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Report No. 1-20

AN INVESTIGATION OF A NUMBER OF LIQUID PROPELLANTS AND A STUDY OF SCALE EFFECT ON JET MOTOR PERFORMANCE

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California Institute of Technology
October 18, 1943

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AIR CORPS JET PROPULSION RESEARCH

GALCIT Project No. 1

Report No. 20

for the

AAF MATERIEL CENTER, AIRCRAFT LABORATORY

AN INVESTIGATION OF A NUMBER OF LIQUID PROPELLANTS

AND

A STUDY OF SCALE EFFECT ON JET MOTOR PERFORMANCE

By

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I. INTRODUCTION AND SUMMARY

A considerable amount of work has been done at the Air Corps Jet Propulsion Research Project with a spontaneously igniting propellant combination consisting of red fuming nitric acid and aniline. (Cf. Ref. 1, 2,3,4,5, and 6) Similar studies have been carried out by the Navy Bureau of Aeronautics Project and by the Aerojet Engineering Corporation. In this report an effort has been made to collect and discuss the results of these investigations and to describe the properties of a number of propellant components.

A desirable propellant is one which gives a high jet velocity and whose components have favorable properties, for example, low vapor pressures and low freezing points. Other properties are discussed in the body of the report. The original propellant, consisting of red fuming nitric acid containing 16% of nitrogen dioxide (NO_2) and aniline, is not completely satisfactory because of the high vapor pressure of the acid and the high freezing point of the aniline. For this and other reasons several modifications of the propellant components have been investigated at the ACJP Project in an effort to obtain a more desirable propellant. These modified propellant components are described in this report and their performance is compared with that of the original propellant.

The oxidizers that have been studied are red fuming nitric acid containing approximately 16% NO_2, red fuming nitric acid containing $6\frac{1}{2}$% NO_2, white fuming nitric acid containing less than 1% NO_2, and a "mixed" acid consisting of 88% white fuming nitric acid and 12% oleum. The fuels are aniline, furfuryl alcohol, and two mixtures of these---one containing 20% furfuryl alcohol, and the other containing 35% furfuryl alcohol.

The tests were made with jet motors delivering a nominal thrust of 1000 lbs. The injectors used were of the multiorifice type. The duration of the individual runs was approximately 20 seconds.

The performance parameters of the various propellant combinations tested are listed in Table I. The characteristic velocity, c^*, is believed to be the most suitable criterion for comparing the performance of the various propellants, since it was found that the thrust coefficient, C_F, was, within the limits of experimental error, the same for all the combinations tested.

The best propellant combination appears to be red fuming nitric acid containing $6\frac{1}{2}\%$ NO_2 and aniline containing 35% furfuryl alcohol. Although this particular combination was not tested, it is recommended because of the low vapor pressure of the acid, and because it is spontaneous and its components remain fluid at ambient temperatures as low as $-22^{\circ}F$. Table I shows that either of the red fuming nitric acids can be used with any of the fuels with no appreciable change in performance.

The Navy Bureau of Aeronautics Project has proposed the propellant combination consisting of "mixed" acid and crude monoethylaniline. The performance of this combination is not yet available so that a comparison with the propellants described in this report cannot be made.

Some of the other liquid propellants under current investigation are briefly described, attention being chiefly directed toward the development of a propellant with higher exhaust velocity. The propellants being studied are gasoline--liquid oxygen and modified gasoline in combination with "mixed" acid.

Tests were made to determine the effect of low ambient temperature on the spontaneity of ignition of a number of propellant combinations. These tests were made with a jet motor delivering approximately 50 lb thrust. The tests showed that above the freezing point of the various combinations the spontaneous ignition property was unaffected. However, it cannot be concluded that the performance of a jet unit will be independent of change in ambient temperature because of changes in physical properties of the propellant components, for example, viscosity, etc. The necessity of considerable investigation of the operation of liquid propellant jet units over the ambient temperature range encountered under service conditions can be anticipated.

Tests have been made at the ACJP Project with jet motors designed to deliver a nominal thrust ranging from 200 lb to 6000 lb at chamber pressures of approximately 300 psi abs and at mixture ratios of from 1 to 5. The motors differed considerably in size; however, they were geometrically similar and all had multi-orifice type injectors. The results obtained show that within the limits of experimental accuracy there is no scale effect on the characteristic velocity c^* and the thrust coefficient C_F, except that in the case of jet motors of around 200 lb thrust the possibility of a 3% to 5% decrease in C_F has not yet been eliminated.

The authors wish to express their appreciation to Dr. Th. von Karman for his direction of the research program, to Dr. B. H. Sage, Dr. E. W. Hough and Mr. J. M. Green for their cooperation in supplying information on the properties of propellant components, to Dr. M. Summerfield for the use of his unpublished data on jet motor performance, to Dr. H. S. Tsien for his helpful suggestions, and to the Navy Bureau of Aeronautics Project for making available data on their work. The skillful assistance of Messrs. R. C. Terbeck and Wm. Stephenson in carrying out the experimental work was invaluable.

II. GENERAL DISCUSSION OF LIQUID PROPELLANTS FOR PURE JET OR ROCKET PROPULSION

A pure jet or rocket propulsion unit is a type of thrust-producing apparatus whose thrust results from the reaction or recoil of a high velocity gas jet. This type of jet unit is distinguished by the fact that it does not utilize atmospheric air, and can operate in a vacuum. The jet gas is generated by the combustion of a propellant in a jet motor. Since atmospheric air is not used, the propellant must contain, in addition to the fuel, an oxidizing agent, generally in the form of a second component. If the propellant consists of a single component, this must contain available oxygen in its chemical structure. The propellant may consist of more than two components, e.g., an inert component may be utilized for reducing combustion temperature.

The propellant can consist of gaseous, liquid or solid components or some combination of the three forms of matter. This report contains the results of investigations on propellants having two components both of which are in liquid form.

The designer of liquid propellant jet units requires a great amount of information on each component of a propellant in addition to the jet motor performance parameters of the propellant. The following information is needed:

1. Physical properties of each propellant component:
    a. Freezing point
    b. Vapor pressure as a function of temperature.
    c. Specific gravity as a function of temperature.
    d. Specific heat as a function of temperature.
    e. Viscosity as a function of temperature.

f. Heat of vaporization.

g. Heat conductivity as a function of temperature.

2. Chemical properties.

a. Heat of formation

b. Effect on materials of construction

c. Effect of diluents and impurities

d. Stability

e. Effect of ambient temperature on ignition.

3. Physiological properties of each propellant component.

a. Effect of inhaled vapors; recommended precautions and treatment.

b. Effect of skin contact; recommended precautions and treatment.

4. Other information on each propellant component.

a. Availability and uniformity of composition.

b. Handling methods

c. Cost

To supply all the information desired on each propellant component represents a vast amount of work, especially because the propellant components utilized in jet propulsion equipment often have not had wide industrial application. This report contains some of the data required on a number of propellant combinations.

In general the components of a liquid propellant should satisfy the following requirements:

1. High heat of combustion per pound of propellant to make possible a high exhaust velocity and therefore low specific propellant

consumption. For a given heat of combustion the exhaust velocity will be higher or the combustion temperature lower if the propellant is such that the molecular weight of the products of combustion is as low as possible.

2. Low freezing point - to allow application over a wide ambient temperature range.

3. Low vapor pressure - to facilitate pump operations, and to minimize storage and transportation problems.

4. Low toxicity and corrosiveness - to reduce hazards to personnel and to simplify requirements on materials of construction.

5. Large availability of raw materials and ease of manufacture - to reduce cost and to simplify logistic problems.

The Air Corps Jet Propulsion Research Project has carried out extensive research with a propellant consisting of red fuming nitric acid as the oxidizer and aniline as the fuel. (Cf. Refs. 1, 2, 3, 4 and 5). This propellant however has a number of limitations, and efforts have been made to improve the properties of each component without creating new disadvantages.

