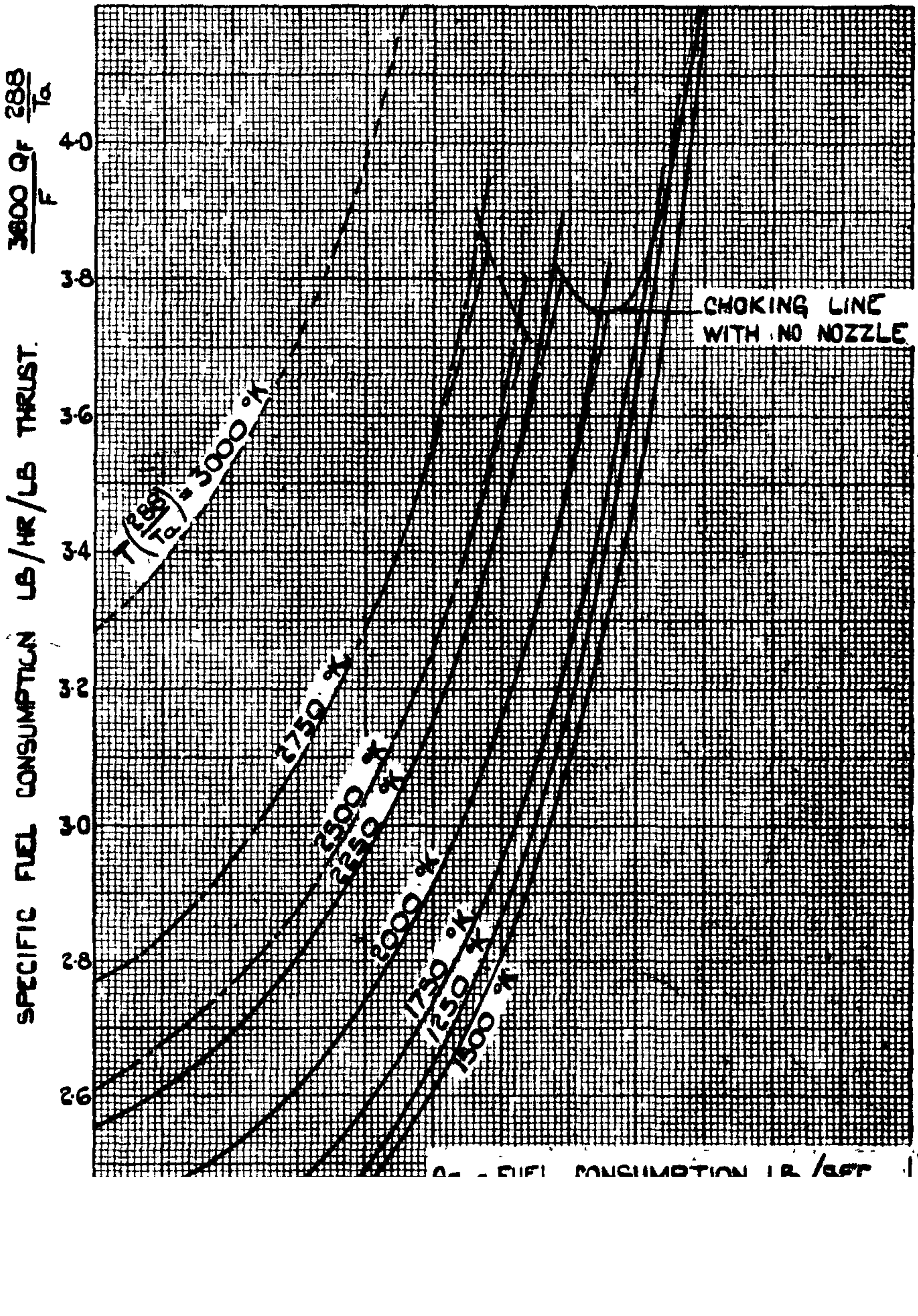
F = NET THRUST LB

FIG. 22.

DESIGN PERFORMANCE OF RAM JETS.

SPECIFIC FUEL CONSUMPTION.

