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AN EVALUATION OF THE HEAT TRANSFER ENCOUNTERED IN A ROCKET MOTOR OPERATING AT HIGH CHAMBER PRESSURES

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by

C. F. W[illegible]

and

M. [illegible]

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AN EVALUATION OF THE HEAT TRANSFER ENCOUNTERED IN A ROCKET MOTOR OPERATING AT HIGH CHAMBER PRESSURES

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Dr. C. F. Warner
and
Dr. M. J. Zucrow

Purdue Research Foundation
and
Purdue University

20 July 1949

TABLE OF CONTENTS

Page

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NOMENCLATURE

I'_{sp} = theoretical specific impulse

q_c = convective heat transfer rate

h_c = convective heat transfer coefficient

T_g = gas static temperature

T_s = wall temperature

Re = Reynolds Number, DG/μ

Pr = Prandtl number, $C_p\mu/K$

D = chamber diameter

G = mass flow density

K = thermal conductivity

μ = gas viscosity

C_p = constant pressure specific heat

T_i = impressed temperature

T_T = gas total temperature

C_v = constant volume specific heat

γ = specific heat ratio, C_p/C_v

q_r = rate of radiant heat transfer

E'_s = effective surface emissivity = $(E_s+1)/2$

E_s = surface emissivity

E_g = emissivity of gas mixture

a = absorptivity of surface

E_{g_c} = emissivity of carbon dioxide at gas temperatures and appropriate value of P_cL. (See Figure 9.)

C_c = correction for effect of total pressure on CO_2 radiation. (See Figure 10)

E_{g_w} = emissivity of water vapor at gas temperature and appropriate value of P_wL. (See Figure 11.)

C_w = correction for effect of total and partial pressure on water vapor radiation. (See Figure 12.)

ΔE_g = correction for superimposed radiation at gas temperature. (See Figure 13.)

P_c = partial pressure of CO_2 in the mixture.

P_w = partial pressure of water vapor in the mixture.

L = radiant beam length through the gas mass

ABSTRACT

The theoretical rates of heat transfer between the walls of a five hundred-pound thrust rocket motor and the combustion gases resulting from the oxidation of octane and aniline by white fuming nitric acid are obtained as a function of the rocket motor chamber pressure. The convective heat transfer coefficients, obtained by using the McAdams correlation, are compared with the rates obtained by the Humble, Lowdermilk, Grele correlation. The radiant heat transfer rates are obtained by the Hottle, Egbert method. It was found that the chamber and nozzle heat transfer rates increase linearly with chamber pressure and that the nozzle heat transfer rates become extremely high at chamber pressures beyond 1500 psia.

AN EVALUATION OF THE HEAT TRANSFER ENCOUNTERED IN A ROCKET MOTOR OPERATING AT HIGH CHAMBER PRESSURES

Dr. C.F. Warner
Dr. M.J. Zucrow

INTRODUCTION

In recent years there has been a general trend both in the United States and Europe to employ for logistics reasons white fuming nitric acid (WFNA) and jet engine fuel as the propellants for various rocket motor applications. At the present time, the experience with these propellants is rather limited when compared with the experience with WFNA and either aniline or furfuryl-alcohol-aniline mixtures. However, the results obtained to date indicate that good performance is obtainable with the newer propellants at the combustion chamber pressures currently employed for the WFNA-aniline system. The problem of developing a satisfactory ignition system for the nonhypergolic WFNA-jet fuel system appears to be well on its way to a solution.

Theoretical studies of the reactions between WFNA and different hydrocarbon fuels have been made by C.H. Trent and M.J. Zucrow.[1] Their results show that the H/C - ratio has only a small influence upon the theoretical specific impulse I'_{sp} and that only a small error is introduced if it is assumed that the reaction between WFNA aand octane is equivalent to that between WFNA and jet engine fuel. Figure 1 illustrates the effect of the combustion chamber pressure on the specific impulse, I'_{sp}, obtainable by reacting WFNA and octane at three different mixture ratios; O/F 4.8, 5.53, and 6.32. It is apparent from the curves that a substantial improvement in performance can be realized at the high combustion chamber pressures.

Undoubtedly the utilization of higher combustion chamber pressures will introduce new operating problems, particularly the problem of cooling the rocket motor combustion chamber and exhaust nozzle. This report is concerned with the estimation of the heat transfer problems of rocket motors operating at combustion chamber pressures up to 2500 psia, and utilizing the WFNA-octane propellant system. For comparison the study was extended to include the WFNA-aniline propellant system.

[1] C.H. Trent and M.J. Zucrow, *The Calculated Performance of Hydrocarbons - White Fuming Nitric Acid Propellants at High Chamber Pressure,* U.S. Navy, Project SQUID, Tech Memo Pur-6, 1949.

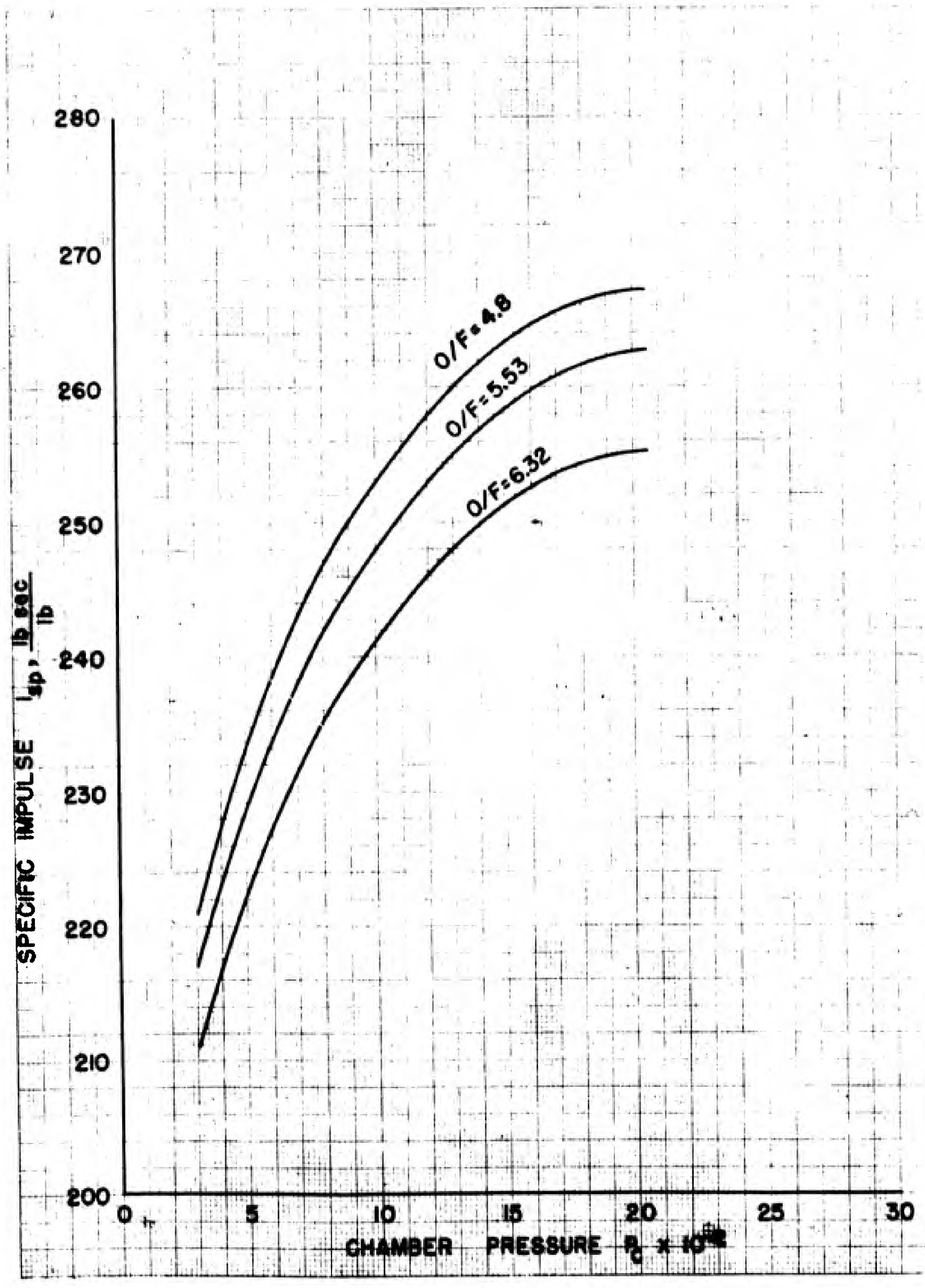

Fig. 1. Average performance of hydrocarbon fuels oxidized by WFNA.

CONVECTIVE HEAT TRANSFER

The heat transfer problems pertinent to rocket motors have been studied by several investigations; notably Seifert, Jet Propulsion Laboratory, California Institute of Technology; Summerfield and Sabersky, Aerojet Engineering Corporation; and Reinhardt,[2] Bell Aircraft Company. In the main the studies have been concerned with the WFNA-aniline system and with combustion chamber pressures close to 300 psia. To the authors' knowledge no studies have been reported which are concerned with WFNA-octane at high combustion chamber pressure.

