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WADC TECHNICAL REPORT 53-119

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A METHOD OF PREDICTING SKIN, COMPARTMENT, AND EQUIPMENT TEMPERATURES FOR AIRCRAFT

LAWRENCE SLOTE
WILLIAM D. MURRAY

NEW YORK UNIVERSITY

JULY 1953

WRIGHT AIR DEVELOPMENT CENTER

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4 Oct 54

WADC TECHNICAL REPORT 53-119 July 1953

ERRATA - September 1954

The following corrections are applicable to WADC Technical Report 53-119, "A Method of Predicting Skin, Compartment, and Equipment Temperature for Aircraft", dated July 1953:

Page vii

Change to read:

Pr	Prandtl Number	$3600 \frac{C_p \mu g}{K}$	dimensionless

Add:

P^*	Ambient Pressure	$\frac{P}{P_\infty}$	atmospheres

Page viii

Change to read:

ρ	density of air	$\frac{slug}{ft^3}$ or $\frac{\# \, sec^2}{ft^4}$

Page 17

Change Equation (19) to read:

$$\frac{h_f D}{K_f} = 0.53 \left[\left(\frac{D^3 \rho_f{}^2 \beta g \Delta T}{\mu_f{}^2} \right) \left(\frac{3600 \, C_p \mu_f g}{K_f} \right) \right]^{1/4}$$

Change to read:

$$\rho_f = \frac{P_\infty}{R T_f} P^*$$

$$\alpha = \frac{g^2 \beta \, C_p \, P_\infty{}^2}{3600 \, \mu_f K_f R^2 T_f{}^2}$$

Substitute P^* for P in equations (20) and (21).

Wright Air Development Center
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United States Air Force
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ERRATA (Continued)

Page 18 Equations 23 and 25

Delete $(P)^{1/2}$, insert $(P^*)^{1/2}$

Page 29 Steps 4 and 5

Delete 81.8, insert 272.7

Page 30

Phase IV - Step 2 change to read:

$$A_c = 2 \frac{A_c \ A_q}{A_c + A_q} = 2 \times \frac{5.4 \times .392}{5.792} = .732 \text{ ft}^2$$

Step 4: Delete P, insert P*

Delete $\sqrt[4]{\frac{P^2}{D}}$, insert $\sqrt[4]{\frac{P^{*2}}{D}}$

Step 5:

Page 32

Delete $\sqrt[4]{\frac{P^2}{D}}$, insert $\sqrt[4]{\frac{P^{*2}}{D}}$

Page 34

Phase II - Step 2 change to read:
$C = 27{,}000$

Page 35

Phase II - Step 6 change to read: $C = 27{,}000$

Step 7 change to read: $C = 27{,}000$

Step 8 change to read: $T_s = 425°R$

Phase III - Step 2 change to read:

$$6.7 = \frac{(T_{si} - 425)}{\frac{.0075}{.45} + \frac{.025}{.0425}}$$

$T_{si} = 436 \ °R$

ERRATA (Continued)

Page 36

Step 4 **change to read:**

Delete P, **insert** p^*

Delete $\sqrt[4]{\frac{P^2}{D}}$, **insert** $\sqrt[4]{\frac{P*^2}{D}}$

Step 6 **change to read:**

$T_q = 468°R$

Page 37

Step 9 change to read: "__________ value of $T_q = 468°R$ is correct."

Phase V -

Step 1 **change to read:** $T_q = 468°R$

Step 5: Delete $\sqrt[4]{\frac{P^2}{D}}$, **insert** $\sqrt[4]{\frac{P*^2}{D}}$

Change to read: $T_q = 468°R$

Page 38

Change to read:

"where $T_q = 468°R$ and _____".

Delete $\sqrt[4]{\frac{P^2}{D}}$, insert $\sqrt[4]{\frac{P*^2}{D}}$

Step 6 **change to read:** $T_c = 441°R$

Pages 58 and 59. Figures 15 and 16

Delete $\left(\frac{P^2}{D}\right)^{1/4}$, **add:** $\left(\frac{P*^2}{D}\right)^{1/4}$

WADC TECHNICAL REPORT 53-119

A METHOD OF PREDICTING SKIN, COMPARTMENT, AND EQUIPMENT TEMPERATURES FOR AIRCRAFT

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William D. Murray

New York University

July 1953

Environmental Criterial Branch
Systems Planning Office
Directorate of Air Weapon Systems
Contract No. AF 33(616)-122
RDO No. 560-76

Wright Air Development Center
Air Research and Development Command
United States Air Force
Wright-Patterson Air Force Base, Ohio

FOREWORD

This report was prepared by the Research Division of the College of Engineering, New York University under Contract AF 33(616)-122. It presents a method for the calculation of the equilibirum surface temperature, compartment air temperature and equipment temperature of bodies in high and low speed flight and contains examples of the temperatures to be expected in typical aircraft section.

The contract was initiated under a project identified by the Research and Development Order 560-76, title unclassified, "Study - Design Temperature Requirements for Operation of USAF Aircraft and Equipment", and was administered by the Environmental Criteria Branch, Systems Planning Office, Directorate of Air Weapon Systems, Wright Air Development Center, with James G. Law as Project Engineer.

ABSTRACT

This report presents a method for the calculation of the equilibrium skin temperature of aerodynamic shapes in steady flight. Graphical methods are presented for the calculation of equipment temperature and compartment air temperature. The numerical and graphical solutions are presented for aircraft flying at speeds to Mach number 5 and for altitudes from 0 to 100,000 feet in the proposed USAF Hot and Cold Atmospheres.

In this report an attempt is made toward simplification of the empirical formulae by graphic presentation of calculations upon typical aerodynamic shapes. Analysis is based upon consideration of a flat plate and the case of an isothermal surface and constant free stream velocity. Methods for applying the graphs to other surfaces are presented.

PUBLICATION REVIEW

FOR THE COMMANDER:

H A Boushey

H. A. BOUSHEY
Colonel, USAF
Director of Air Weapon Systems

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LIST OF ILLUSTRATIONS

NOMENCLATURE

a	speed of sound	ft/sec
A_t	total area (surface)	ft^2
A_c	surface area of compartment	ft^2
A_q	surface area of equipment	ft^2
A_e	effective area $= 2 \frac{A_c A_q}{A_c + A_q}$	ft^2
A_p	projected area	ft^2
c	specific heat of skin	BTU/lb °R
c_p	specific heat of air at constant pressure	BTU/lb °R
D	characteristic dimension	ft
g	acceleration due to gravity	32.2 ft/sec^2
G_a	nocturnal irradiation from above the surface	BTU/hr ft^2
G_b	nocturnal irradiation from below the surface	BTU/hr ft^2
G_β	nocturnal irradiation $= 1/2(G_a + G_b)$	BTU/hr ft^2
G_s	solar irradiation	BTU/hr ft^2
h_s	outer surface coefficient of convective heat transfer	BTU/hr ft^2 °R
h_f	convective (air film) coefficient of heat transfer for single horizontal cylinders	BTU/hr ft^2 °R
h_q	convective (air film) coefficient of heat transfer for single horizontal cylinders $= 1/2\ h_f$	BTU/hr ft^2 °R
J	mechanical equivalent of heat	778 ft lb/BTU
k	thermal conductivity of skin	BTU/hr ft^2 (°R/ft)

k	thermal conductivity of air	BTU/hr ft^2(°R/ft)
L	total distance from leading edge	ft
M	Mach Number $= \frac{V}{a}$	dimensionless
Nu	Nusselt Number $\frac{hx}{k}$	dimensionless
P	ambient pressure	lb/ft^2
Pr	Prandtl Number $\frac{c_p \mu g}{k}$	dimensionless
q_e	equipment heat release	BTU/hr
q_{cv}	heat dissipated by convection	BTU/hr
q_r	heat dissipated by radiation	BTU/hr
r	recovery factor	dimensionless
R	universal gas constant	1711 lb ft/slug °R
Re	Reynold's Number $\frac{V \rho x}{\mu}$	dimensionless
s	slant length on cone	ft
t	time	hours
T	ambient temperature	°R
T_e	adiabatic surface temperature	°R
T_c	compartment air temperature	°R
T_q	equipment surface temperature	°R
T_s	outer equilibrium skin temperature	°R
T_{si}	inner equilibrium skin temperature	°R
ΔT	temperature difference	°R
T'	reference temperature	°R
V	velocity	ft/sec

W	weight	lbs
w	specific weight of skin material	lb/ft^3
x	local distance from leading edge	ft
β =	coefficient of thermal expansion $= \frac{1}{T}$,	$\frac{1}{°R}$
γ	ratio of specific heats = 1.4	dimensionless
α_β	nocturnal absorbtivity	dimensionless
α_s	solar absorbtivity	dimensionless
$\epsilon = \dfrac{\epsilon_1 \epsilon_2}{\epsilon_1 + \epsilon_2 - \epsilon_1 \epsilon_2}$	mean emissivity,	dimensionless
ϵ_s	emissivity of outer surface	dimensionless
ϵ_1	emissivity of hot surface	dimensionless
ϵ_2	emissivity of cold surface	dimensionless
σ	Stefan-Boltzmann constant 17.3×10^{-10}	BTU/hr ft^2 $°R^4$
ρ	density of air	lb/ft^3
τ	skin thickness	ft
μ	absolute viscosity	lb sec/ft^2
θ_s	1/2 vertex angle	degrees

no	subscript denotes local condition
o	subscript denotes free stream condition
∞	subscript denotes sea level condition
x	subscript denotes local or point value
m	subscript denotes mean value
f	subscript refers to air film

INTRODUCTION

As actual and proposed speeds and altitudes of flight increase, the problem of maintaining a satisfactory environment for airborne equipment becomes increasingly difficult.

The skin and internal temperatures of an aircraft travelling through the atmosphere at a high Mach number may become excessive unless a sufficient amount of cooling is supplied. Conversely, in low speed aircraft operating under extremely low temperature conditions it is essential that artificially produced heat maintain the temperature above a minimum design value (presently -80°F non-operating, -65°F operating). Calculations of skin and equipment temperatures are therefore essential to the development of high altitude aircraft and to the problem of controlling the actual temperature of airborne equipment to a value that will provide satisfactory life.

This report is the result of an investigation to determine by literary search and computations those properties of the atmosphere which affect the heating and cooling of a body in motion at various altitudes. This information is used to determine stabilized skin temperatures, equipment temperatures, and typical compartment air temperatures.

This report examines the problem of aerodynamic heating of typical aerodynamic shapes in steady subsonic and supersonic flight to determine the value and significance of the pertinent parameters with the aim of using these parameters to calculate the temperatures that can be expected as a function of altitude and speed. An attempt is made toward the simplification of empirical formulae by graphic presentation of calculations. In our analysis we shall consider two basic shapes, planes and cones. These are of interest for aircraft for the following reasons.

Generally, much of the body of an aircraft is cylindrical. Since the body diameter is very large compared to the boundary layer thickness, conditions on the surface of this cylinder are similar to those on a flat plane, except for the effects of the distrubance produced farther upstream by the nose of the aircraft.

The nose of the aircraft may be conical or ogival. In the latter case the nose may be considered a series of truncated cones, so procedures used in dealing with cones are applicable to an ogival nose to a reasonable degree of accuracy.

The usual assumptions made in analyzing boundary layers are applicable here. The system considered is one in which the temperature over any surface considered is assumed to be uniform over the entire surface. The boundary layer thickness is negligible compared to the length of the surface. It is further assumed that boundary layer conditions which exist with subsonic velocities are not affected by increase of speed to supersonic values, except for effects on boundary layer air which are considered in this report. Also, the system considered is one which has a boundary layer under zero pressure gradient beginning from some known point with zero thickness.

With this concept in mind appropriate general aerodynamic parameters may be established for the determination of equilibrium skin temperatures. Although a determination of the accuracy of temperatures obtained by methods presented herein can only be made after checking the results of an experimental investigation, it is estimated that the following maximum errors will result from these graphical methods: $T_e \pm 5\%$, $|(T_s - T_e)| \pm 10\%$, $|(T_q - T_s)| \pm 20\%$, $|(T_c - T_q)| \pm 20\%$. Accuracy will be discussed in more detail later in the report.

A METHOD OF PREDICTING SKIN, COMPARTMENT, AND EQUIPMENT TEMPERATURES FOR AIRCRAFT

Analysis

Heat Balance Equation

If the initial conditions and the specified flight trajectory are known, skin temperature can be evaluated as a function of time by means of the following differential equation:

$$A_t c \gamma w \frac{dT_s}{dt} = h_s A_t (T_e - T_s) + A_t \alpha_\beta G_\beta + A_p \alpha_s G_s + q_e - \epsilon_s \sigma A_t (T_s)^4 \qquad (1)$$

The equilibrium skin temperature T_s, is the skin temperature which will be reached after a sufficient lapse of time under steady flight conditions at constant altitude, 1/. Equation (1) defines the value of T_s, since in the limit as $t \rightarrow \infty$, $\frac{dT_s}{dt} \rightarrow 0$ and Equation (1) reduces to

$$h_s A_t (T_e - T_s) + A_t \alpha_\beta G_\beta + A_p \alpha_s G_s + q_e - \epsilon_s \sigma A_t (T_s)^4 = 0 \qquad (2)$$

or

$$h_s (T_e - T_s) + \left(\alpha_\beta G_\beta + \frac{A_p}{A_t} \alpha_s G_s\right) + \frac{q_e}{A_t} - \epsilon_s \sigma (T_s)^4 = 0 \qquad (3)$$

The thermal balance of a given area of the skin is the resultant of the following sources of heat flow:

1. Heat gained by the body due to aerodynamic heating, $h_s(T_e - T_s)$
2. Heat gained by the body due to solar and atmospheric irradiation, $\left(\alpha_\beta G_\beta + \frac{A_p}{A_t} \alpha_s G_s\right)$
3. Heat release of the equipment, $\frac{q_e}{A_t}$

4. Heat lost by the body due to radiation to outer space, $-\epsilon_s \sigma (T_s)^4$

Adiabatic Surface Temperature, T_e

In order to calculate equilibrium surface temperatures, adiabatic surface temperatures are first computed from the following equation:

$$T_e = T + \frac{r V^2}{2 g J c_p} \tag{4}$$

or

$$T_e = T \left(1 + \frac{\gamma - 1}{2} r M^2\right) \tag{5}$$

If a fluid is brought from a state of motion to a state of rest by friction, as occurs at the surface of a flat plate oriented parallel to the air stream, the temperature of the fluid is raised to a value different from that obtained by bringing the fluid to rest adiabatically. The ratio of the temperature rise due to friction to the temperature rise due to adiabatic compressure is known as the recovery factor. 2/, 3/, 4/. The value of the recovery factor depends on whether the flow is laminar or turbulent. For laminar flow, the recovery factor, r, can be taken as $(Pr)^{1/2}$ and for turbulent flow, the recovery factor, r, can be taken as $(Pr)^{1/3}$.

In this study, the value of the recovery factor, r, is taken to be 0.87 which is an average value obtained from existing information on subsonic and supersonic flow, both laminar and turbulent, on flat plates and cones. γ, the ratio of specific heats for air, is 1.4. Equation (5) then becomes:

$$T_e = T (1 + 0.174 M^2) \tag{5a}$$

Equation (5a) is presented in graphical form as Figure 4.

Temperature for Evaluating Air Properties, T'

6/, 7/ The reference temperature T' is obtained from Crocco's solution of the boundary layer equations covering Mach numbers 0 to 5. It is the temperature at which the physical properties of the air are evaluated so as to eliminate the effect of small variations in the variables Pr, M and T_s.

$$T' = 0.42\,T\,(1 + .076\,M^2) + 0.58\,T_s \qquad (6)$$

Since Equation (6) contains the term T_s which is the equilibrium surface temperature that we are eventually trying to determine, we will substitute in its place the adiabatic surface temperature T_e in order to obtain a close approximation of T'. When T_e is substituted for T_s up to a Mach number of 5, these values are close and therefore this substitution is acceptable. This is also the condition for an insulated plate. If q_e in Equation (3) is small in comparison with the other terms this then approaches the case of an insulated plate.

Since varying the reference temperature causes the Nusselt number and the Reynold's number to change in the same direction, the effects of an improper choice of temperature are greatly minimized Equation (6) then becomes:

$$T' = T\,(1 + 0.03192\,M^2 + 0.116\,r\,M^2) \qquad (6a)$$

This simplification is also justified from the results obtained by Hantzsche and Wendt 8/ who have shown in their experiments on a flat plate the effects of varying fluid properties resulting from temperature variations in the boundary layer. Their results indicate that at a Mach number of 5, the exact unit thermal conductance differs from one for isothermal flow by less than 5% for ratios of surface to fluid absolute temperatures ranging from 1 to 8. The effect of variable properties should even be much less pronounced at low Mach numbers.

Substituting the value of the recovery factor, $r = 0.87$ into Equation (6a), we obtain:

$$T' = T\,(1 + 0.133\,M^2) \qquad (6b)$$

Figure 5 shows the plot of Equation (6b). Some investigators use the adiabatic surface temperature T_e as the temperature to evaluate the fluid properties and their experimental findings are also in good agreement with data obtained empirically. In order to illustrate the amount of variation due to using T_e and T' as the reference temperature, an example is shown in the appendix.

Prandtl's Number, Pr, and Thermal Conductivity, K

Prandtl's number is defined as:

$$Pr = 3600\,\frac{g\mu c_p}{k} \qquad (7)$$

It can be regarded as the ratio of momentum diffusivity to thermal diffusivity. As the temperature increases, the value of Prandtl's number decreases as shown in Figure 3. Figure 3 also shows the variation of thermal conductivity of air, k, with temperature.

Reynold's Number, Re

Reynold's number is defined as:

$$Re = \frac{V\rho x}{\mu} \qquad (8)$$

In this report we will use what we call a modified Reynold's number, the Reynold's number without a distance parameter, which is defined as:

$$\frac{Re}{x} = \frac{\Phi\,(P)(T)^{1/2}(M)}{(T')^{1.7}} \qquad (8a)$$

This expression is derived as follows:

$$\text{Reynold's Number,} \quad Re = \frac{V\rho x}{\mu}$$

$$\text{From the perfect gas law,} \quad \rho = \frac{P}{R'}$$

Using the viscosity law, $\frac{\mu}{\mu_\infty} = \left(\frac{T'}{T_\infty}\right)^{.7}$

and $$\frac{Re}{x} = \frac{V\,P}{R\,T'\,\mu_\infty\,(T'/T_\infty)^{.7}}$$

$$= \frac{V\,P}{\frac{R\,\mu_\infty}{(T_\infty)^{.7}}\;(T')^{1.7}}$$

Now $V = M\,a$ where a is the speed of sound and is equal to $a_\infty (T/T_\infty)^{1/2}$. Then:

$$\frac{Re}{x} = \frac{P\,a_\infty\,(T/T_\infty)^{1/2}\,M}{\frac{R\,\mu_\infty}{(T_\infty)^{.7}}\;(T')^{1.7}}$$

$$= \frac{a_\infty (T_\infty)^{.2}}{R\,\mu_\infty} \times \frac{P\,(T)^{.5}\,M}{(T')^{1.7}}$$

Let $$\frac{a_\infty (T_\infty)^{.2}}{R\,\mu_\infty} = \phi$$

Therefore,
$$\frac{Re}{x} = \phi\,\frac{(P)(T)^{1/2}(M)}{(T')^{1.7}} \qquad (8a)$$

In this form Re/x can be presented as a nomograph as shown in Figure 11. By use of this modification of

Reynold's number a parameter is found which remains constant with x along a continuous surface.

Equation (8a) is a convenient way of expressing the modified Reynold's number as a function of a specific atmosphere (P & T), speed of aircraft (M), and the reference temperature (T'). The magnitude of this modulus is an indication of the rate of new fluid arriving in the thermal boundary layer to carry heat to or from the surface. The critical Reynold's number is that value of Re at which the flow characteristics of the fluid change from laminar to turbulent conditions. This value will be taken as 5×10^5 for purposes of computation in this report, that is, $Re > 5 \times 10^5$ will be considered as turbulent flow and $Re < 5 \times 10^5$ will be considered as laminar flow.

Other investigators have used other values of Re for transition. These values range from 1×10^5 to 2.5×10^6. The value used herein has been found to agree with most experimental results available. It should be noted that many investigators have used free stream conditions for the evaluation of Reynold's number and thus have obtained values higher than those used in this report.

Heat Transfer Coefficient, h_s

9/ It is now possible to introduce the Nusselt number, for the flat plate which combines the heat transfer coefficient, Prandtl number, and the Reynold's number into an extremely useful parameter defined by:

$$(Nu)_x = \frac{h\,x}{k} = 0.33\ (Re)^{.5}\ (Pr)^{.33} \qquad (9)$$

and

$$(Nu)_x = \frac{h\,x}{k} = 0.03\ (Re)^{.8}\ (Pr)^{.33} \qquad (10)$$

Equation (9) is applicable only to a laminar boundary layer while Equation (10) is applicable to a turbulent boundary layer. Equations (9) and (10) can be re-written as follows:

$$(h)_x = 0.33\ \frac{k}{(x)^{.5}} \left(\frac{Re}{x}\right)^{.5}\ (Pr)^{.33} \qquad (11)$$

$$(h)_x = 0.03 \frac{k}{(x)^{.2}} \left(\frac{Re}{x}\right)^{.8} (Pr)^{.33} \qquad (12)$$

Equations (11) and (12) are presented as nomograms in figures 12 and 13.

For a flat plate, free stream conditions of velocity and pressure are used in the determination of the reference temperature T', and based on this value of T' the properties of the fluid are evaluated. These properties are, thermal conductivity, density, and viscosity. Prandtl's number is also determined for this reference temperature T'.

For any other aerodynamic shape, the free stream conditions of velocity, pressure and temperature are replaced by the local conditions just outside of the boundary layer at that point. These properties can be found once the pressure distribution around the body in question is known. 13/, 14/ This data can be determined from wind tunnel experiments or from free flight tests and, in some cases, computed. The change in these properties for a cone at various speeds can be found from Figures 7,8,9, and 10, which make use of the fact that velocity is constant along the surface of a cone. For ogives we can assume series of truncated cones or a cone faired into a cylinder.

In addition, it has been shown in reference 8 that the local heat transfer coefficient in laminar flow for the cone must be multiplied by $(3)^{1/2}$ to account for curvature. For the case of turbulent flow on a cone, the heat transfer coefficient as obtained by Gazley, 16/ appears to be 1.15 times larger than the local heat transfer coefficient as obtained by the flat plate equations. This reasoning will also be applied to other bodies of revolution.

It should be noted that equations (9) through (12) are the local values of Nusselt number and heat transfer coefficient at x. The mean value over any continuous flat surface for Nusselt number and the heat transfer coefficient are given by the following expressions:

$$(Nu)_m = 2 \ (Nu)_x \qquad (9a)$$

$$(Nu)_m = 1.25 \ (Nu)_x \qquad (10a)$$

$$(h)_m = 0.66 \frac{k}{(L)^{.5}} \left(\frac{Re}{L}\right)^{.5} (Pr)^{.33} \quad (11a)$$

$$(h)_m = 0.0375 \frac{k}{(L)^{.2}} \left(\frac{Re}{L}\right)^{.8} (Pr)^{.33} \quad (12a)$$

where L = the total distance from the leading edge to the point in question.

The relation between the local and mean heat transfer coefficient for a cone with laminar flow is shown to be:

$$(h)_m = \frac{4}{3}(h)_x$$

and similarly for turbulent flow:

$$(h)_m = \frac{10}{9}(h)_x$$

Radiation Exchange

11/ Another factor affecting surface temperature is radiation. A heated body emits radiant energy at a rate and of a quality dependent on the temperature of the body. This energy relationship is shown by the following expression:

$$\frac{dq}{dt} = \epsilon_s \sigma A_t (T_s)^4 \quad (13)$$

where the emissivity ϵ_s, varies with wavelength (or temperature of radiation), degree of roughness and, if a metal, with the degree of oxidation. 10/ Table I gives the emissivities of various surfaces for various emission temperatures. Although the values in this table apply strictly to normal radiation from the surface, they may be used with negligible error for hemispherical emissivity except in the case of well polished metal surfaces, for which the hemispherical emissivity is 15 to 20 per cent higher than the normal value.

Values in Table I also hold for absorbtivity which is dependent upon the nature of the incident radiation and should be evaluated at the color temperature of the irradiating source. For purposes of computation, the value of absorbtivity due to nocturnal irradiation will be taken as that of a source whose temperature is 100°F.

The radiation gained from the sun, space and earth is shown by the second and third terms of Equation (3) where,

$$\frac{dq}{dt} = \left(\alpha_\beta G_\beta + \frac{A_p}{A_t} \alpha_s G_s \right) \quad (14)$$

is the irradiation to the surface from the sun assumed to be at its zenith and from the earth and surrounding atmosphere and which is expressed as a function of altitude in Figure 6. Above 50,000 feet these values are assumed to be constant. During a night flight the only irradiation factor involved is that due to the earth and the surrounding atmosphere.

Since all the terms of Equation (3) except those for solar radiation use total surface area and this term uses the projected surface area or effective irradiation area to the sun, the term A_p/A_t is introduced. The projected surface area A_p, for a horizontal flat plate (wing) is 1/2 the total surface area A_t. For a body of revolution A_p is $1/\pi$ times the the total surface area, A_t.

Equipment Heat q_e

A quantity of heat is dissipated from internal components (electronic tubes, resistors, motors, etc.) operating at a given load. This value of heat, q_e, is the net heat released from all such components and is divided by A_t in order to be expressed in terms of BTU per square foot of compartment surface. If such components are artificially cooled the value of q_e will be reduced and in the case where cooling exceeds heat output of equipment q_e will have a negative value.

Method of Solution

All charts and graphs necessary to achieve solution of the specified parameters are contained in the appendix of this report. The following steps are taken for solution of the parameters to be used in the heat balance equation (3).

Step	Fig. or Table	Result
1. Altitude, Speed, Atmosphere	Given	Alt, M_o
2. For given altitude and speed determine ambient pressure and temperature from atmosphere to be used for solution	1,2	P_o, T_o
3. Determine local conditions adjacent to the boundary layer		
For a <u>flat plate</u> local conditions are same as free stream (ambient) conditions	1,2	P,T,M
For a <u>cone</u> in supersonic flow when the shock wave is detached use free stream conditions as an approximation and when the shock wave is attached use the following figures to determine local conditions	7,8,9,10	P,T,M
4. Determine adiabatic surface temperature for given speed and local temperature	4	T_e
5. Determine reference temperature for property evaluation for given speed and local temperature	5	T'
6. Determine Modified Reynold's No.	11	Re/x
7. Multiply value of Re/x as found in step 6 by the distance in ft. from the leading edge to the section being investigated. If the value is $>5 \times 10^5$ turbulent conditions exist and if the value is $<5 \times 10^5$ laminar conditions exist		Re

Step	Fig. or Table	Result
8. Determine value of thermal conductivity and Prandtl number for the determined value of T'	3	k, Pr.
9. Knowing k, Pr, x, Re/x determine value of heat transfer coefficient for proper flow conditions	12,13	h_s
10. Determine absorbtivity of material for solar and nocturnal irradiation	Table I	α_β α_s
11. Determine emissivity of surface	Table I	ϵ_s
12. Determine irradiation on surface at the given altitude. At night only nocturnal irradiation is present and during the day both nocturnal and solar irradiation are present. For symmetrical bodies $G_\beta = 1/2(G_a+G_b)$	6	G_β, G_s
13. Determine net heat exchanged from equipment per square foot of compartment surface		$\frac{q_e}{A_t}$

From the above thirteen steps, the various parameters necessary to solve Equation (3) have been obtained. We can now solve Equation (3):

$$h_s(T_e - T_s) + (\alpha_\beta G_\beta + \frac{A_p}{A_t} \alpha_s G_s) + \frac{q_e}{A_t} - \epsilon_s \sigma (T_s)^4 = 0 \qquad (3)$$

Sample Calculation

An example is presented to illustrate the application of the method of solution of the various parameters required for solution of Equation (3). The example is for an aluminum (24ST) flat plate (i.e. wing) with zero angle of attack, flying at Mach number 2 at a constant altitude of 40,000 feet in the 12/ proposed USAF Cold Atmosphere during the day. The problem is to determine the parameters necessary to solve Equation (3) for the steady state equilibrium temperature T_s.