FLIGHT MACH NO = 3·0

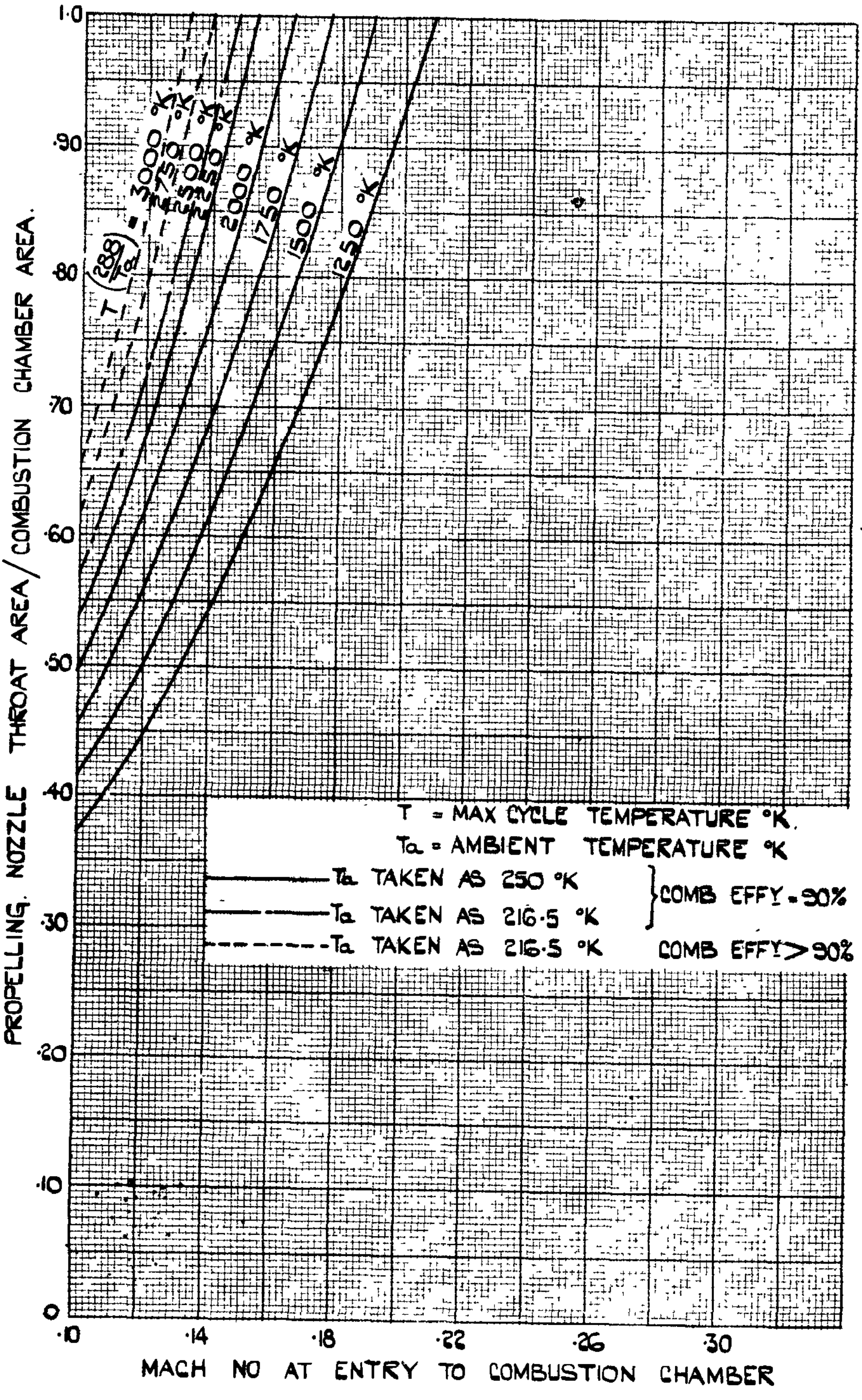


Q_F = FUEL CONSUMPTION LB./SEC

DESIGN PERFORMANCE OF RAM JETS.