The modifications in the oxidizer have been aimed largely at reducing its vapor pressure. This is brought about by the use of a fuming nitric acid with a smaller NO_2 content. The original oxidizer contained about 16% of NO_2; the modified oxidizers are (1) red fuming nitric acid with 6.5% of NO_2 and (2) white fuming nitric acid with less than 1% of NO_2. Changes in the fuel have been made in an attempt to widen the temperature range of applicability by developing a fuel with low freezing point. The original fuel was aniline; the other fuels studied are (1) aniline containing 20% of furfuryl alcohol, (2) aniline containing 35%

of furfuryl alcohol, and (3) furfuryl alcohol.

A further departure from these combinations has been proposed by the Navy Bureau of Aeronautics Project. The propellant proposed consists of a "mixed" acid, made up of nitric and sulfuric acids, as the oxidizer, and crude monoethylaniline as the fuel. The "mixed" acid contains 88% of white fuming nitric acid (95% or more HNO_3) and 12% of oleum (with 20% SO_3). This oxidizer is interesting because of its low vapor pressure, its low rate of corrosion of mild steel, and its probable low sensitivity to water content.

The use of liquid nitrogen tetroxide, N_2O_4, as an oxidizer has been considered, and some experimental work with this material has been done at GALCIT Project No. 1. However, nitrogen tetroxide has certain physical properties which make it an undesirable propellant component. Its boiling point is about 70°F., and hence its vapor pressure at ambient temperatures is much too high for convenient operation. Furthermore, this material has a relatively high freezing-point, 15°F., and hence it can be used only over a very limited temperature range.

For higher exhaust velocities, fuels with higher heats of combustion must be used. Gasoline is such a fuel, and it is being studied at GALCIT Project No. 1 in conjunction with liquid oxygen.

The use of gasoline, with red fuming nitric acid as the oxidizer, has been studied at some length at GALCIT Project No. 1 (Cf. Refs. 7,8,9, and 10). This work was suspended when serious difficulties were encountered with ignition and steady combustion. Efforts are now being made either to make this propellant combination spontaneously ignitable or to reduce its reaction time so that auxiliary ignition is reliable. This is being done by the addition of various materials to the gasoline. Mixed xylidines (suggested by Dr. Ewing, of the Aerojet Engineering Corporation), mixed toluidines, and monoethylaniline are among the materials under investigation. As a part of this program, plans are being made to test these gasoline mixtures with the "mixed" acid.

III. LIQUID PROPELLANT JET MOTOR PERFORMANCE PARAMETERS

The merit of a jet motor is judged by the effective exhaust velocity attained by the products of combustion. The thrust is given by the equation:

$$F = \frac{W}{g} c$$

so that the effective exhaust velocity is

$$c = \frac{Fg}{W}$$

The expression for effective exhaust velocity can be rewritten as:

$$c = \frac{Fg}{W} = \left(\frac{p_c f_t g}{W}\right) \left(\frac{F}{p_c f_t}\right) = c^* C_F$$

In this equation the characteristic velocity, c^*, and the thrust coefficient, C_F, are defined. These parameters of jet motor performance involve only the primary experimental measurements of thrust, chamber pressure, exhaust nozzle throat area, and rate of propellant consumption.

The theory of jet motor performance yields corresponding expressions for c^* and C_F. It is assumed that the gases in the combustion chamber follow the perfect gas laws and undergo an isentropic expansion.

$$c^* = \frac{p_c f_t g}{W} = \frac{1}{\left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{2(\gamma-1)}}} \sqrt{\frac{g R T_c}{\gamma M}}$$

$$C_F = \frac{F}{p_c f_t} = \sqrt{\frac{2\gamma^2}{\gamma-1} \left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{(\gamma-1)}} \left[1 - \left(\frac{p_o}{p_c}\right)^{\frac{(\gamma-1)}{\gamma}}\right]}$$

Where:

f_t = Exhaust nozzle throat area.

g = Acceleration due to gravity.

M = Average molecular weight of the products of combustion.

p_c = Chamber pressure (absolute).

p_o = External pressure (absolute).

R = Universal gas constant [1544 (ft lb)/ (lb mole) (^oF)]

T_c = Absolute temperature of the products of combustion.

w = Rate of propellant consumption.

$\gamma = \frac{C_p}{C_v}$ ratio of specific heats of the products of combustion.

The expression for C_F is for an exhaust nozzle expanded to the optimum area ratio. The effect of a change in area ratio can also be calculated (Cf Ref 11).

The design of a liquid propellant jet motor depends on a knowledge of the interrelation of the parameters c, c*, and C_F. These three parameters depend in turn on such variables as the burning volume in the combustion chamber, the chamber pressure, the composition and the mixture ratio of the propellant components, the effectiveness of the mixing of the propellant components brought about by the injection process, and the contour of the exhaust nozzle passage.

Theoretically, C_F depends only on the pressure ratio, the area ratio, and the ratio of specific heats of the exhaust gases for exhaust nozzles of reasonably smooth contour. The agreement of experimental results with theoretical values is very close, as will be shown in the discussion of the test data.

The experimental determination of the characteristic velocity, c*, of a propellant is affected by a variety of factors. The volume of the combustion chamber must be adequate to insure complete burning of the propellant, and the ratio of the chamber volume to the exhaust nozzle throat area, L*, is used to define this volume. The geometry of the combustion chamber is further defined by the fact that it is desirable to have its diameter several times larger than the exhaust nozzle throat diameter in order to reduce the gas velocity and the consequent rate of heat transfer to the combustion chamber walls. It is also necessary to allow a certain axial distance between the injector and the exhaust nozzle, though this depends on the type of injection used.

The temperature and equilibrium composition of the products of the propellant combustion are affected by the combustion pressure and by the propellant components and their mixture ratio. Propellants having high heats of combustion and yielding products of combustion having low molecular weights are most favorable for optimum jet motor performance. Such a combination gives the highest exhaust velocity for a given heat of combustion or for a given combustion temperature.

This general conclusion follows from a consideration of the equations obtained by considering the behavior of perfect gases (Cf. Ref. 11).

$$c = \sqrt{2g\left[1-\left(\frac{p_o}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]H_c} \qquad (6)$$

where

$$H_c = \left(\frac{\gamma}{\gamma-1}\right)\frac{R}{M}T_c \qquad (7)$$

In these equations H_c is the absolute enthalpy per unit weight of the products of combustion in the chamber and T_c is their absolute temperature. In ordinary practice the heat of combustion of the propellant, H_p, is often used as an approximation for H_c. The term $\left[1-\left(\frac{p_o}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]$ in equation (6) is the ideal thermodynamic efficiency of the isentropic process by which the heat energy of the gas in the combustion chamber is transformed to kinetic energy while expanding through the exhaust nozzle to the exit pressure, $p_e = p_o$, and assumes a correctly formed nozzle.

The average molecular weight of the products of combustion is usually related to the gas constant, γ , in such a manner that the product

$\left(\frac{\gamma}{\gamma-1}\right)\frac{1}{M}$ varies inversely with M. A low value of M tends to increase γ and increase the thermodynamic efficiency of the process in equation (6). From expression (7), it is apparent that for a given absolute enthalpy, or heat of combustion, the temperature will decrease with a decrease in M, or, conversely, a propellant with a higher heat of combustion can be used without exceeding a given chamber temperature if the average molecular weight of the products of combustion is low.

The method of propellant injection has an important effect on the combustion process and on the optimum combustion chamber geometry. The direction and velocity of the propellant streams should be such that good mixing and dispersion are produced and no large droplets hit the combustion chamber walls. The minimum combustion volume required and the minimum length of the combustion chamber are affected by these factors.

It has been shown experimentally that the characteristic velocity of a propellant is not affected by the expansion of the exhaust nozzle (Cf. Ref. 4). This means that, for the exhaust nozzles tested, the process of combustion or heat liberation is not influenced by the expansion in the nozzle. In other words, the expansion of the gases in the nozzle approaches the theoretical adiabatic process very closely. However it is possible that by using very long nozzles the time interval during which the expansion is carried out can be greatly lengthened. Then the gases in the exhaust may be able to reach new chemical equilibriums corresponding to the prevailing temperatures and pressures in the nozzle. Such a shift in equilibrium is usually accompanied by heat liberation, and therefore the expansion in the nozzle can be other than adiabatic. In such cases, the variation in expansion process in the nozzle will change the characteristic velocity and the thrust coefficient, and the complete separation of the combustion process and the expansion process is no longer possible.