Step 1 40,000 feet, Mach No. 2, Cold Atmosphere

Step 2 From Figs. 1 & 2, $P_o = 334$ lb/ft^2 $T_o = 375$ °R

Step 3 $M = 2$ $P = 334$ lb/ft^2 $T = 375$ °R

Step 4 From Fig. 4, $T_e = 650$ °R

Step 5 From Fig. 5, $T' = 570$ °R

Step 6 From Fig. 11, $Re/x = 2.5 \times 10^6$

Step 7 for $x = 1$ ft, $Re = 2.5 \times 10^6$ Since this value is greater than 5×10^5, turbulent conditions exist

Step 8 From Fig. 3, $k = .0158$ BTU/hr ft^2 (°F/ft), $Pr = .71$

Step 9 From Fig. 13, $h_s = 65$ BTU/hr ft^2 °F

Step 10 From Table I, $\alpha_s = .53$ $\alpha_\beta = .26$

Step 11 From Table I, $\epsilon_r = .30$

Step 12 From Fig. 6, $G_\beta = 36$ BTU/hr ft^2

$G_r = 410$ BTU/hr ft^2

Step 13 No equipment heat dissipation $q_e = 0$

The previous example was for a wing (flat plate) but it can also be applied to aircraft body (cone) by using the corrections of charts, Figures 7,8,9, and 10, as shown in the following example for a 30° cone under the same flying conditions.

Step 1 40,000 feet, Mach No. 2, Cold Atmosphere

Step 2 From Figs. 1 & 2 $P_o = 334$ lb/ft^2, $T_o = 375$ °R

Step 3	From Figs. 7,8,9,10 30° cone shock wave is attached $M = 1.75$ $P = 501$ lb/ft^2 $T = 420$ °R
Step 4	From Fig. 4 $T_e = 660$ °R
Step 5	From Fig. 5 $T' = 600$ °R
Step 6	From Fig. 11 $Re/x = 3.2 \times 10^6$
Step 7	For $x = 1$ ft $Re = 3.2 \times 10^6$ Since this value is greater than 5×10^5 turbulent conditions exist
Step 8	From Fig. 3, $k = .016$ BTU/hr ft^2 (°F/ft) $Pr = .695$
Step 9	From Fig. 13, $h_s = 77$ BTU/hr ft^2 °F For a cone in turbulent flow multiply this value by 1.15 or $h_s = 89$ BTU/hr ft^2 °F
Step 10	From Table I, $\alpha_s = .53$ $\alpha_\beta = .26$
Step 11	From Table I, $\epsilon_s = .30$
Step 12	From Fig. 6, $G_\beta = 36$ BTU/hr ft^2 $G_s = 410$ BTU/hr ft^2
Step 13	No equipment heat dissipation $q_e = 0$

Graphical Solution of Heat Transfer Equation

Up to this point existing information on aerodynamic heating was reviewed and the parameters involved in the heat transfer Equation (3) were analyzed and set up in such a way that they are amenable to simple graphical determination. It is believed that this type of presentation will reduce the great amount of work required of those having need for this information. Likewise, it is logical that having presented the graphical solution for the parameters involved in the heat transfer Equation (3), the solution of this equation for the equilibrium skin temperature should also be presented as a graphical determination.

Rearranging Equation (3) as follows:

$$\sigma (T_s)^4 + \frac{h_s T_s}{\epsilon_s} - \frac{\left[\left(\alpha_\beta G_\beta + \frac{A_p}{A_t} \alpha_s G_s\right) + \frac{q_e}{A_t} + h_s T_e\right]}{\epsilon_s} = 0 \qquad (15)$$

and performing the following substitutions:

$$A = \sigma$$

$$B = \frac{h_s}{\epsilon_s}$$

$$C = \frac{\left[\left(\alpha_\beta G_\beta + \frac{A_p}{A_t}\,\alpha_s G_s\right) + \frac{q_e}{A_t} + h_s T_e\right]}{\epsilon_s}$$

Equation (15) becomes:

$$A\,T_s^4 + B\,T_s - C = 0 \qquad (16)$$

This form of fourth degree equation can be readily solved by graphical methods. This graph is constructed in Figure 14 of the appendix.

The manner in which this graph is used to determine the equilibrium skin temperature T_s is as follows:

1. For the given flight conditions, compute values of B & C.
2. On scale B, mark off the computed value of B (note scale factors on graph).
3. Draw a straight line from the origin 0-0 to value of B previously determined in step 2.
4. On scale C, mark off the computed value of C (note scale factors on graph).
5. Through this value of C draw a straight line parallel to the line drawn in step 3.
6. Where the straight line from step 5 intersects the proper AT_s^4 curve, drop a line perpendicular to the temperature axis and where this perpendicular intersects the temperature axis is the value of T_s, the equilibrium skin temperature.

Figure 14 in the appendix of this report shows the actual detail working graph for the solution of the equilibrium skin temperature T_s. An example is presented to illustrate the use of this figure in the solution of Equation (3) for the equilibrium skin temperature, T_s.

Sample Calculation

Using the data obtained from the example for the wing on page 12 we obtain the following:

$$B = \frac{h_s}{\epsilon_s} = \frac{65}{.3} = 216.7$$

$$C = \frac{\left[(\alpha_\beta G_\beta + \frac{A_p}{A_t}\alpha_s G_s) + \frac{q_e}{A_t} + h_s T_e\right]}{\epsilon_s}$$

$$= \frac{(.26 \times 36 + 1/2 \times .53 \times 410) + 0 + 65 \times 650}{.3}$$

$$C = 141,400$$

Noting the scale factors on Figure 14, draw a line from origin 0 - 0 to B = 216.7. Then through value of C = 141,400 draw a line parallel to the line drawn from origin to B. Where this line intersects curve (1) drop a perpendicular to the temperature axis and read the value of the equilibrium skin temperature T_s = 650 °R.

Now, having determined the equilibrium skin temperature T_s, we are in a position to compute the value of the equipment temperature and the compartment air temperature.

Inside Skin Temperature, T_{si}

When the walls are poor conductors there exists a temperature gradient within the wall. If this gradient exists then the inside skin surface temperature T_{si} must be determined so that it can be used in the determination of equipment temperature and compartment air temperature, T_c. The following is the expression for determining the inside skin equilibrium surface temperature, (T_{si}).

$$\frac{q_e}{A_c} = \frac{k_s}{\tau} (T_{si} - T_s) \tag{17}$$

where,

$\frac{q_e}{A_c}$ = heat dissipated by equipment, BTU/hr ft^2 of compartment surface

k_s = thermal conductivity of skin, BTU/hr ft^2 (°F/ft)

τ = skin thickness, feet

T_s = outside equilibrium skin temperature, °F

T_{si} = inside equilibrium skin temperature, °F

Equipment Temperature, T_q

To determine the temperature of components buried within the interior of electronic equipment located within a compartment is a very complicated matter because of the complex nature of the heat transfer. In this analysis, for reasons of simplification, it is convenient to assess an average temperature of the equipment by considering it as a smooth heat generating package separated from the skin by a considerable blanket of air across which the equipment is mounted by vibration mounts which offer good thermal insulation to the compartment wall. The heat lost from the equipment by conduction through the vibration mounts will thus be assumed to be a very small factor.

The following is the equation for a heat balance on an assumed piece of equipment.

$$q_e = q_{cv} + q_r \tag{18}$$

In this analysis, the chassis and the components enclosed are considered at identical temperature and because of the complex mass of the equipment, it will be treated as a simple smooth package such as a small boiler tube hundles in a box for which data may be obtained from references 17 and 18.

From Mc Adams 17/ the heat transfer coefficient by convection from single horizontal cylinders is given by the following expression:

$$\frac{h_f D}{k_f} = 0.53 \left[\left(\frac{D^3 \rho_f^2 \beta g \Delta T}{\mu_f^2} \right) \left(\frac{C_p \mu_f}{k_f} \right) \right]^{1/4} \qquad (19)$$

Let $\rho = \frac{P_\infty}{R\, T_f}$

$$\alpha = \frac{g \beta C_p P_\infty^2}{\mu_f\, k_f\, R^2\, T_f^2}$$

Equation (19) becomes:

$$h_f = \frac{0.53\, k_f}{D^{1/4}} (P)^{1/2} (\alpha)^{1/4} (\Delta T)^{1/4} \qquad (20)$$

The coefficient of heat transfer h_f is for the air film to which all the terms with subscript f refer, and the constant is the shape factor.

Equation (20) gives the value of the heat transfer coefficient h_f for single horizontal cylinders which approximate the heat transfer coefficient for small rectangular boxes, flat plates, etc.

The suppression of convection by proximity of the shell to the component bundle may be neglected if the convection is handled in the manner suggested in reference 18/ where ΔT is the difference between hot and cold surfaces and convective transfer is 50% of normal convection to free air. Equation (20) becomes:

$$h_q = \frac{0.27\, k_f}{D^{1/4}} (P)^{1/2} (\alpha)^{1/4} (\Delta T)^{1/4} \qquad (21)$$

As a means of simplification all terms having the subscript f are evaluated at the temperature T_f which will be taken as the average between the equipment temperature and the equilibrium inside skin surface temperature.

The heat transferred by convection is 17/:

$$q_{cv} = h_q \, A_e \, \Delta T \tag{22}$$

or

$$q_{cv} = \frac{0.27 \, k_f \, A_e}{D^{1/4}} \, (P)^{1/2} \, (\alpha)^{1/4} \, (T_q - T_{si})^{5/4} \tag{23}$$

Here effective area, A_e, accounts for the difference in area of the equipment and inside wall.

In Equation (18) the radiative heat transfer q_r is given by the following expression:

$$q_r = \epsilon \sigma A_e \, (T_q^4 - T_{si}^4) \tag{24}$$

Substituting Equations (23) and (24) in Equation (18) the following expression is obtained:

$$q_e = \frac{0.27 \, k_f \, A_e}{D^{1/4}} \, (P)^{1/2} \, (\alpha)^{1/4} \, (T_q - T_{si})^{5/4} + \epsilon \sigma A_e (T_q^4 - T_{si}^4) \tag{25}$$

Equation (25) can be solved by a trial and error method but this requires a great deal of laborious procedure. The computations are reduced to a minimum by resorting to graphs. The first step in the graphical solution to Equation (25) is to plot Equations (23) and (24) as functions of T_q and T_{si}. In this way we will have two graphs, one for convective heat transfer as a function of T_q and T_{si} and one for radiative heat transfer as a function of T_q and T_{si}.

By making an estimate of the equipment temperature T_q, we can obtain from these graphs the values of q_r and q_{cv}. Since we know q_e, the total heat dissipated by the equipment, the sum of the computed q_r and q_{cv} should equal q_e. If they do not, the initially assumed value of T_q was in error. After a few trials we should be able to obtain a very close value of equipment temperature T_q which satisfies Equation (25). This graph is shown as Figure 15 in the appendix of this report. Examples showing the use of this graph are also contained in the appendix.

After having determined the equipment temperature T_q we know the value of the convective heat transfer, q_{cv}. Since it is this heat transfer that is heating up the compartment air, we can solve for the compartment air temperature, T_c.

Compartment Air Temperature, T_c

The compartment air is heated by the convective heat from the equipment since radiation from the equipment passes through the air to the outer skin:

$$q_{cv} = h_f \, A_q \, \Delta T \tag{22}$$

The heat transfer coefficient h_f is determined by Equation (19) where ΔT is equal to $(T_q - T_c)$ and $T_f = \frac{T_q + T_c}{2}$

Equation (20) can be solved by graphical means. A graph is constructed of q_{cv} as a function of T_q and T_c. Since T_q and q_{cv} are known, the point at which these two values intersect on the graph is the value of T_c that satisfies Equation (7). Figure 16 and an illustrative example of its use are contained in the appendix.

Conclusions

The continuously increasing airplane performance and the recent achievements in the design of miniaturized electronic equipment has created the broad problem of aerodynamic heating which has come into focus comparatively recently. This inevitable merging of miniaturized designs with sonic region environments necessitates evaluation of the combined effects upon not only the aircraft structure but also on the mechanical and electronic components. Because of the complexity of the modern aircraft which may contain many distinct systems, it becomes self evident that these systems must operate satisfactorily when exposed to the ever increasing problems of aerodynamic heating and of cooling at high altitudes.

In view of the above, an attempt has been made to present a concise simplified graphical solution of the aerodynamic heating problem as it applies to the solution of the equilibrium skin temperature, equipment temperature and compartment air temperature. It is believed that the methods presented in this report will aid electronic designers and applications engineers in determining and evaluating electronic equipment for any value of air temperature, altitude, and installation environment.

Aside from the internal heat dissipation, the most important factor in establishing component temperature is frequently the temperature of the surrounding surfaces such as the aircraft structure. This one fact alone allows a test set-up to be made where the walls of the test cell can be maintained at the equilibrium skin temperature as determined by the methods of this report and conditions of altitude can also be simulated. The piece of equipment under question can then be inserted into this simulated environmental aircraft compartment and its operation and performance studied. From these tests which simulate actual compartment environments in high performance aircraft, specifications for equipment to be installed in aircraft can be based on the conditions to be expected and not on conditions which exist in relatively low performance aircraft of the past. This test set-up is more realistic and practical than any hypothetical analysis presented for determination of equipment temperature.

The example of equipment temperature has been restricted to a single, rather elementary unit in a relatively uniform environment under ideal conditions, and unfortunately most practical problems encountered are not nearly this simple. The surface temperature of the equipment is distinctly non-uniform, and it must be considered that the hottest point on the equipment is the one that determines its useful life.

Aside from the thermal environment and its effect on electronic equipment, there are two main considerations pertaining to the influence of elevated temperatures upon the mechanical performance of high speed aircraft and missiles; that is, the effects upon the structure (strength and rigidity), and upon fuel and interior equipment. The structural design problem involves the selection of materials of satisfactory elevated temperature strength and rigidity. A secondary and just as important, structural design problem is the determination of thermally induced stresses. Similarly the graphical solution of the aerodynamic heating problem of equilibrium skin temperature as presented in this report will aid the aircraft and missile designer in evaluating structures, ordinance and mechanical equipment for any value of air temperature, altitude and installation environment.

As can be seen in the final examples of a typical fuselage compartment (Figures 17, 18, 19), at the higher velocities the difference between component, compartment, and equilibrium skin temperatures is negligible, except when equipment heating per unit area is extremely large ($q_e/A_e > 5000$ BTU/hr ft^2). Conversely, at these speeds a great deal of cooling is required to significantly reduce the equipment temperature. It is interesting to note that at the lower altitudes, again except when equipment heating or cooling are of a very high magnitude, the value of equilibrium skin temperature is equal to the adiabatic surface temperature. As altitudes increase above 40,000 feet, there is a departure from the adiabatic surface temperature. With equipment heat being produced, as in the example shown, at low Mach numbers the skin temperature is higher than the adiabatic surface temperature, due to this equipment heat. At the high Mach numbers, however, the skin temperature is lower than the adiabatic surface temperature, due to heat loss from the skin by radiation.

This example will serve to indicate the range of temperatures to be encountered by aircraft and aircraft equipment flying up to 100,000 feet at speeds from M = 0 to M = 5. Three curves for skin and equipment temperature for a typical fuselage section are shown at each of three speeds. At M = 5, the central curve indicates temperature for little heat being produced; the low temperature curve indicates the effect of cooling ($q_e/A_e = -5000$ BTU/hr ft^2); and the higher temperature curve shows temperatures to be attained with a given amount of equipment heating ($q_e/A_e = 5000$ BTU/hr ft^2).

An idea of the problems of keeping aircraft skins cool at the high Mach numbers is shown by a calculation of cooling required for the surface of this typical compartment to be maintained below 1500°R under steady flight conditions. In this section at M = 5, approximately 90,000 BTU/hr of cooling are required per square foot of surface area. When a value of this order is multiplied by the total aircraft surface area the magnitude of cooling is shown to be fantastically large. The most applicable method of maintaining internal equipment below reasonable temperature limits at these speeds, even at high altitudes, is to use a layer of good thermal insulation in conjunction with an efficient cooling system. For example, a two inch layer of insulation with a conductivity of .15 BTU/hr ft^2 °F, would require that only 250 BTU/hr ft^2 of cooling be supplied to maintain internal equipment at approximately 610°R (150°F) at M=5, 60,000 ft, with the sample section investigated. Although this does not solve the problem of excessive skin temperature and the associated decrease in strength, equipment in the aircraft can be held at reasonable temperatures with this system.

Although it is possible to show an example of ranges to be expected of skin and equipment temperature, it is impossible at the present time to make recommendations regarding equipment temperature specifications. This is due to the wide range of temperatures to be encountered depending on speeds, part of aircraft, type, size and shape of components, and relationship to other components. It can be stated, however, that equipment should operate down to temperatures of the atmosphere encountered in the altitude range of the aircraft. This is especially important, as can be seen in Figure 17 with low speed aircraft where equipment temperatures are very close to free air temperatures.

With increase in aircraft velocities high temperature difficulties become pronounced. A use of the methods presented in this report will serve to offer a range of temperatures to be encountered by specific equipment systems operating in specific parts of specific aircraft, or of the cooling requirements to maintain temperature below some limit under such conditions. In general, it is recommended that the system of using a test chamber of the same size and configuration as the aircraft compartment and with walls at temperatures determined in this report be used to experimentally determine equipment temperatures and to evaluate high temperature life of equipment. It is also recommended that equipment be designed to operate at as high temperatures as possible in high speed aircraft in order to reduce the requirements for cooling and the excessive weight imposed by large capacity cooling systems.

A rough evaluation of accuracy to be obtained using the methods of this report has been carried out. The wide range of temperatures encountered prohibits statement of an expected accuracy for the total method. However, mean accuracy of portions of the system can be estimated as follows:

T_e - accurate to $\pm$ 5%. This is based on variation of recovery factor experienced under actual conditions with the average recovery factor of this report. Under most conditions the value of T_e will be accurate to within this value.

$|T_e - T_s|$ - accurate to $\pm$ 10%. The accuracy of this difference is estimated from a knowledge of errors generally associated with a method of solution of the type presented.

$|T_s - T_q|$, $|T_q - T_c|$ - accurate to $\pm$ 20%. Here accuracy is estimated from general accuracy obtainable with solutions of problems of natural convective heating.

Although the accuracy of the methods of solving for temperature differences does not appear good, a review of the magnitude of these differences compared to the value of adiabatic surface temperature, T_e, shows that errors introduced are of smaller magnitude than those in T_e, and thus the 5% accuracy in determination of T_e is the major factor in determining accuracy of the overall method of solution.

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Appendix I

1. Example Comparing T_e and T':

An example comparing the values of heat transfer coefficient using T' and T_e as the reference temperatures to evaluate the properties of air is as follows:

$P = 100$ lb/ft^2 $\quad$ $T = 400$ °R $\quad$ $M = 5$

From Figures 4 and 5:

$T_e = 2140$ °R $\quad$ $T' = 1730$ °R

Reference Temperature	$T_e = 2140$ °R	$T' = 1730$ °R
Re/x (Fig. 11)	1.8×10^5	2.4×10^5
k (Fig. 3)	.0483	.0404
Pr (Fig. 3)	.64	.65
$(h)_x$ (x=10 ft Fig. 13) (turbulent)	14.0	14.0
$(h)_x$ (x=1 ft Fig. 12) (laminar)	6.5	6.0

It can be seen from the above example that our choice of reference temperature T' as defined in this report produces results in the determination of the heat transfer coefficient comparable to the results of some other investigators when using the reference temperature T_e. It is believed that use of T' is, however, more realistic and will offer better accuracy at the higher speeds.

2. Example of Variation of Recovery Factor

Based on the findings of various investigators, the results of their experiments on flat plates and ogives have shown recovery factor to have values from .82 to .96. These values were obtained for conditions of subsonic and supersonic flow both laminar and turbulent. Based on these findings for flat plates, cones, and ogives, we have assumed for purposes of the development of the method presented herein, that the value of the recovery factor was equal to the average of 0.87. Many values of recovery factors have appeared in the literature and 0.87 was determined as an average from reviewing the results of the various investigators. Although it is known that this assumption reduces the accuracy of the results, its effects are shown to be small.

$M = 5$ $P_o = 100$ lb/ft $T_o = 400$ °R

r	.82	.87	.96
T_e	2040	2140	2320
% Δ	-4.58%	-	+8.41%

r	.82	.87	.96
T'	1670	1728	1833
% Δ	-3.36%	-	+5.51%

Should further accuracy be required, a more exact value of recovery factor can be used in a second solution for T_e and T'. The error will be reduced considerably for lower values of Mach number.

Appendix II

1. Examples of Methods Presented

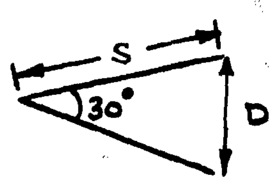
a. The following example will show the use of all the figures and charts in this report.

Data

(1) 40,000 feet Cold Atmosphere Daytime

(2) Mach Number 2

(3) Conical nose compartment, thin wall, $\tau = .091"$



D = 16" S = 31"

Surface area of compartment = $\pi rs = \pi \times 8 \times 31 = 778\ in^2 = 5.4\ ft^2$.

(4) Material of compartment aluminum (24ST)

(5) Electronic Equipment
horizontal cylinder 3" diameter, 6" long
surface painted black, surface area = $.392\ ft^2$.

(6) Heat dissipated by electronic equipment =
540 BTU/hr = 100 BTU/hr ft^2 of compartment surface.

Phase I - Determination of Heat Transfer Parameters

Step 1 40,000 feet Mach No. 2, Cold Atmosphere, daytime

Step 2 From Fig. 1 & 2 $P_o = 334\ lb/ft^2$ $T_o = 375\ °R$.

Step 3 From Figs. 7,8,9,10 30° Cone Shock wave attached
$M = 1.75$ $P = 501\ lb/ft^2$ $T = 420.0\ °R$

Step 4 From Fig. 4 $T_e = 660\ °R$

Step 5 From Fig. 5 $T' = 600\ °R$

Step 6 From Fig. 11 $Re/x = 3.2 \times 10^6$

Step 7 For $x = L = 31" = 2.58$ ft $Re = 8.26 \times 10^6$
Since this value is greater than 5×10^5 turbulent conditions exist.

Step 8 From Fig. 3 $k = .016$ BTU/hr ft^2 (°F/ft)
$Pr = .695$

Step 9 From Fig. 13 $(h)_x = 64$ BTU/hr ft^2 °F
for a cone in turbulent flow, multiply this value by 1.15 or $(h)_x = 73.6$ BTU/hr ft^2 °F. In this particular case, we will use the mean heat transfer coefficient $(h)_m$ which is $\frac{10}{9}(h)_x$ or $\frac{10}{9} \times 73.6 =$ 81.8 BTU/hr ft^2 °F

Step 10 From Table I $\alpha_s = .53$ $\alpha_{\beta} = .26$

Step 11 From Table I $\epsilon_s = .3$

Step 12 From Fig. 6 $q_{\beta} = 36$ BTU/hr ft^2
$q_s = 410$ BTU/hr ft^2

Step 13 Equipment heat dissipation $\frac{q_e}{A_c} = 100$ BTU/hr ft^2

Phase II - Determination of Equilibrium Skin Temperature, T_s

Step 1 $B = \frac{h_s}{\epsilon_s} = \frac{81.8}{.3} = 272.7$

Step 2 $$C = \frac{\left[\left(\alpha_\beta\, G_\beta + \frac{A_p}{A_t}\, \alpha_s\, G_s \right) + \frac{q_e}{A_t} + h_s\, T_e \right]}{\epsilon_s}$$

$$= \frac{(.26 \times 36 \times \frac{1}{\pi} \times .53 \times 410) + 100 + 81.8 \times 660}{.3}$$

$C = 180{,}730$

Step 3 Using Figure 14 and computed values of B and C

Step 4 On Scale B, mark off the computed value of B = 81.8 (note scale factor on graph)

Step 5 Draw a straight line from the origin 0 - 0 to value of B = 81.8

Step 6 On scale C, mark off the computed value of C = 180,730 (note scale factor on graph)

Step 7 Through this value of C = 180,730 draw a straight line parallel to the line drawn in Step 5

Step 8 Where the straight line from Step 7 intersects the proper curve (curve 3, note scale factor on graph) drop a line perpendicular to the temperature axis and where this perpendicular intersects the temperature axis is the value of T_s, the equilibrium skin temperature = 660 °R

Phase III - Inside Skin Temperature

Since the wall of the compartment is very thin, the inside skin temperature T_{si} is the same as the outside equilibrium skin temperature T_s $T_{si} = T_s = 660$ °R

Step 1 $$\frac{q_s}{A_c} = \frac{k_s}{\tau} (T_{si} - T_s)$$

$k_s = 145$ BTU/hr ft^2 (°F/ft) (aluminum)

$\tau = .091" = .0075$ ft

Step 2 $100 = \dfrac{145}{.0075}(T_{si} - 660)$

$\dfrac{.75}{145} = (T_{si} - 660)$

Step 3 $T_{si} = 660$ °R

Phase IV - Equipment Temperature

Step 1 $q_e = 540$ BTU/hr

Step 2 $A_e = 2\dfrac{A_c\ A_q}{A_c + A_q} = \dfrac{5.4 \times .392}{5.792} = .732\ \text{ft}^2$

effective area $A_e = .732\ \text{ft}^2$

Step 3 Using Figure 15 and knowing

$$\frac{q_e}{A_e} = \frac{q_{cv}}{A_e} + \frac{q_r}{A_e}$$

$$\frac{540}{.732} = \frac{q_{cv}}{A_e} + \frac{q_r}{A_e}$$

$$738 = \frac{q_{cv}}{A_e} + \frac{q_r}{A_e}$$

Step 4 $P = .1559$ atmospheres

$D = 3" = 1/4$ ft

$$\sqrt[4]{\frac{P^2}{D}} = \sqrt[4]{\frac{(.1559)^2}{1/4}} = .558$$

Step 5 $\epsilon_1 = .9 \quad \epsilon_2 = .9$

$$\epsilon = \frac{\epsilon_1 \epsilon_2}{\epsilon_1 + \epsilon_2 - \epsilon_1 \epsilon_2} = \frac{.81}{1.8 - .81} = .818$$

Step 6 Assume value of T_q and compute

$\frac{q_{cv}}{A_e}$ & $\frac{q_r}{A_e}$

$T_q = 1000$ °R

Step 7 $\frac{q_{cv}}{A_e} = 120$

Step 8 $\frac{q_r}{A_e} = 1050$

Step 9 $\frac{q_{cv}}{A_e} + \frac{q_r}{A_e} = 1170$ which is greater than $\frac{q_e}{A_e}$

therefore, assumed value of T_q is too large

Step 10 Assume $T_q = 940$ °R

Step 11 $\frac{q_{cv}}{A_e} = 83$

Step 12 $\frac{q_r}{A_e} = 655$

Step 13 $\frac{q_{cv}}{A_e} + \frac{q_r}{A_e} = 83 + 655 = 738$

which is equal to $\frac{q_e}{A_e} = 738$

$T_q = 940$ °R

Phase V - <u>Compartment Air Temperature</u>

Step 1 $T_q = 940\ °R$ and $\frac{q_{cv}}{A_e} = 83$

Step 2 $A_e = 2\frac{A_c\ A_q}{A_c + A_q} = .732\ ft$

Step 3 $q_{cv} = 83 \times .732 = 60.7$ BTU/hr

Step 4 $\frac{q_{cv}}{A_q} = \frac{60.7}{.392} = 155$ BTU/hr ft^2

Step 5 Using Figure 16

$$\sqrt[4]{\frac{P^2}{D}} = .558$$

$$T_q = 940\ °R$$

$$\frac{q_{cv}}{A_q} = 155$$

where $T_q = 940\ °R$

$\frac{q_{cv}}{A} = 187$ at $\sqrt[4]{\frac{P^2}{D}} = .558$

intersect is the value of T_c

Step 6 $T_c = 680\ °R$

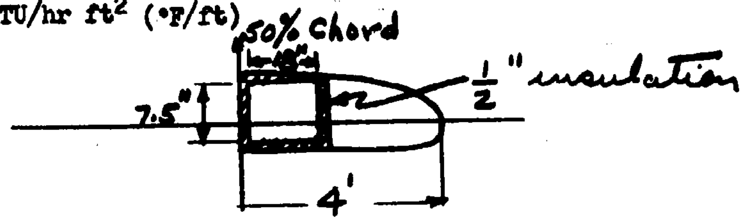
Another example for a wing section is as follows.