THROAT AREA

FLIGHT MACH NO = 1·0



DESIGN PERFORMANCE OF RAM JETS.

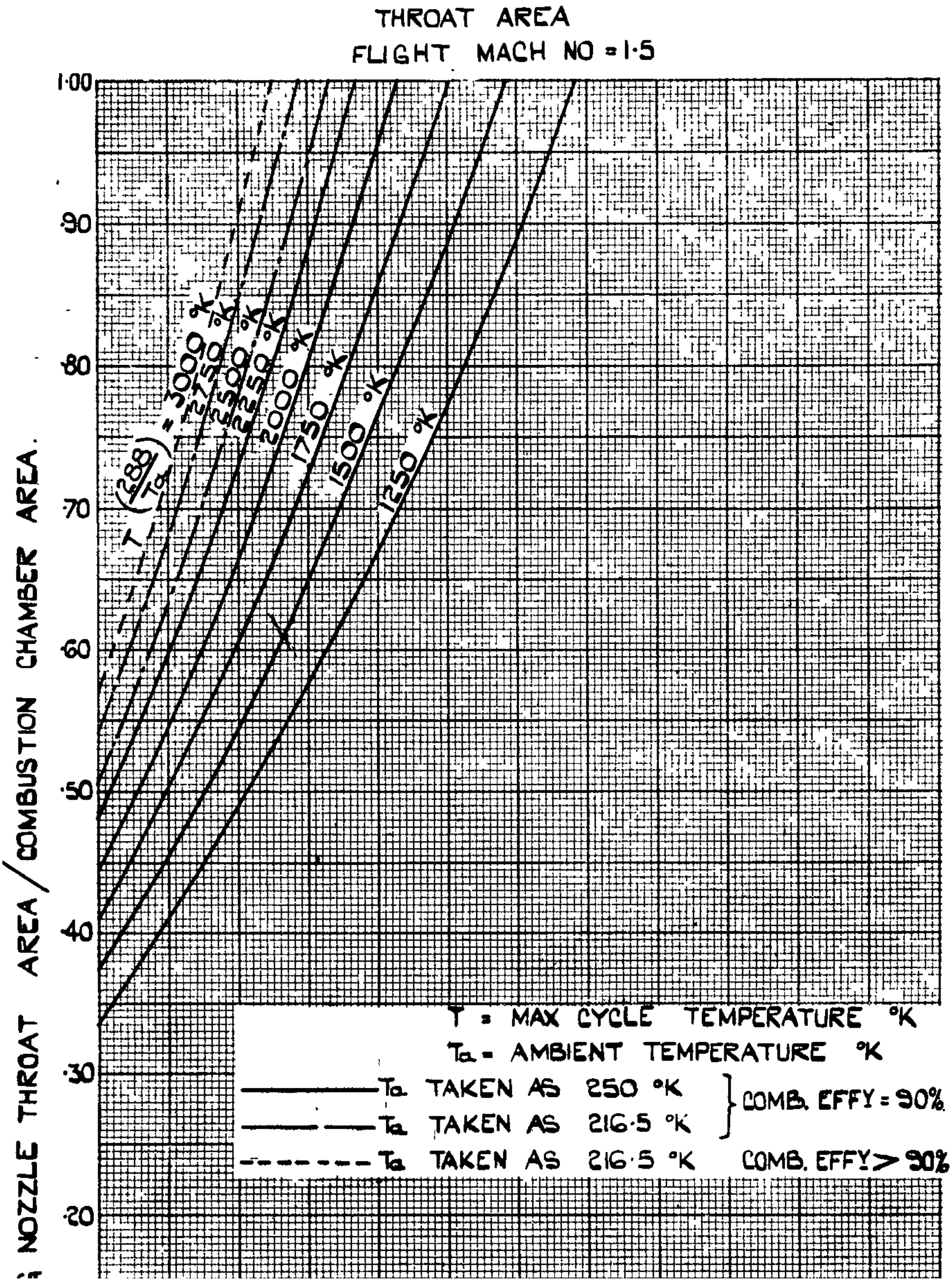
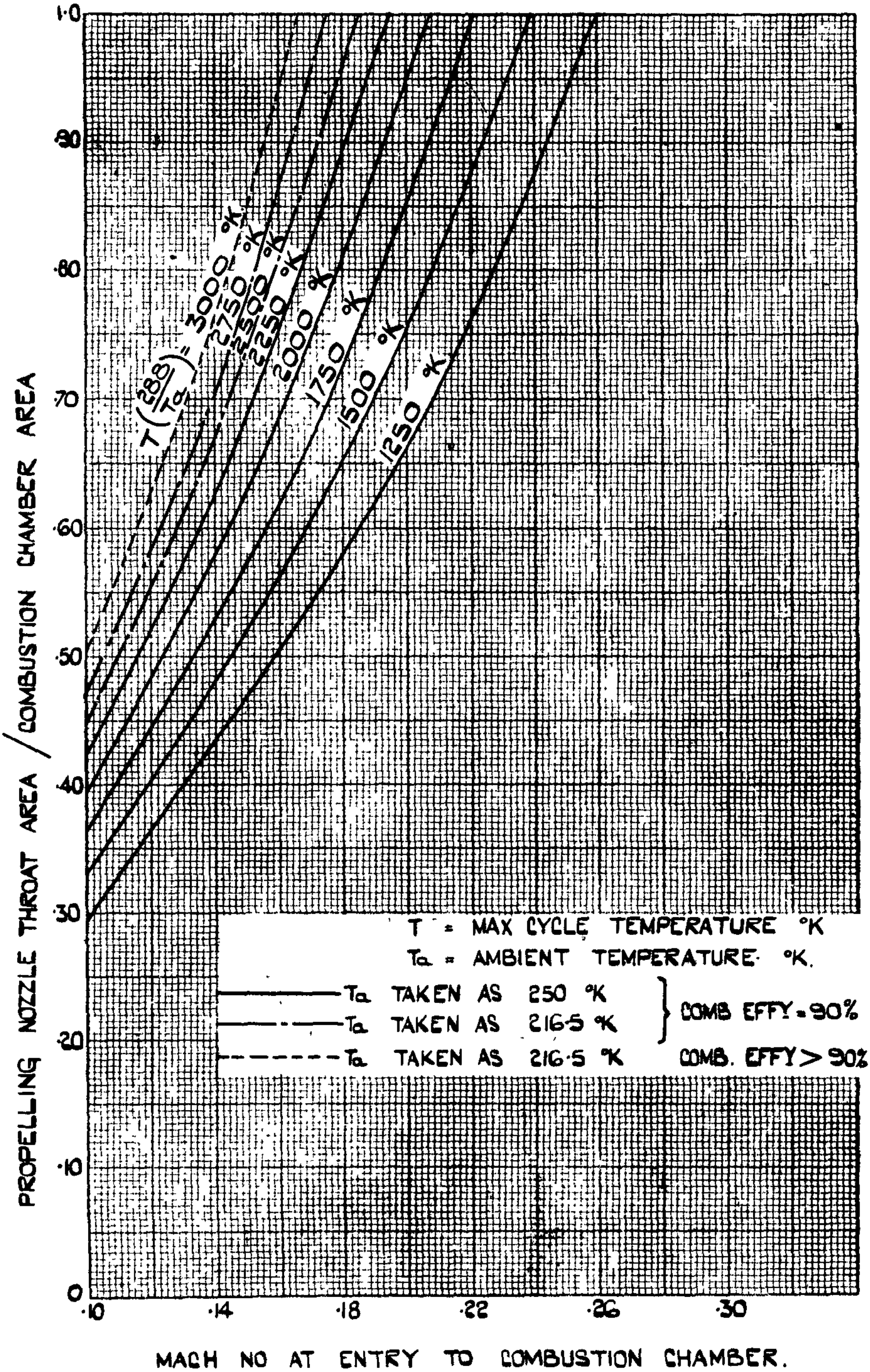