The heat transfer from the hot combustion products to the walls of a rocket motor is difficult to analyze even from an overall point of view. Fairly good agreement has been obtained by various laboratories, however, between the analytical and the experimentally determined values for the overall heat transfer at various sections of the combustion chamber and exhaust nozzle for rocket motors operating at three hundred pounds per square inch chamber pressure. It must be realized, however, that overall values are not criteria of the ability of the rocket motor to operate without *burnout*, since there is ample evidence indicating that the design of the propellant injection system can exercise a major effect in producing local *hot spots*. Furthermore, due to the fact that the injector design and chamber configuration affect the overall heat transfer, close agreement between the calculated and the experimental values cannot be expected in a specific case. Consequently, although the calculated overall values of heat transfer may indicate that the motor has an adequate cooling system, there is no guarantee that the values will not be exceeded in a given motor and that there will be no localized *hot spot* failures.

Despite the inadequacies of the calculations, the calculated and experimentally determined values of the overall heat flux densities for the critical sections of the motor are instructive and constitute the basic information for designing the rocket motor cooling system.

Heat is transferred from the hot radiant gases to the relatively cool walls of the motor by both forced convection and radiation. The rates of heat transfer by each mode of heat transfer can be computed separately and then combined to give the overall rate of heat transfer.

When applying the conventional convective heat transfer equations to the combustion chamber of a rocket motor and its nozzle, the assumption is made that their configuration satisfies the requirements of the Reynolds analogy: a long tube through which hot gases are flowing with uniform velocity.

[2]T.F. Reinhardt, *Regenerative Rocket Cooling*, Meteor Report No. BAC-7, 1947.

The present theory of convective heat transfer is based on the concept that heat is transferred from a moving body of gas to the confining metal wall through a more or less stagnant film adjacent to the wall. The mechanism by which the heat is transferred from the fluid core through the film to the wall is similar to the mechanism underlying the conduction of heat through a solid homogeneous body. Consequently the equation for the convective heat transfer, q_c, can be written in a form similar to that for heat transfer by conduction. Thus, using the notation presented on page *v*, the convective heat transfer is given by

$$q_c = h_c(T_g - T_s) \tag{1}$$

The theoretical evaluation of h_c is quite difficult, since it is a function of a large number of variables, such as wall surface, surface roughness, gas velocity, temperature, the physical properties of the gas, and perhaps several others. The relationship between these variables is so complex that no completely satisfactory theoretical relationship has been developed. It has been demonstrated experimentally, however, that h_c may be expressed as a function of Reynolds and Prandtl numbers for incompressible gas flow in long tubes. One such relationship for the cooling of fluids is presented below.[3]

$$h_c = 0.0265\frac{K}{D}(Re)^{0.8}(Pr)^{.3} \tag{2}$$

Equation 2 represents the correlation of a large amount of experimental data for values of Reynolds number below 500,000 and under conditions of relatively small temperature differences between the fluid and the wall. The physical properties used in Eq. 2 are evaluated at the static temperatures of the fluid. This equation neglects the effect of compressibility.

At first glance, it would seem that Eq. 2 is inadequate for application to the flow conditions existing in a rocket motor. At the present time, few experimental data are available in the range of high Mach numbers under conditions of large temperature gradients. A recent literature survey by A. Ramachandran[4] revealed that the Mach number has little or negligible effect upon the heat transfer coefficient when the static temperature of the gas, T_g, in Eq. 1, is replaced by the *impressed* or *adiabatic wall* temperature; the latter is defined as the temperatures assumed by the wall in the absence of heat transfer and may be evaluated by the use of the equation

$$\alpha = \frac{T_i - T_g}{T_T - T_g} \tag{3}$$

where α, termed the recovery factor, is evaluated by Eq. 4.

$$\alpha = (Pr)^{1/3} \tag{4}$$

[3] M.H. McAdams, *Heat Transmission*, Second Edition, McGraw-Hill Co., New York, N.Y., 1942, p. 168.

[4] A. Ramachandran, *A Forced Convection Heat Transfer Study*, Unpublished Master's Thesis, Purdue University, Lafayette, Indiana, 1947.

Most of the reported investigations of heat transfer coefficients for high velocity fluid flow were performed with relatively small temperature gradients between the fluid and wall, while in a rocket motor, the temperature gradient is several thousand degrees Fahrenheit. McAdams and others have shown that, for high subsonic velocities in heated tubes, the effective mean heat transfer coefficient for air is independent of the temperature difference between the fluid and wall.

Another correlation recently proposed by Humble, Lowdermilk, and Grele[5] obtained from a study of air flow in a heated tube at high surface temperatures, $2060^{\circ}R$, is presented by Eq. 5.

$$\frac{hD}{K_s} = 0.023 \left(\frac{\rho_s V_b D}{\mu_s} \right)^{0.8} \left(\frac{C_{p_s} \mu_s}{K_s} \right)^{0.4} \tag{5}$$

The subscripts s and b indicate that the fluid property is to be evaluated at the surface and bulk . uid temperature respectively. Whether or not this correlation is applicable to the reverse case of a gas being cooled needs experimental verification.

The turbulence existing in the combustion chamber of a rocket motor is difficult to define. The Reynolds number for the chamber based upon the mass velocity indicates a highly turbulent flow. In view of the intense combustion reaction taking place, it would seem that the flow in the combustion chamber must be in a highly turbulent state.

It should be apparent that while Eqs. 2 and 5 are at best approximations, they will have to suffice for the present because of our lack of knowledge. In the absence of adequate information upon the subject, and with full realization that the calculated results will be approximate, Eq. 2 and Eq. 5 were used to calculate the convective heat transfer coefficient.

The evaluation of the fluid physical properties for use in Eq. 2 and Eq. 5 poses an additional problem due to the lack of data at the temperatures involved. The data pertaining to the physical properties of the gas mixture were calculated from the weighted values for the individual gases making up the mixture.

The combustion chamber pressure was chosen as the independent parameter. From thermochemical combustion equilibrium calculations, the combustion gas temperatures, the mean molecular weights, and the molar compositions were obtained for stoichiometric WFNA-octane and WFNA-aniline reactions. Figures 2 and 3 present the molal composition of these gases. Unfortunately,

---

[5] L.V. Humble, W.H. Lowdermilk, M.D. Grele, *Heat Transfer Coefficients and Friction Factors for Air Flowing in a Tube at High Surface Temperatures,* Heat Transfer and Fluid Mechanics Institute, A.S.M.E., 1949.