<u>Data</u>

(1) 40,000 feet, Cold Atmosphere, Daytime

(2) Mach Number 0.8

(3) Wing compartment, skin-aluminum (24ST), .091" thick
1/2" insulation whose thermal conductivity is
.025 BTU/hr ft^2 (°F/ft)



Dimensions of Compartment: Length = 18"
Width = 12"
Depth = 7.5"

$$\text{Surface Area of Compartment} = 2 \times \frac{(18" \times 12")}{144} = 3 \text{ ft}^2$$

(4) Electronic Equipment - horizontal cylinder
3" diameter, 6" long, surface painted black
Surface area = .392 ft^2

(5) Heat dissipated by electronic equipment
= 20 BTU/hr = 6.7 BTU/hr ft^2 of compartment surface

Phase I - <u>Determination of Heat Transfer Parameters</u>

Step 1 40,000 feet Mach No. = 0.8 Cold Atmosphere Daytime

Step 2 From Fig. 1 & 2 P_o = 334 lb/ft^2 T_o = 375 °R

Step 3 M = 0.8 P = 334 lb/ft^2 T = 375 °R

Step 4 From Fig. 4 T_e = 420 °R

Step 5 From Fig. 5 $T' = 415$ °R

Step 6 From Fig. 11 $Re/x = 1.2 \times 10^6$

Step 7 Take x as the mid point of the compartment as measured from the leading $x = 3.25$ ft $Re = 3.9 \times 10^6$ Since this value is greater than 5×10^5 turbulent conditions exist.

Step 8 From Fig. 3 $k = .012$ BTU/hr ft^2 (°F/ft)
$Pr = .735$

Step 9 From Fig. 13 $h_s = 19$ BTU/hr ft^2 °F

Step 10 From Table I $\alpha_s = .53$ $\alpha_\beta = .26$

Step 11 From Table I $\epsilon_s = .3$

Step 12 From Fig. 6 $G_\beta = 36$ BTU/hr ft^2
$G_s = 410$ BTU/hr ft^2

Step 13 Heat dissipated by equipment = 6.7 BTU/hr ft^2 of compartment surface.

Phase II - Determination of Equilibrium Skin Temperature, T_s

Step 1 $$B = \frac{h_s}{\epsilon_s} = \frac{19}{.3} = 63.3$$

Step 2 $$C = \frac{\left[\left(\alpha_\beta G_\beta + \frac{A_p}{A_t} \alpha_s G_s\right) + \frac{q_e}{A_c} + h_s T_e\right]}{\epsilon_s}$$

$$= \frac{(.26 \times 36 + \frac{1}{2} \times .53 \times 410) + 6.7 + 19 \times 420}{.3}$$

$C = 30,797$

Step 3 Using Fig. 14 and computed values of B and C

Step 4 On scale B, mark off the computed value of B = 63.3 (note scale factor on graph)

Step 5 Draw a straight line from the origin 0 - 0 to value of B = 63.3

Step 6 On scale C, mark off the computed value of C = 30,797 (note scale factor on graph)

Step 7 Through this value of C = 30,797 draw a straight line parallel to the line drawn in Step 5

Step 8 Where the straight line from Step 7 intersects the proper curve (curve 3, note scale factor on graph) drop a line perpendicular to the temperature axis and where this perpendicular intersects the temperature axis is the value of T_s, the equilibrium skin temperature T_s = 490 °R

Phase III - Inside Skin Temperature

Step 1

$$\frac{q_e}{A_c} = \frac{k_s}{\tau}(T_{si} - T_s)$$

k_s = 145 BTU/hr ft^2 (°F/ft) aluminum

τ = .091" = .0075 ft

k_s = .025 BTU/hr ft^2 (°F/ft) insulation

τ = .5" = .0425 ft

Step 2

$$6.7 = \frac{(T_{si} - 490)}{\frac{.0075}{145} + \frac{.025}{.0425}}$$

T_{si} = 501 °R

Phase IV - Equipment Temperature

Step 1 q_e = 20 BTU/hr

Step 2 $\quad A_e = 2\frac{A_c\ A_q}{A_c+A_q} = \frac{3 \times .392}{3.392} = .794\ ft^2$

Effective area $A_e = .794\ ft^2$

Step 3 $\quad$ Using Fig. 15 and knowing

$$\frac{q_e}{A_e} = \frac{q_{cv}}{A_e} + \frac{q_r}{A_e}$$

$$\frac{20}{.794} = \frac{q_{cv}}{A_e} + \frac{q_r}{A_e}$$

$$\frac{q_{cv}}{A_e} + \frac{q_r}{A_e} = 28.8$$

Step 4 $\quad P = .1559$ atmospheres

$D = 3" = 1/4$ ft

$$\sqrt[4]{\frac{P^2}{D}} = .558$$

Step 5 $\quad \epsilon_1 = .9 \qquad \epsilon_2 = .9$

$$\epsilon = \frac{\epsilon_1\ \epsilon_2}{\epsilon_1 + \epsilon_2 - \epsilon_1\epsilon_2} = \frac{.81}{1.8 - .81} = .818$$

Step 6 $\quad$ Assume value of T_q and compute

$$\frac{q_{cv}}{A_e}\ \&\ \frac{q_r}{A_e}$$

$T_q = 530\ °R$

Step 7 $\frac{q_{cv}}{A_e} = 13$

Step 8 $\frac{q_r}{A_e} = 15.8$

Step 9 $\frac{q_{cv}}{A_e} + \frac{q_r}{A_e} = 13 + 15.8 = 28.8$

which is equal to $\frac{q_e}{A_e} = 28.8$

therefore, assumed value of $T_q = 530$ °R is correct.

Phase V - <u>Compartment Air Temperature</u>

Step 1 know $T_q = 530$ °R and $\frac{q_{cv}}{A_e} = 13$

Step 2 $A_e = 2 \frac{A_c\ A_q}{A_c + A_q} = .794\ ft^2$

Step 3 $q_{cv} = 13 \times .794 = 10.3$ BTU/hr

Step 4 $\frac{q_{cv}}{A_q} = \frac{10.3}{.392} = 26.3$ BTU/hr ft^2

Step 5 Using Fig. 16

$$\sqrt[4]{\frac{P^2}{D}} = .558$$

$$T_q = 530\ °R$$

$$\frac{q_{cv}}{A_q} = 26.3$$

where T_q = 530 °R and $\frac{q_{cv}}{A_q}$ = 26.3 intersect at

$\sqrt[4]{\frac{P^2}{D}}$ = .558 is the value of T_c

Step 6 T_c = 505 °R

Appendix III

1.

TABLE I

Total Normal Mean Emissivity and Absorbtivity

Description	Mean Effective ϵ_s and α		
Metals	α_s 100°F	ϵ_s 1000°F	α_s Solar
Aluminum			
Polished	.04	.08	.10
Oxidized	.11	.18	
Chromium	.08	.26	.49
Copper			
Polished	.04	.18	.26
Oxidized	.18	.18	
Oxidized & Corroded	.38		
Iron			
Polished	.06	.13	.45
Oxidized	.74	.78	
Nickel			
Polished	.04	.10	.40
Oxidized	.39	.67	
Zinc			
Polished	.02	.04	.46
Oxidized		.11	
Alloys			
Aluminum			
3 SO	.24	.24	
53 SO, Weathered	.73	.55	
24 ST	.052		.53
24 ST, Weathered	.26	.30	
75 ST, Polished	.07	.15	
75 ST, Anodized	.56	.32	
75 ST, Unpolished	.04	.08	
Inconel			
Clean and smooth	.14		
Dull	.21		

TABLE I

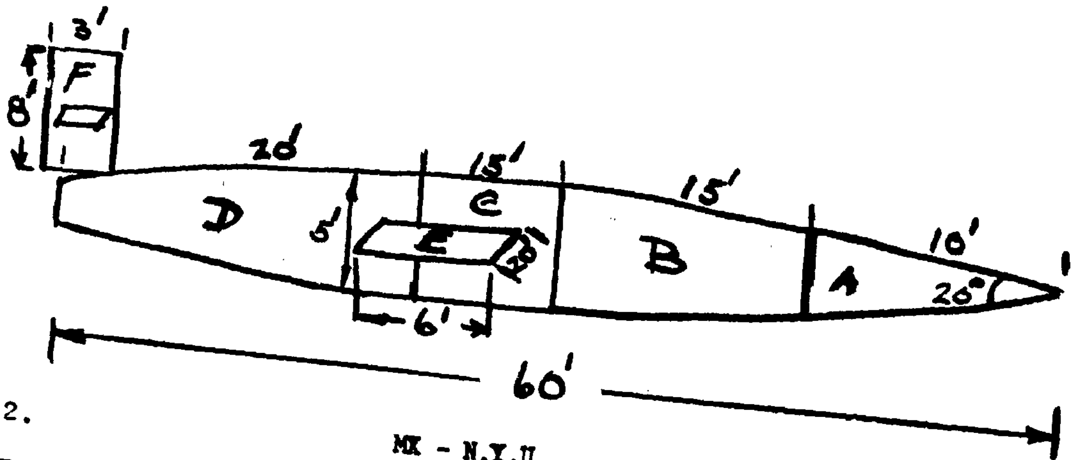
Total Normal Mean Emissivity and Absorbtivity

Description	Mean Effective ϵ_s and α		
Alloys (Cont'd)	α_B 100°F	ϵ_s 1000°F	α_s Solar
Magnesium			
J.H. Magnesium alloy (Weathered)	.58	.55	
M.H. 42 Magnesium alloy	.70	.40	
F.S.A. Dowmetal	.38	.38	
Nickel			
Monel, Clean	.12	.10	.43
Monel, Oxidized		.46	
Chromenickel	.64		
Steel			
18-8 Stainless, Polished	.12	.27	
18-8 Stainless, Unpolished	.14	.32	
18-8 Stainless, Weathered	.62	.62	
18-8 Stainless, Sand Blasted	.84	.84	
18-8 Stainless, Oxidized	.85	.85	
Rolled Sheet	.66		
Rough Sheet	.94	.97	
Tin			
Speculum Metal	.08	.13	.39
Miscellaneous Surfaces			
Shiny Bakelite, Clean	.89		
Plexiglass	.89		
Glass	.85		
Painted Surfaces			
Aluminum painted with Black Decoret	.89		
Bakelite painted with Black Decoret	.94		

TABLE I

Total Normal Mean Emissivity and Absorbtivity

Description	Mean Effective ϵ_s and α		
	α_B	ϵ_s	α_s
Painted Surfaces (Cont'd)	100°F	1000°F	Solar
External air drying enamel W.P.Fuller, D-70-6342	.85	.84	
Aluminized Lacquer 1234 over 75 ST Alclad	.65		
Clear Lacquer 1234 over 75 ST Alclad	.53		
Pigments	125°F	750°F	Solar
Lampblack paint	.96	.97	.97
Camphor Soot	.98	.99	.99
Acetylene Soot	.99	.99	.99
Platinum black	.91	.95	.98
Lampblack	.94	.94	
Flat Black Lacquer	.96	.98	
Black Lacquer	.80	.95	
Blue (Co_2O_3)	.87	.86	.97
Black (CuO)		.85	
Red (Fe_2O_3)	.96	.70	.74
Green (Cu_2O_3)	.95	.67	.73
Yellow (PbO)	.74	.49	.48
Yellow ($PbCrO_4$)	.95	.59	.30
White (Al_2O_3)	.98	.79	.16
" (Y_2O_3)	.89	.66	.26
" (ZnO)	.97	.91	.18
" (CnO)	.96	.78	.15
" ($MgCO_3$)	.96	.89	.15
" (ZrO_2)	.95	.77	.14
" (ThO_2)	.93	.53	.14
" (MgO)	.97	.84	.14
" ($PbCO_3$)	.89	.71	.12
White Enamel	.92		
Dark Glossy Varnish	.89		
Spirit Varnish	.83		



2.

MX - N.Y.U.

Example of temperatures attained on typical aircraft sections operating under steady conditions at a fixed altitude.

TABLE II

Example Temperature Calculations for Typical Aircraft Sections

Flight Conditions:

Alt. 40,000 feet $\quad M_o = 3 \quad P_o = 449$ lb/ft^2 $\quad T_o = 415$ °R

USAF Hot Atmosphere, Daytime

$\alpha_B = .26 \quad \alpha_s = .53$

$G_B = 36$ BTU/hr ft^2

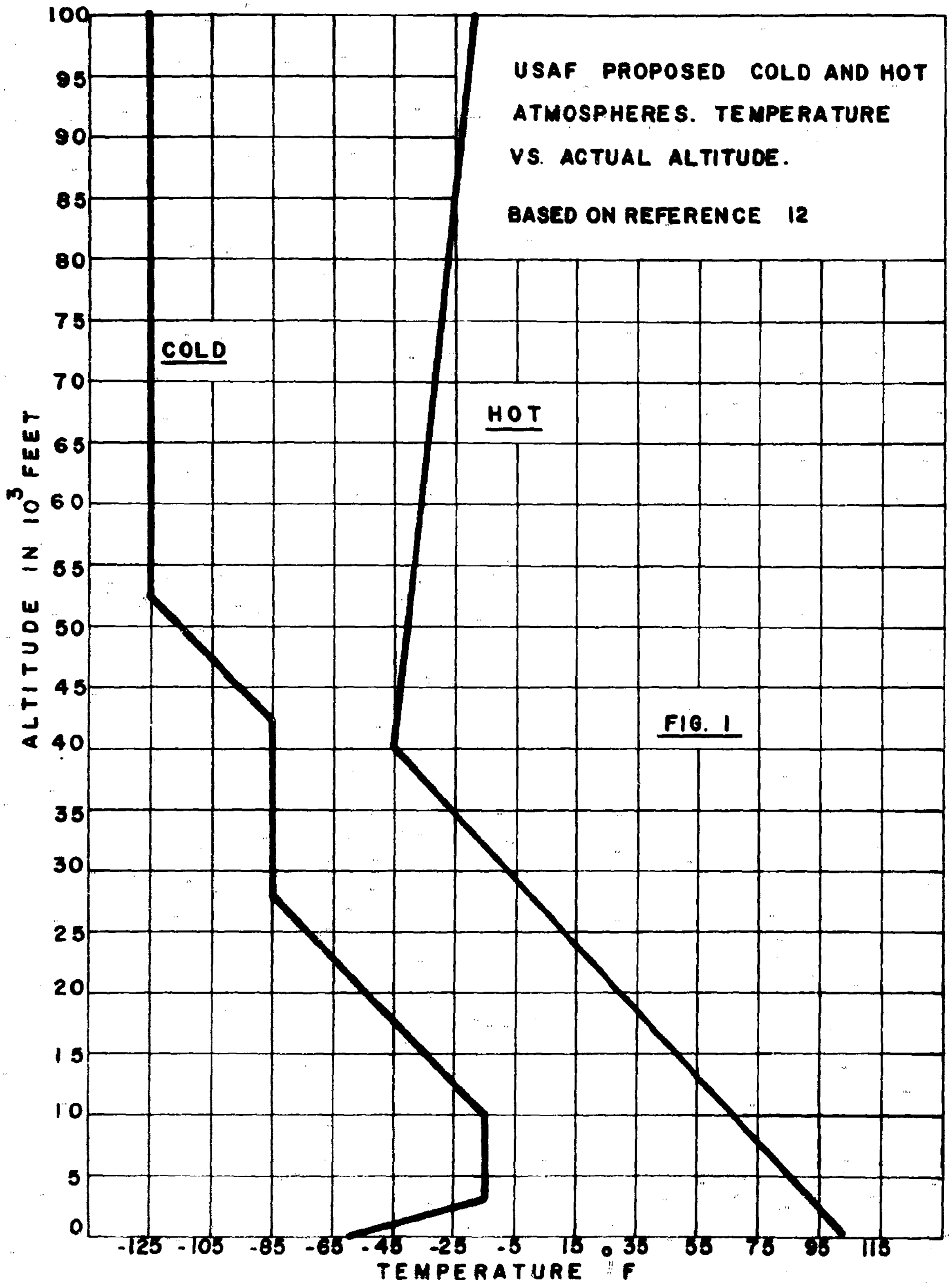
$\epsilon_s = .3 \quad \epsilon_1 = .9 \quad G_s = 410$ BTU/hr ft^2 $\quad \epsilon_2 = .9$

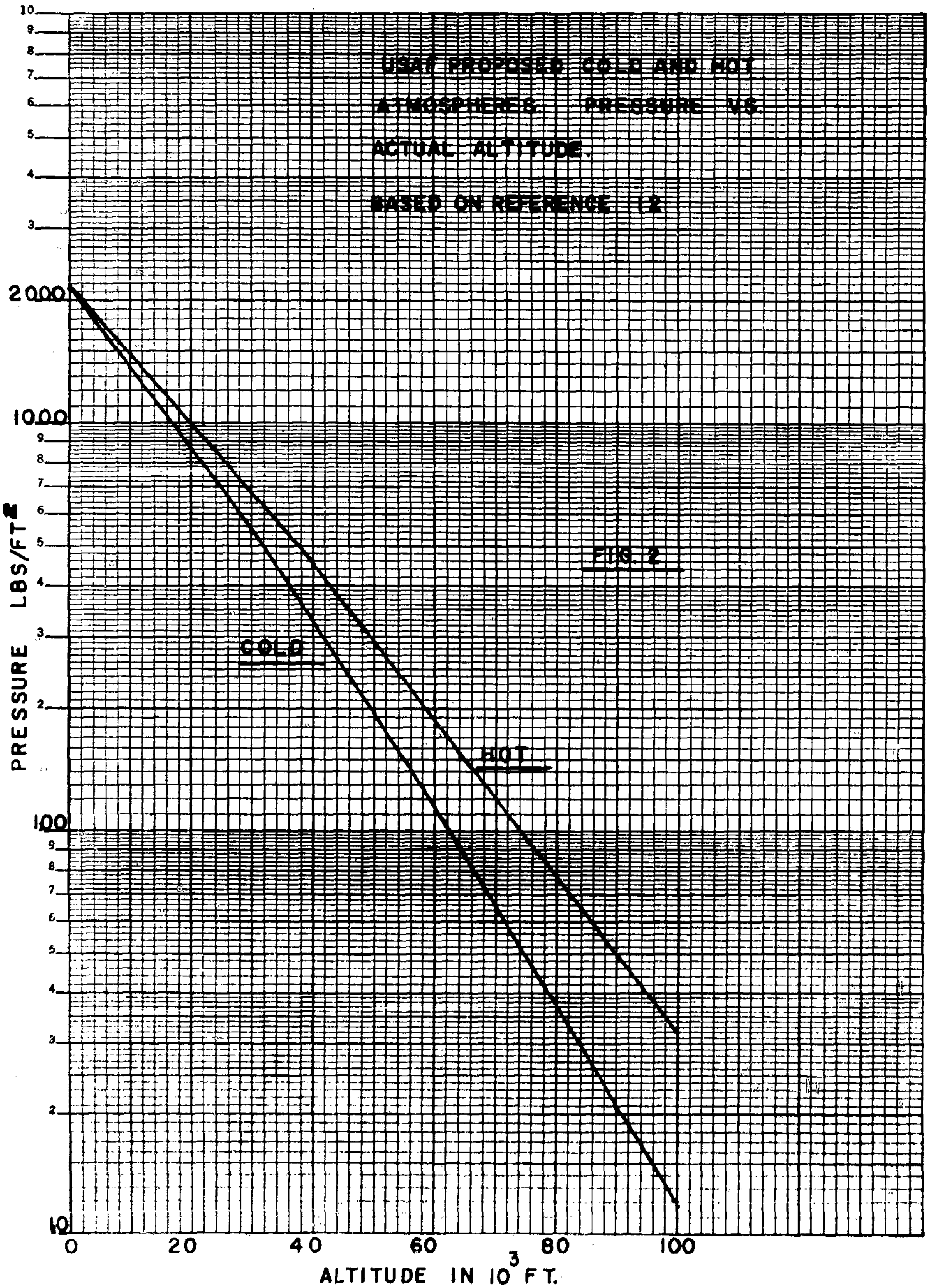
$k = .023$ BTU/hr ft^2 (°F/ft) $\quad Pr = .67$

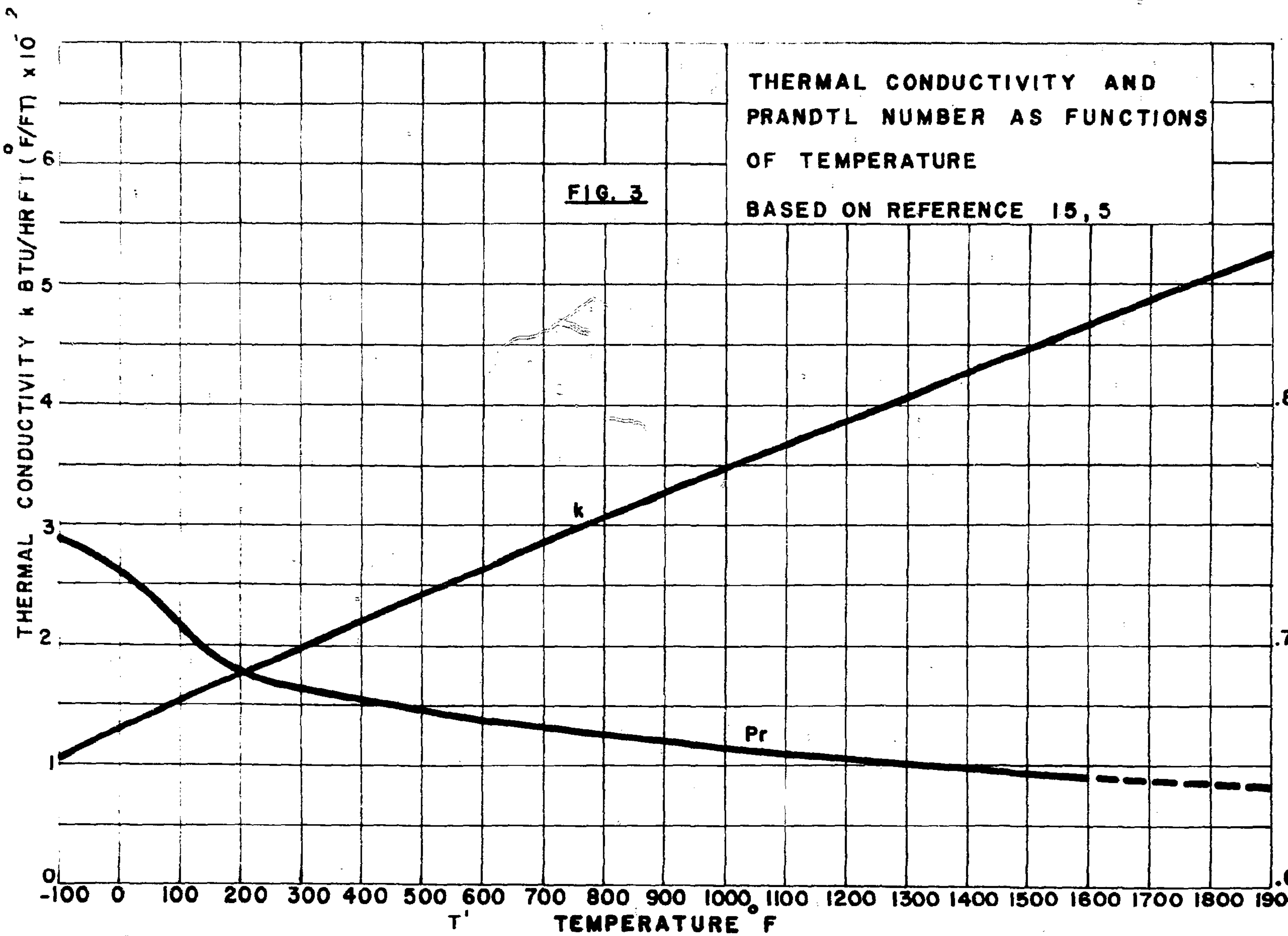
Parameters

Station	A	B	C	D	E	F
Area A_t, ft^2	60	180	220	240	100	240
x or L ft	L=5	x=17.5	x=32.5	x=50	L=6	L=3
q_e=BTU/hr	2000	4000	3000	200,000	-	-
A_q, ft^2	10	200	100	120	-	-
A_c, ft^2	60	180	220	240	-	-
D, ft	2	4	4	4	-	-
M	2.70	2.70	2.70	2.70	3	3
T, °R	469	469	469	469	415	415
P, lb/ft^2	697	697	697	697	449	449
T_β, °R	1065	1065	1065	1065	1065	1065
T, °R	925	925	925	925	910	910
$\frac{Re}{x}$ or $\frac{Re}{L}$	4.5×10^6	4.5×10^6	4.5×10^6	4.5×10^6	3×10^6	3×10^6

Station	A	B	C	D	E	F
$(h)_x$ or $(h)_m$	$(128)_m$	$(92)_x$	$(81)_x$	$(72)_x$	$(88)_m$	$(100)_m$
T_s, °R	1050	1060	1060	1080	1060	1055
T_q, °R	1055	1065	1065	1240	-	-
T_c, °R	1052	1062	1060	1120	1060	1055


USAF PROPOSED COLD AND HOT ATMOSPHERES. TEMPERATURE VS. ACTUAL ALTITUDE.
BASED ON REFERENCE 12
COLD
HOT
FIG. 1
ALTITUDE IN 10^3 FEET
100
95
90
85
80
75
70
65
60
55
50
45
40
35
30
25
20
15
10
5
0
-125 -105 -85 -65 -45 -25 -5 15 35 55 75 95 115
TEMPERATURE °F


USAF PROPOSED COLD AND HOT
ATMOSPHERES. PRESSURE VS.
ACTUAL ALTITUDE.
BASED ON REFERENCE 12
FIG. 2
COLD
HOT
PRESSURE LBS/FT²
ALTITUDE IN 10³ FT.