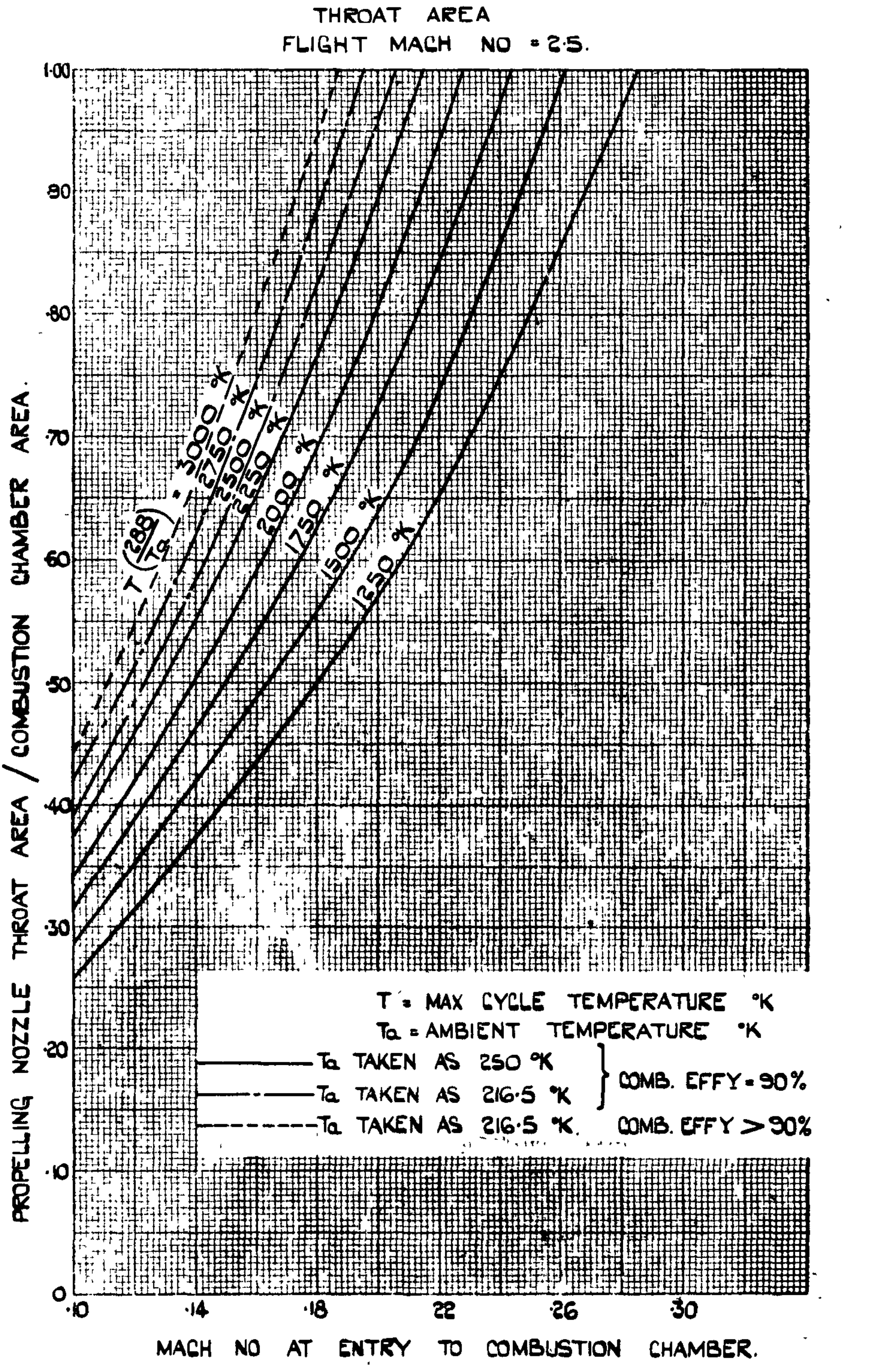
FIG.