The fortunate choice of short nozzles enables the completely separate determination of c* and C_F from the experimental data and makes it possible to break down the jet motor performance into two distinct parts, propellant effectiveness and exhaust nozzle characteristics.

The performance of various propellants is thus most easily judged by comparing the values of c* obtained over a common range of combustion chamber pressure, assuming that the injector mixing is satisfactory and the combustion volume is adequate.

IV. DESCRIPTION OF EXPERIMENTAL WORK

A. Test Equipment

1. Test Pit

The tests were performed in Pit B on the Project premises. This pit has a concrete walled test floor and an observation and control room with windows through which the jet motor can be observed. The pit is described in detail in Ref. 4.

2. Test Equipment Circuit

The basic circuit diagram of the test equipment is shown in Fig. 1. This shows the rotating thrust-jack piston and the water cooling system used with the aluminum cooled motor. These two features were not available during many of the tests.

The motors and injectors are described elsewhere in this report. Information on the other items shown on the circuit diagram, such as the feed pressure regulator and the control valves, can be obtained from the various drawings mentioned on Fig. 1.

The propellants were forced into the combustion chamber from the propellant tanks by nitrogen pressure.

3. Test Motors

Three jet motor assemblies were used during the series of tests. The first motor, shown on Fig. 2, was similar to that used in the tests of Ref 4, except for a slightly thicker combustion chamber wall and a longer chamber which provided more burning volume. The injector and the exhaust nozzle were held in place by threaded collars. This motor could not be operated for more than 20 seconds at a time with the hotter propellant combinations without overheating the exhaust nozzle. Even with these short runs the exhaust nozzles gradually deformed by plastic flow and had to be replaced.

The second motor, shown in Figs 3 and 4 was assembled in a different manner. The external collars at each end of the chamber were abandoned and the injector and the exhaust nozzle were threaded directly into the chamber. A reinforcing ring at each end of the chamber kept the threaded portions from warping. All interior surfaces of this motor were chrome plated. The motor was mounted from the flange at the injector end of the chamber. This motor gave good service, but it too was subject to overheating and could be used only for short duration runs.

The third motor was adapted from a design originally developed for regenerative cooling (Cf Ref 5). The threaded injector and exhaust nozzle feature was retained because of its simplicity and effectiveness. The assembly of this motor is shown on Figs 5 and 6. Water was supplied to the cooling passage through the chamber at the rate of approximately 1.8 pounds per second.

The aluminum cooled motor permitted runs of 30 seconds or more duration to be made according to the propellant tank capacity, and thus enabled the average rate of propellant consumption to be determined with satisfactory accuracy.

The injectors used in the motors are shown on Figs 7 and 8.

The exhaust nozzles used are shown on Figs 9, 10 and 11. All the exhaust nozzles were made of copper with chrome plate on the inside surface.

It should be mentioned here that all motors used developed faults in the tests, and none of them can be recommended as production designs. The details of motor design and construction are not considered essential to the subject of this report.

Lt. Stiff of the Navy Bureau of Aeronautics Project has kindly made available the drawings shown in Figs 12 and 13. The original design of this type of motor was carried out by Lt. Truax of the same Project. The chamber, exhaust nozzle and injector of this motor, designed to deliver 1500 lb thrust at a chamber pressure of 300 psi, differ radically in geometry from the types tested at the ACJP Project. The performance of this motor will be discussed in Part VII, Section C. A typical chamber pressure curve is shown in Fig 14.

B. Experimental Measurements

1. Thrust Measurement

The jet motor was mounted on top of a parallelogram frame, and the motion of the motor was restrained by a piston type hydraulic thrust jack. The piston had an area of 1 square inch, so that the thrust in pounds was equal to the pressure in psi of the fluid in the chamber of the thrust jack.

The hydraulic lines were carefully bled, so that there was little play in the system. This minimized the forces introduced by the deflection of the propellant lines.

The thrust was measured with a Bourdon tube pressure gage reading from 0 to 2000 psi. The accuracy in reading this gage from the film was about ± 5 psi, or approximately ± ½% of the normal thrust of 1000 pounds. The overall accuracy of the thrust measuring system was approximately ±1% when the piston of the thrust jack was rotated and the propellant valves were elastically mounted.

Previous to run 536 the piston was not rotated, and friction of the piston in the cylinder accounted for errors of up to -5%. Previous to run 624 the lines from the propellant valves to the motor were restrained in such a manner that errors in thrust of -3% were sometimes present. Figs 4, 6 and 15 show the thrust jack and the motor and gears used to rotate the piston.

2. Chamber Pressure

The combustion chamber pressure was measured with a Bourdon tube pressure gage. The gage had a 0 to 1000 psi range and could be read to within ±3 psi, or approximately ±1% of the normal reading of 300 psi.

3. Exhaust Nozzle Throat Area

The diameter of the throat of the exhaust nozzle was measured before and after each run with a telescope gage and micrometer calipers. The average of these two readings was used in computing the throat area.

The diameter of the throat immediately after a run, when the nozzle was still hot, was up to ½% greater than the diameter measured before the run, when the nozzle was cold. The uncertainty as to the actual diameter during the run and the difficulty of taking measurements of the throat make an accuracy in throat area of not more than ±2% probable.

4. Propellant Consumption

The total propellant consumption during a run was found by measuring the change in level in the cylindrical propellant supply tanks. The level was indicated on a sight glass which was shut off from the tank while pressure was applied. The specific gravity of each propellant component was measured before each run. From this data, combined with the calibration factors of the tanks, the total weight of the propellant used could be determined to an accuracy of approximately ±2%.

5. Time Measurement

The duration of the runs was determined from an electric clock with a sweep second hand which was photographed with the pressure gages. The method of determining the effective time of the run is described in Part IV, Section C. The accuracy of this determination varied from approximately ±1.2% to ±.8%, according to the length of the run.

C. <u>Reduction of Test Data</u>

The steps in the reduction of the test data were as follows:

1. The film was read to give uncorrected values of chamber pressure, p_c, thrust jack pressure gauge reading, p_f, and time, t.

Regulated feed pressure, p_r, was also recorded.

2. The propellant tank sight glass measurements and the measured exhaust nozzle throat diameter were recorded on the run data sheet. Also recorded on this sheet were rough gauge readings and other miscellaneous data pertinent to the run, such as the appearance of the jet and notes on the operation of the equipment.

3. The total propellant consumption was computed from the tank measurements, the tank calibration factors, and the specific gravity of the propellants. The exhaust nozzle throat area was computed from the average of the measured diameters.

4. The uncorrected values of p_c and p_f were plotted against time (Cf Fig 16).

5. From the plotted curves, the time interval during which approximately constant conditions prevailed was determined and an arithmetical average of the values of p_c and p_f over this period was computed.

6. The effective duration of the run was determined by dividing the total area under the p_c curve by the average steady state value of p_c obtained in step 5. A planimeter was used to obtain the area under the p_c curve. The curve of p_c was used for this purpose because there was less lag in these pressure indications and because the readings were usually more steady.

7. The pressure gauge calibrations were applied to the average values of p_c and p_f. An average atmospheric pressure of 14.0 psi was used to obtain the absolute chamber pressure. The corrected value of p_f was multiplied by the thrust jack

calibration factor to obtain the average thrust during the run.

8. These corrected quantities were used to compute the propellant performance characteristics as follows:

$$w = \frac{W_f + W_o}{t_{eff}} \qquad c^* = \frac{p_c f_t}{w\, g}$$

$$r = \frac{W_o}{W_f} \qquad C_F = \frac{F}{p_c f_t}$$

$$c = \frac{F g}{w \cancel{g}} = c^* C_F$$

The approximate accuracy of the experimental measurements and of the computed quantities is as follows. These percent errors apply to a run of 25 seconds duration with a thrust of 1000 lbs and a chamber pressure of 300 psi absolute. As the time, thrust, or chamber pressure decrease the percent errors of the measurements can be expected to increase.