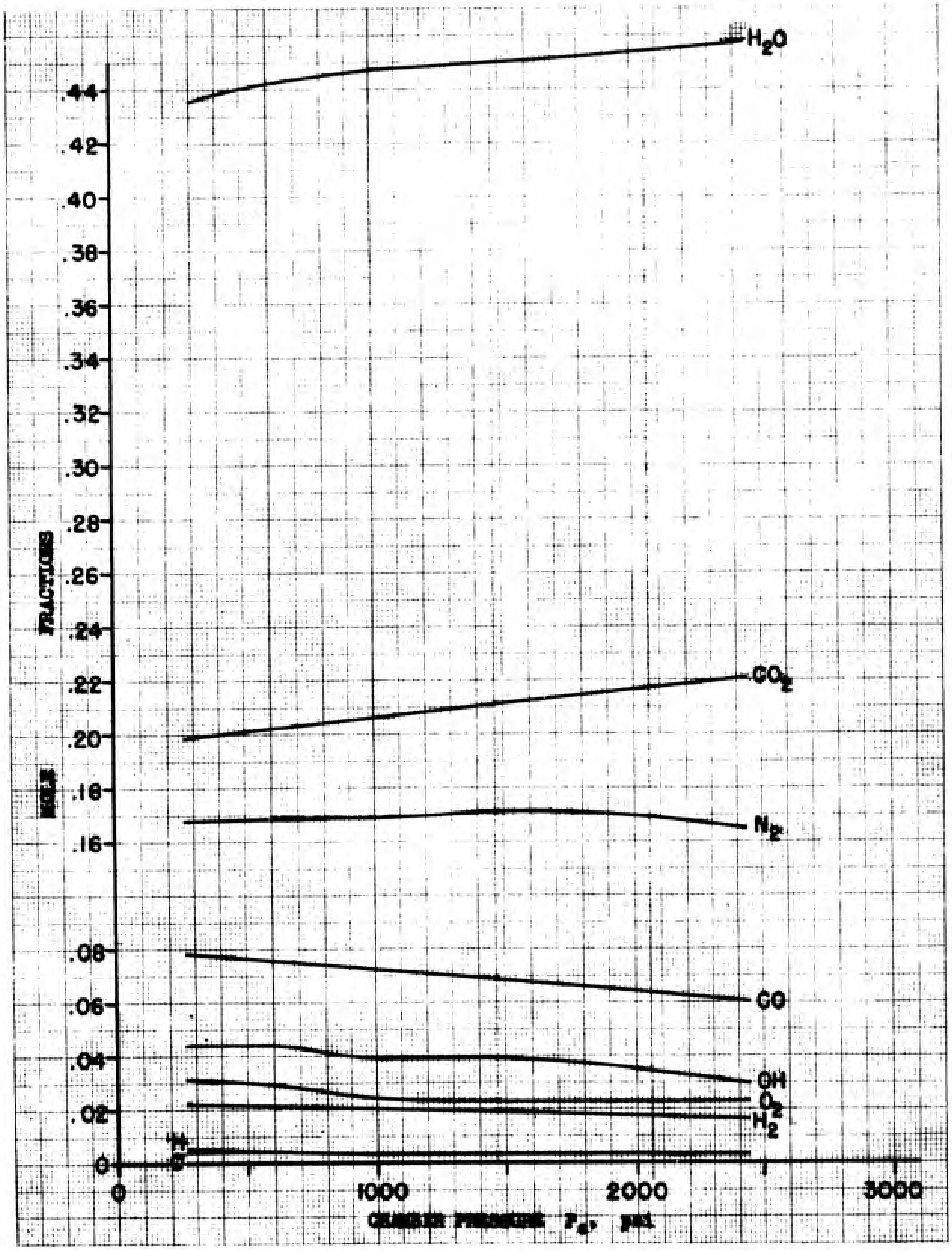

Fig. 2. Molal composition of combustion gas mixtures resulting from the oxidation of octane with white fuming nitric acid.

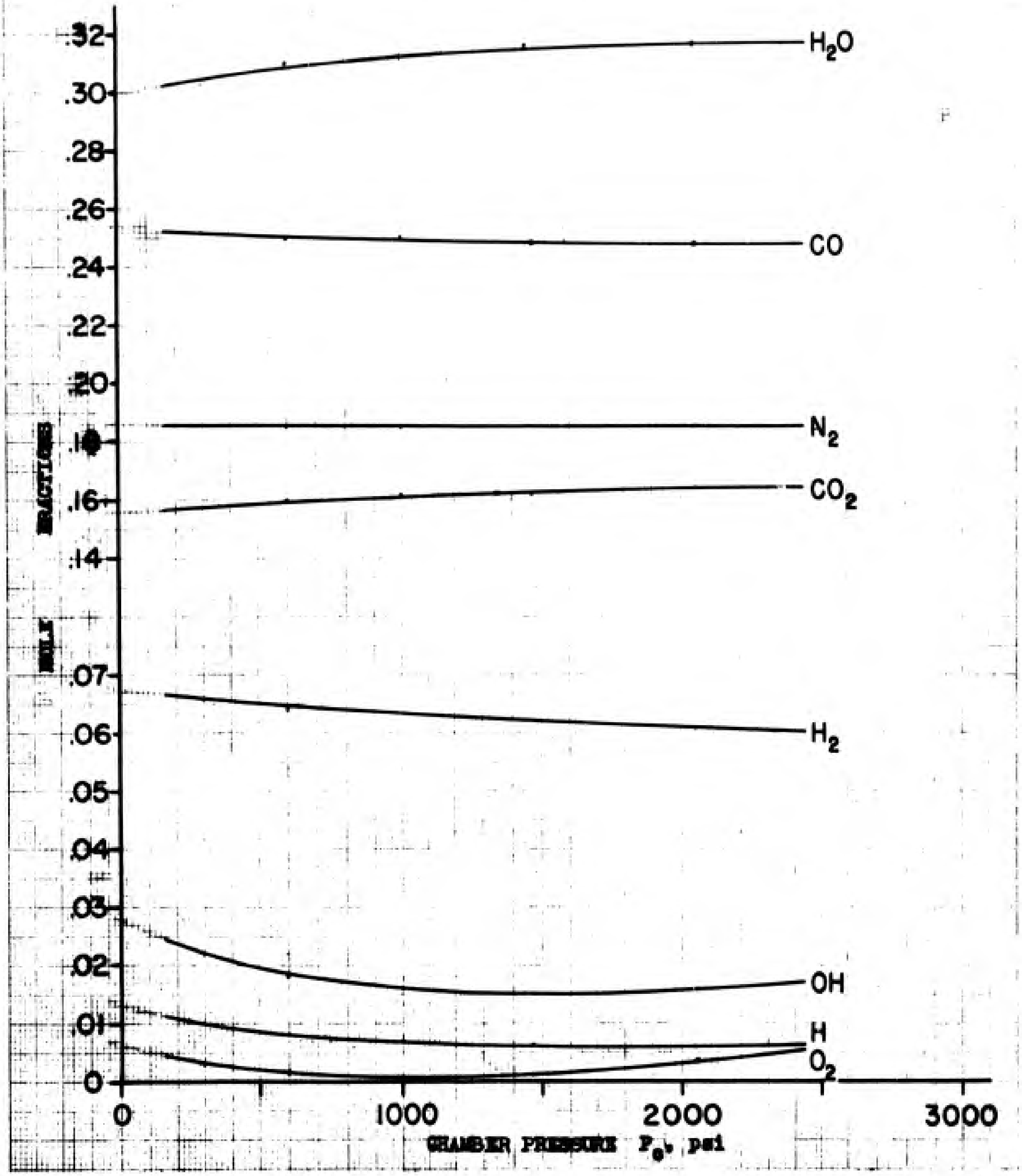

Fig. 3. Molal composition of combustion gas mixtures resulting from oxidation of aniline with white fuming nitric acid.

the temperatures and pressures encountered in the liquid propellant rocket motors considered here exceed those for which experimental physical property data are available. The temperatures of 5000 to 7000°F encountered in rocket motors operating at high combustion pressures are well above the critical temperatures of the individual gas components forming the combustion gas mixture, yet the combustion chamber pressures are moderate compared to the corresponding critical pressures. It seemed reasonable, therefore, to assume that under the aforementioned conditions, the pertinent physical properties are functions of temperature alone. (That assumption was made.)

Because of the few experimental data available, the values of thermal conductivity of the component gases were calculated by the kinetic theory relationship.[8]

$$K = \bar{E} C_V \mu \tag{6}$$

where

$$\bar{E} = 1/4(9\gamma - 5)$$

The values obtained for the individual gaseous components using Eq. 6 compare favorably with estimates made by other laboratories; such as, the M.W. Kellogg Company, and the University of California, Los Angeles. Figure 4 presents the thermal conductivity of the combustion gas (based upon the weighted average of the individual values for the individual gaseous components) as a function of temperatures.

No published experimental data are available for the viscosity of gases over the temperature range encountered in rocket motors. It was necessary, therefore, to extrapolate the published data. The Sutherland equation was used to extrapolate the viscosities of the individual gas components with full cognizance that this procedure is open to question. Figure 5 presents the viscosities of the combustion gases as a function of temperature.

Figure 6 presents the constant pressure specific heat and specific heat ratio of the combustion gases. For the determination of the gas velocities in the chamber and throat the gas density is required. The gas density was evaluated by assuming the perfect gas equation of state to hold for the combustion gas mixture. The mass flow density G and the combustion chamber diameter D were determined from dimensions, thrust, and the mixture ratio for the motor under consideration.

[8] L.B. Loeb, *Kinetic Theory of Gases*, Mc-Graw-Hill Co., New York, N.Y., 1934.