THERMAL CONDUCTIVITY AND PRANDTL NUMBER AS FUNCTIONS OF TEMPERATURE
BASED ON REFERENCE 15, 5
k
Pr
THERMAL CONDUCTIVITY k BTU/HR FT (°F/FT) x 10^-2
PRANDTL NUMBER Pr
T' TEMPERATURE °F
-100 0 100 200 300 400 500 600 700 800 900 1000 1100 1200 1300 1400 1500 1600 1700 1800 1900
0 1 2 3 4 5 6
.6 .7 .8
FIG. 3

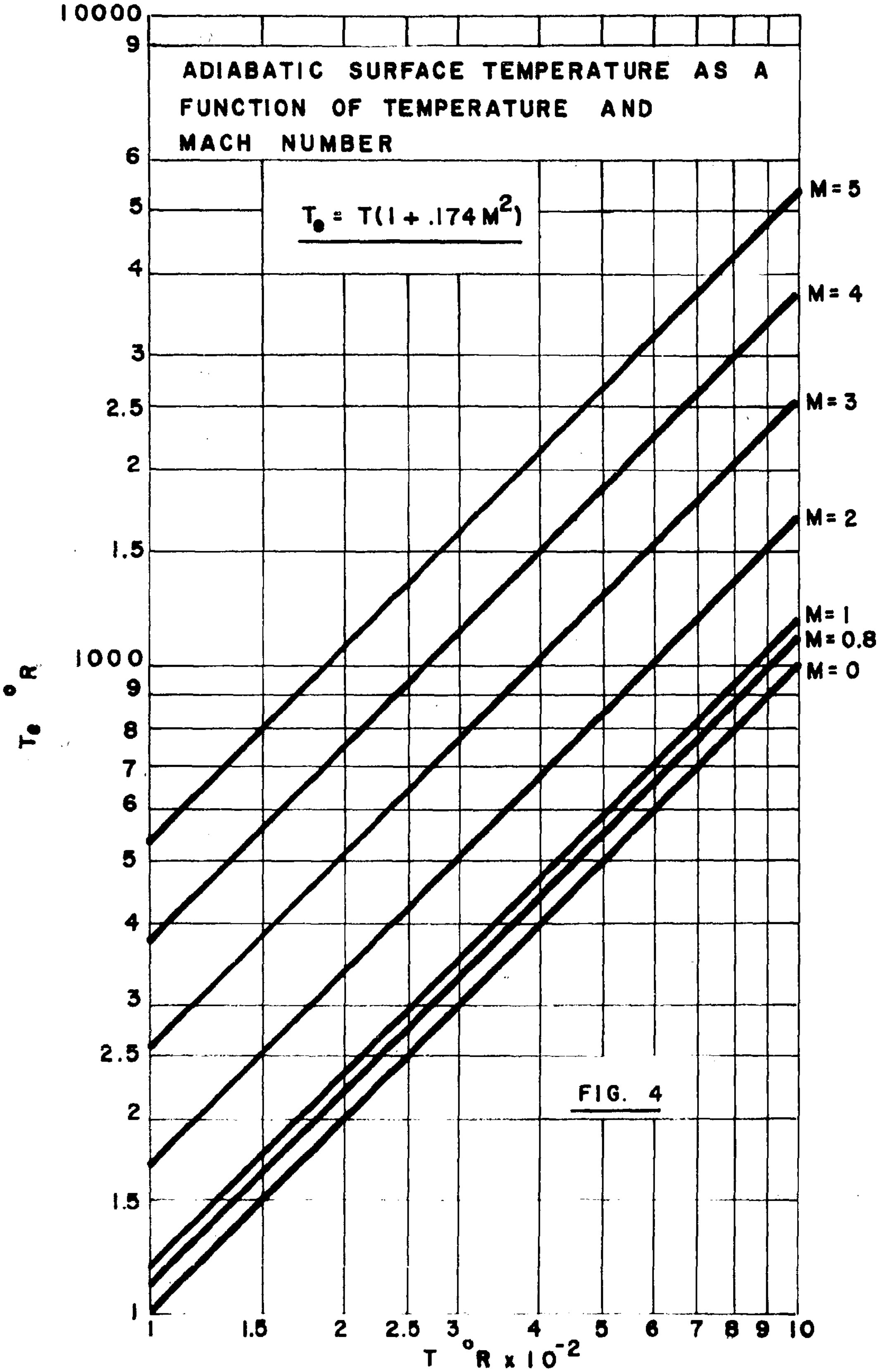

ADIABATIC SURFACE TEMPERATURE AS A FUNCTION OF TEMPERATURE AND MACH NUMBER
$T_e = T(1 + .174 M^2)$
M = 5
M = 4
M = 3
M = 2
M = 1
M = 0.8
M = 0
10000
9
6
5
4
3
2.5
2
1.5
1000
9
8
7
6
5
4
3
2.5
2
1.5
1
T_e °R
1
1.5
2
2.5
3
4
5
6
7
8
9
10
T °R x 10^{-2}

FIG. 4

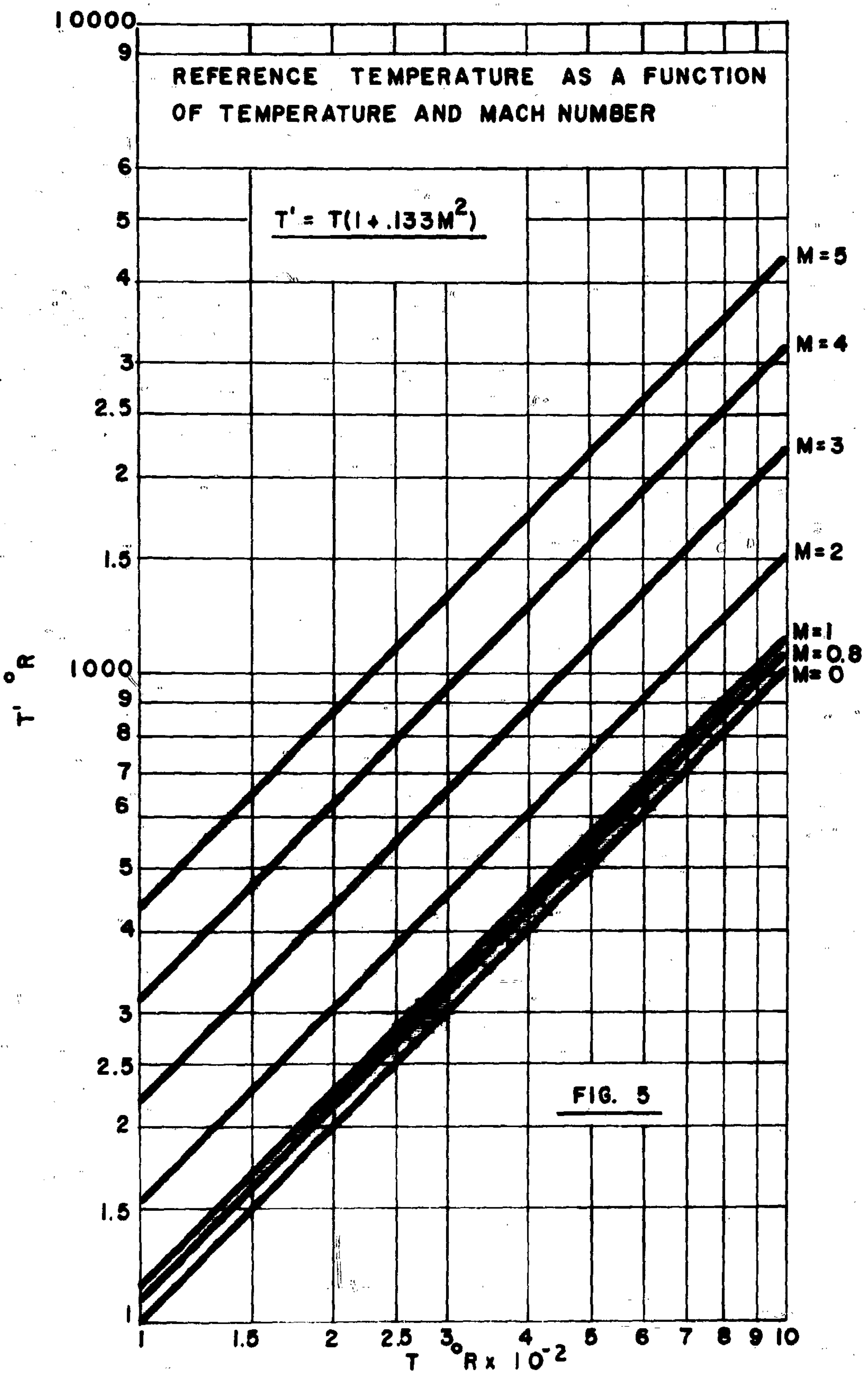
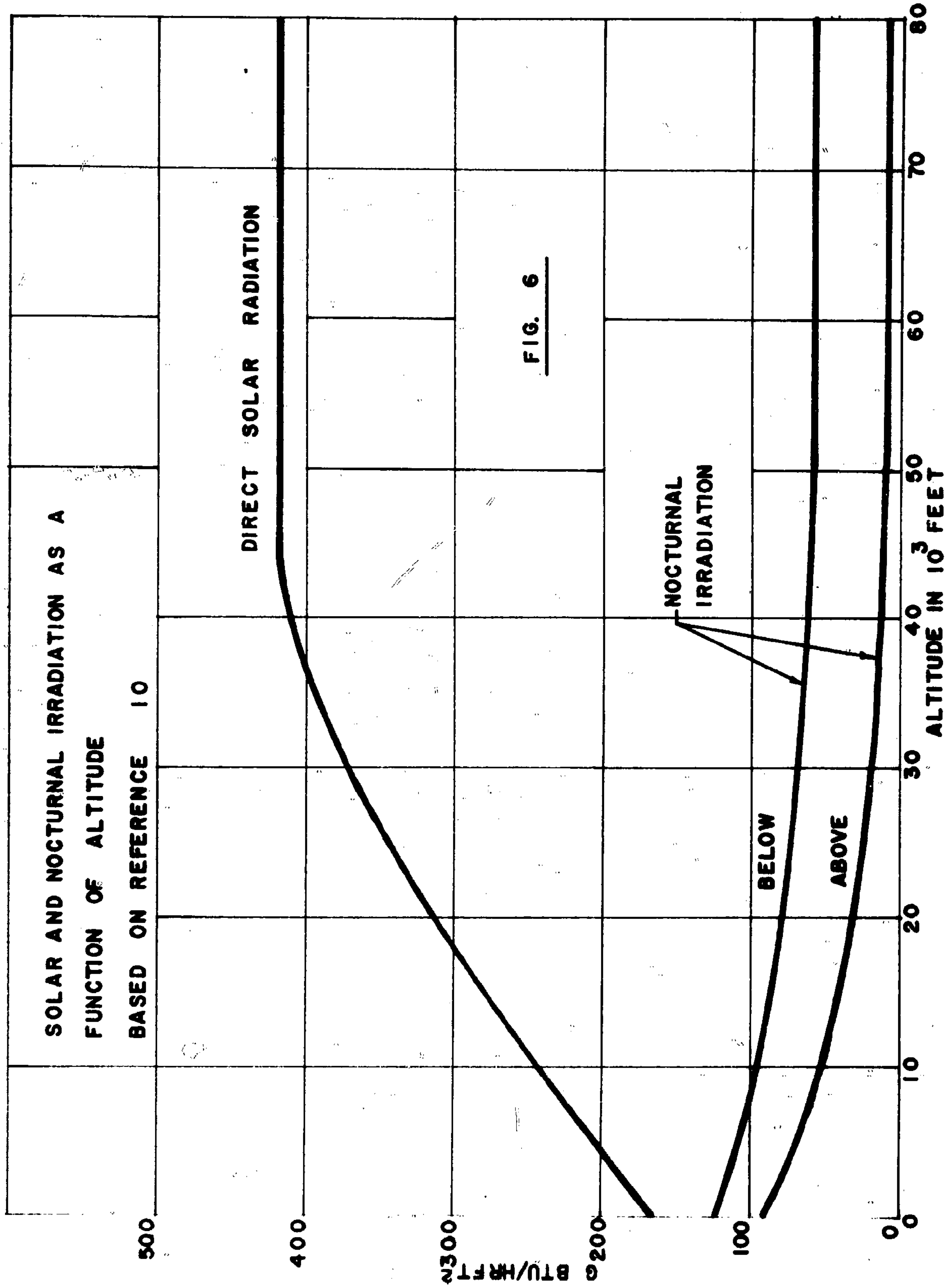
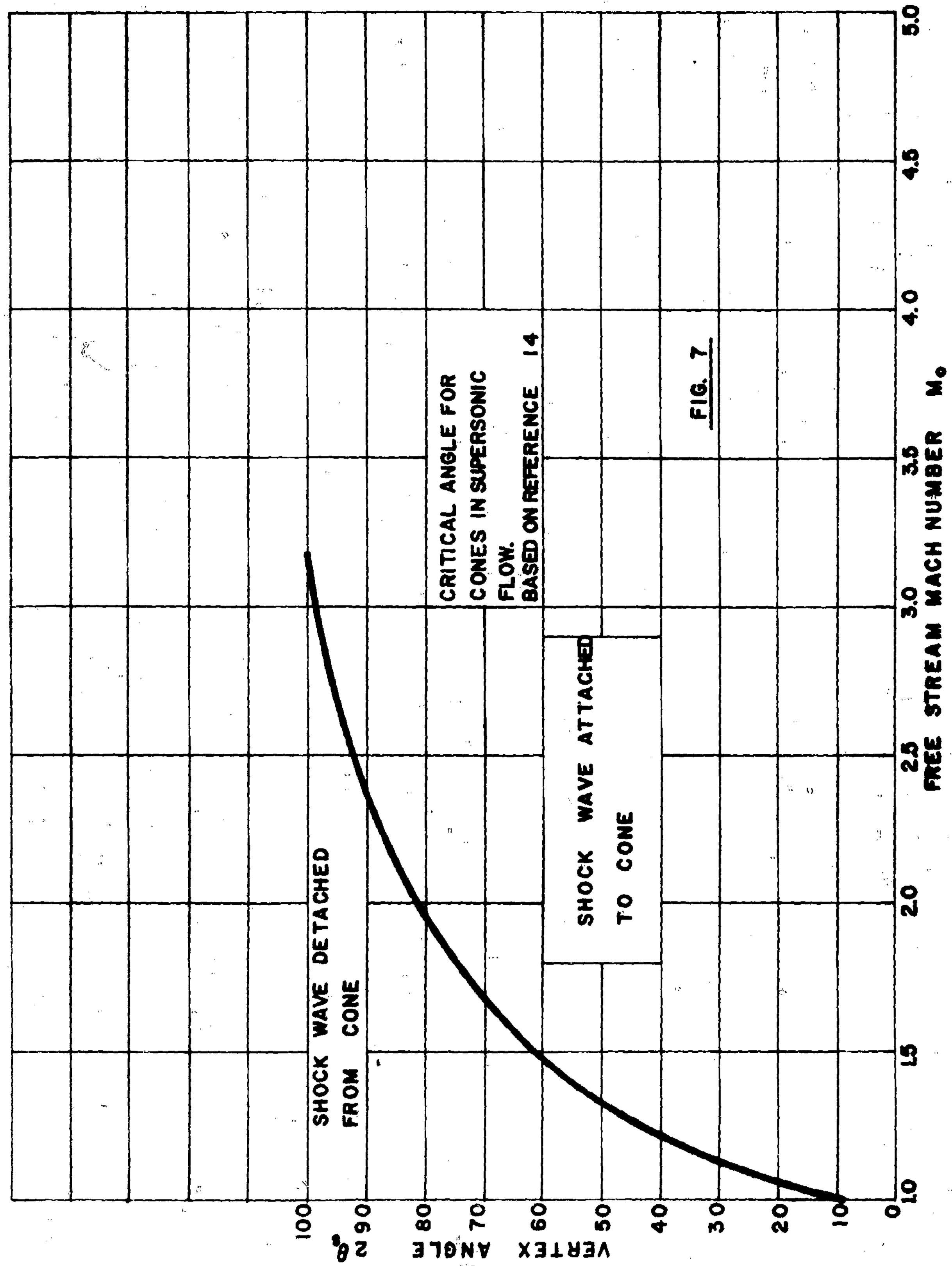

REFERENCE TEMPERATURE AS A FUNCTION OF TEMPERATURE AND MACH NUMBER
T' = T(1 + .133M²)
M=5
M=4
M=3
M=2
M=1
M=0.8
M=0
T' °R
T °R x 10⁻²
10000
1000
1
1.5
2
2.5
3
4
5
6
7
8
9
10

FIG. 5



SOLAR AND NOCTURNAL IRRADIATION AS A FUNCTION OF ALTITUDE BASED ON REFERENCE 10

FIG. 6


CRITICAL ANGLE FOR
CONES IN SUPERSONIC
FLOW.
BASED ON REFERENCE 14
FIG. 7
SHOCK WAVE DETACHED
FROM CONE
SHOCK WAVE ATTACHED
TO CONE
FREE STREAM MACH NUMBER Mo
VERTEX ANGLE 2θs
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
5.0
0
10
20
30
40
50
60
70
80
90
100

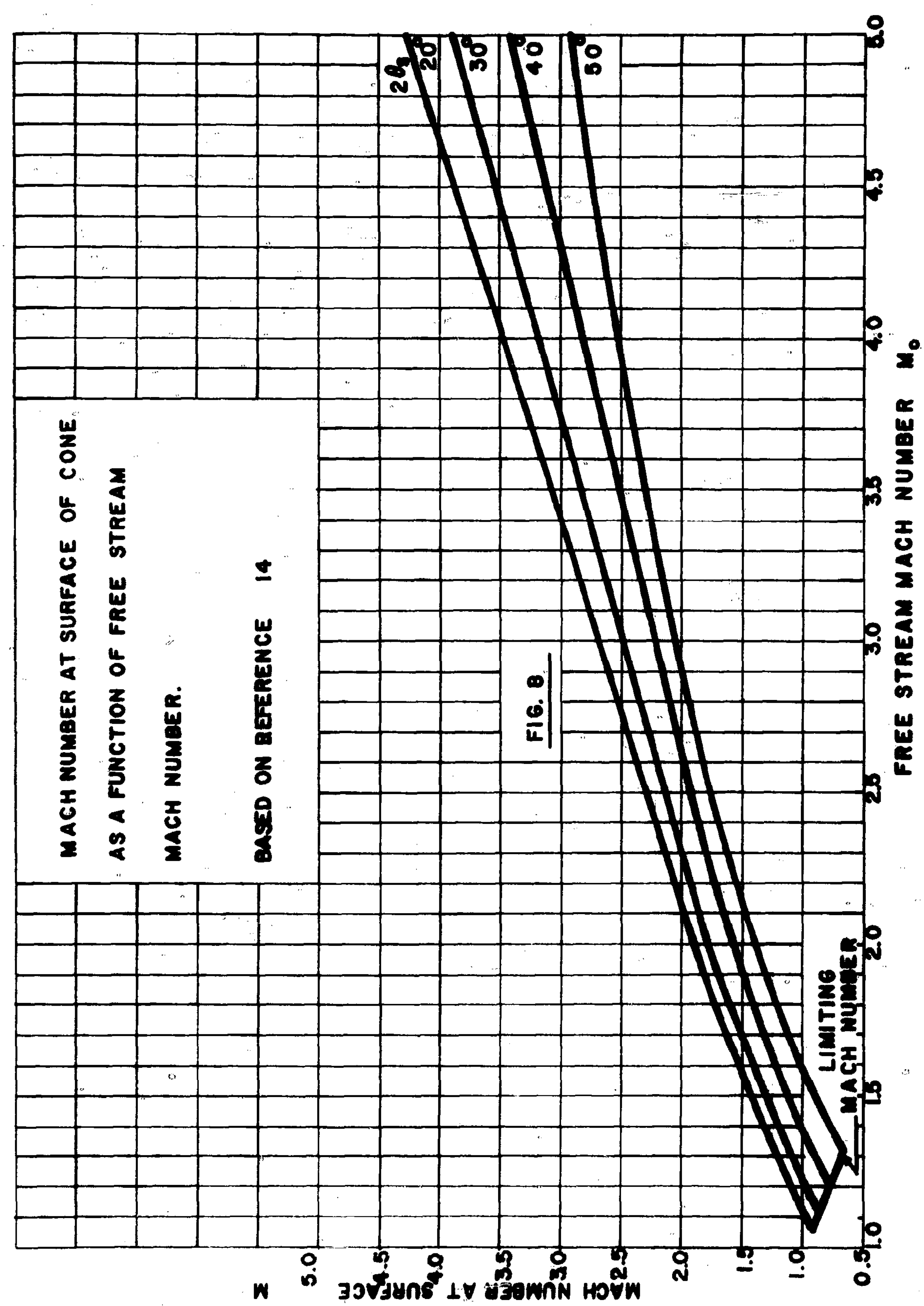

MACH NUMBER AT SURFACE OF CONE
AS A FUNCTION OF FREE STREAM
MACH NUMBER.
BASED ON REFERENCE 14
20°
30°
40°
50°
LIMITING
MACH NUMBER
FREE STREAM MACH NUMBER M_o
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
5.0
MACH NUMBER AT SURFACE M
0.5
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
5.0

FIG. 8

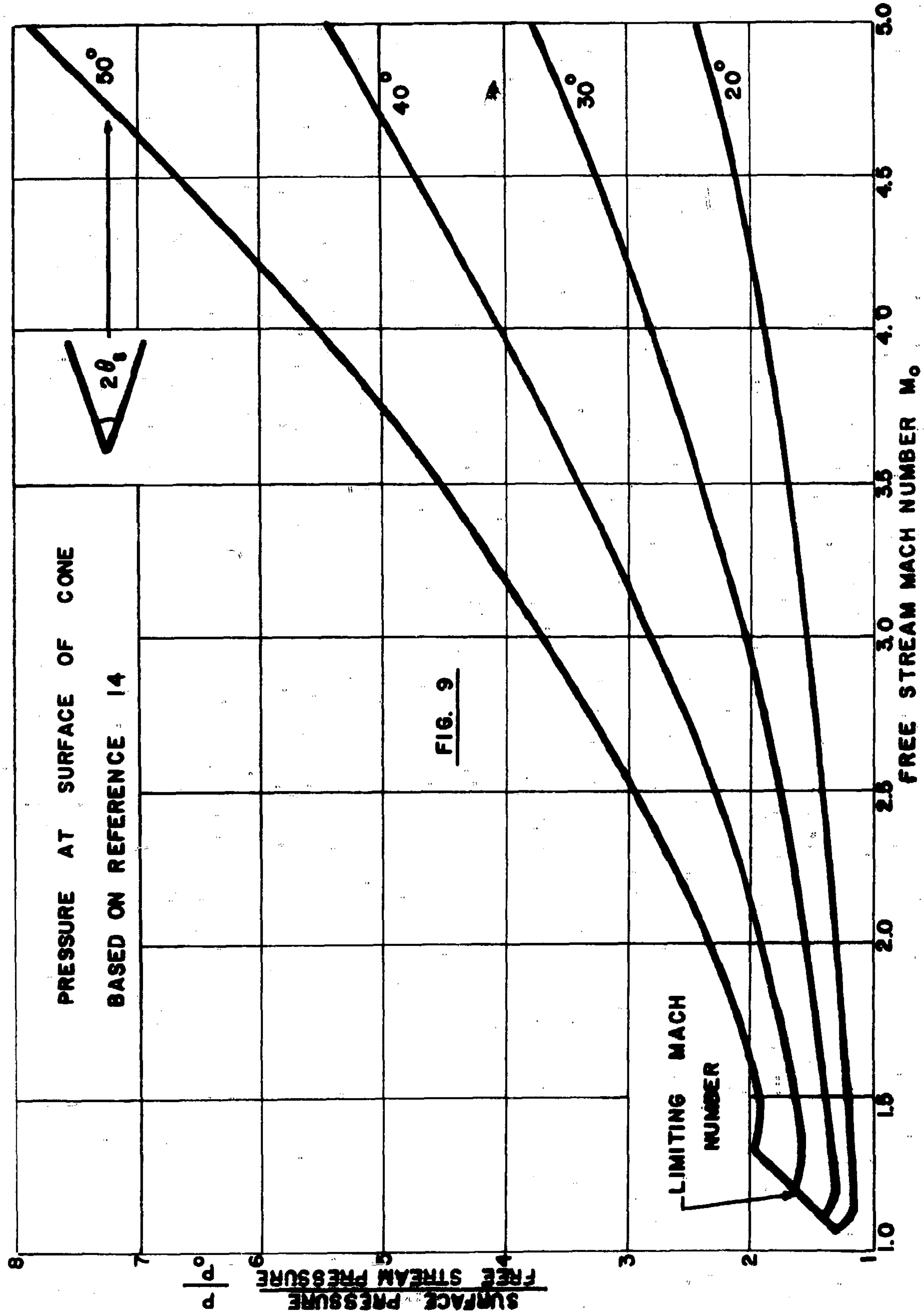

PRESSURE AT SURFACE OF CONE
BASED ON REFERENCE 14
$2\theta_s$
50°
40°
30°
20°
LIMITING MACH NUMBER
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
5.0
FREE STREAM MACH NUMBER M_o
1
2
3
4
5
6
7
8
$\frac{p}{p_o}$
SURFACE PRESSURE / FREE STREAM PRESSURE

FIG. 9

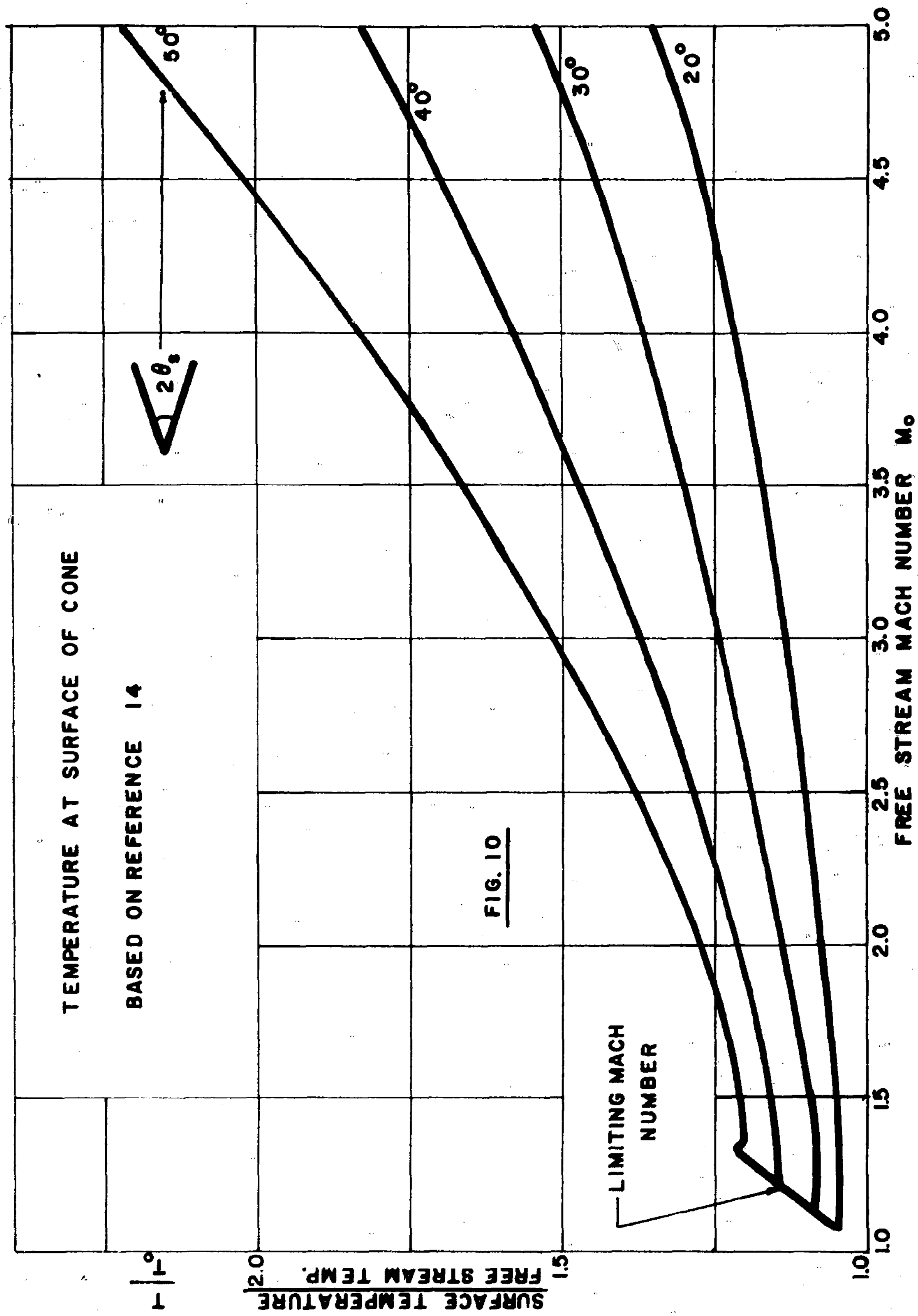
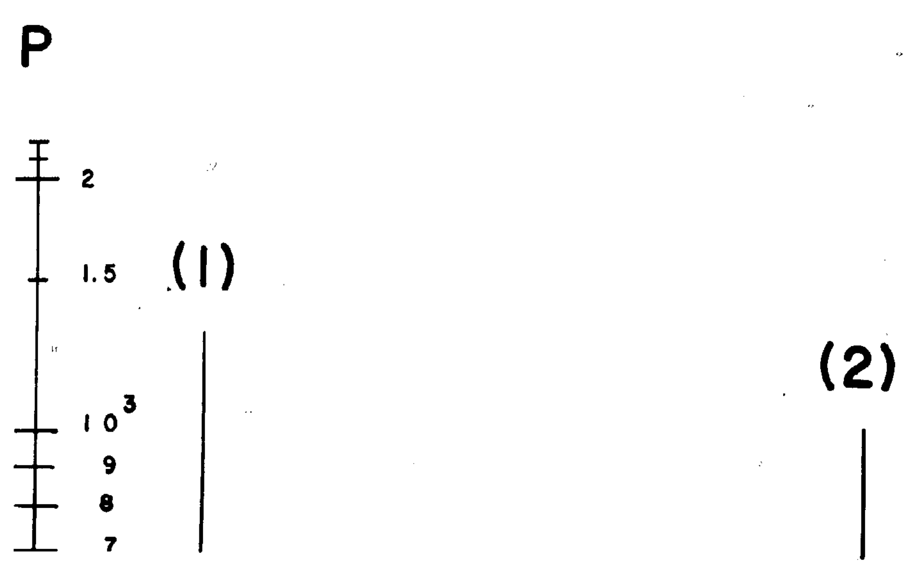

TEMPERATURE AT SURFACE OF CONE
BASED ON REFERENCE 14
LIMITING MACH NUMBER
$2\theta_s$
50°
40°
30°
20°
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
5.0
FREE STREAM MACH NUMBER M_0
$\frac{T}{T_0}$
$\frac{\text{SURFACE TEMPERATURE}}{\text{FREE STREAM TEMP.}}$

FIG. 10



P
2
1.5
(1)
(2)
10 3
9
8
7

$\underline{Re}$

X

(2)