Quantity	Accuracy (probably error)
F	± 1% (without systematic errors)
p_c	± 1%
f_t	± 2%
W	± 2%
t	± 1%
w	± 2.3%
r	± 2.9%
c^*	± 3.2%
C_F	± 2.5%
c	± 2.5%

Attention is again directed to the fact that fairly large systematic errors were possible in the thrust measurement throughout the greater part of this series of tests. These errors affect the accuracy of the determination of C_F and c. Only random errors of the magnitude indicated are believed to enter into the determination of c*.

V. PROPERTIES OF THE PROPELLANT COMBINATIONS INVESTIGATED AND THEIR PERFORMANCE IN JET MOTORS*

A. 16% Red Fuming Nitric Acid and Aniline

1. Properties

a. Red fuming nitric acid

The specifications for this acid include a minimum of 13% NO_2 and a maximum of 3% water. The acid whose performance is described here has varied in NO_2 content from 16% to 20%, and in water content from 3% to 5%. It will hereinafter be called 16% RFNA. Manufacturing problems connected with the absorption of NO_2 with nitric acid made it difficult to keep the water content consistently below 3%. It is believed that the performance of this acid is significantly impaired only when the water content exceeds 4-5%.

The bubble-point pressure, specific weight, and viscosity of this acid, at various temperatures, are given in Figs. 17, 18 and 19 (Cf. Ref. 3). In addition, some thermodynamic data for pure nitric acid and for nitrogen tetroxide are given in Tables III, IV, V and VI (Cf. Ref. 9, 12 and 13).

*Most of the properties listed for the propellent components have been determined, or assembled from the literature, by B. H. Sage, E. W. Hough, and J. Green, of the Department of Chemical Engineering, California Institute of Technology.

The bubble-point pressure of 16% NO_2 acid is quite high, being 50 psi abs at 155.8°F., and estimated to be 15 psi abs at slightly over 100 F. This high vapor pressure, even at ambient temperatures, is a disadvantage in pumping operations and presents difficulties in transportation and storage of the acid. In addition this acid gives off extensive NO_2 fumes, which are hazardous to personnel.

The high specific weight of this oxidizer is a definite advantage in jet propulsion, since it reduces the size of tank needed for a given weight of oxidizer.

The freezing-point of the acid, which is below -43.6°F., is sufficiently low to insure a wide ambient temperature range of operation.

The use of red fuming nitric acid as a jet propulsion oxidizer presents certain difficulties with materials of construction because of its highly corrosive action on most metals and on almost all organic materials. Stainless steel, aluminum, chrome-plate, and duriron have been found to be most resistant to the acid at various temperatures (Cf. Ref. 6). Among the organic materials, a plastic called Saran is somewhat resistant, and has limited applicability (Cf. Ref. 14).

b. Aniline

The bubble-point pressure, specific weight, and viscosity of aniline, at various temperatures, are given in Figs. 20, 21 and 22, (Cf. Refs. 3, 15, 16). In addition, some thermodynamic properties are given in Table VII.

The chief disadvantages in the use of aniline are its relatively high freezing point (21.02°F.) and its toxicity. The freezing point of a propellant component establishes the lower ambient temperature limit of operation, and a lower limit of 21°F. does not permit a wide enough range of applicability. The toxicity of aniline is such that definite precautions must be taken against absorption through the skin, nose and mouth. Inhaling of aniline vapor is dangerous; also, aniline can be absorbed through the unbroken skin, and such contact should be avoided.

A property of aniline that makes it useful as a jet propulsion fuel is its spontaneous ignition with various fuming nitric acids. This eliminates the necessity of an ignition device, and all of its attendant difficulties.

2. Reaction of Propellant Components

A detailed theoretical study of the reaction of 15% red fuming nitric acid and aniline has been made by Hough, Green and Sage, (Cf. Ref. 3). In this study it was assumed that equilibrium is attained. The calculated products of combustion at various mixture ratios and at chamber pressures of 300 and 600 psi are shown in Table VIII, and in Fig. 23, (the latter for 300 psi chamber pressure only). The calculated reaction temperature, exhaust velocity, and specific impulse are also shown in Table VIII and in Fig. 24.

These calculations show that, as the mixture ratio increases, the mole fractions of water and of carbon dioxide in the products of combustion increase, and the mole fractions of hydrogen and of carbon monoxide decrease. At a mixture ratio of 5, the mole fraction of free oxygen in the products of combustion becomes appreciable.

It is interesting to note that, according to the calculations, there is only a very slight increase in reaction temperature (about 100°F.) when the chamber pressure is increased from 300 psi to 600 psi. There is, however, about a 10% increase in exhaust velocity due to the increase in the thermodynamic efficiency of the expansion process (Cf. Part III, equations 6 and 7).

A complete discussion of these calculated results can be found in Reference 3.

Fig. 25, (Cf. Ref. 21) shows how mixture ratio influences various properties of the reaction products -- γ , average molecular weight, and heat capacity. Here again, as the chamber pressure is changed from 300 psi to 600 psi, there is very little change in these properties.

3. Performance

The performance of the 16% red fuming nitric acid and aniline was determined in the motor shown in Figs. 3 and 4. The results are given in Table IX and plotted in Fig. 26. The performance parameters were determined over a chamber pressure range from 227 to 337 psi abs and a nominal mixture ratio of 2.6. These results are discussed in more detail in Part VII.

B. <u>6-1/2% Red Fuming Nitric Acid and Aniline</u>

1. Properties

a. Red Fuming Nitric Acid

The specifications for this acid stipulate an NO_2 content of 6-1/2 ($\pm$1)% and a maximum water content of 1½%. However, manufacturing problems make it difficult to keep the water content down to the specified maximum, and the acid used has had between 1.7% and 2.2% of water. The acid whose properties are given here was commercial grade, and contained, by analysis,

91.50%	HNO_3
6.41%	NO_2
2.09%	H_2O

The water is determined by difference, with the assumption that these three are the only constituents.

The bubble-point pressure, specific weight, and viscosity of this acid are given in Figs. 17, 18 and 19. (Cf. Ref. 20)

The bubble-point pressure of this acid is somewhat lower than that of the high NO_2 acid described in Part V, Section A, 1, a, at temperatures below 150°F. At 110°F., the vapor pressure of the 6-1/2% NO_2 acid is about 5 psi abs. Pumping operations are more easily carried out, and transportation and storage problems are less severe than with acid of higher vapor pressure. In addition, the acid with lower vapor pressure fumes less extensively when exposed to the atmosphere, and the hazard to personnel is somewhat reduced.

It is believed that the decrease in NO_2 content of the acid from 16% to 6-1/2% will not adversely affect the rates of corrosion of the various metals used in contact with the acid. There may actually be a decrease in the rates of corrosion.

b. Aniline

The properties of aniline are discussed in Part V, Section A, 1, b.

2. Performance

The performance parameters of this propellant combination were investigated to determine the effect of reducing the NO_2 content in the nitric acid from a nominal value of 16% to 6-1/2%. The motor used in these tests is shown in Figs. 3 and 4. The results obtained with the 6-1/2% NO_2 content nitric acid and aniline are given in Table IX, and C_F, c^* and c are plotted against chamber pressure range from 135 to 335 psi abs and a nominal mixture ratio of 2.65.

C. White Fuming Nitric Acid and Aniline

1. Properties

a. White fuming nitric acid

White fuming nitric acid is essentially pure HNO_3. However, the acid frequently contains a small amount of water that has not been removed in the manufacture of the acid; also, a small amount of NO_2 is often present from the decomposition of HNO_3. The fumes which appear when the acid is opened to the air are actually a fog of nitric acid droplets produced by the dissolving of HNO_3 vapor in atmospheric water.

The white fuming nitric acid used in the jet motor tests described below was commercial grade, and contained about 0.3 to 0.6% of NO_2 and about 1.5% of water. This is somewhat different in composition from the acid whose physical properties are here listed; this acid was a chemically pure white fuming acid containing 0.54% of NO_2 and 13% of water.

The bubble-point pressure, specific weight, and viscosity of this acid are given in Figs. 17, 18 and 19, (Cf. Ref. 3).