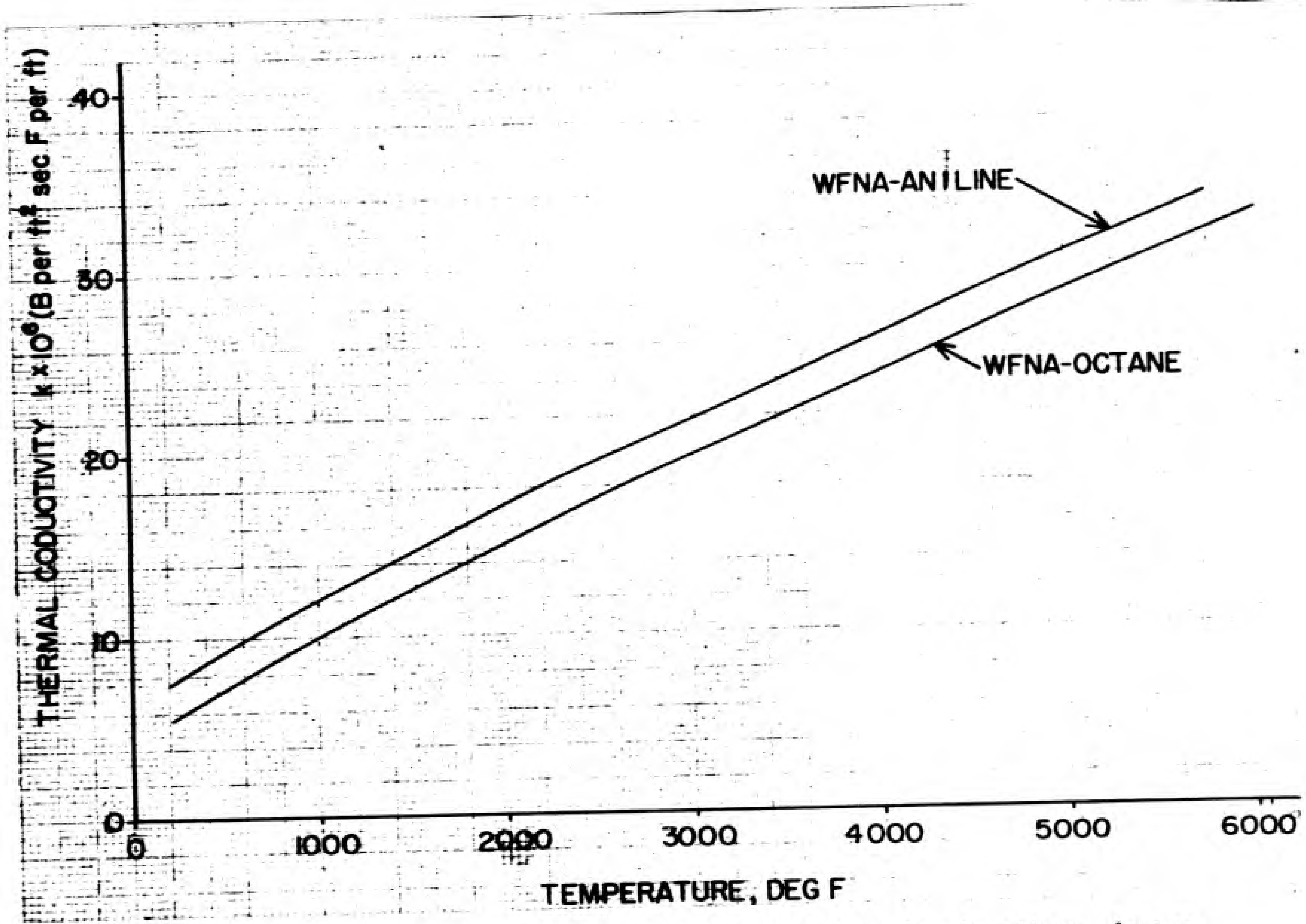

Fig. 4. Thermal conductivity of combustion gas mixtures resulting from the oxidation of octane and aniline with white fuming nitric acid.

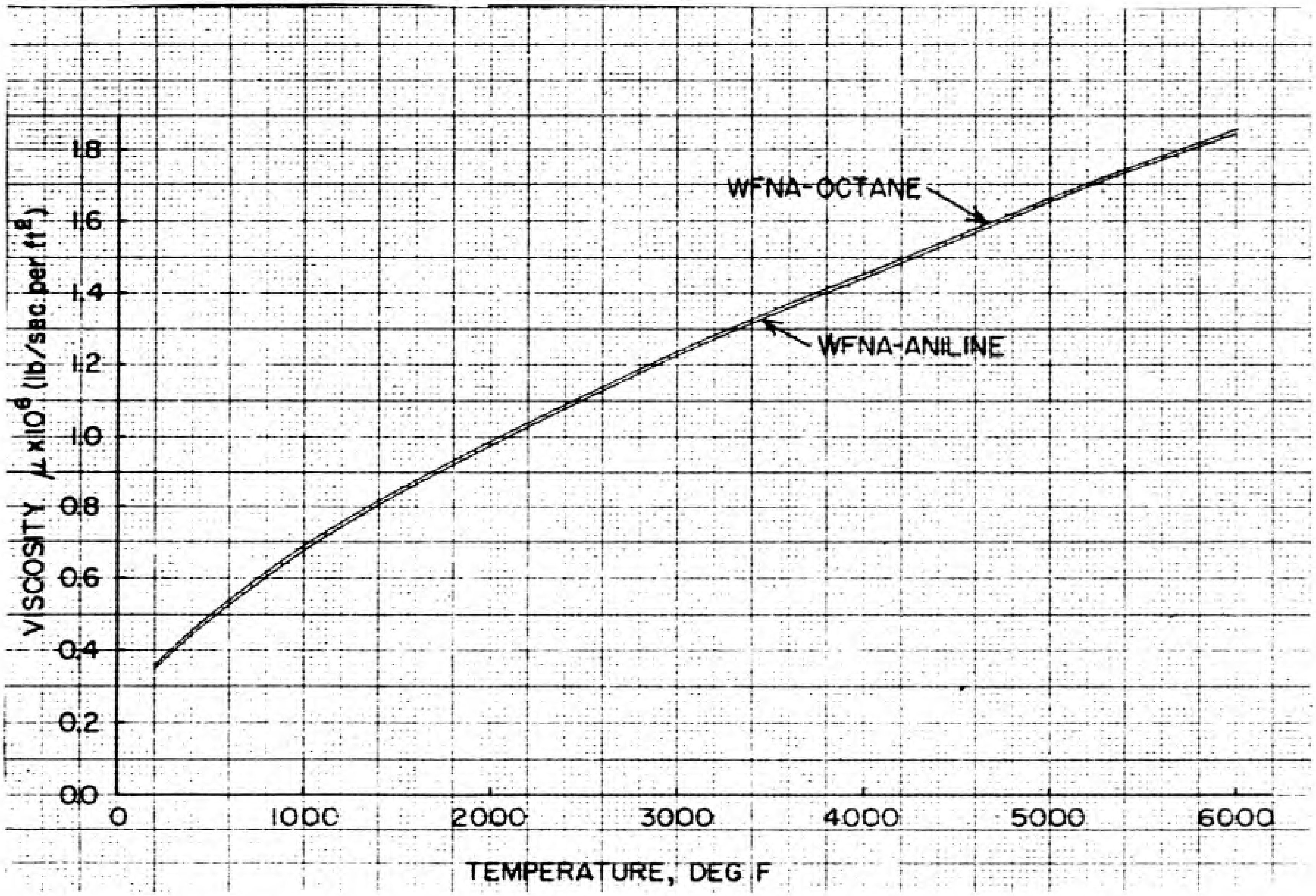

Fig. 5. Dynamic viscosity of combustion gas mixtures resulting from the oxidation of octane and aniline with white fuming nitric acid.