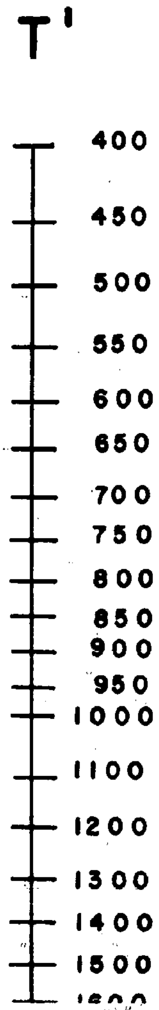
M

λ

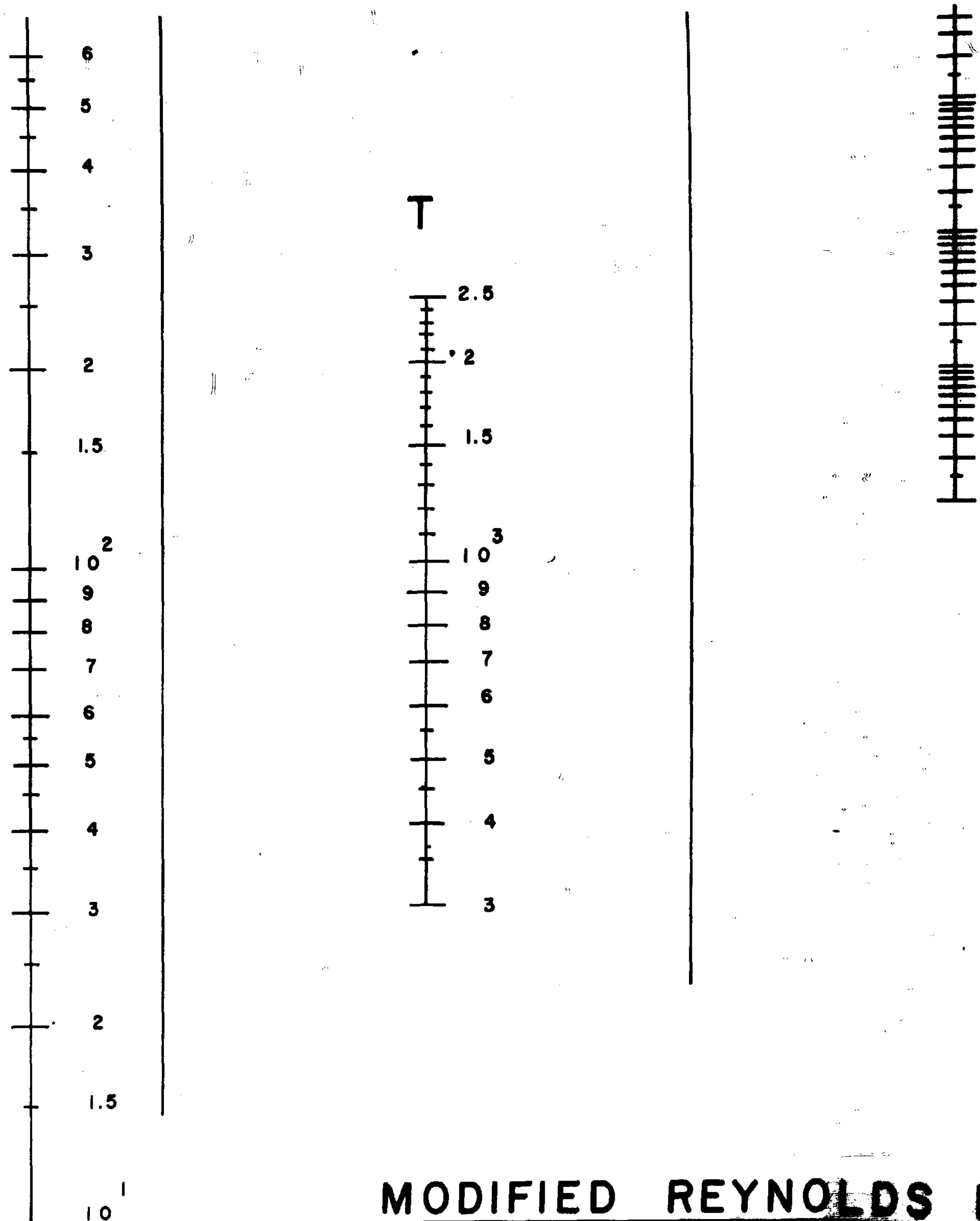
10^8

6

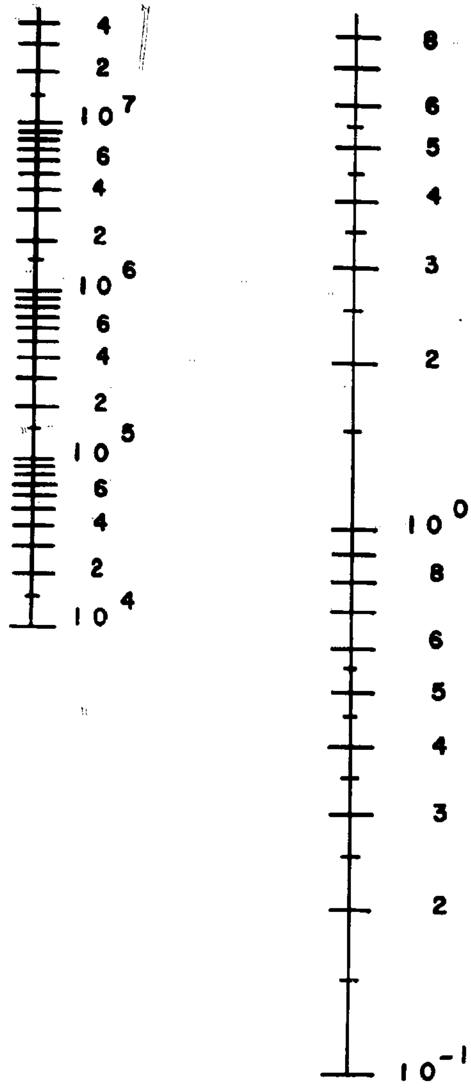
10^1


T'
400
450
500
550
600
650
700
750
800
850
900
950
1000
1100
1200
1300
1400
1500

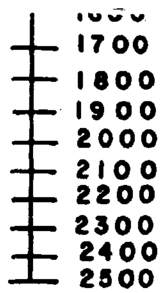
3

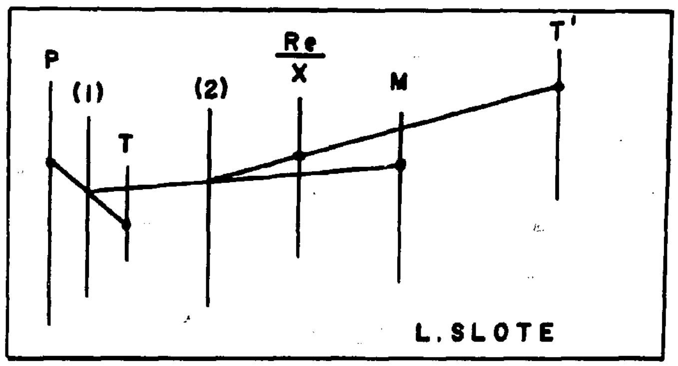


MODIFIED REYNOLDS



REYNOLDS MODULUS FOR AIR





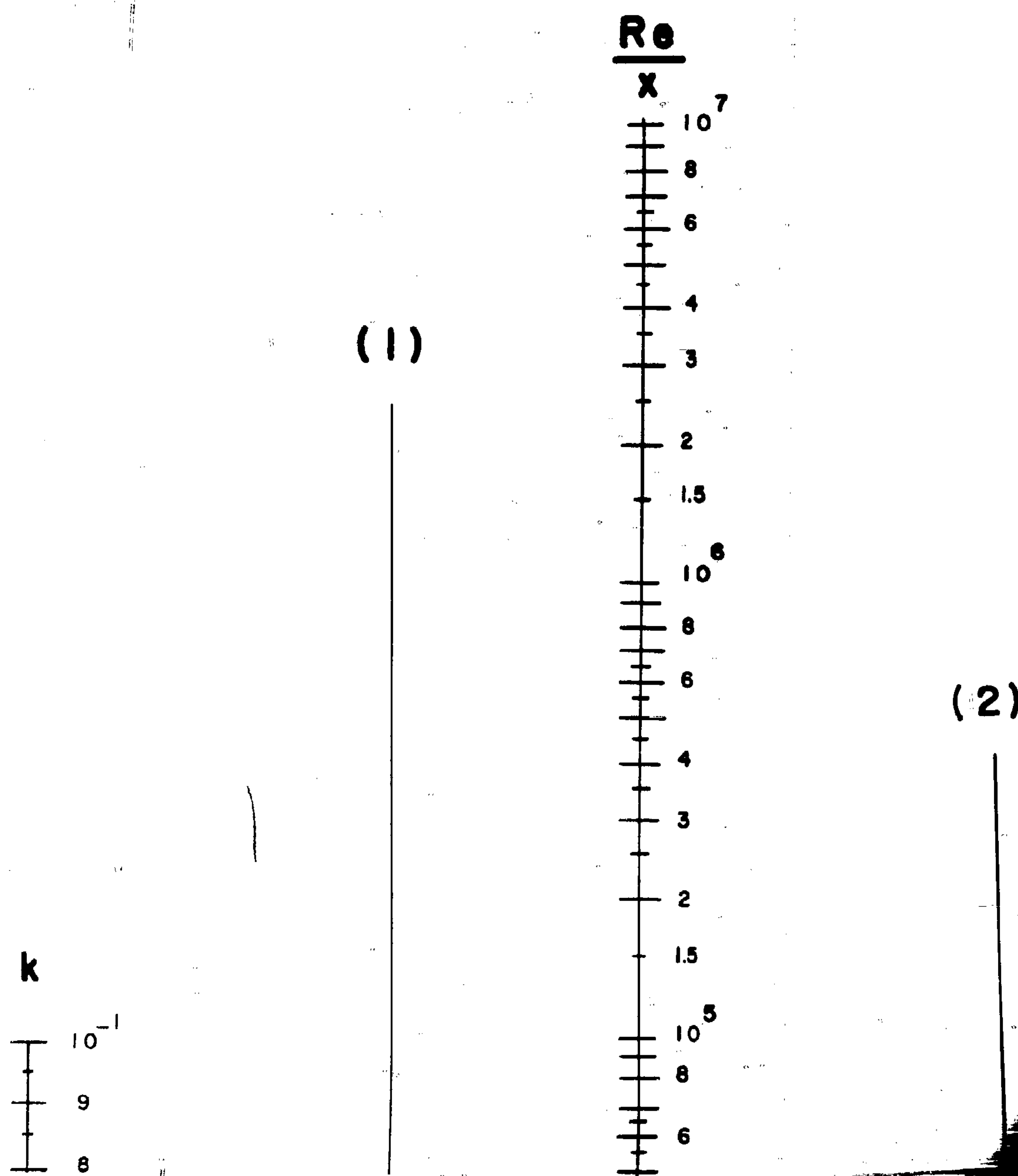
PROCEDURE

a FOR A GIVEN P AND T FIND INTERSECTION ON (1)

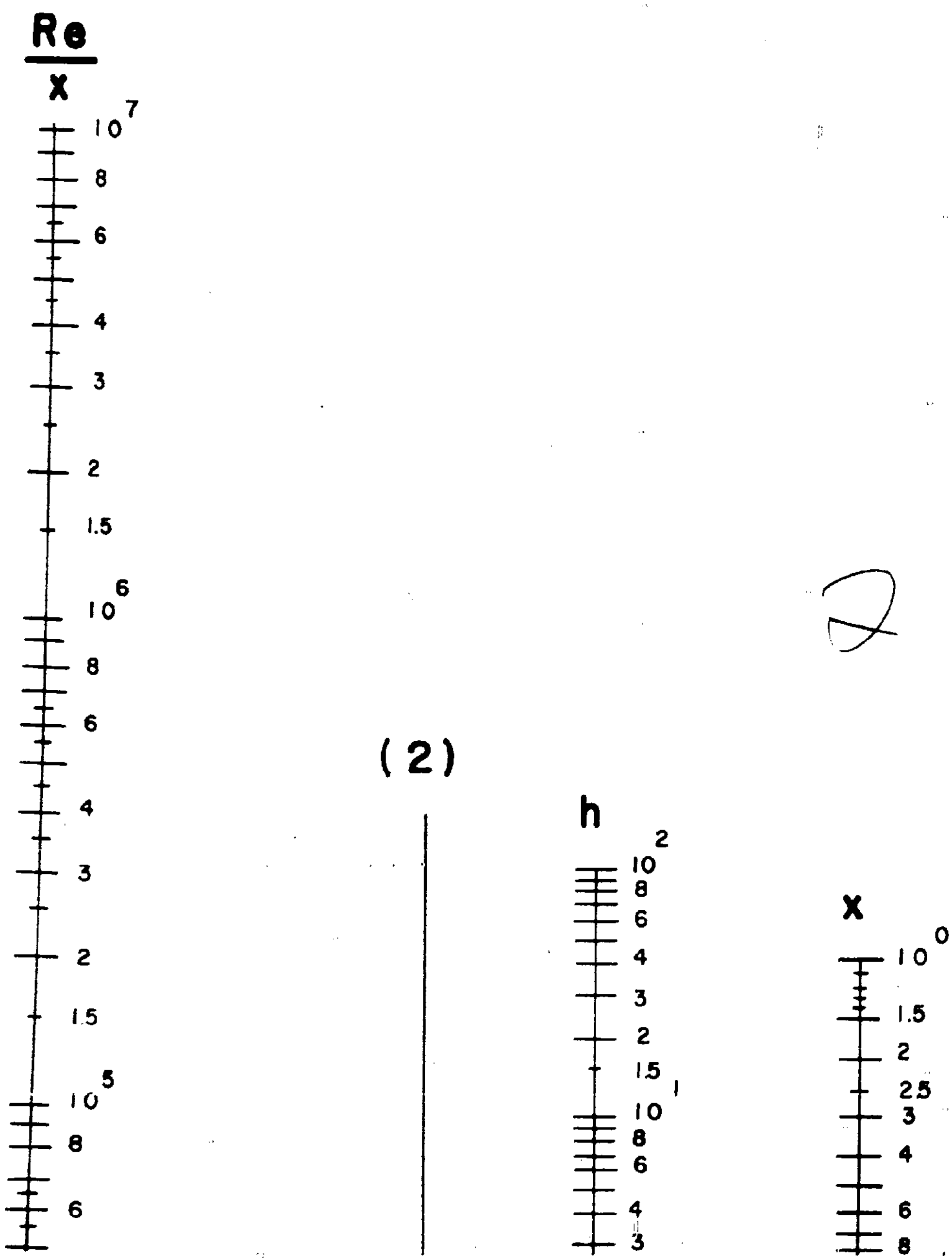
b ALIGN INTERSECTION ON (1) WITH GIVEN M AND FIND INTERSECTION ON (2)

c ALIGN INTERSECTION ON (2) WITH T' TO DETERMINE $\frac{Re}{X}$

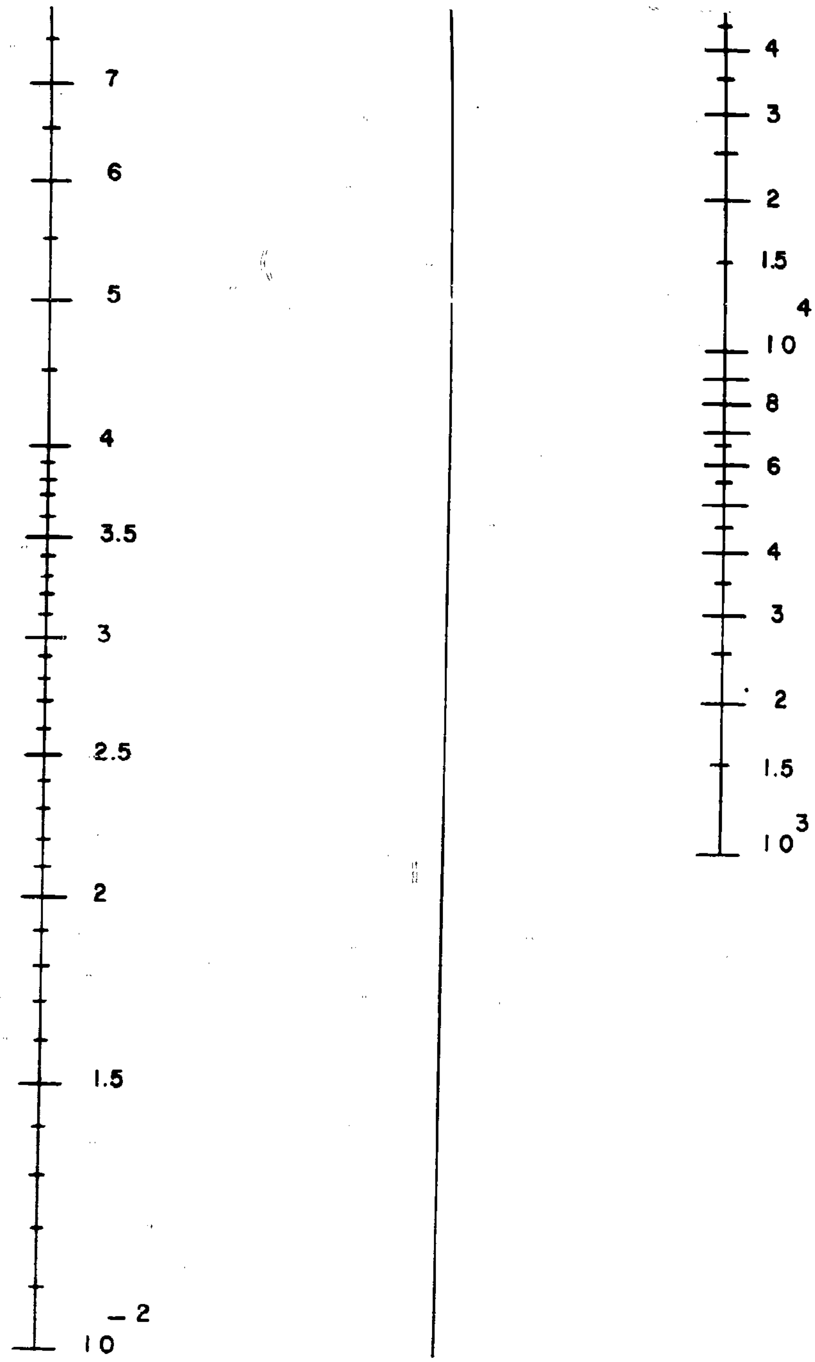
FIG. 11



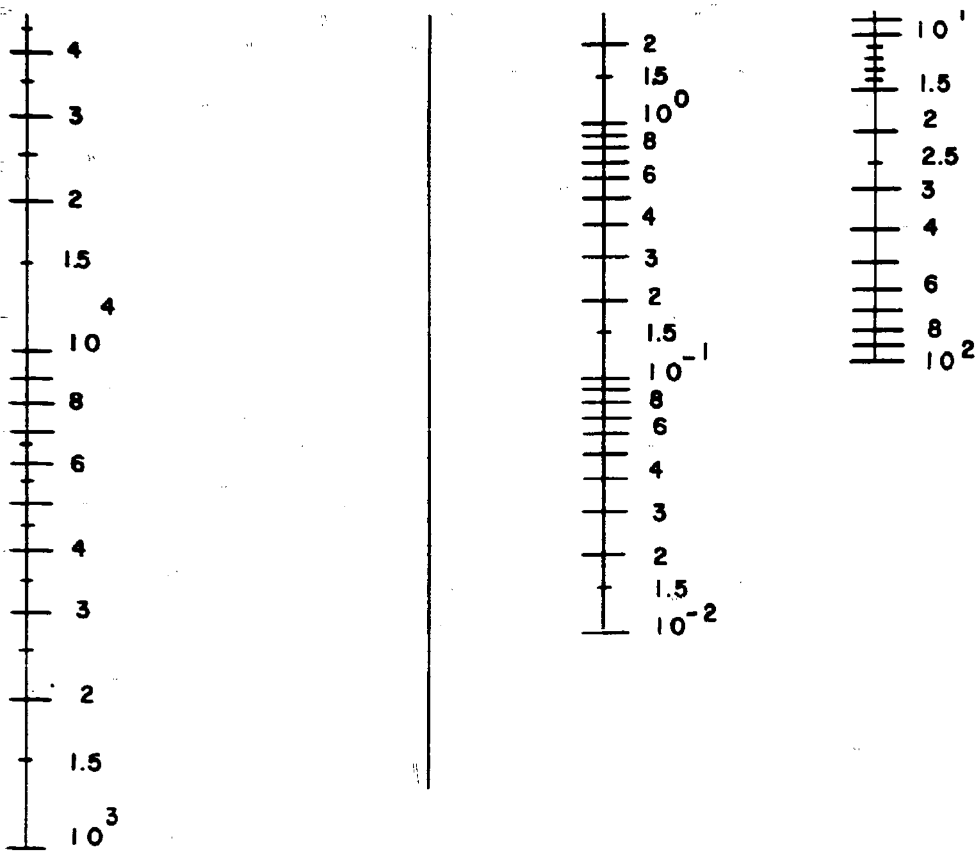
Re
x
10^7
8
6
4
3
2
1.5
10^6
8
6
4
3
2
1.5
10^5
8
6
(1)
(2)
k
10^{-1}
9
8



Re/x
10^7
8
6
4
3
2
1.5
10^6
8
6
4
3
2
1.5
10^5
8
6
(2)
h
10^2
8
6
4
3
2
1.5
10^1
8
6
4
3
x
10^0
1.5
2
2.5
3
4
6
8

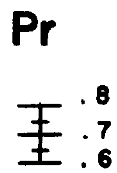


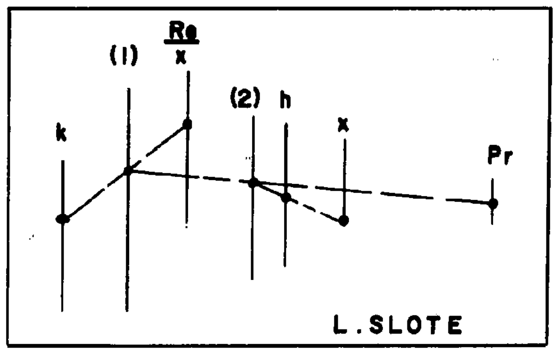
HEAT TRANSFE



EAT TRANSFER IN LAMINAR FLOW

4





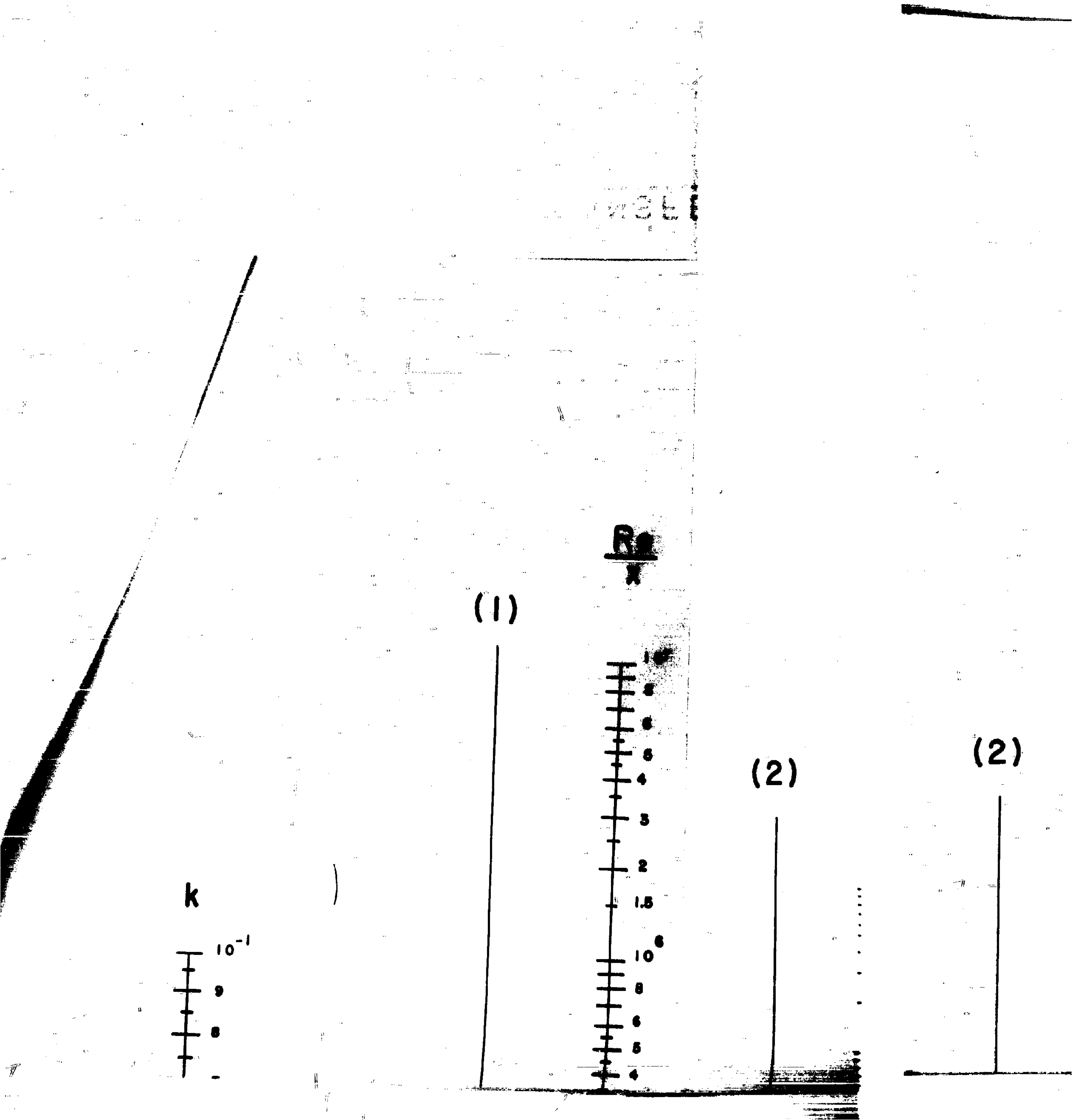
PROCEDURE

a FOR A GIVEN k AND $\frac{Re}{x}$ FIND INTERSECTION ON (1)