Because of the very low NO_2 content, the bubble-point pressure of this acid is considerably lower than that of red fuming acids. This would be an davantage in jet motor operation. However, there seems to be evidence that the ignition characteristics of an acid of very low NO_2 content are more sensitive to slight changes in water content; this would be a definite disadvantage.

Another disadvantage is brought about by the fact that the specific weight of white fuming acid is about 5% lower than that of 16% NO_2 red fuming acid. This would make necessary slightly larger acid tanks for a given weight of oxidizer.

The investigation of white fuming nitric acid has been discontinued, and the study of the mixed acid oxidizer described in Part II has been substituted. This latter oxidizer should have the same advantage of low vapor pressure, and it should be superior to white fuming nitric acid in that slight increases in water content have less effect on spontaneous ignition.

b. Aniline

The properties of aniline are discussed in Part V, Section A, 1, b.

2. Performance

The performance parameters of the white fuming nitric acid-aniline combination were determined in the jet motors shown in Figs. 3 and 4. The first motor had an $L^* = 100$ in. and $\epsilon = 3.5$, the second $L^* = 45.5$ and $\epsilon = 4.0$. The results obtained are given in Table IX, and C_F, c^* and c are plotted as functions of chamber pressure in Fig. 28. The tests were carried out over a chamber pressure range from 235 to 486 psi abs and a nominal mixture ratio of 2.6.

D. "Mixed" Acid and Aniline

1. Properties

a. "Mixed" acid

The "mixed" acid is made by adding 12 parts by weight of oleum (fuming sulfuric acid) to 88 parts of white fuming nitric acid. The oleum contains 20% of sulfur trioxide, and the fuming nitric acid contains 95% or more of HNO_3. When chemically pure materials are used, the resulting acid is a clear solution: However, with technical grade acids, a greenish-white gelatinous precipitate is formed when the acids are mixed. The formation of this precipitate has been found to depend upon dissolved iron as an impurity in either or both of the acids, and can be eliminated almost completely by using acids with little or no iron. It is believed that the precipitate is some form of ferric sulfate. The room temperature corrosion rate of steel in "mixed" acid is very slight, and the very smallamount of iron that dissolves does not lead to the formation of much precipitate. "Mixed" acid can be shipped in mild steel containers.

In the "mixed" acid runs described in this report, no attempt was made to remove the small amount of precipitate, and its presence in the acid produced no difficulty. However, it may be desirable to introduce a filter in the filling system.

b. Aniline

The properties of aniline are discussed in Part V, Section A, 1, b.

2. Performance

The performance parameters of the "mixed" acid-aniline combination were determined in the jet motor shown in Figs. 3 and 4. The results obtained are given in Table IX, and C_F, c* and c are plotted as functions of chamber pressure in Fig. 29.

The tests were carried out over a chamber pressure range from 100 to 287 psi abs and a nominal mixture ratio of 2.6.

E. 6½% Red Fuming Nitric Acid and Furfuryl Alcohol

1. Properties

a. 6½% Red Fuming Nitric Acid

The properties of this acid have been discussed in Part V, Section B, 1, a.

b. Furfuryl Alcohol

The vapor pressure, specific weight, and viscosity of furfuryl alcohol at various temperatures are given in Figs. 20, 21 and 22, (Cf. Refs. 21 and 22).

The freezing point of furfuryl alcohol (-23.8°F) (Cf. Refs. 21 and 23) is low enough to permit a wider range of operation than is possible with aniline. The vapor pressure of furfuryl alcohol is quite similar to that of aniline. The slightly higher specific weight is an advantage, in that smaller tanks could be used.

Polymerization of furfuryl alcohol, reported by Dunlop and Peters (Cf. Ref. 24) might be a source of difficulty, since it may lead to the formation of solid or gummy particles which can clog small orifices and other fine clearances. However, Dunlop and Peters have shown that the addition of very small quantities (0.1%) of piperidine, or other organic bases, inhibits polymerization very markedly.

2. Performance

The performance parameters of the 6½% red fuming nitric acid-furfuryl alcohol combination were determined for two values of mixture ratio. Tests at a nominal mixture ratio of 1.6 were carried out over a chamber pressure range from 128 to 326 psi abs in the jet motor shown in Figs. 3 and 4. The results obtained are given in Table IX, and C_F, c* and c are plotted as functions of chamber pressure in Fig. 30.

Tests at a nominal mixture ratio of 2.6 were made in motors of two different geometries. The first series of tests was made in the same motor used for a mixture ratio of 1.6 (Cf. Fig. 3) over a chamber pressure range from 131 to 314 psi abs. The second series was made in the same motor equipped with the exhaust nozzle shown in Fig. 11 over a chamber pressure range from 249 to 490 psi abs. The results obtained are given in Table IX, and C_F, c* and c are plotted against chamber pressure in Fig. 31.

For the first series of tests the motor had L* = 45.5 in. and ϵ = 4.0; for the second series the motor had L* = 100 in., and ϵ = 3.5. It is seen in Fig. 31 that two distinct curves for C_F versus chamber pressure were obtained.

The reason for this effect is believed to lie entirely with the thrust measuring apparatus. Immediately after this series of tests was completed a similar series was made with the same injector, chamber, and exhaust nozzles and a propellant consisting of white fuming nitric acid and aniline. In addition, the restraint on the propellant lines leading to the motor was lessened by elastically mounting the propellant valves. These tests yielded values for C_F for the exhaust nozzle with an area ratio of 3.5 which were substantially higher than the results under discussion (Cf. Figs. 28 and 31).

On Fig. 31 a dotted extension of the C_F curve for the exhaust nozzle with $\epsilon = 4.0$ is proposed as a conservative approximation to the C_F values to be expected from the exhaust nozzle with $\epsilon = 3.5$. At pressure ratios from 20 to 40 the change in ϵ from 4.0 to 3.5 should have little effect on C_F, as will be discussed in Part VII, D, 1. This assumed C_F is used in calculating exhaust velocity and in making comparisons of this propellant with others.

Fig. 31 shows that the characteristic velocity of this propellant remains constant over a wide range of chamber pressure.

F. <u>6½% Red Fuming Nitric Acid and Aniline Containing 20% of Furfuryl Alcohol.</u>

1. Properties

a. 6½% red fuming nitric acid

The properties of this acid are described in Part V, Section B, 1, a.

b. Aniline containing 20% of Furfuryl Alcohol

It has been pointed out above that the high freezing point of aniline (21.02°F) places a definite limitation on the use of aniline as a jet motor fuel. The freezing point can be lowered by various additive agents, and many of these have been investigated (Cf. Refs. 3 and 20). Of all those studied, the best seem to be orthotoluidine and furfuryl alcohol when the following are considered:

(1) Efficiency of freezing-point lowering

(2) Low vapor pressure

(3) Cost and availability

(4) Spontaneity with red fuming nitric acid.

Examination of the freezing-point curves of the aniline-orthotoluidine system and the aniline-furfuryl alcohol system (Figs. 32 and 33 taken from References 3 and 20) indicates several advantages in the use of furfuryl alcohol as the additive agent. The lowest temperature of operation possible with aniline-orthotoluidine is about 0°F (20% of orthotoluidine) unless a mixture is used that is largely orthotoluidine. Furthermore, care would be necessary to see that the composition remained within a limited range, since the freezing-point curves for the aniline-orthotoluidine system rise sharply from the eutectic points. Since these objections do not hold with furfuryl alcohol, it was decided to use this material. A mixture containing 45% of furfuryl alcohol remains fluid and reacts spontaneously with red fuming nitric acid at temperatures down to -45°F.

A possible disadvantage is the polymerization of furfuryl alcohol, mentioned previously, to form objectionable solid resinous masses. As pointed out above, the addition of organic bases to furfuryl alcohol reduces considerably the tendency to polymerize (Cf. Ref. 24). Since aniline is an organic base, it was believed that polymerization would not occur in aniline-furfuryl alcohol mixtures. This was investigated by heating the mixture, in a sealed chamber, in contact with various materials - glass, steel, and copper. In each case, the mixture was heated to a maximum temperature of about 450°F., and was maintained at a temperature of 300°F. or above for 3 hours, and, there was no evidence of objectionable polymerization. In the actual jet-motor tests with aniline-furfuryl alcohol mixtures, there has been no sign of formation of undesirable gummy or solid residues.