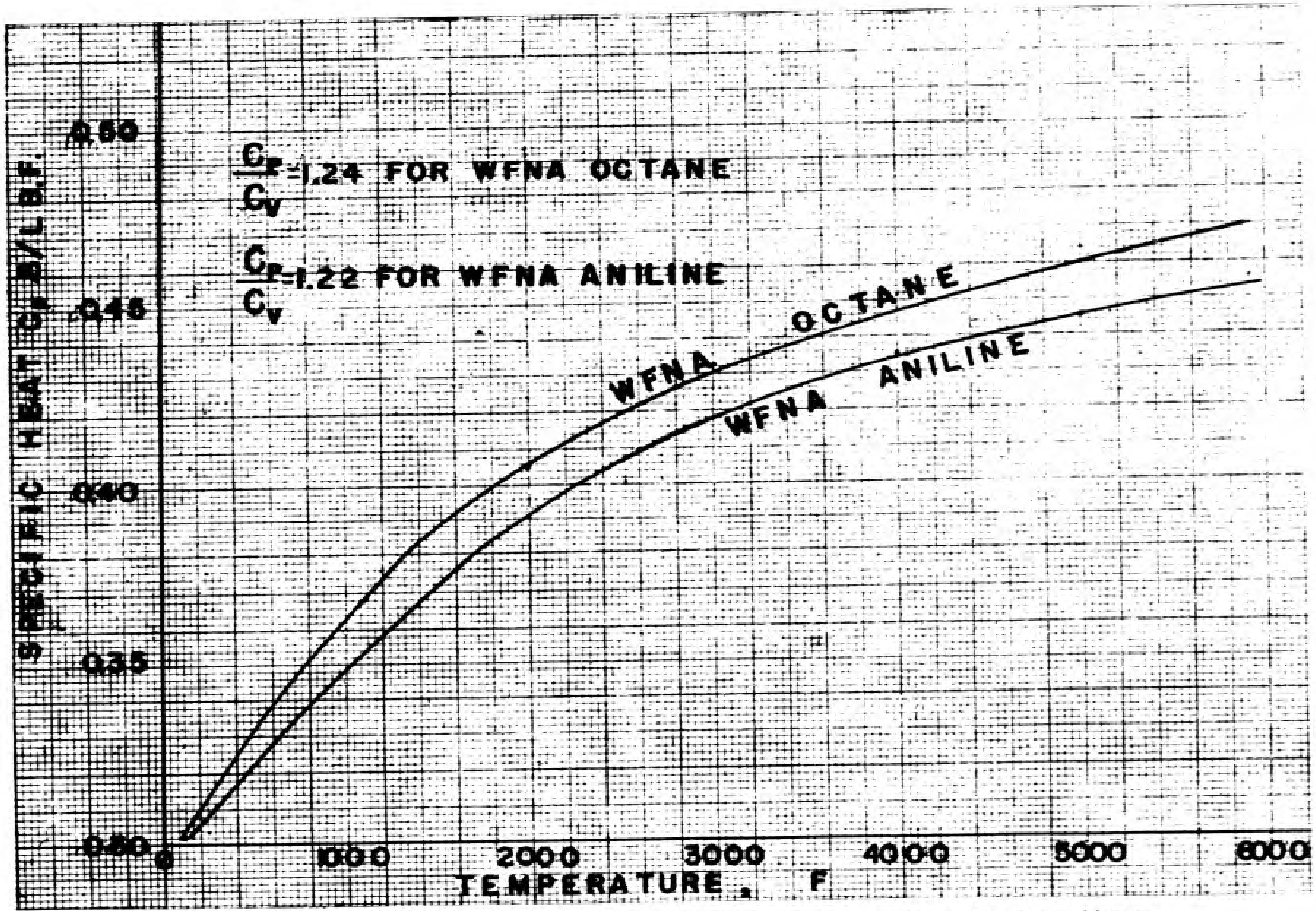

Fig. 6. Constant pressure specific heat of combustion gas mixtures resulting from the oxidation of octane and aniline with white fuming nitric acid.

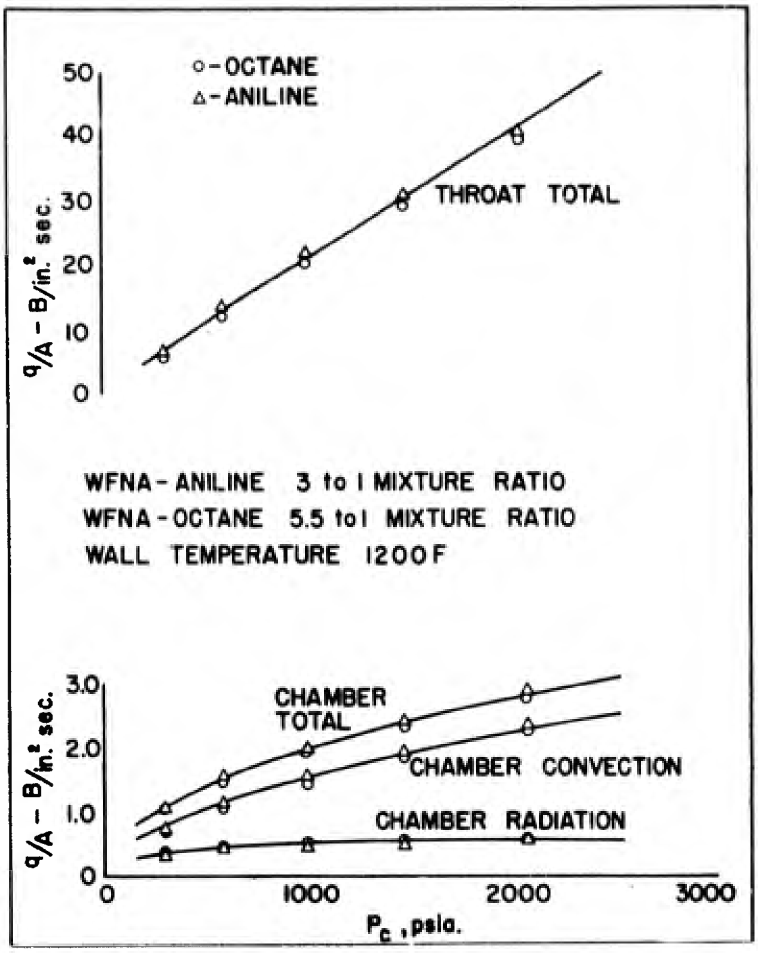

Fig. 7. Heat transfer rates as a function of chamber pressure for a 500-pound thrust motor.

The motors were assumed to have a constant thrust of 500 pounds, an L* of 100 inches, a length to diameter ratio of 2, and variable chamber and throat diameters. From these data the values of h_c were determined by employing Equations 2 and 5.

In the calculation of the convective heat transfer coefficient for the rocket motor nozzle throat, it was assumed for convenience that there was no change in the chemical aggregation and physical properties of the gases from that for the combustion chamber.

The values of the heat flux density were calculated for a 500-pound thrust motor for the reactions of WFNA-octane and WFNA-aniline for combustion chamber pressures varying from 300 to 2500 pounds per square inch.

Figure 7 and Table I present the calculated values of the convective heat transfer coefficient as a function of chamber pressure for the two propellant combinations.

RADIANT HEAT TRANSFER

For the purpose of this study, it will be assumed that the radiant portion of the total amount of heat transferred is contributed primarily by the water vapor and carbon dioxide components of the combustion gases.

The radiation from CO was neglected in the final calculations. A trial calculation was made, however, to determine the effect of CO radiation upon the results. It was assumed in making the trial calculation that the emissivity of CO is equal to half of that for CO_2; an assumption based on McAdams.[7] The results of the trial calculation showed that at combustion

[7] M.H. McAdams, *Heat Transmission,* Second Edition, McGraw-Hill Co., New York, N.Y., 1942, p. 72.

Table I. Comparison of convective heat transfer rates.

	Combustion Chamber		Nozzle Throat	
Pressure	McAdams	Humble	McAdams	Humble
		WFNA and Octane		
300 psia	0.83 B/in² sec.	1.37 B/in² sec	6.6 B/in² sec	10.1 B/in² sec
600	1.24 B/in²	2.10 B/in²	12.5 B/in²	18.9 B/in²
1000	1.69 B/in²	2.87 B/in²	20.0 B/in²	30.8 B/in²
1470	2.22 B/in²	3.71 B/in²	29.5 B/in²	44.0 B/in²
2058	2.77 B/in²	4.62 B/in²	40.6 B/in²	61.0 B/in²
		WFNA and Aniline		
300 psia	0.89	1.60	6.57	11.7
600	1.39	2.42	12.60	21.5
1000	1.82	3.30	21.10	38.0
1470	2.05	4.12	30.00	50.5
2058	2.80	5.12	41.70	70.0

Wall temperatures assumed to be 1200°F

McAdams: $$\frac{hD}{K_b} = 0.0265 \left(\frac{\rho_b V_b D}{\mu_b}\right)^{.8} \left(\frac{C_{p_b}\mu_b}{K_b}\right)^{.3}$$

Humble: $$\frac{hD}{K_s} = 0.023 \left(\frac{\rho_s V_b D}{\mu_s}\right)^{.8} \left(\frac{C_{p_s}\mu_s}{K_s}\right)^{.4}$$

s = surface temperature

b = bulk temperatures

chamber pressure of 300 psia with WFNA-aniline the inclusion of CO radiation increased the heat transfer in the exhaust nozzle throat section by less than 2 percent. The propellants WFNA-octane produce a much smaller amount of CO than do the aforementioned propellants. It was therefore concluded that no appreciable error is introduced by neglecting the effect of CO radiation.

A literature survey indicated that the method developed by Hottel and Egbert[8] was the most suitable one for estimating the heat transferred by gas radiation. The method of Hottel and Egbert was developed for application to furnaces operating at moderate temperatures and low pressures, and it was necessary to extrapolate their curves to the pressures encountered in rocket motors. A great many objections can be raised to the application of this procedure to the problem at hand, and the authors will welcome any suggestions which will improve the accuracy of the estimation of radiant heat transfer.

[8] H. C. Hottel, and R.B. Egbert, 'Radiant Heat Transmission from Water Vapor,' *Trans. A.I.Ch.E.*, Vol. 38, 1942, p. 531.