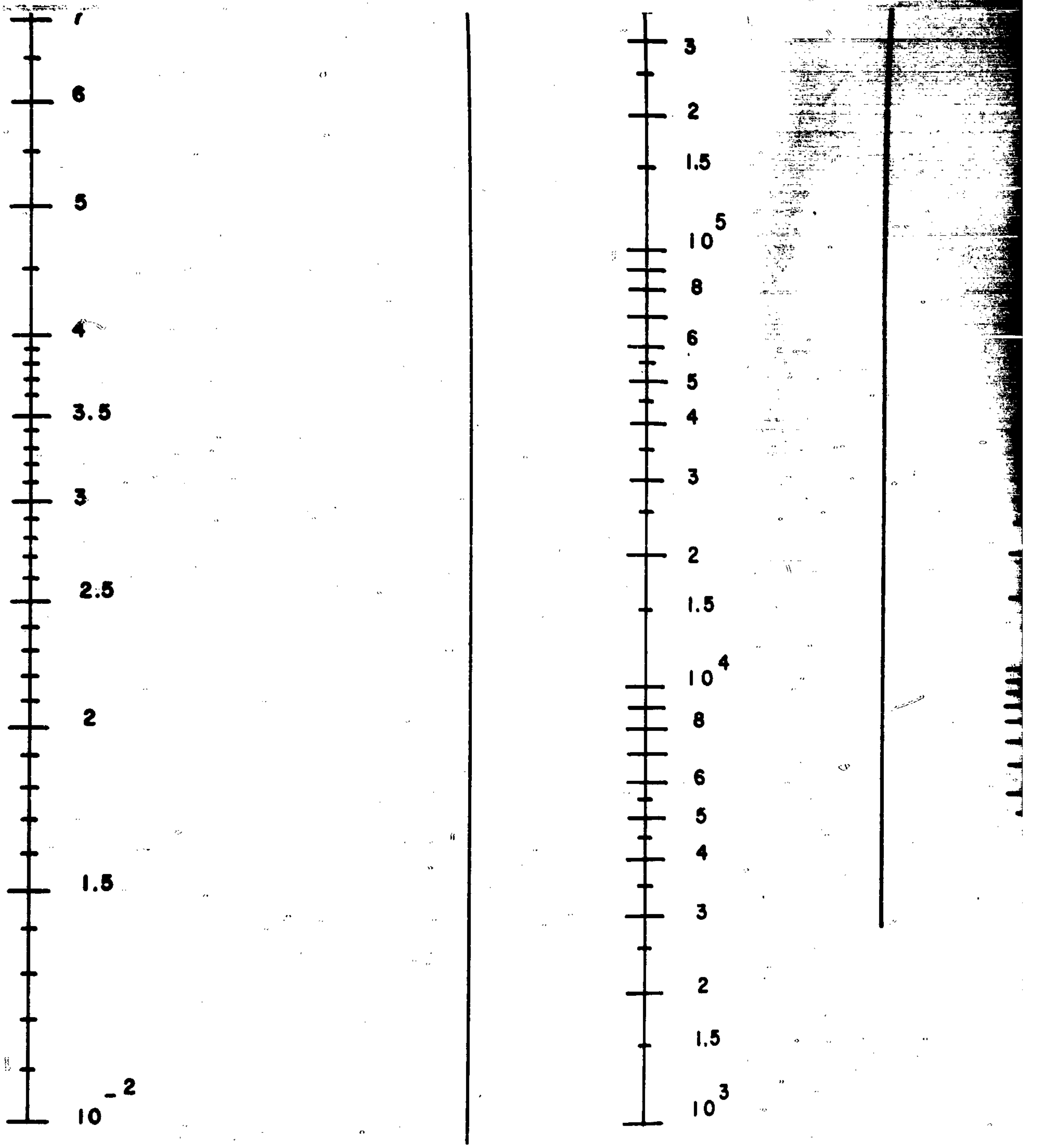
b ALIGN INTERSECTION ON (1) WITH GIVEN Pr AND FIND INTERSECTION ON (2)

c ALIGN INTERSECTION ON (2) WITH x TO DETERMINE h

FIG. 12

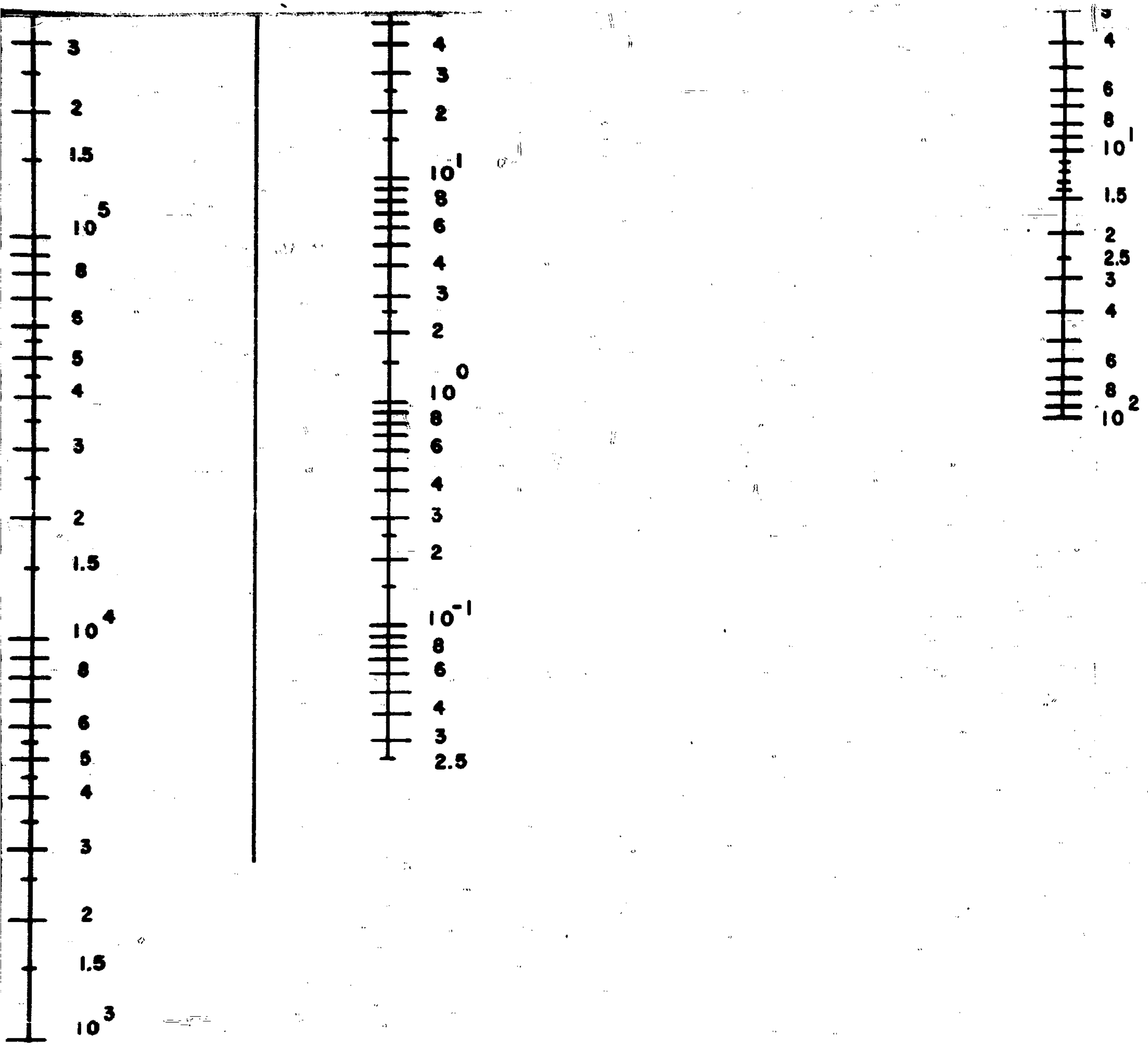


k
10^{-1}
9
8
(1)
$\frac{Re}{x}$
8
6
5
4
3
2
1.5
10^{6}
8
6
5
4
(2)
(2)



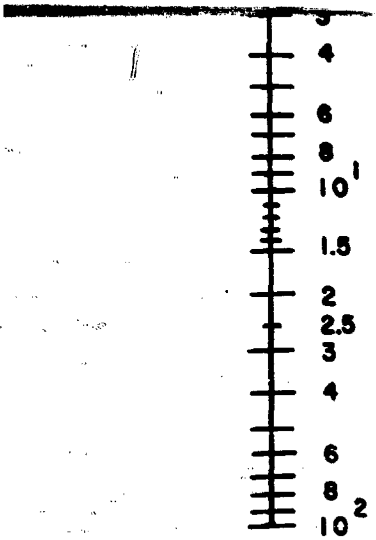
HEAT TRANSFER IN TU

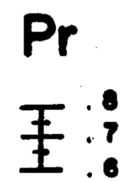
3

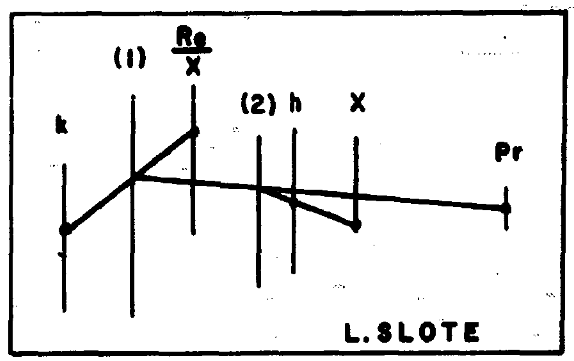


NSFER IN TURBULENT FLOW

41







PROCEDURE

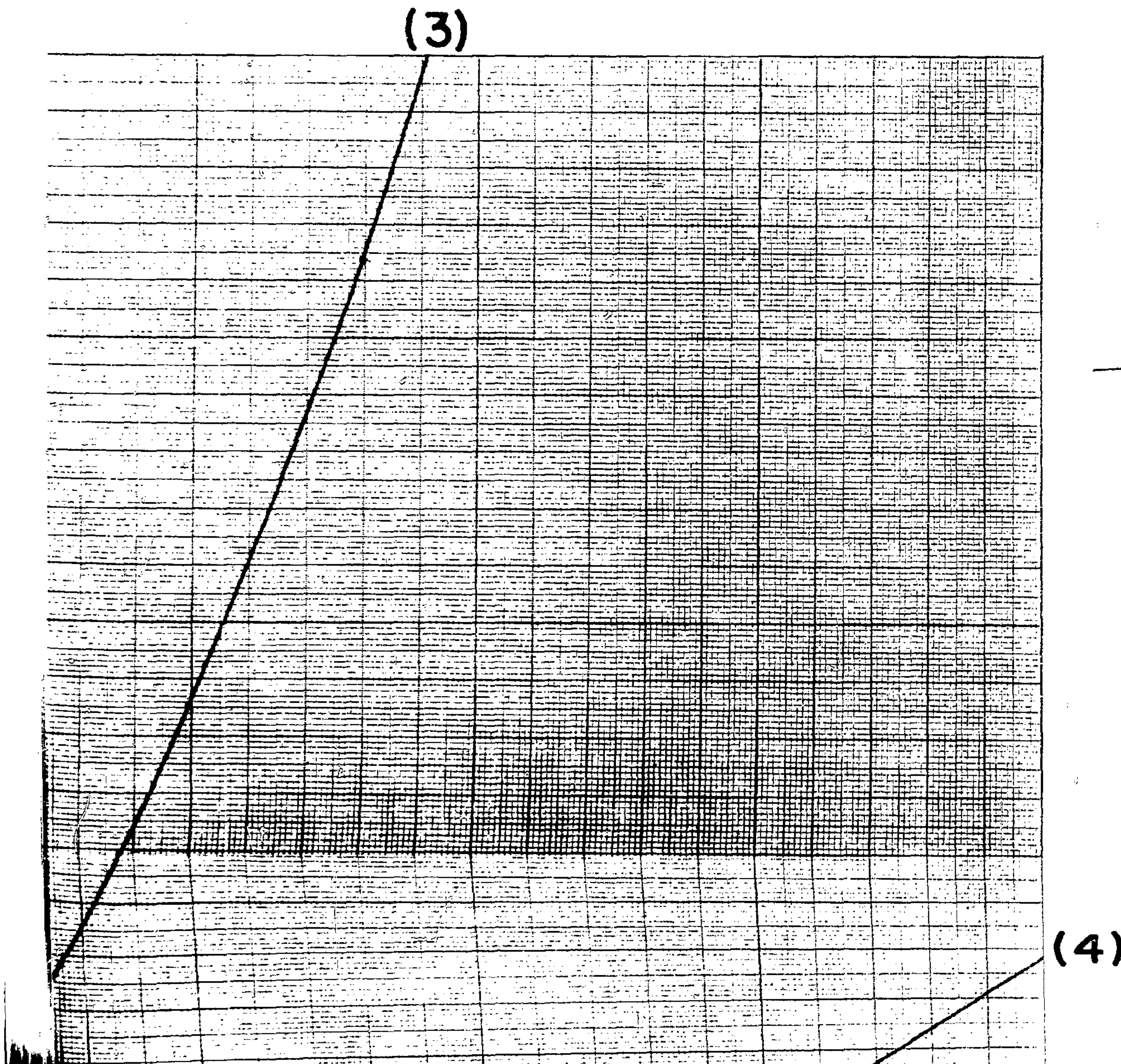
a FOR A GIVEN k AND $\frac{Re}{X}$ FIND INTERSECTION ON (1)

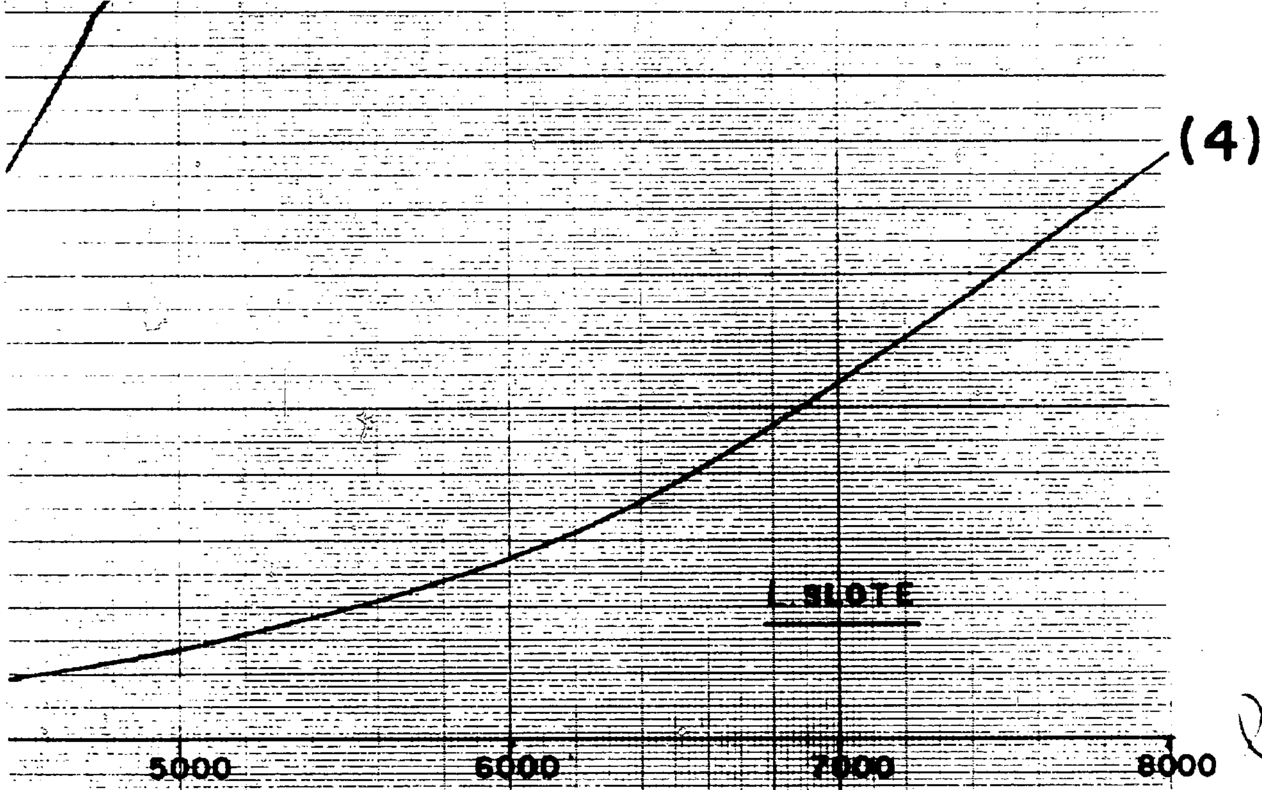
b ALIGN INTERSECTION ON (1) WITH GIVEN Pr AND FIND INTERSECTION ON (2)

c ALIGN INTERSECTION ON (2) WITH X TO DETERMINE h

FIG. 13

5


(3)
(4)



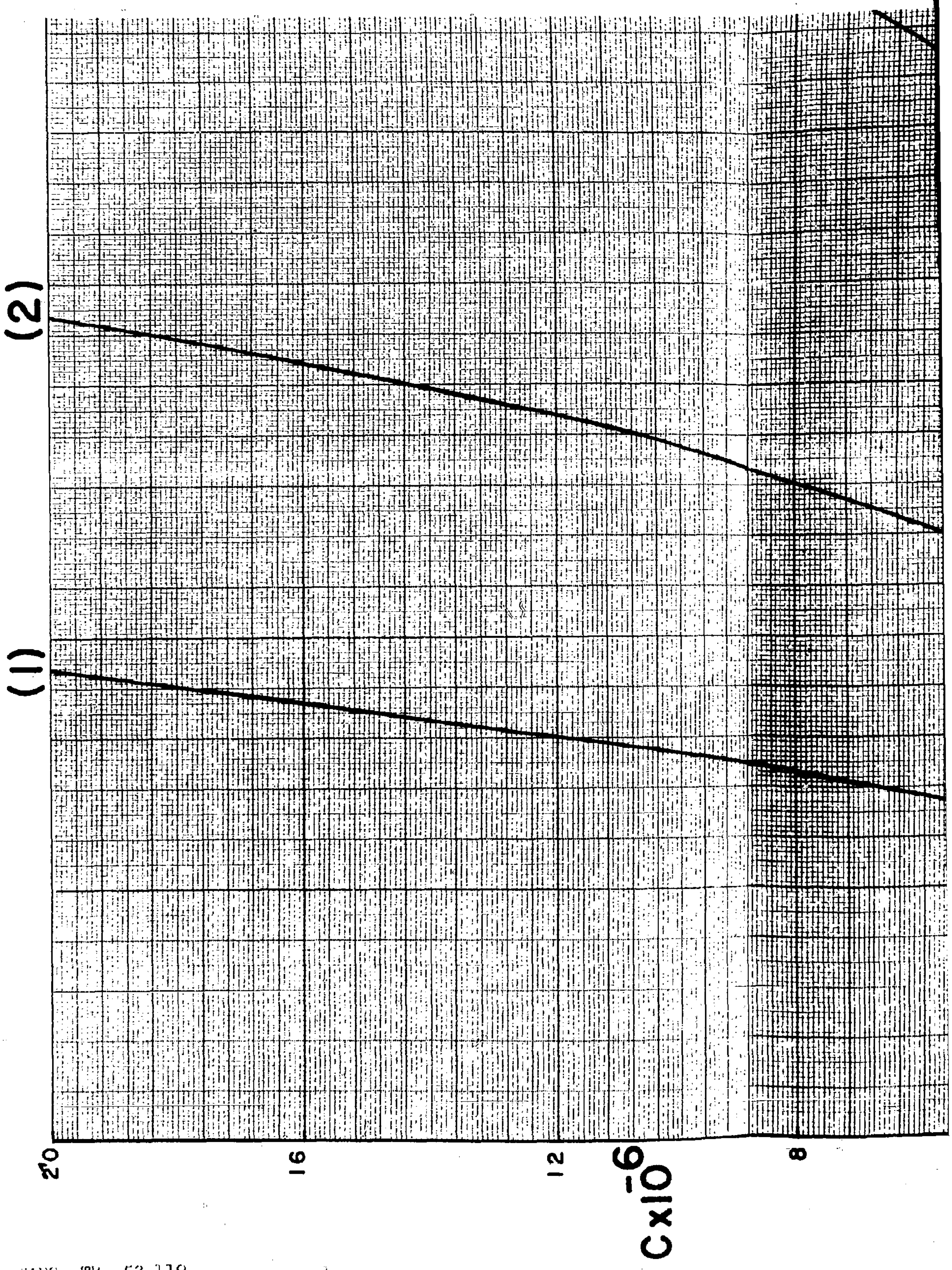
NOTE

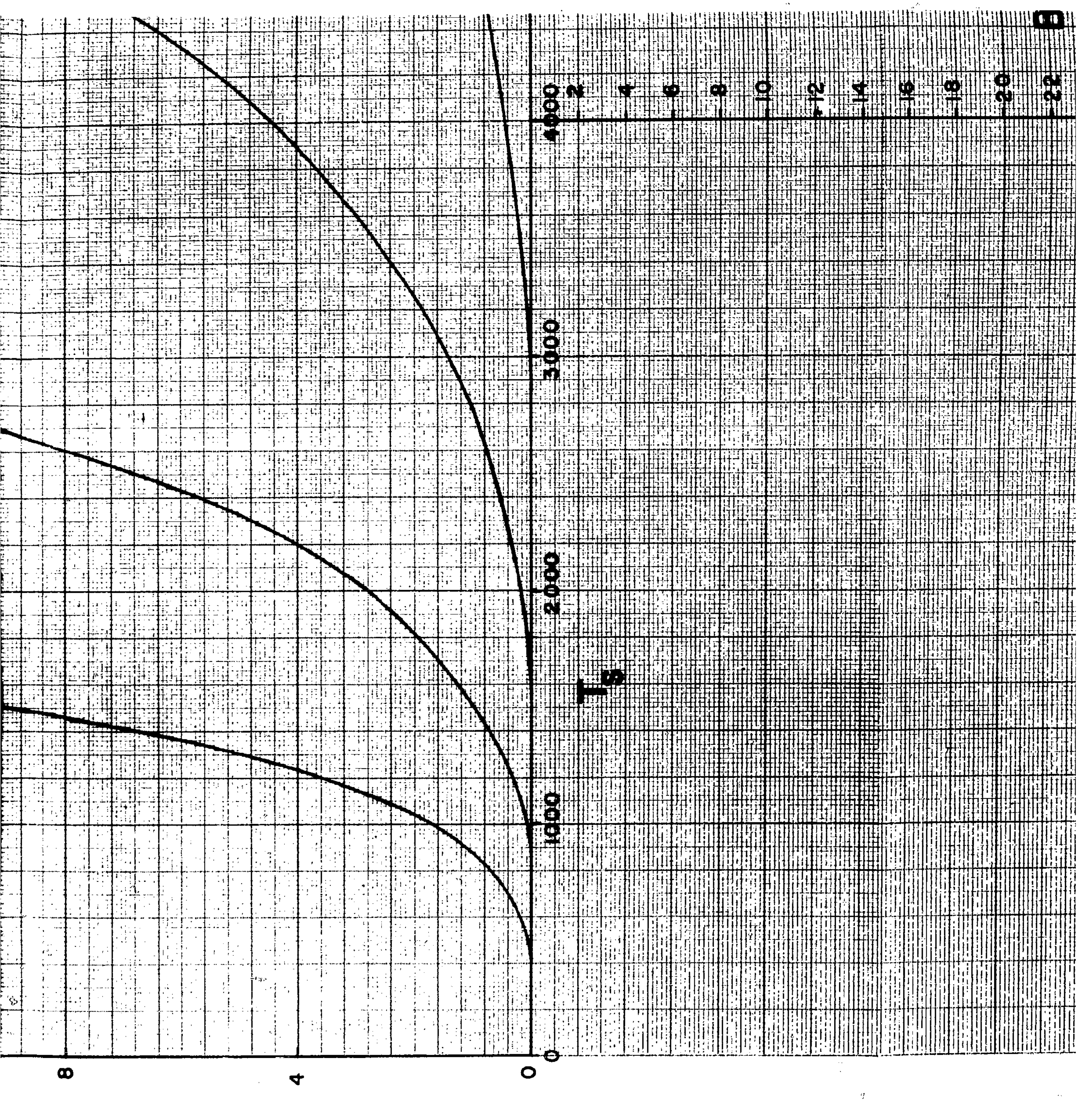
WITH CURVE	MULTIPLY SCALE $C \times 10^6$ BY	MULTIPLY SCALE B BY
(1)	.001	.1
(2)	.01	1
(3)	.1	10
(4)	1	100

NOTE

WITH CURVE	MULTIPLY SCALE $C \times 10^6$ BY	MULTIPLY SCALE B BY
(1)	.001	.1
(2)	.01	1
(3)	.1	10
(4)	1	100

FIG. 14


(2)
(1)
20
16
12
8
Cx10^{-6}



0
4
8
1000
2000
3000
4000
T_s
2
4
6
8
10
12
14
16
18
20
22

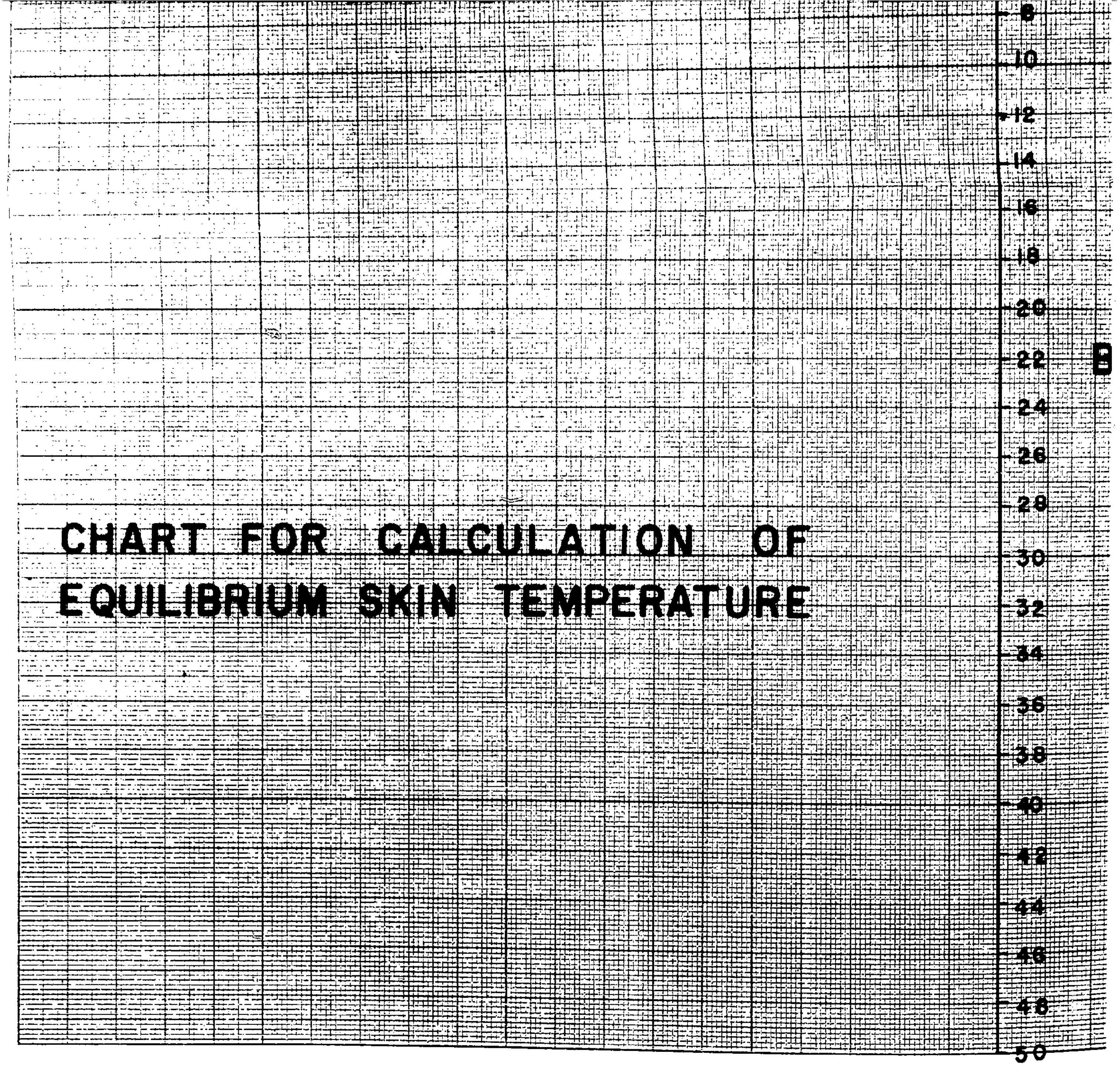
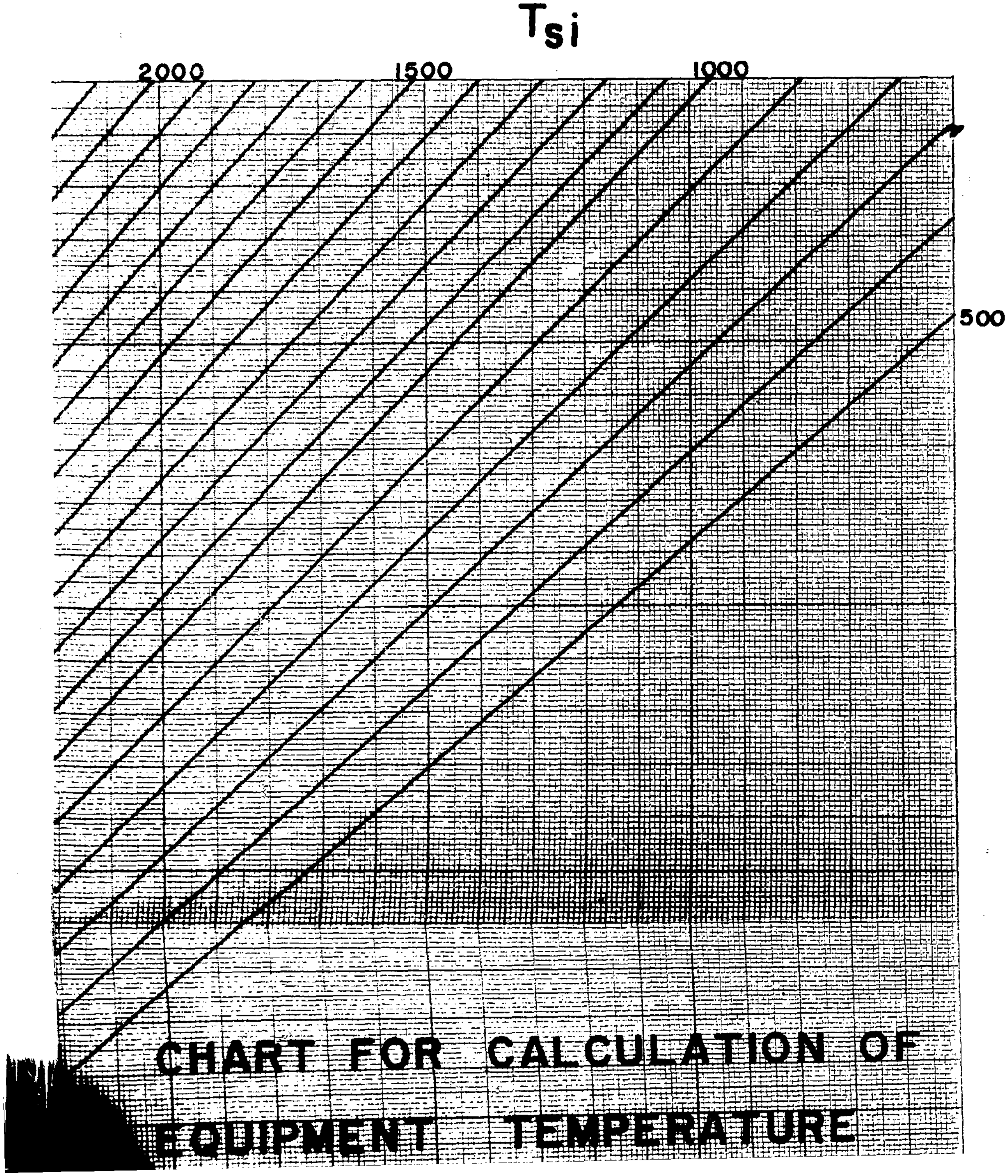
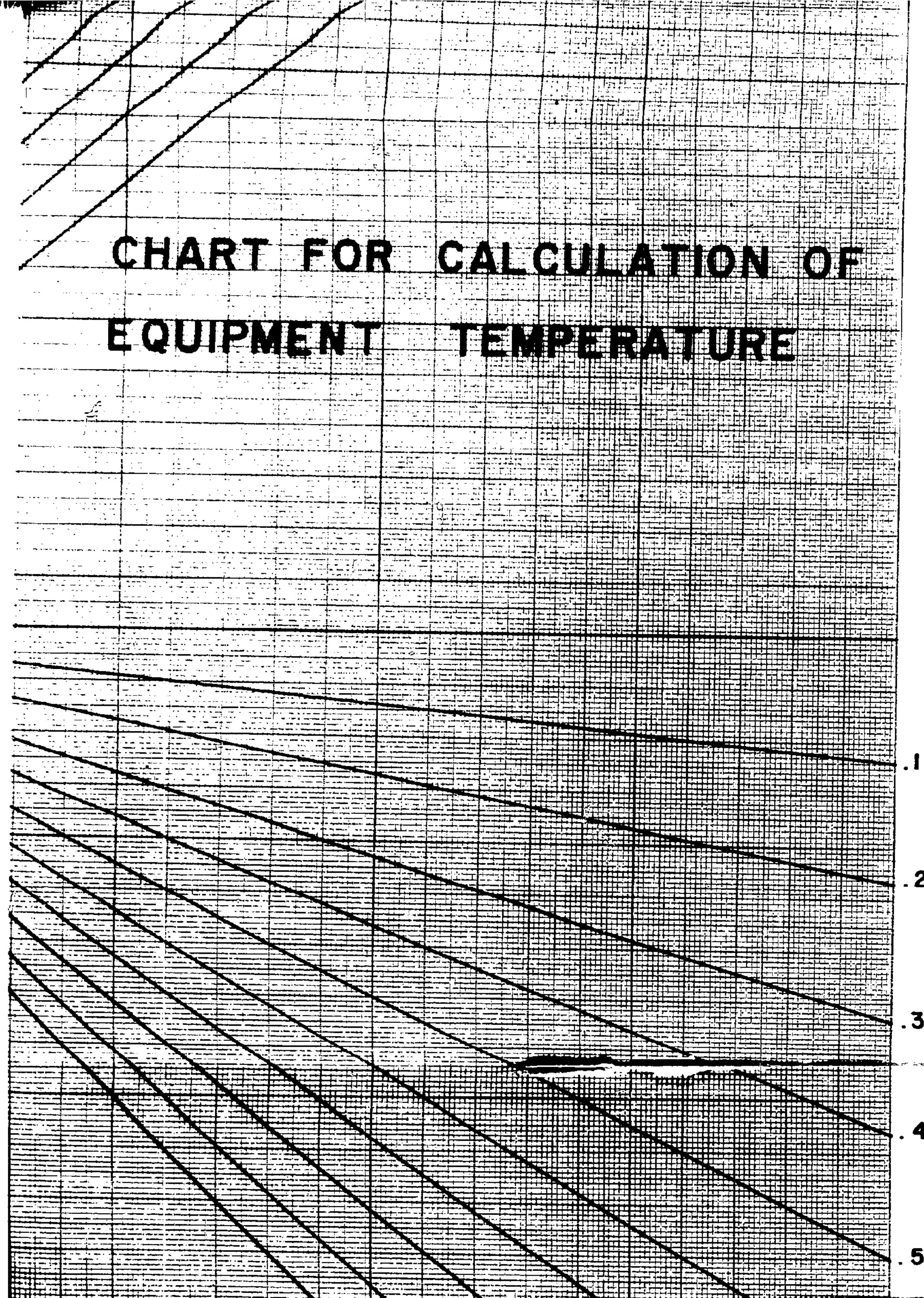