The bubble-point pressure, specific weight, and viscosity of this fuel at various temperatures are given in Figs. 20, 21 and 22, (Cf. Ref. 20).

The freezing-point of this fuel, as indicated on the freezing-point curve, Fig. 33, is about -1°F. Since the freezing-point curve was obtained with purified aniline and furfuryl alcohol, this temperature is somewhat higher than the temperature at which a corresponding mixture of commercial grade materials begins to solidify. Thus, the temperatures indicated on the freezing-point curve represent conservative lower limits of fluidity, the actual limits with commercial grade substances being somewhat lower.

2. Performance

The performance parameters of the 6½% red fuming nitric acid - 80% aniline - 20% furfuryl alcohol combination were determined for two values of mixture ratio. Tests at a nominal mixture ratio of 1.4 were carried out over a chamber pressure range from 207 to 272 psi abs in the jet motor shown in Fig. 2. The results obtained are given in Table IX, and C_F, c^* and c are plotted as functions of chamber pressure in Fig. 34.

Tests at a nominal mixture ratio of 2.5 were made in different motors having $L^* = 45.5$, 54.6 and 64.2 in. The motors used are shown in Figs. 2, 3, and 5. The results obtained are given in Table IX, and C_F, c^* and c are plotted as functions of chamber pressure in Fig. 35.

The fact that the characteristic velocity at low chamber pressures is lower for the motor with L* = 45.5 in. than for the motor with L* = 54.6 may indicate that the combustion volume of the former motor was marginal or inadequate, though the differences may also be ascribed to experimental scatter.

G. 16% Red Fuming Nitric Acid and Aniline Containing 35% of Furfuryl Alcohol.

1. Properties

a. 16% red fuming nitric acid

The properties of this acid have been discussed in Part V, Section A, 1, a.

b. Aniline containing 35% of furfuryl alcohol

The reasons for the selection of furfuryl alcohol as the additive agent to lower the freezing point of aniline have been discussed in Part V, Section F, 1, b.

The bubble-point pressure, specific weight, and viscosity of this fuel are given in Figs. 20, 21 and 22, (Cf. Ref. 20).

The freezing point of a mixture of aniline and furfuryl alcohol containing 35% of the latter is about -22°F., as indicated on the freezing-point curve, Fig. 33. However, as explained in Part V, Section E, 1, b, -22°F. is a conservative lower limit of fluidity; commercial grade materials solidify at somewhat lower temperatures.

2. Performance

The performance parameters of the 16% red fuming nitric acid-65% aniline - 35% furfuryl alcohol combination were determined for a nominal mixture ratio of 2.5. The two motors used are shown in Figs. 3 and 2 and had L* = 45.5 and 64.2 in. respectively.

The results obtained are given in Table IX, and C_F, c^* and c are plotted as functions of chamber pressure in Fig. 36.

The change in L^* during these tests did not affect the characteristic velocity, which shows only a slight increase with increasing chamber pressure.

VI. EFFECT OF LOW AMBIENT TEMPERATURE ON SPONTANEOUS IGNITION

Tests were made to determine the effect of low ambient temperature on the spontaneity of ignition of a number of propellant combinations. For these tests, a small-scale jet motor, designed to give approximately 50 lb thrust, was used, (Cf. Fig. 37). Other than chamber pressure, no measurements were taken in this setup, its chief function being to indicate qualitatively ignition and combustion characteristics.

The entire unit, i.e., propellant tanks, propellant feed lines, and motor, was surrounded by a cooling bath, with only the exhaust nozzle protruding as shown in Figs. 38 and 39. For temperatures down to about 5°F., ice and ice-salt mixtures were utilized as the cooling agent, (Cf. Fig. 40). For lower temperatures, the cooling bath contained dry ice and carbon tetrachloride, or dry ice and carbon tetrachloride-trichloroethylene mixtures. In almost all cases, the propellant components were cooled beforehand and then poured into the propellant tanks. The temperature of the propellant components in the tanks was measured immediately before the run, and it is this temperature that appears in Table X, where the various tests are described.

The fuel mixtures used in these tests do not correspond exactly to the fuels that are described in Part V of this report, because the low-temperature tests had been made before the fuel compositions were definitely fixed.

The results indicate that low ambient temperature has no effect on the spontaneous ignition of the propellant. There was no noticeable increase in ignition lag - even at a temperature of -40°F.

It is seen from the table that some of the fuel mixtures remained liquid at temperatures below the freezing points indicated in Fig. 33. There are several reasons for this. First, mixtures containing aniline frequently have a tendency to supercool, resulting in a metastable condition in which the liquid is at a temperature below its true freezing point. Secondly, the freezing-point curves were obtained with purified materials, whereas the low-temperature tests were made with commercial-grade aniline and furfuryl alcohol. Impurities tend to lower the freezing point of the mixture. Since in actual application the impure commercial-grade materials will be used, the range of applicability will extend somewhat below the lower limit of operation prescribed for purified materials in Fig. 33.

Although these small scale tests of the various propellant combinations showed no effect of low ambient temperature on their spontaneous ignition it cannot be concluded that no effects are to be expected on jet motor performance. The Aerojet Engineering Corporation has reported that at low ambient temperatures large scale units with uncooled jet motors exhibit materially altered performance. A motor designed to give 1000 lb thrust for 25 seconds when operated at an ambient temperature of 80°F showed a decrease in thrust and in increase in duration when it was operated within 20°F of the freezing point of the fuel mixture.

Ambient temperature variations can be expected to influence performance, since the propellant components undergo changes in their physical properties, for example, viscosity, etc. These changes will probably especially influence the rate of propellant discharge through an injector and the mixture ratio of the propellant components.

The necessity of considerable investigation of the operation of liquid propellant jet units at low ambient temperatures can be anticipated.

VII. DISCUSSION OF EXPERIMENTAL RESULTS

A. Comparison of various acid-aniline combinations tested.

One of the main purposes of carrying out the experimental program described in the preceding parts of the report was to determine the effect on performance parameters of various nitric acid type oxidizers in combination with aniline. The mixture ratio of the propellant components was in the neighborhood of 2.6 for most of the tests.

In Fig. 41 the faired curves of C_F, c* and c vs chamber pressure of the acid-aniline propellants are compared. Each curve is reproduced from the curves drawn through the experimental points plotted in Part V. The values of the thrust coefficient, C_F, obtained from the different tests are practically the same, the differences being well within the experimental accuracy of the data. This is in agreement with the discussion in Part VII, D, 1 where it is pointed out that the changes in the thermodynamic properties of the products of combustion and in the area ratio of the exhaust nozzles used can have only a negligible effect on the thrust coefficient.

The curves of characteristic velocity, c*, show some differences in the performance of the various propellants. Slight changes of mixture ratio in the neighborhood of r = 2.6 should have no effect on c*, according to the discussion in Part VII, D, 2. These differences are then actual or due to changes in the test conditions or errors in the experimental measurements. It is believed that the accuracy of the faired curves of c* is well within the limit of ±3.2% discussed in Part IV.

The two red fuming acids have identical performance. The white fuming acid has a performance which appears lower than that of the red fuming acids, but at equal chamber pressures the injector jet velocity of the white acid was lower than that of the red acids, so that the efficiency of the combustion process may have been affected. This same change in test conditions occurred during the tests of red fuming acid and aniline containing 20% furfuryl alcohol, but no effect on c* was found. It therefore appears that the white fuming acid performance is somewhat lower than that for the red fuming acids, though the difference seems to decrease at higher chamber pressures.

During the tests it was observed that rougher starts were obtained with the white acid than with any of the other acids. At chamber pressures of approximately 125 psi abs with all the propellants a throbbing occurred in the motor operation, but with the white acid this roughness of operation seemed more severe.