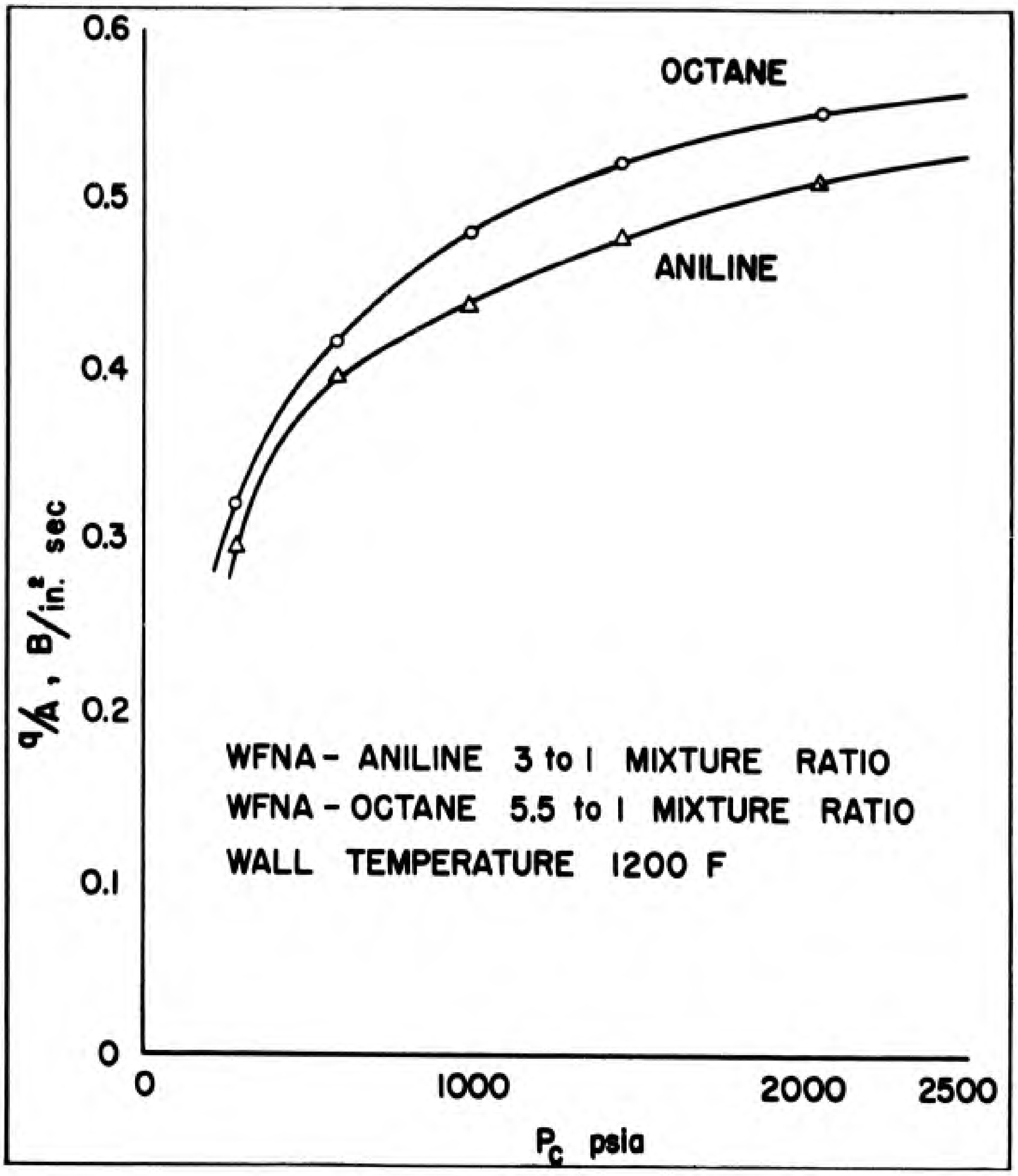

Fig. 8. Chamber radiant heat transfer as a function of chamber pressure for a 500-pound thrust motor.

The general equation for the heat radiated by hot gases to the walls of an enclosure is

$$q_r = \frac{.1723}{144 \times 3600} E'_s \left[E_g \left(\frac{T_g}{100} \right)^4 - a \left(\frac{T_s}{100} \right)^4 \right] \tag{7}$$

The emissivity and absorptivity of the gas mixture may be calculated from the values for the individual gas components by the following equations

$$E_g = (E_{g_C} \cdot C_C + E_{g_W} \cdot C_W - \Delta E_g) \tag{8a}$$

$$a = (a_C + a_W - \Delta E_S) \approx (E_{S_C} \cdot C_C + E_{g_W} \cdot C_W - \Delta E_S) \tag{8b}$$

when the ratio of the absolute gas temperature to the absolute wall temperature is greater than 1.25. The values of the gas emissivities and their correction factors were obtained from the Hottel charts presented in the appendix. In evaluating these terms, the partial pressures of the water vapor and carbon dioxide were obtained from the gas composition based on thermochemical calculations.[10] The length of the radiant beam through the gas mass is equal to the diameter of the combustion chamber for a rocket motor.

Figure 8 presents the results of the radiant heat transfer calculations for the rocket motor. The rate of radiant heat transfer in the throat was assumed to be equal to that in the chamber.

TOTAL RATE OF HEAT TRANSFER

The total rate of heat transfer q_T for the combustion chamber and nozzle throat were obtained by adding the rates of convective and radiant heat transfer. Thus

$$q_T = q_c + q_r \tag{9}$$

The results obtained for the total rate of heat transfer are presented in Figure 7. The values obtained by the methods presented in this study are in good agreement with those obtained experimentally for WFNA-aniline operating at low pressures as presented by Sutton.[9]

CONCLUSIONS

The results obtained from this study must be regarded as being approximations in view of the large number of assumptions made, particularly these pertaining to the conductivity and viscosity of the combustion gases, the estimation of radiant heat transfer, and the formulas for correlating heat transfer data. Nevertheless, the study does show qualitatively the following facts. The rate of heat transfer to the combustion chamber and nozzle increases with the combustion chamber pressure. The heat transfer to the combustion chamber is small enough, however, to indicate that regenerative cooling should be adequate for that portion of the rocket motor. The heat transfer to the nozzle throat section becomes so large at combustion chamber pressure above possibly 1500 psia that it seems doubtful that it can be cooled regeneratively.

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[9] G.P. Sutton, *Rocket Propulsion Elements,* John Wiley and Sons, Inc., New York, 1948, p. 140.

[10] Zucrow and Trent, Project Squid Tech. Memo. Pur-6.

The substitution of gasoline for aniline as the fuel component in a WFNA system does not aggravate the problem of cooling the rocket motor. There is a decided deviation between the results obtained from the Humble, and the McAdams correlation equations; the limited available experimental data seem to agree more favorably with the McAdams correlation. More reliable data on the physical properties of combustion gases as a function of temperature are needed. Experimental data on the heat transfer by radiation are needed.

APPENDIX

The charts presented in Figures 9 to 13 are based upon the original charts of Hottel and Egbert,[11] but extended by the authors. It is extremely doubtful that the factor C_c continues to increase above about 10 atm or 20 atm at 5000°F as shown.

[11] W.C. Hottel and R.B. Egbert, 'Radiant Heat Transmission from Water Vapor,' *Trans. A.I.Ch.E.*, Vol.38, 1942, p. 531.

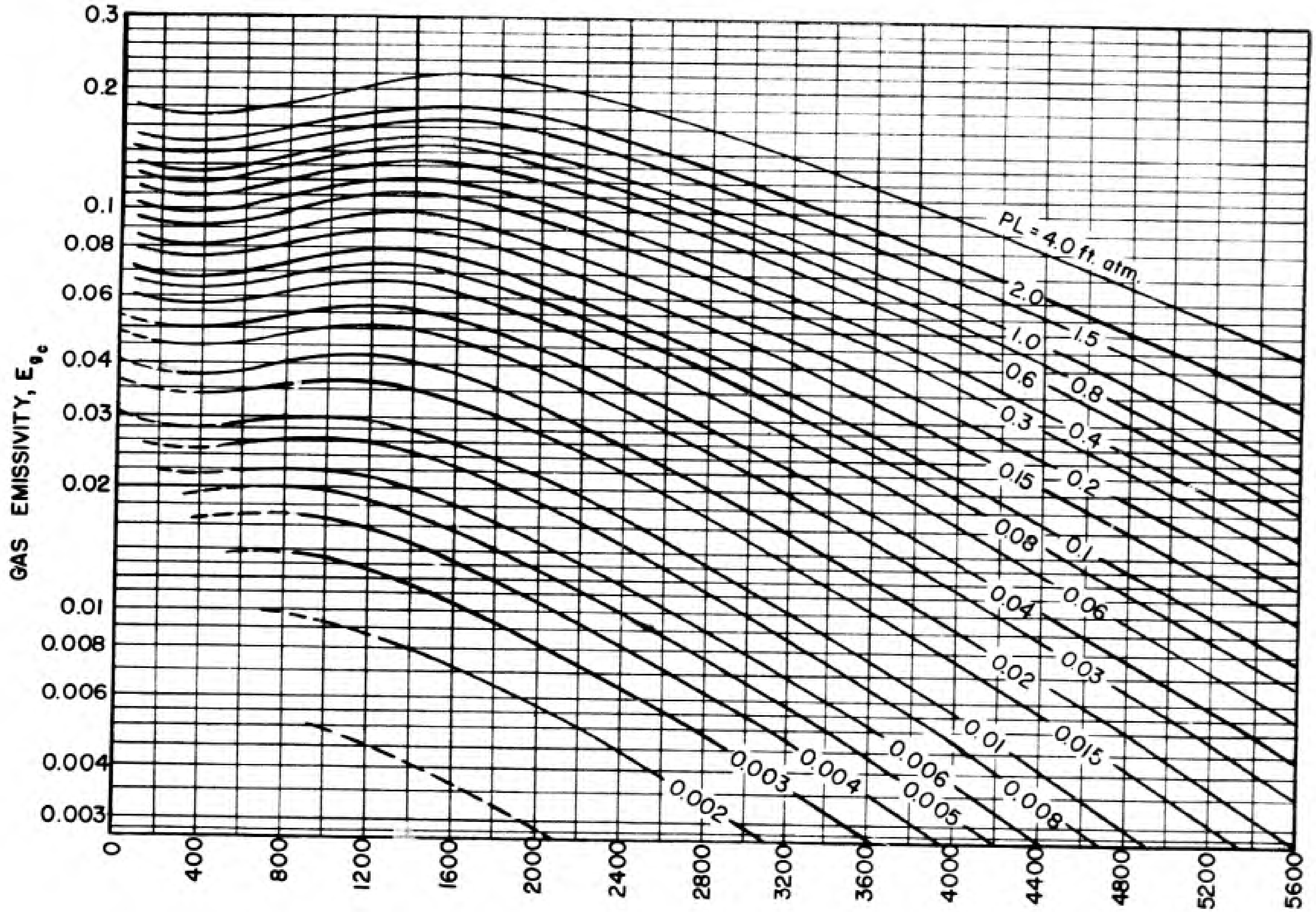

Fig. 9. Emissivity of carbon dioxide.