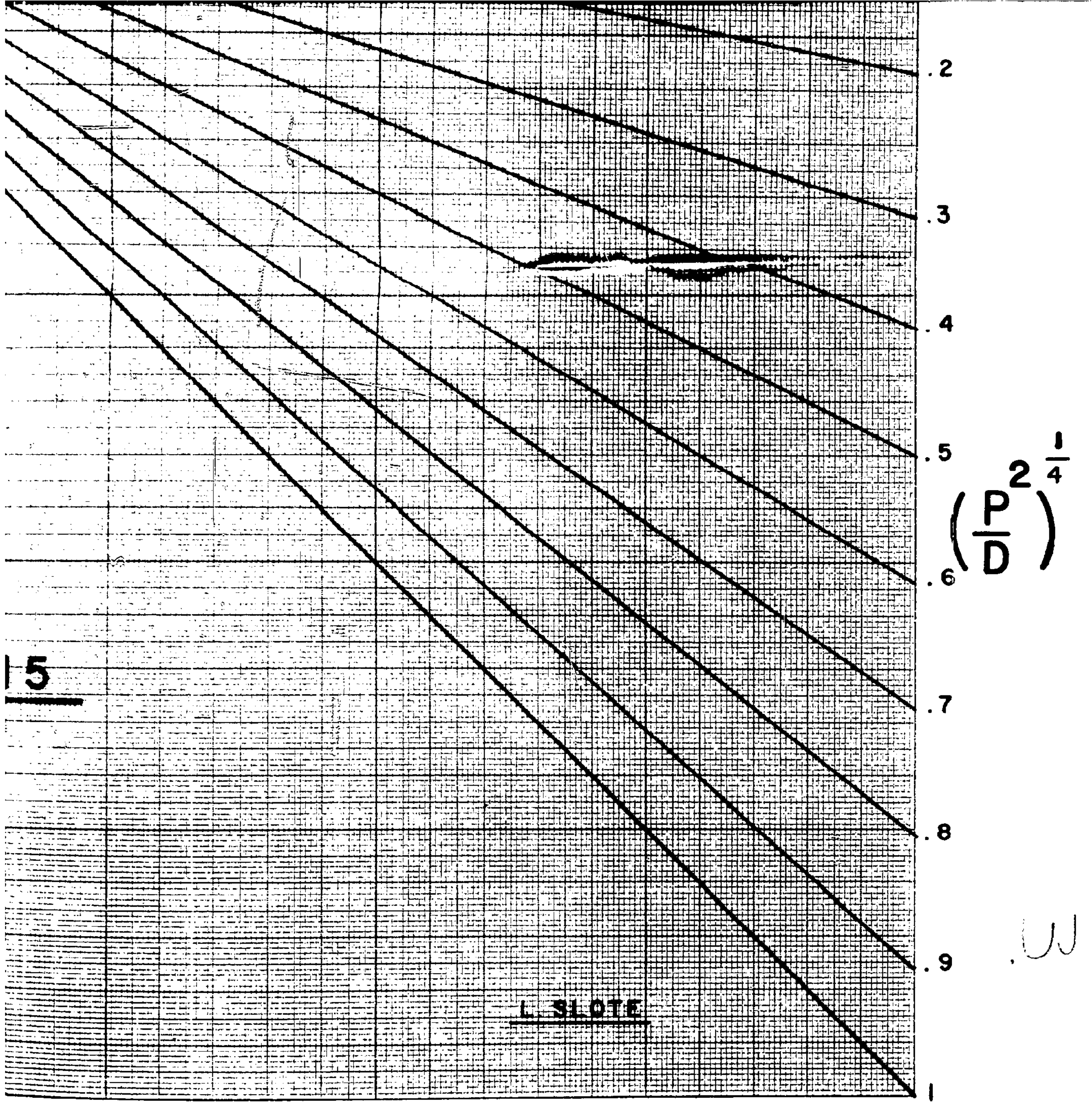
CHART FOR CALCULATION OF
EQUILIBRIUM SKIN TEMPERATURE
8
10
12
14
16
18
20
22
24
26
28
30
32
34
36
38
40
42
44
46
48
50

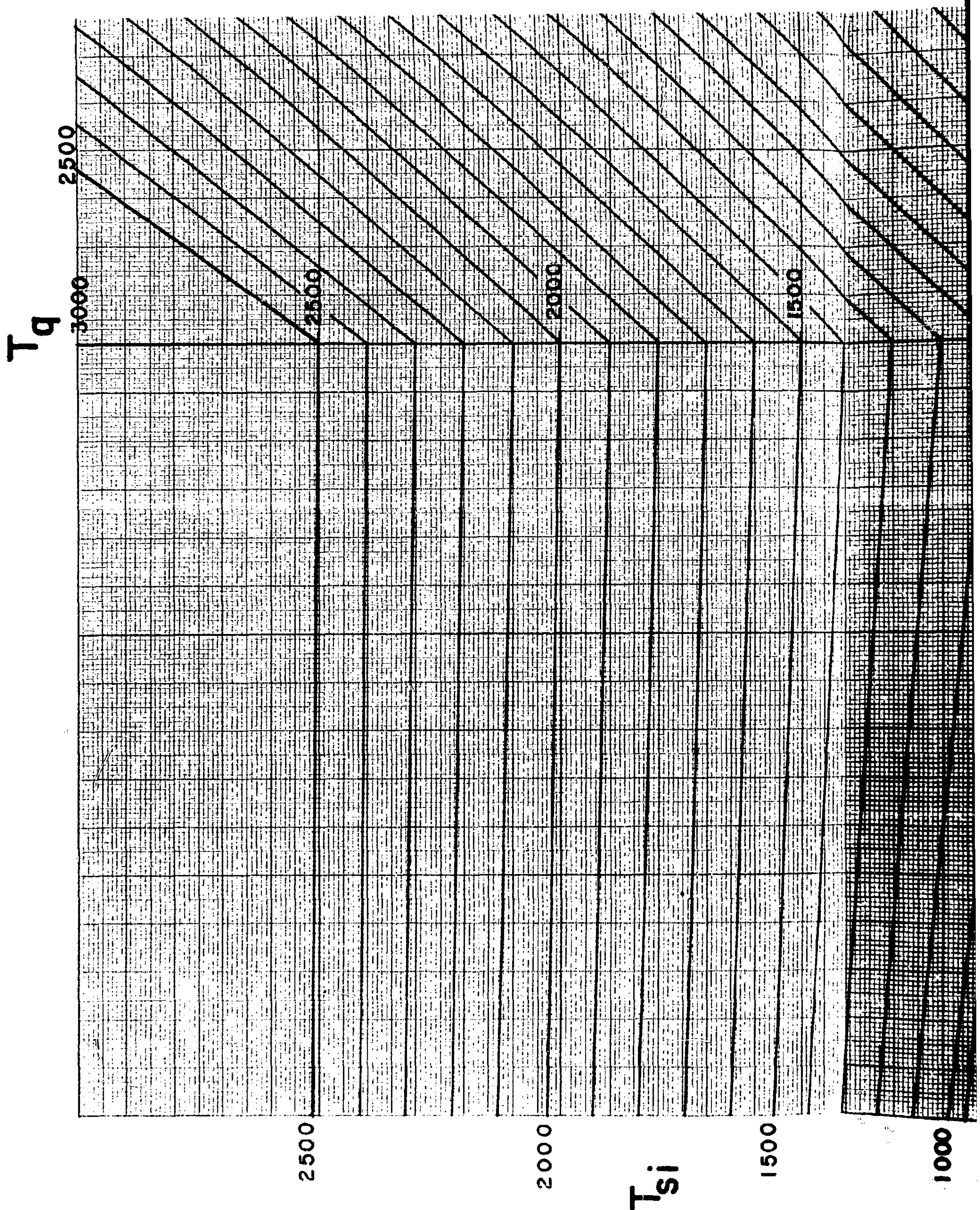


T_{si}
2000
1500
1000
500
CHART FOR CALCULATION OF
EQUIPMENT TEMPERATURE

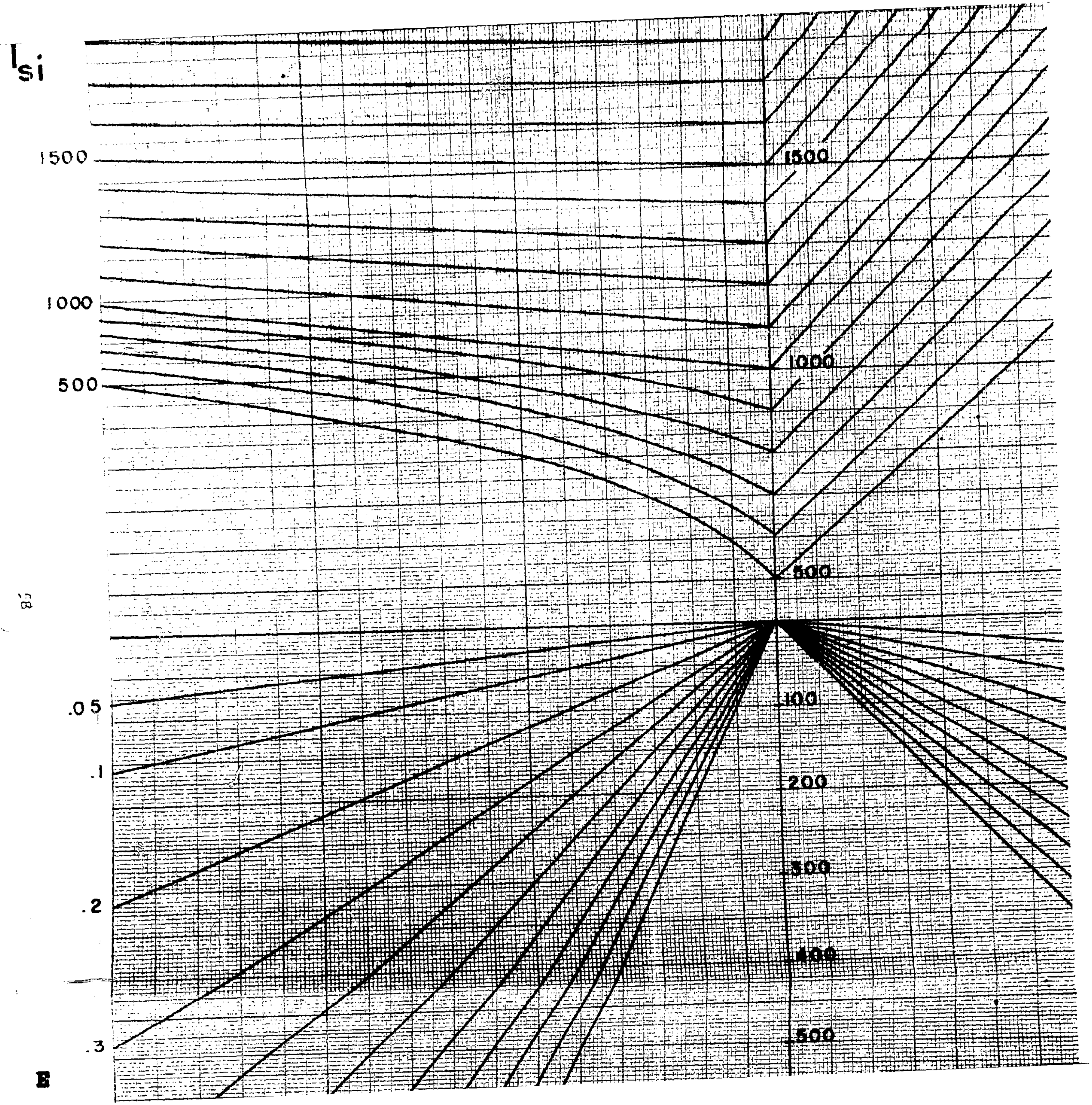


$2\frac{1}{4}$

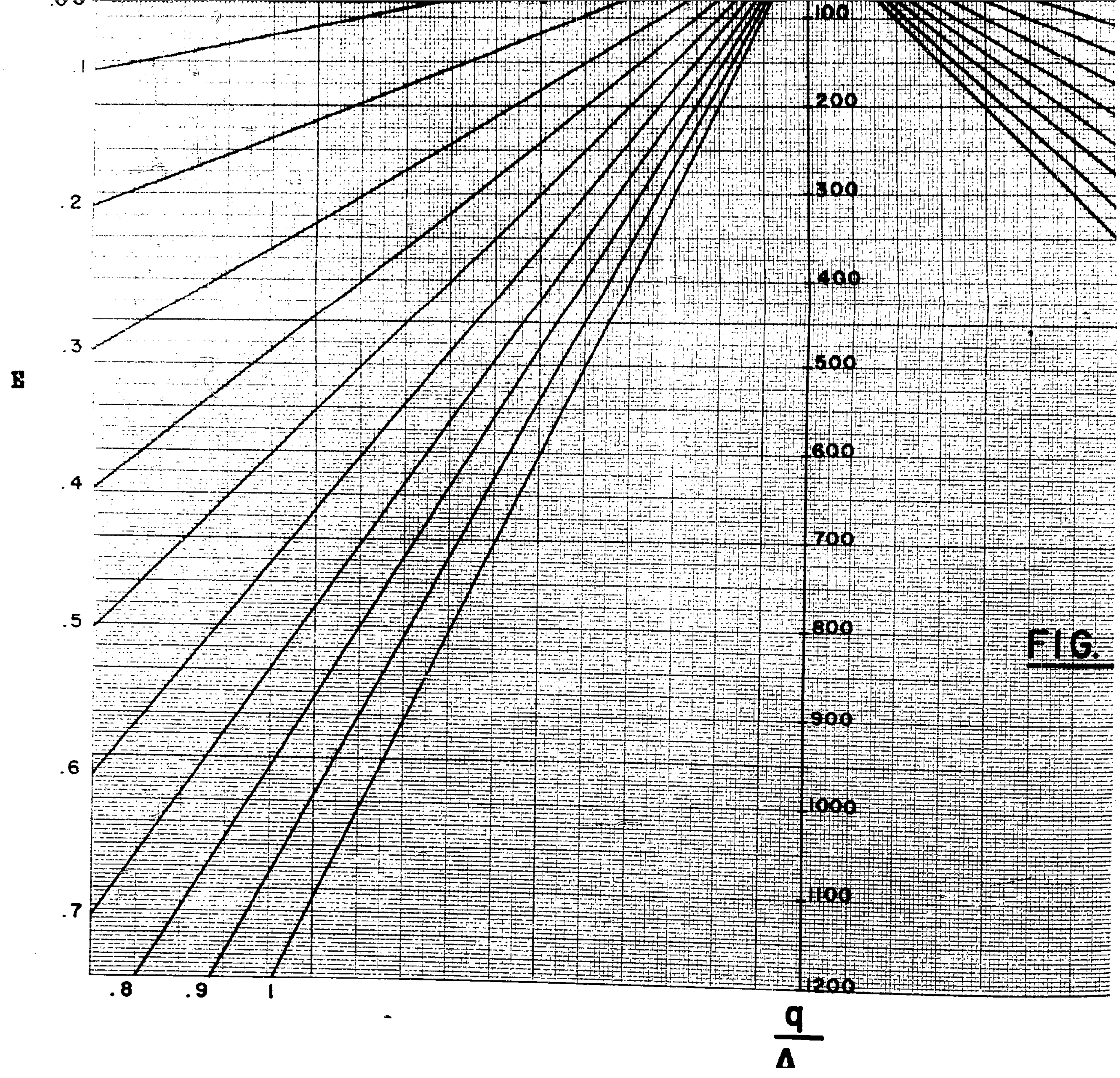

.2
.3
.4
.5
.6
.7
.8
.9
1
$\left(\frac{P^2}{D}\right)^{\frac{1}{4}}$
15
L SLOTE
.W



T_q
2500
3000
2500
2000
1500
2500
2000
1500
1000
T_{si}

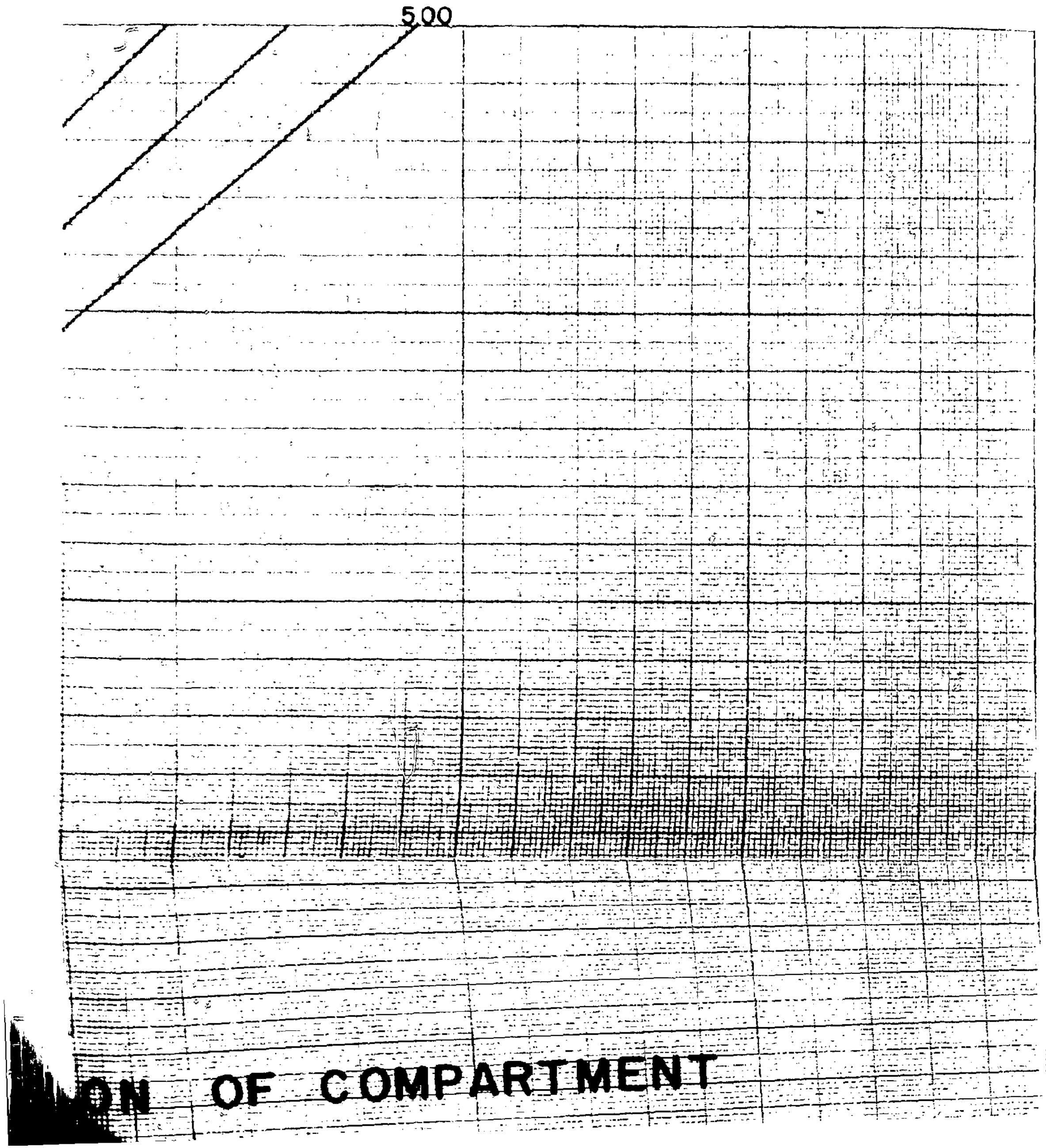


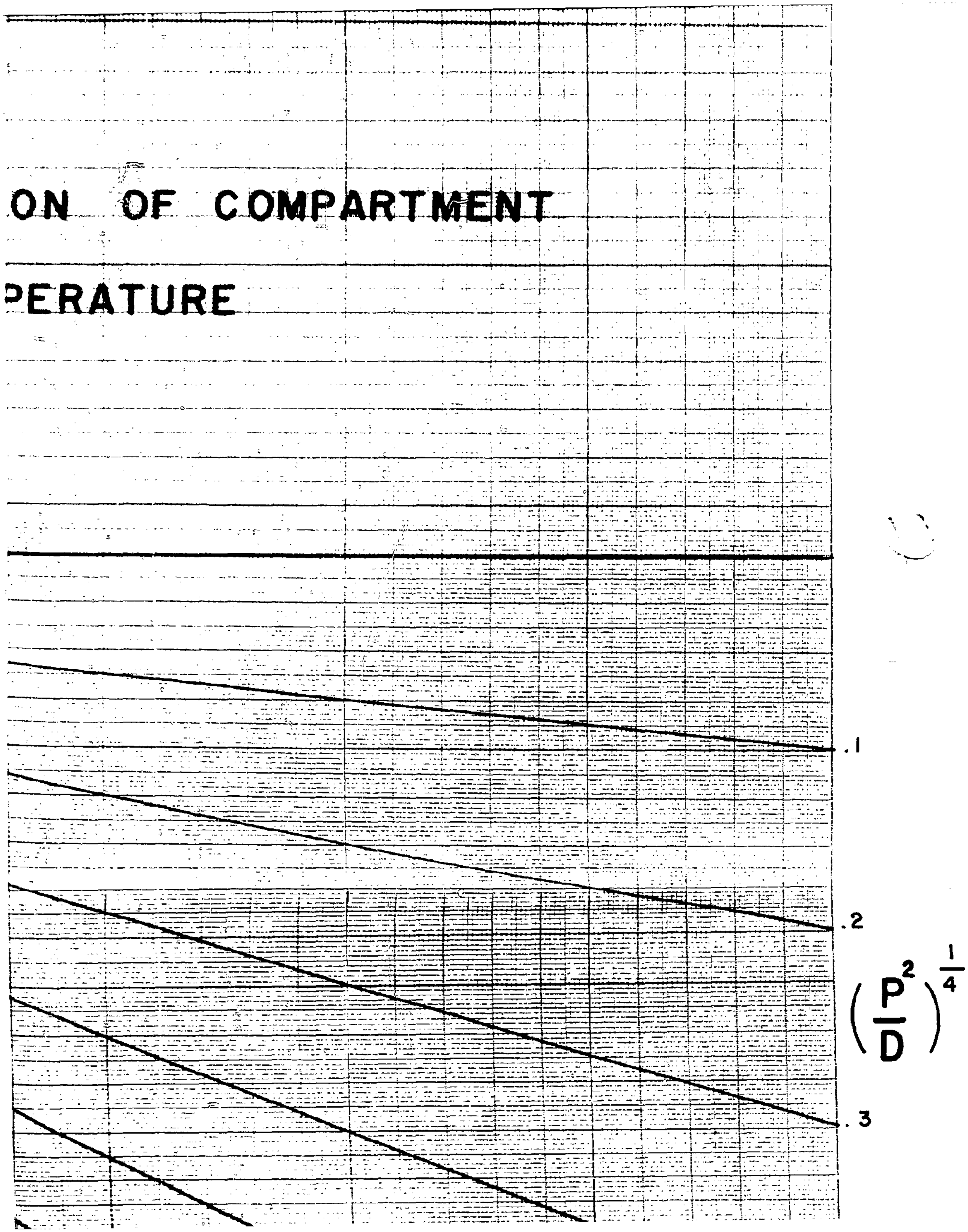
I_{si}
1500
1000
500
.05
.1
.2
.3
E
1500
1000
500
100
200
300
400
500

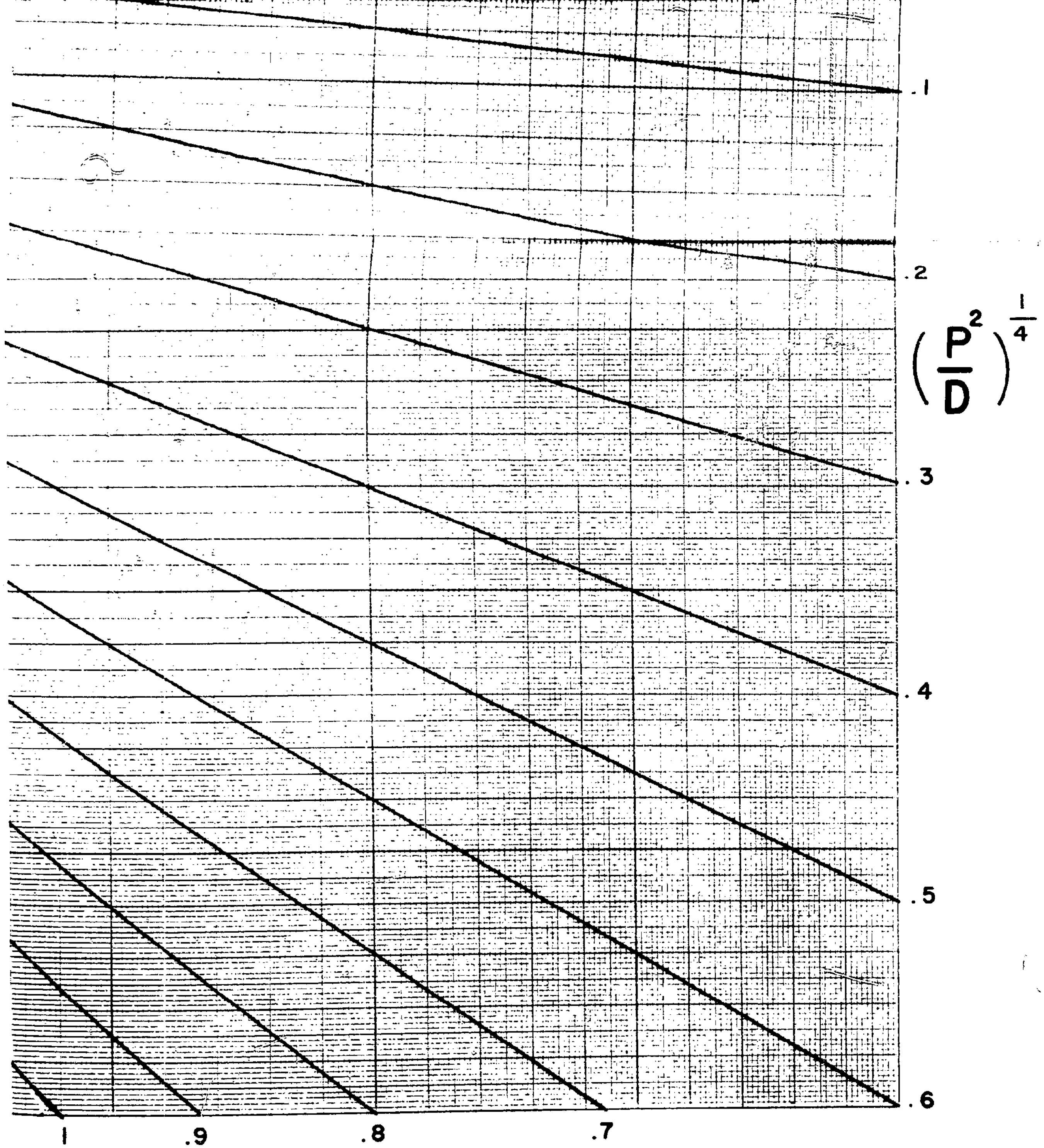


E
.1
.2
.3
.4
.5
.6
.7
.8
.9
1
100
200
300
400
500
600
700
800
900
1000
1100
1200
q / Δ

FIG.