The effect of a change in the NO_2 content of the fuming nitric acids was studied theoretically by Sage, Hough, and Green (Cf Ref 21). Their calculations, made for a mixture ratio of 3 and a chamber pressure of 300 psi abs, showed that dropping the NO_2 content from 15% to 0% would cause a drop in chamber temperature from 5065°F to 4900°F, corresponding to approximately a $1\frac{1}{2}$% decrease in c*.

A decrease in the NO_2 content of the nitric acid is desirable, as it reduces the vapor pressure of this propellant component. The change from 16% NO_2 to $6\frac{1}{2}$% NO_2 involved no change in performance. The further change to white fuming nitric acid resulted in a small drop in performance and a roughness of operation. In addition slight increases in water content of the white fuming acid affect the spontaneity of the reaction. For these reasons it does not appear advisable at present to go below an NO_2

content in the neighborhood of $6\frac{1}{2}\%$.

The "mixed" acid-aniline propellant remains to be discussed. Fig. 41 shows that the characteristic velocity of this combination is approximately $5\frac{1}{2}\%$ lower than that obtained with the red fuming acids and aniline under identical test conditions in the neighborhood of 300 psi abs chamber pressure. At lower chamber pressures the performance of the other propellants falls off more rapidly, so that below 170 psi abs the "mixed" acid-aniline propellant appears to be superior to the others. It is possible that this phenomenon depends on the test conditions and may be affected by variations in L* and injector jet velocity as well as the change of propellant.

The "mixed" acid has a low vapor pressure and other desirable physical properties. Various observers have noticed that a jet motor appears to start more smoothly with this combination than with any of the others described in this report. Besides the lower performance, it has the disadvantage of containing a finely divided flocculent precipitate. (See Part V, D, 1). Sufficient data is not yet on hand to judge its suitability as a jet oxidizer, but because of the promise it has shown, further tests are being carried out.

In Fig. 41 the exhaust velocity, c, is shown as a function of the chamber pressure for the various acid-aniline propellant combinations. Since the various propellants have approximately the same C_F curves, and since c is equal to the product of C_F and c*, the differences in exhaust velocities are due to differences in characteristic velocities, and the discussion above of c* can be used in comparing the exhaust velocity curves.

B. Comparison of All the Propellant Combinations Tested.

In Fig. 42 the performance parameters of all the propellant combinations tested are compared. The curves of C_F, c*, and c drawn through the experimental points in Part V are reproduced. The tests of all the combinations shown were made within a mixture ratio range from 2.4 to 2.65, a region where mixture ratio is believed to have only a small effect on c*.

Three propellants not shown in Fig. 41 are included in the comparison in Fig. 42. These are red fuming nitric acid - pure furfuryl alcohol and red fuming acid with two mixtures of aniline and furfuryl alcohol, one containing 20% and the other 35% furfuryl alcohol.

The C_F curves show very small differences between the various combinations, and it is believed that the differences can be considered within experimental error although at lower pressures distinct changes in combustion characteristics might bring about real differences in C_F.

The characteristic velocity of the propellant using red fuming acid with pure furfuryl alcohol as the fuel is slightly lower than that of the similar propellant using aniline as the fuel, but it lies above the corves obtained with the white fuming acid-aniline and the "mixed" acid-aniline propellants. The optimum mixture ratio of the furfuryl alcohol propellant has not yet been determined, but it is believed that this fuel is only slightly inferior to aniline and that the results obtained here are near the optimum.

The performance of the two propellants using aniline-furfuryl alcohol mixtures is identical to that of the red fuming nitric acid-aniline propellants previously discussed.

The addition of furfuryl alcohol to the aniline is desirable because it lowers the freezing point of the fuel. In addition it was observed during the tests that the propellants using furfuryl alcohol in the fuel started more easily and operated more smoothly than those using aniline alone. The aniline-furfuryl alcohol mixtures appear to be the most desirable of the jet fuels studied to date.

The exhaust velocity curves of the various combinations are also drawn on Fig. 42, though the comparison of propellants is probably better made on the basis of characteristic velocity.

C. Comparison of Results Obtained by Various Investigators with the 16% RFNA-Aniline Propellant Combination.

The performance of jet motors utilizing a propellant composed of red fuming nitric acid containing approximately 16% NO_2 with aniline as the fuel has been investigated to a considerable extent. Experiments have been made with different test equipment and with motors designed to give a wide range of thrust at the Air Corps Jet Propulsion Research Project by Summerfield, Powell, Seifert and the present authors (Cf Refs 2, 4 and 5). In addition the Navy Bureau of Aeronautics Project and the Aerojet Engineering Corporation have independently determined the performance characteristics of this propellant combination.

The available results of the above investigations are compared in Fig. 43 for a nominal mixture ratio of 2.5. The results obtained with 1000 lb thrust production type jet units by the Aerojet Engineering Company are in good agreement with those of the present authors.

The most comprehensive study of this propellant combination was made by Summerfield. The values of the characteristic velocity of all investigators mentioned are seen in Fig. 43 to be in excellent agreement with those of Summerfield.

The values of C_F obtained by the present authors are considerably higher than those obtained by other experimenters. It is believed that these most recent results are more accurate than those of Summerfield and Seifert. The use of a rotating thrust jack piston has eliminated the experimental error introduced by friction which was present in the older thrust measuring equipment. The somewhat lower values of C_F reported by Stiff may be caused by the exhaust nozzle contour used in the Truax motor shown in Fig. 12.

The values of exhaust velocity obtained by the various investigators are also plotted in Fig. 43. The spread in exhaust velocity results is due to the differences in C_F discussed above.

D. Discussion of the Performance Parameters C_F and c^*

The following discussion of the performance parameters C_F and c^* is made with special reference to the red fuming nitric acid-aniline propellant combination. However the various factors influencing them are believed to hold generally, unless otherwise noted, for the various combinations discussed in this report.

1. The Thrust Coefficient C_F.

As noted in Parts V and VII of this report the thrust coefficient was apparently not affected by the various propellant combinations tested; in other words the thermodynamic properties of the products of combustion were insensitive to changes in propellant components and mixture ratio. Summerfield studied the effect of mixture ratio over the range from 1.3 to 4.6 for the 16% RFNA-aniline combination and also found that mixture ratio had no apparent influence on C_F, though his results were probably subject to a systematic error due to friction in the thrust measuring apparatus.

Figs. 44 and 45 are ideal thrust coefficient diagrams based on Malina's universal ideal thrust diagram. (Cf Ref 11). They show the theoretical optimum value of C_F as a function of pressure ratio and exhaust nozzle area ratio for two values of the ratio of specific heats of the exhaust gases. Fig. 46 is an exhaust nozzle design chart for obtaining the theoretically optimum C_F and nozzle area ratio for various values of the pressure ratio and specific heat ratio.

In Fig. 47 Tsien's curve showing the effect of exhaust nozzle exit angle on the loss of jet momentum and Summerfield's experimental verification are reproduced. (Cf Ref 11). It is interesting to note that the experimental curve shows a continuous loss in jet momentum only until the nozzle angle reaches approximately 40°. At this angle it appears that the jet separates from the nozzle wall and increasing the nozzle angle does not cause further momentum loss.

The curves of Figs. 44 and 45 show that C_F decreases with a decrease in the pressure ratio for an exhaust nozzle with a given area ratio. They also show that the effect of using an exhaust nozzle with an area ratio of 3.5 instead of one with an area ratio of 4.0 is negligible over the chamber pressure range from 250 to 550 psi abs.

Fig. 46 shows that the change in C_F with a change in the ratio of specific heats of the products of combustion is small. Results of calculations of the effect of mixture ratio on γ are shown in Fig. 25. The change in C_F to be expected by varying the mixture ratio from 1 to 4 is less than 1% at a chamber pressure of 300 psi. Changes in C_F due to the modifications of the propellant components discussed in this report should also not be larger than this magnitude.

The average experimental value of C_F is compared in Fig. 48 with the theoretical value computed by means of Eq (5), assuming an external pressure of 14.0 psi abs. In the calculation an average value of 1.25 was used for the ratio of the specific heats of the products of combustion. The curves show that if the area ratio of an exhaust nozzle is held constant the deviation of the experimental from the optimum C_F increases as the chamber pressure decreases. This is to be expected with an exhaust nozzle which has the proper area ratio for the higher pressure ratio, and is due to overexpansion of the gases.