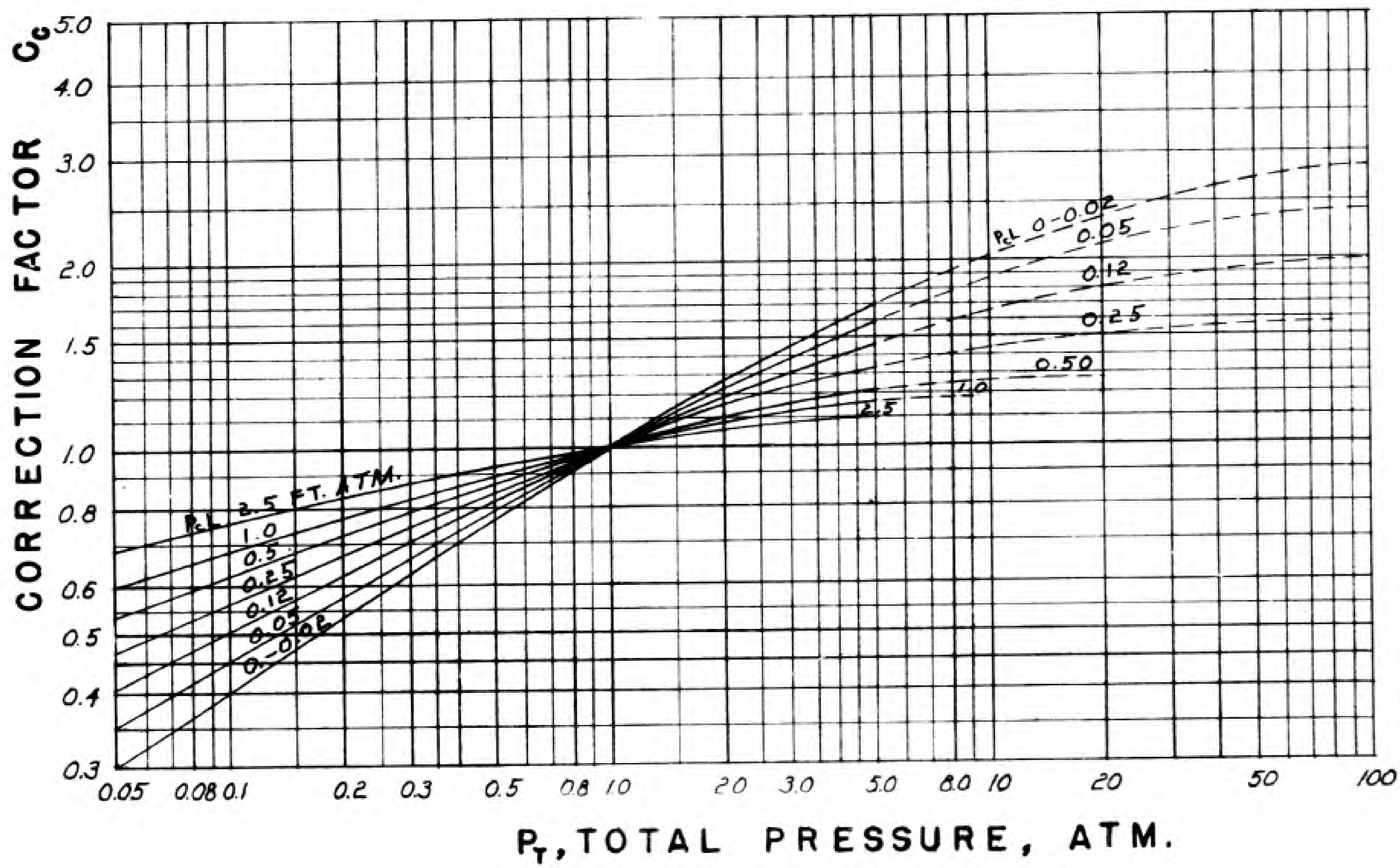

Fig. 10. Effect of total pressure on carbon dioxide radiation.

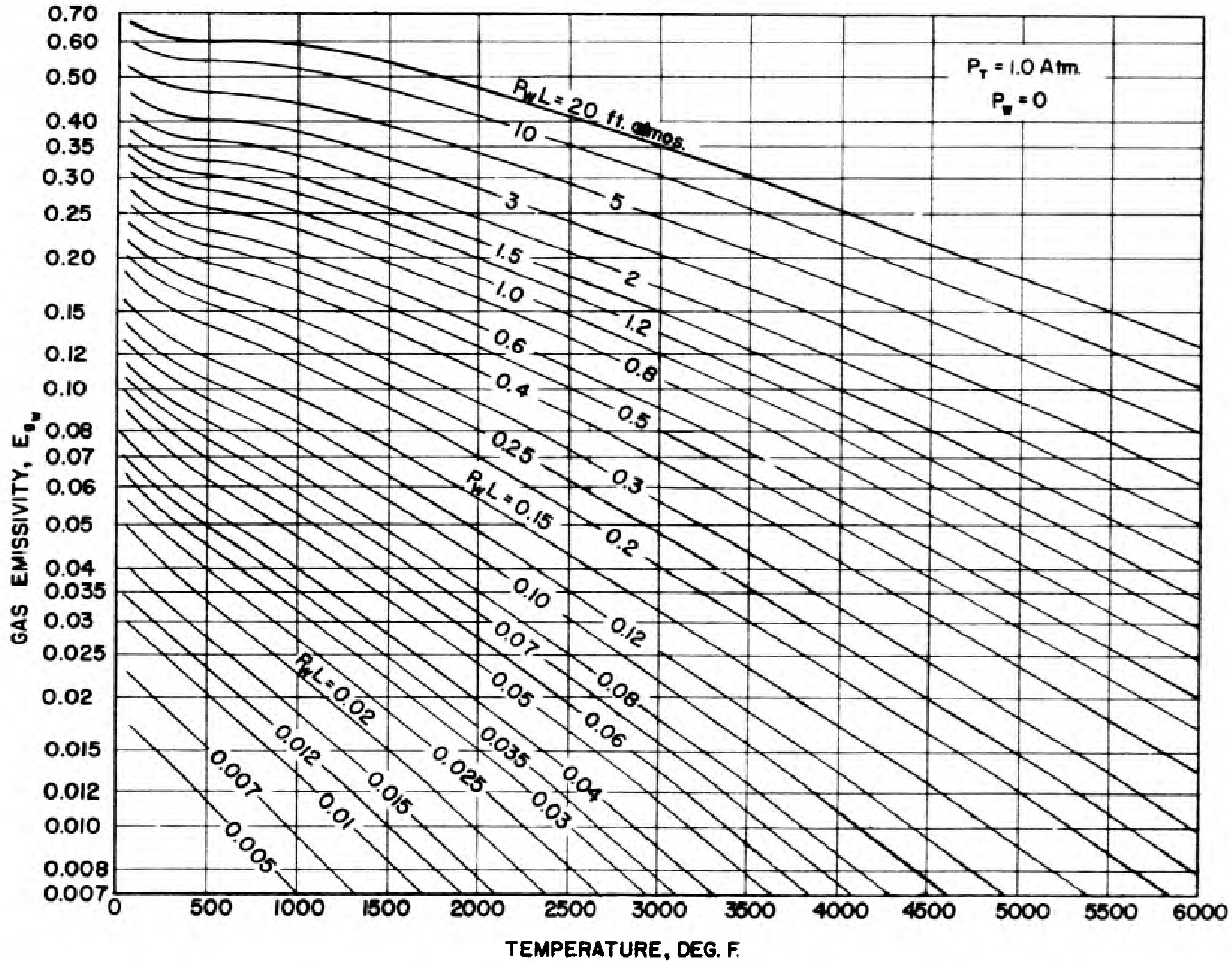

Fig. 11. Emissivity of water vapor.