DESIGN PERFORMANCE OF RAM JETS.

THROAT AREA
FLIGHT MACH. NO = 2·0



DESIGN PERFORMANCE OF RAM JETS.



DESIGN PERFORMANCE OF RAM JETS.

THROAT AREA.

FLIGHT MACH NO = 3·0

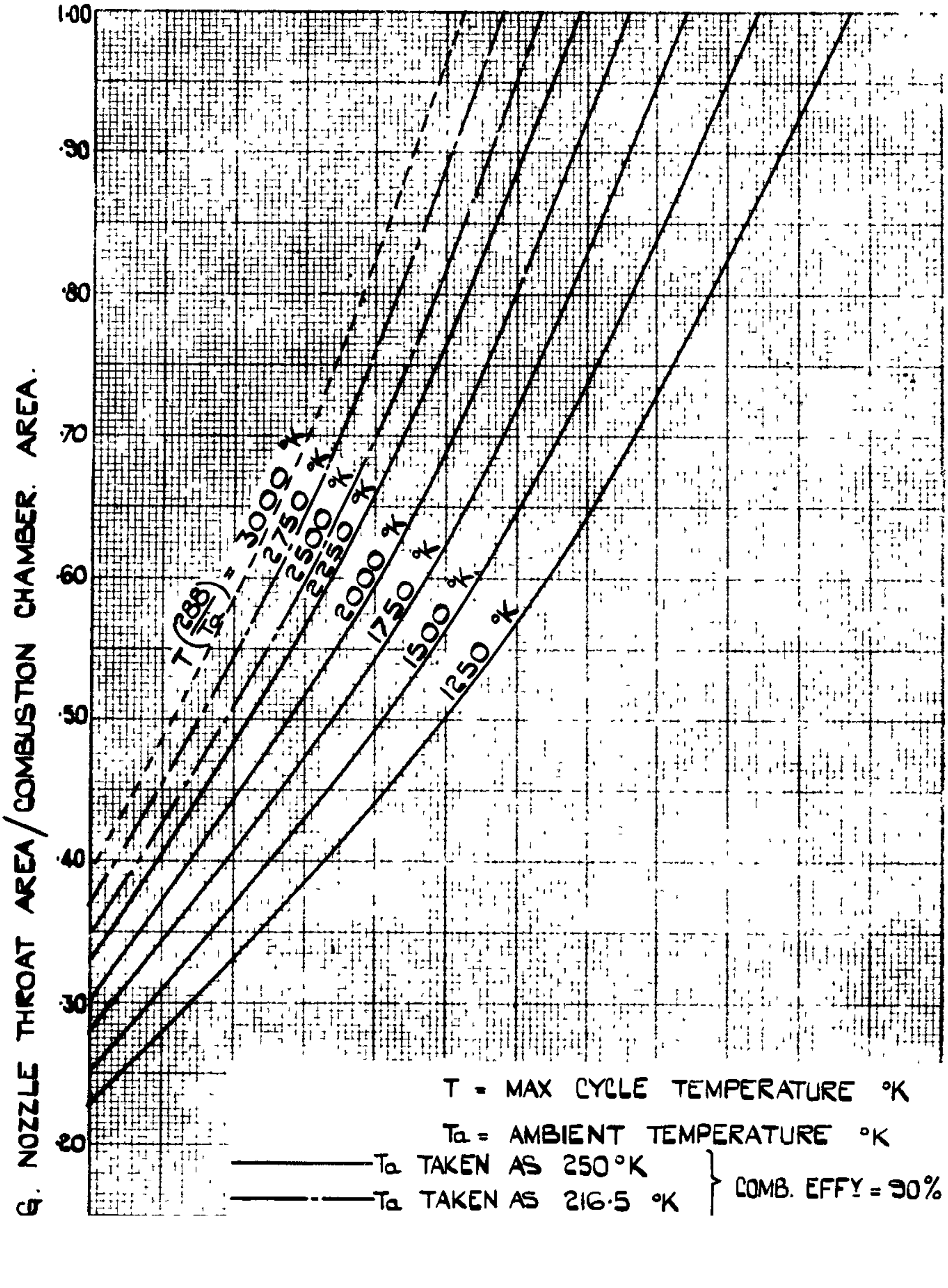


FIG. 28.

COMPARISON OF PERFORMANCE OF RAM JET HAVING OBLIQUE OR NORMAL SHOCK INTAKE.