500
ON OF COMPARTMENT


ON OF COMPARTMENT
ERATURE
.1
.2
.3
$\left(\frac{P^2}{D}\right)^{\frac{1}{4}}$


.1
.2
.3
.4
.5
.6
.7
.8
.9
1
$\left(\frac{P^2}{D}\right)^{\frac{1}{4}}$

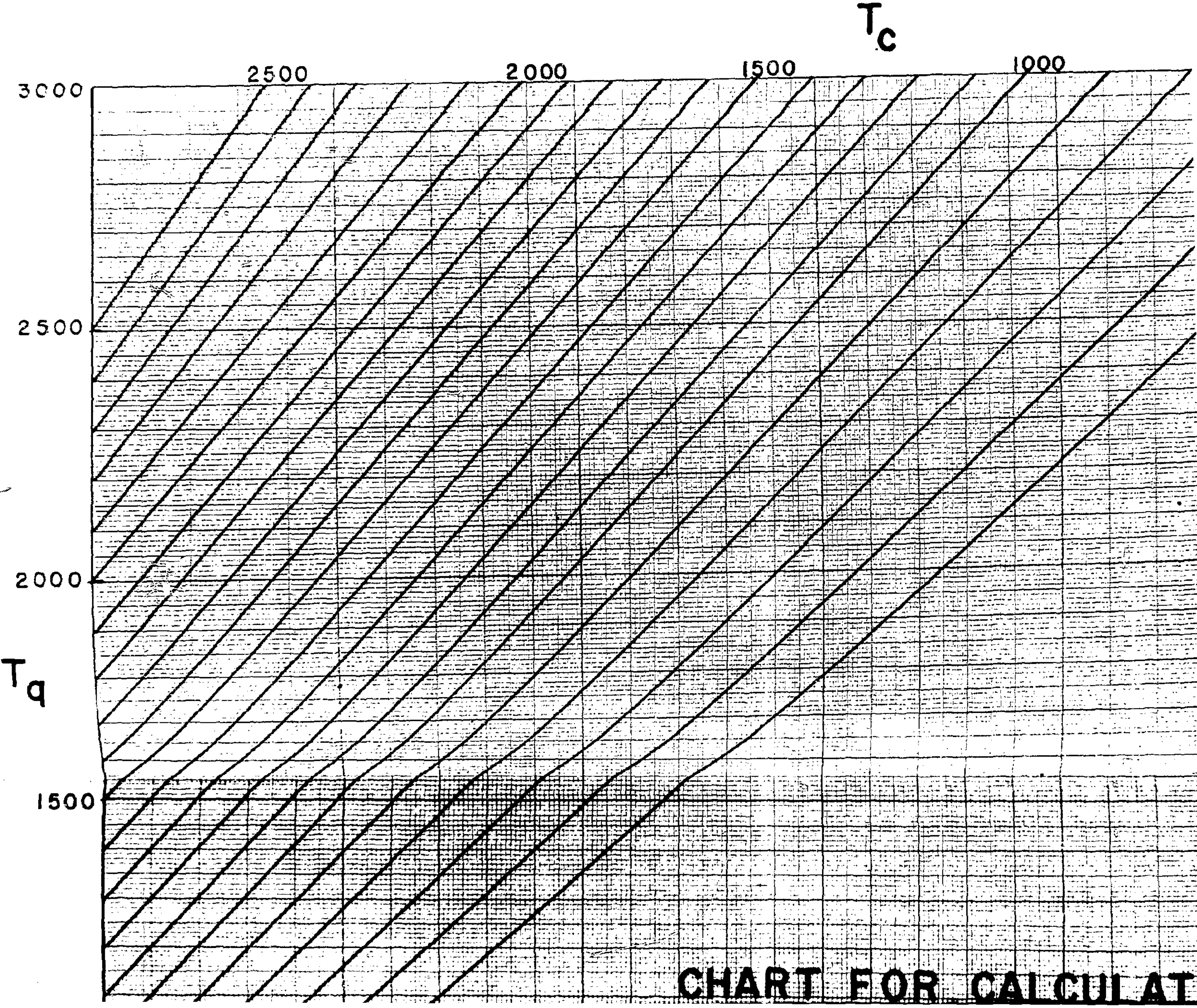


CHART FOR CALCULATI

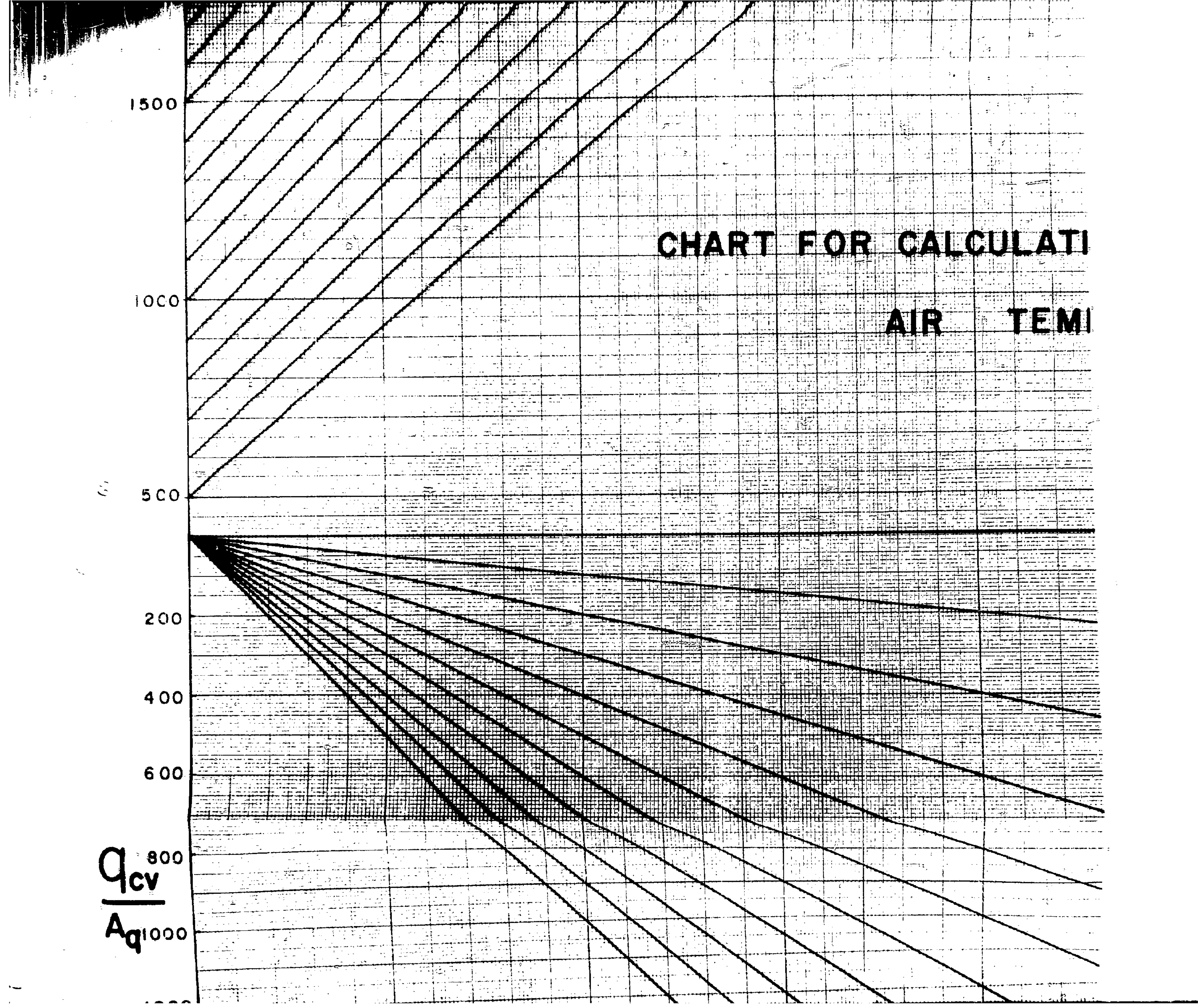
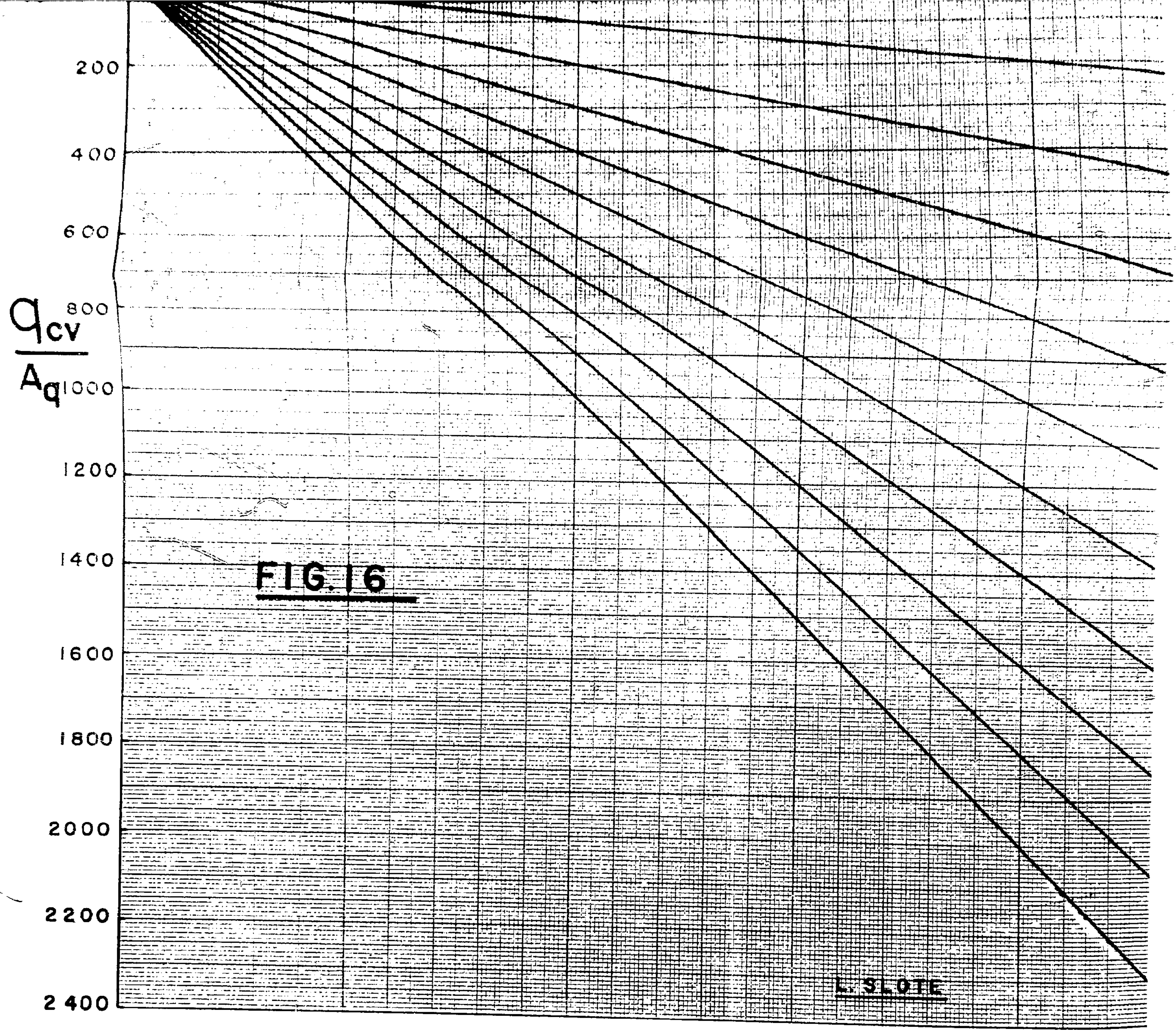
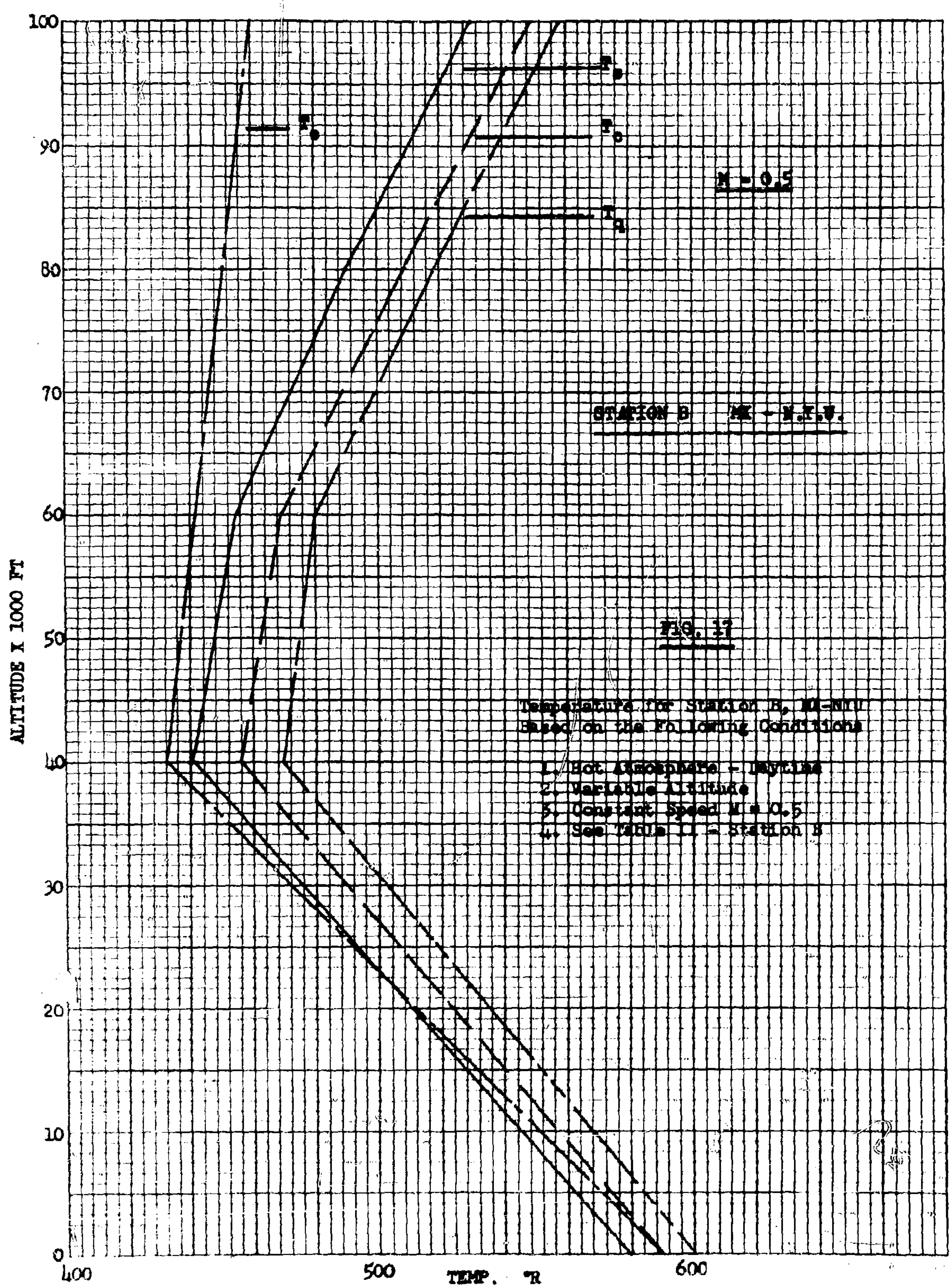


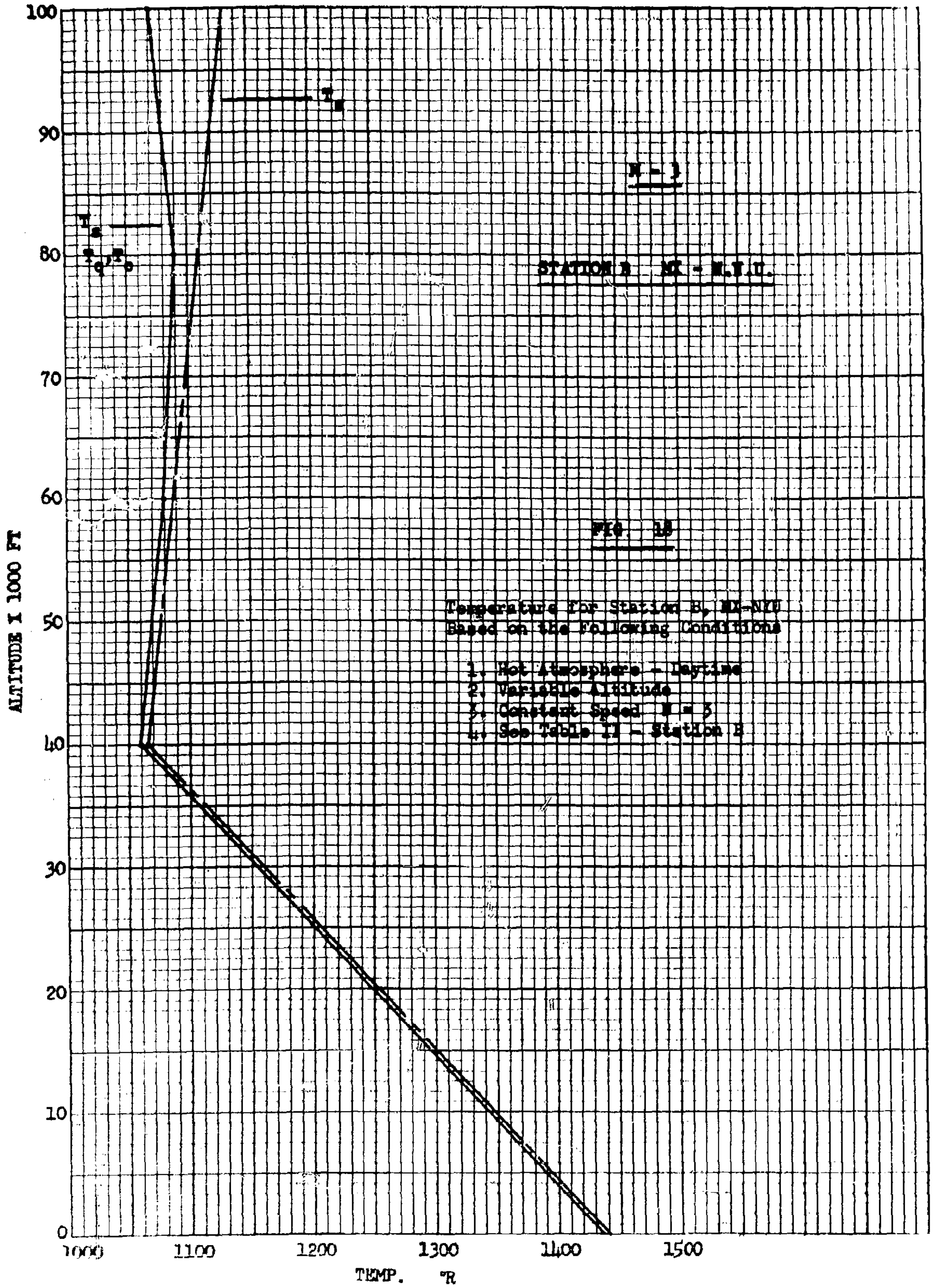
CHART FOR CALCULATI
AIR TEMI
1500
1000
500
200
400
600
800
1000
$\frac{q_{cv}}{A_q}$



200
400
600
800
1000
1200
1400
1600
1800
2000
2200
2 400
$\frac{q_{cv}}{A_q}$
L. SLOTE

FIG. 16


M = 0.5
STATION B   MK - N.Y.U.
FIG. 17
Temperature for Station B, MK-NYU
Based on the Following Conditions
1. Hot Atmosphere - Daytime
2. Variable Altitude
3. Constant Speed M = 0.5
4. See Table II - Station B
ALTITUDE X 1000 FT
TEMP. °R
0
10
20
30
40
50
60
70
80
90
100
400
500
600


ALTITUDE X 1000 FT
100
90
80
70
60
50
40
30
20
10
0
1000
1100
1200
1300
1400
1500
TEMP. °R
STATION B MX - N.Y.U.
Temperature for Station B, MX-NYU
Based on the Following Conditions
1. Hot Atmosphere - Daytime
2. Variable Altitude
3. Constant Speed
4. See Table II - Station B

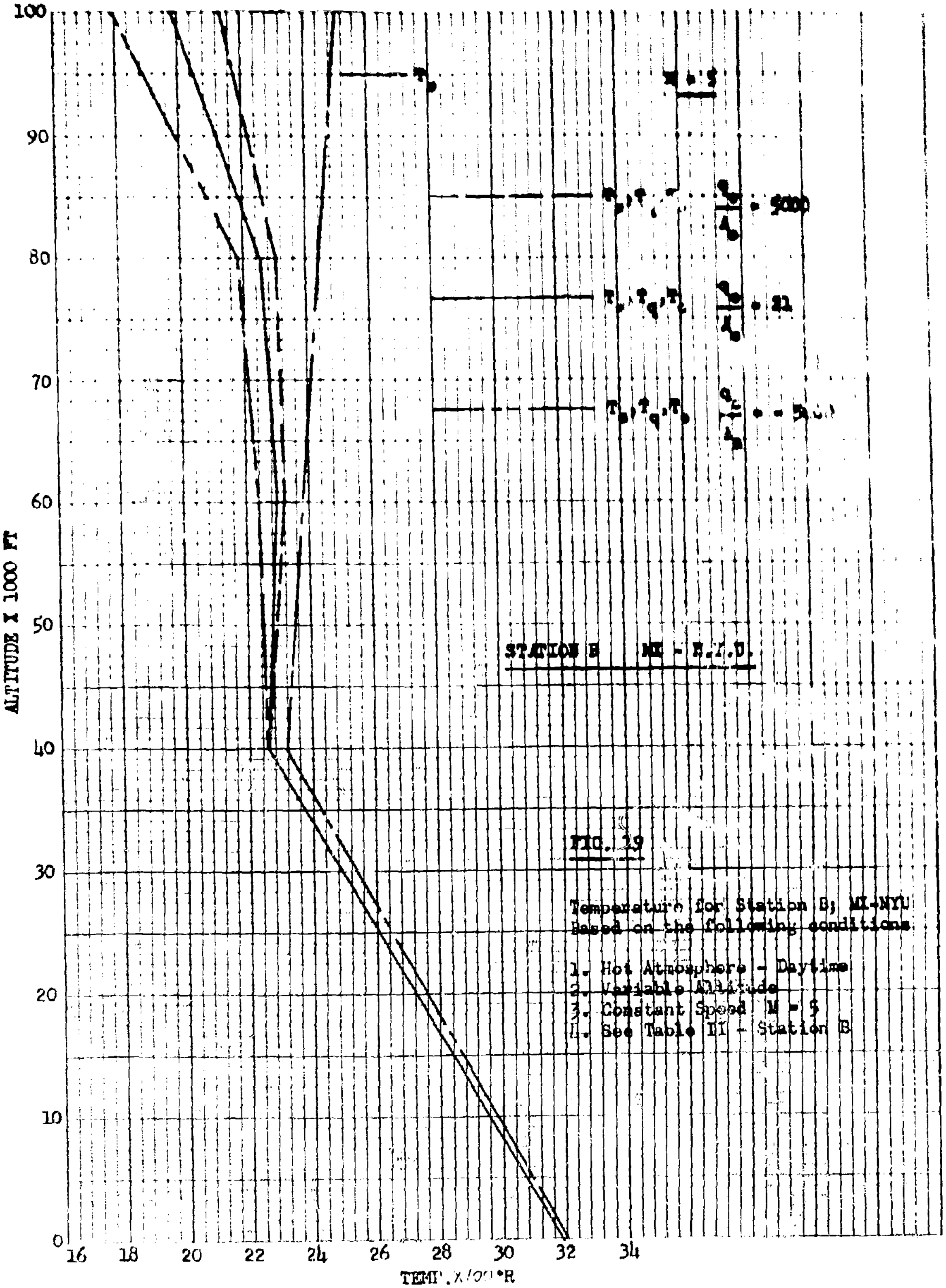


FIG. 19

Temperature for Station B; MI-NYU
Based on the following conditions

1. Hot Atmosphere - Daytime
2. Variable Altitude
3. Constant Speed M = 5
4. See Table II - Station B