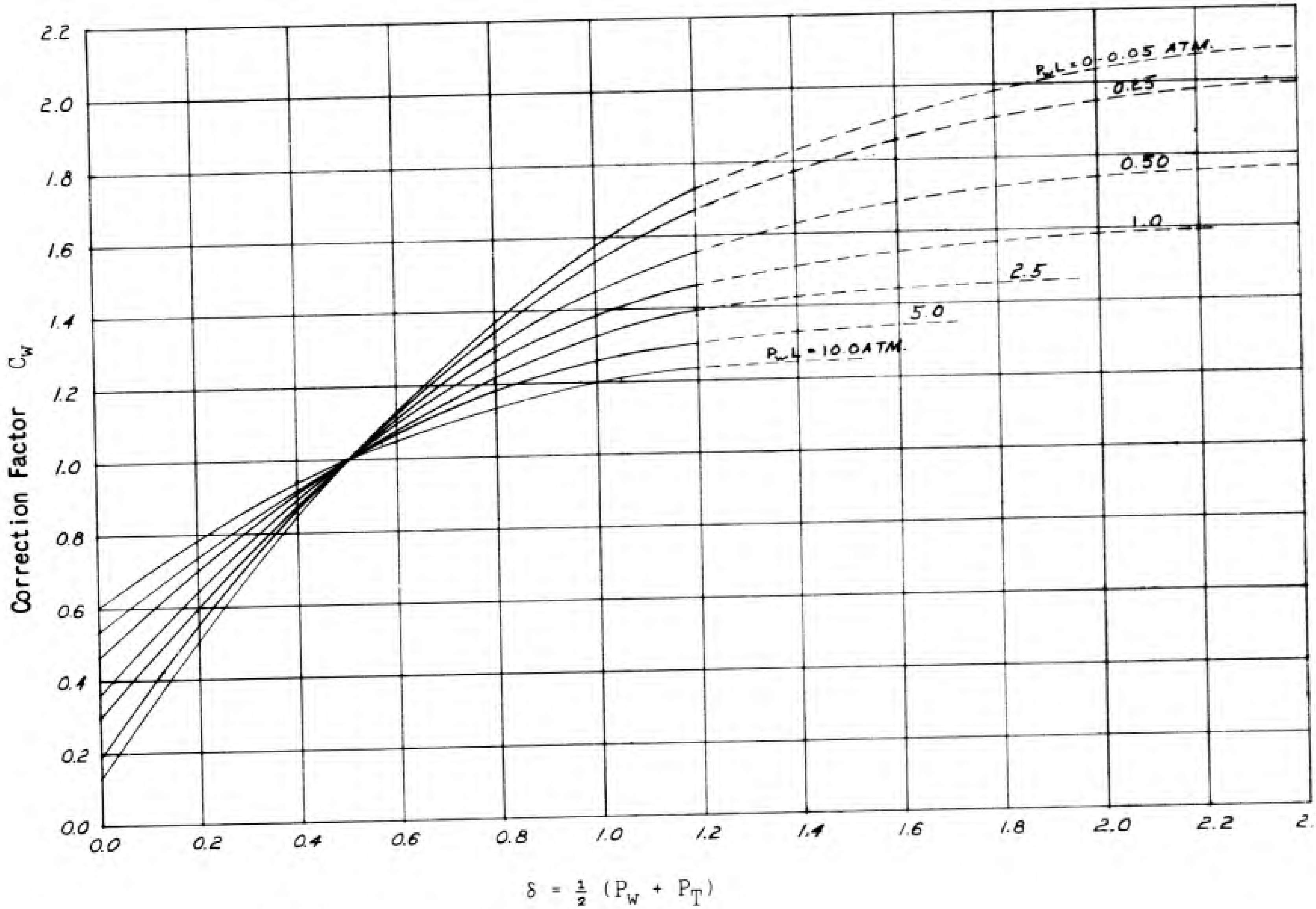

Fig. 12. Effect of total pressure and partial pressure on water vapor radiation.

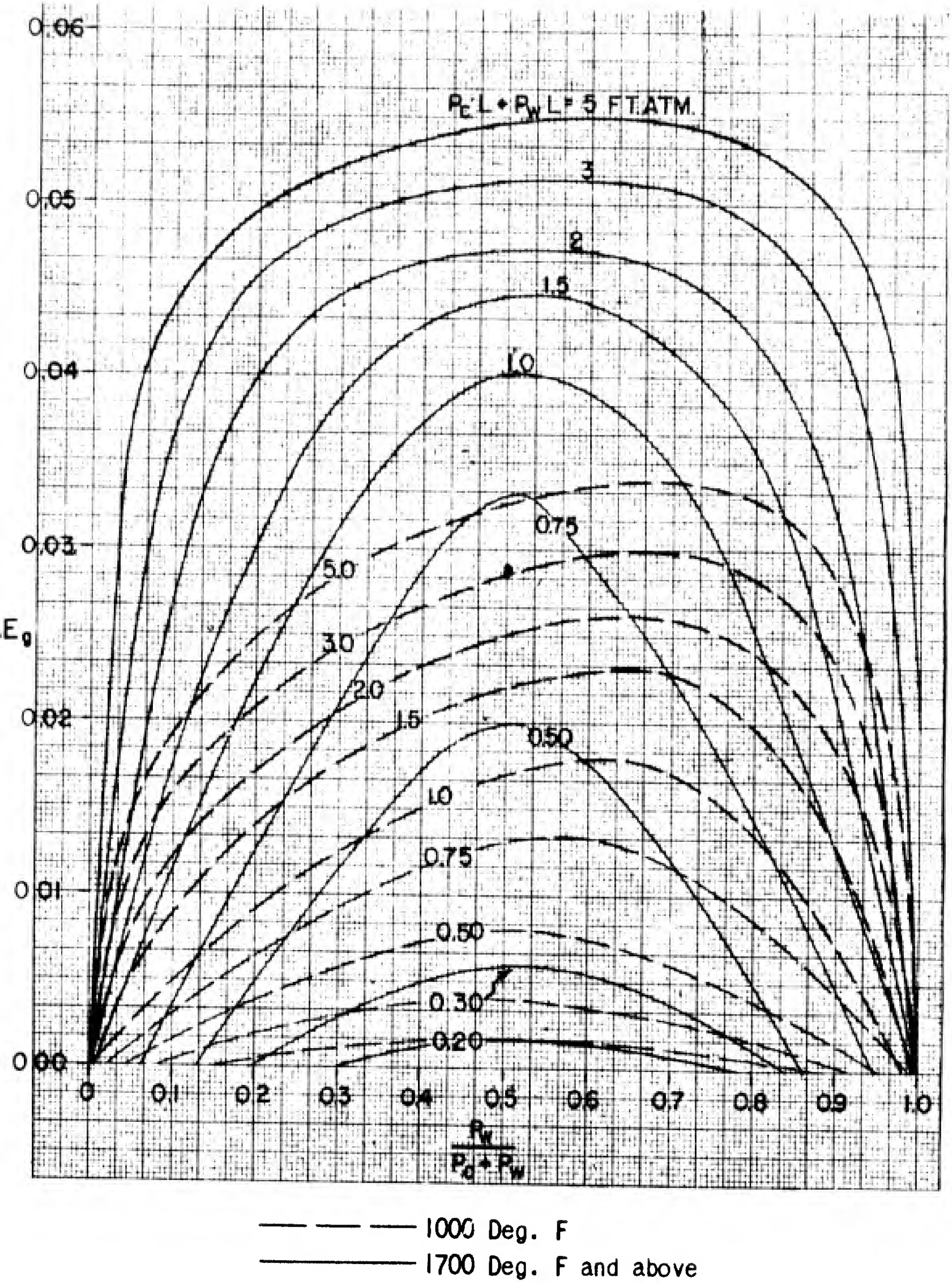

Fig. 13. Correction for superimposed radiation from mixtures of carbon dioxide and water vapor.

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TITLE: An Evaluation of the Heat Transfer Encountered in a Rocket Motor Operating at High Chamer Pressures - and Appendix (Technical Report)

AUTHOR(S) : Warner, C.F.; Zucrow, M.J.

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ABSTRACT:

The heat-transfer problems of rocket engines operating at combustion-chamber pressures up to 2500 psis and utilizing the WFNA (white fuming nitric acid) and octane propellant system were studied. For comparison, the study is extended to include the WFNA-aniline propellant system. The theoretical rates of heat transfer between the walls of a 500-lb thrust rocket engine and the combustion gases resulting from the oxidation of octane and aniline by WFNA are obtained as a function of chamber pressure. The convective-heat transfer coefficients, obtained by using the McAdams correlation, are compared with the rate obtained by the Humble, Lowdermilk, Grele correlation. The radiant-heat transfer rates are obtained by the Hottle, Egbart method. It was found that the chamber and nozzle heat transfer rates increase linearly with chamber pressure and that the nozzle heat transfer rates become extremely high at chamber pressures beyond 1500 psia.

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