THRUST COEFFICIENT V FLIGHT MACH Nº

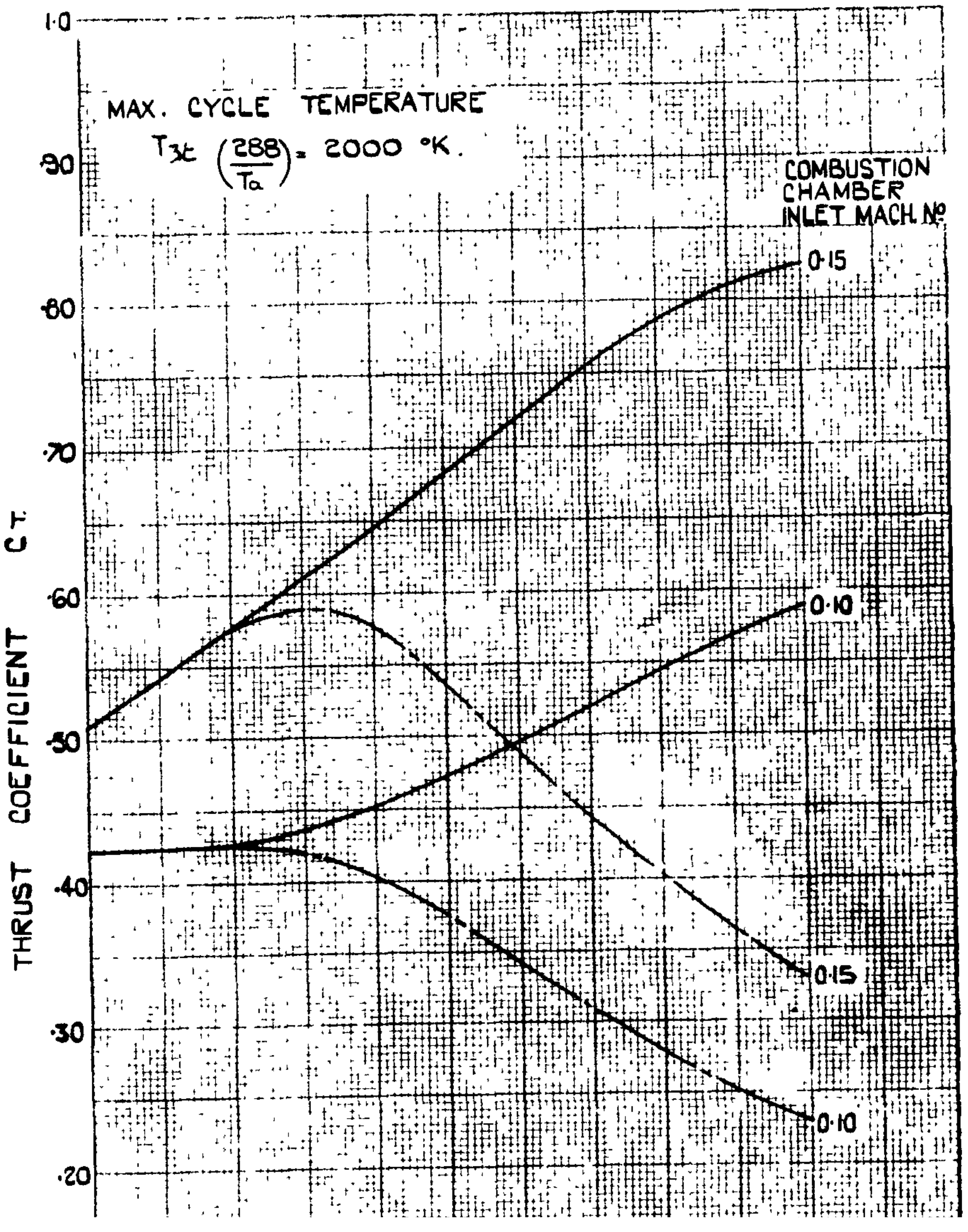
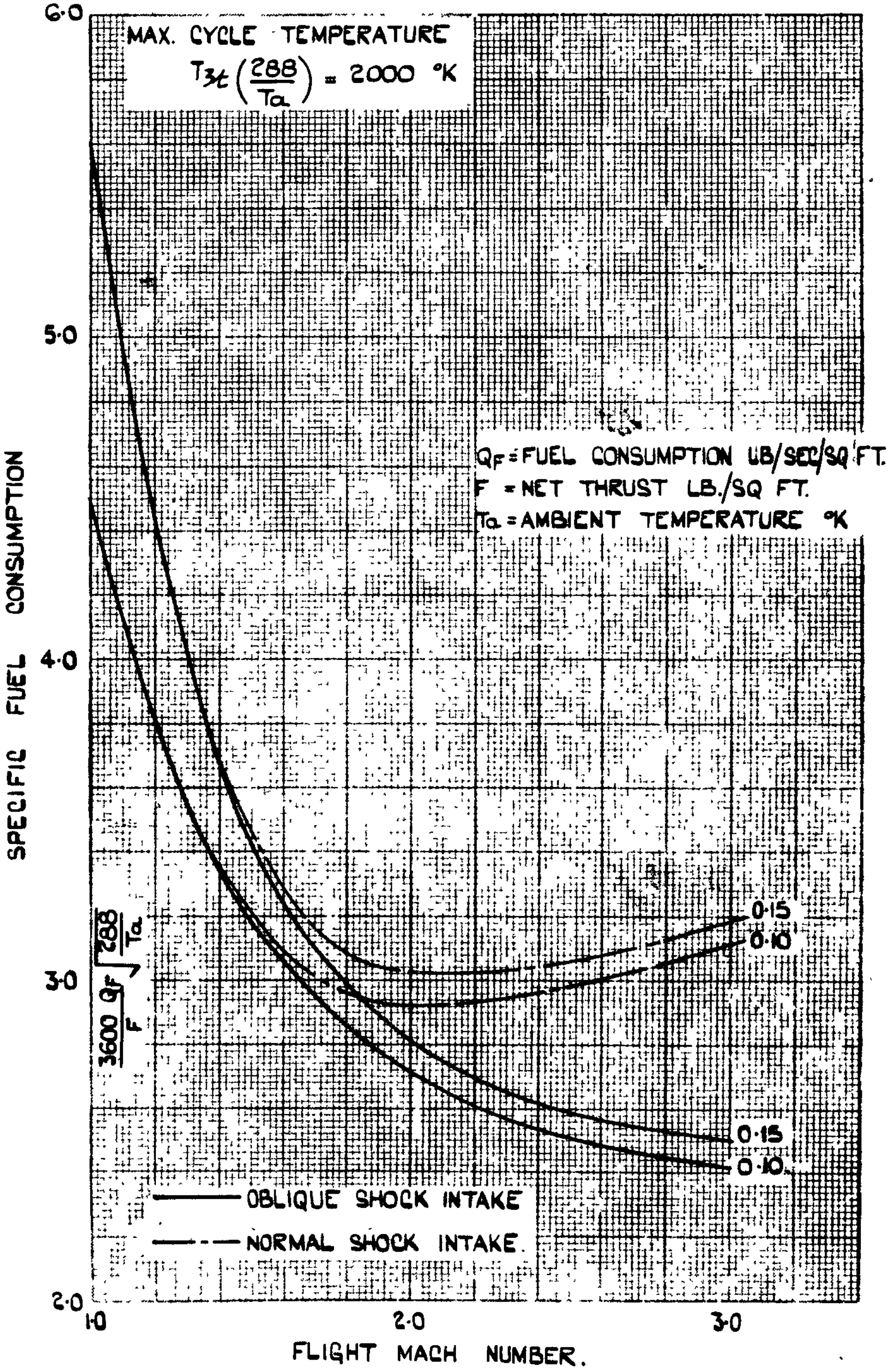


FIG. 29.

COMPARISON OF PERFORMANCE OF RAM JET HAVING OBLIQUE OR NORMAL SHOCK INTAKE.

SPECIFIC FUEL CONSUMPTION V. FLIGHT MACH Nº



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