At 300 psi abs the following comparison is obtained:

$$\frac{p_c}{p_o} = \frac{300}{14.0} = 21.4 \qquad C_{F\,exp} = 1.37 \qquad \epsilon_{actual} = 4.0$$

$$C_{F\,opt} = 1.41 \qquad \epsilon_{opt} = 3.8$$

The following relation can be written for the thrust coefficient, $C_{F_{exp}}$.

$$C_{F\,exp} = \lambda\, C_d\, C_{F\,opt}$$

where λ is Tsien's correction factor and C_d is a discharge coefficient that accounts for friction and slight deviations from the adiabatic expansion process.

For the present example the nozzle angle was 15°, with a corresponding value of .983 for λ; therefore the discharge coefficient has the value:

$$C_d = \frac{C_{F\,exp}}{\lambda C_{F\,opt}} = \frac{1.37}{.983 \times 1.41} = .987$$

This calculation indicates that a carefully designed exhaust nozzle can be expected to approach very closely to the theoretical thrust coefficient.

It is believed that the short nozzles used in the tests reported herein do not allow the products of combustion sufficient time to shift equilibrium during the expansion process as discussed in Part III of this report; therefore the comparison of experimental results with theoretical values calculated by assuming an adiabatic expansion should be valid.

2. The Characteristic Velocity, c*

A few remarks have been made in Part III on the significance of the characteristic velocity, c*, and the various factors that affect it. The experimentally determined values of c* for the propellant combinations recently studied are presented in Part V and compared in Part VII, Sections A, B, and C.

In order to check the reliability of the data obtained by the present authors the results have been compared with those reported by other investigators. In Fig. 49 the variation of c* with mixture ratio is shown for the 16% RFNA-aniline combination at a chamber pressure of 300 psi abs. The results compared were obtained with motors delivering from 200 to 6000 lbs thrust. (It will be shown in Part VIII that there is no apparent scale effect on the characteristic velocity, c*).

Summerfield has carried out the most comprehensive study of the effect of mixture ratio and chamber pressure on the propellant performance. From Fig. 49 it is seen that the results of all investigators are in good agreement with his determinations of c*. The curve that he faired through his experimental points has been revised to be in agreement with all the experimental data now available, and this is the curve drawn on Fig. 49.

A revised set of characteristic velocity curves which show its variations with mixture ratio and chamber pressure is drawn in Fig. 50.

The present authors carried out experiments with 6½% RFNA and a fuel consisting of 80% aniline plus 20% furfuryl alcohol at nominal mixture ratios of 1.4 and 2.5 over a range of chamber pressures. The effect of mixture ratio on c* is shown in Fig. 51. The tests were not carried to completion but the results show a loss in c* with a decrease in mixture ratio similar to that obtained with aniline alone used as a fuel.

Summerfield also determined the effect of the characteristic length, L* on the jet motor performance parameters for the 16% RFNA-aniline propellant. He found that the thrust coefficient C_F, was not affected over the range of values he studied; however the characteristic velocity, c*, appeared to drop when L* was made too small. In Fig. 52 the variation of c* with L* is shown, including data obtained from more recent tests. All the results in this figure were obtained with motors of similar geometrical shape and using multi-orifice injectors. (Cf Figs 2, 3, 5, 7 and 8).

Any decrease in c* with L* presumably means that the propellant does not remain in the chamber long enough to burn completely. Since it is desirable to maintain a ratio between the exhaust nozzle throat diameter and the diameter of a cylindrical combustion chamber of from 2.5 to 4.0 in order to reduce heat transfer, a constant value of L* in effect prescribes a constant combustion length, l_c , for all combustion chambers. In jet motors of high thrust this leads to very squat chamber proportions, and ordinary multi-orifice type injectors do not make it possible to utilize all the chamber volume as combustion volume; thus the criteria L^* and $\frac{d_c}{d_t}$ are not sufficient to completely define the combustion chamber volume required in a jet motor.

The nature of the propellant injection influences the effective combustion volume in a cylindrical combustion chamber. Ideally the propellant components should be injected and thoroughly mixed uniformly over the back face of the chamber. Actually, finite streams of each propellant component are arranged to impinge at some distance from the face of the injector, so that at least the volume of the chamber between the injector and the point of impingement cannot be considered as combustion volume. In addition the propellant components are not perfectly mixed or uniformly distributed throughout the chamber cross-sectional area at the point of impingement, so that an additional part of the chamber volume becomes ineffective as burning volume.

The mixing and distribution of the propellant throughout the chamber depends greatly on the velocity and direction of the impinging propellant component streams issuing from the face of a multi-orifice type injector (Cf Figs 7 and 8). The injector shown in Fig. 8 has a pressure drop across the orifices of approximately 125 psi when delivering propellant to a motor giving 1000 lb thrust. Other injectors (Cf Fig 7) have been designed to operate at approximately 200 psi pressure difference.

The exact manner in which chamber pressure, combustion volume and injector characteristics affect the characteristic velocity is not clear. Experimentally there appears to be a definite rise in c* with increasing chamber pressure and injector stream velocity, at least up to a certain point. It also appears that at a given chamber pressure poor injection can be compensated for by an increase in combustion chamber volume, provided that poor mixing does not promote localized overheating and erosion of the jet motor.

The latter point is illustrated by the results shown in Fig. 31 for the $6\frac{1}{2}\%$ RFNA and furfuryl alcohol combination. Two sets of points are plotted. The lower chamber pressure group was obtained with the motor shown in Fig. 3 which had an L* of 45.5 in. The higher chamber pressure group was obtained with the same chamber and injector, but with an exhaust nozzle of decreased throat area which increased the L* of the motor to 100 in. (Cf Fig 11).

The pressure drop across the injector was approximately 125 psi at a chamber pressure of 300 psi abs when L* was 45.5 in. When the L* was increased to 100 in. by using the exhaust nozzle with the smaller throat area it became necessary to increase the chamber pressure to 550 psi abs in order to obtain the same propellant flow and injector pressure drop as was previously obtained at a chamber pressure of 300 psi abs. Thus when the L* = 100 in motor was operating at 300 psi abs chamber pressure the velocity of the injector streams was considerably lower than for the L* = 45.5 in motor. Nevertheless the characteristic velocity was the same for both motors at 300 psi abs chamber pressure. Most of the propellants showed a consistent increase in c* with chamber pressure; however the $6\frac{1}{2}\%$ RFNA furfuryl alcohol propellant described above (Cf Fig 31) and the "mixed" acid aniline propellant (Cf Fig 29) exhibited no increase in c* above a chamber pressure of approximately 250 psi abs. It must be pointed out, though, that an increase of chamber pressure during the tests of a given motor was always accompanied by a corresponding increase in the injector stream velocity. Thus the increases in c* with chamber pressure shown on the plotted test results may be due to the increase in chamber pressure, the increase in injector stream velocity, or to a combination of these influences.

Jet motors whose geometry is similar to those described herein and having multi-orifice type injectors with pressure drops of at least 125 psi appear to give good performance with L* values in the neighborhood of 50 inches.

Theoretical calculations of the combustion of 15% RFNA and aniline at several mixture ratios and two chamber pressures have been made by Sage, Hough and Green. The exhaust velocity obtained by expansion to atmospheric pressure, the combustion temperature, and the composition and properties of the reaction products have been calculated (Cf Table VIII, and Figs 23, 24 and 25).

The theoretical value of c* can be calculated by dividing the theoretical exhaust velocity by the theoretical thrust coefficient. This has been done and the results are in Table XI and on Fig 44, where c* is plotted as a function of mixture ratio. The theoretical thrust coefficient is obtained from Fig 46, using the proper pressure ratio and the value of γ obtained from Fig 25 for the given mixture ratio and pressure ratio.

These calculations show that the effect of chamber pressure on the theoretical value of c* is negligible; therefore only one theoretical curve is given on Fig 44. The comparison on Fig 44 shows that the theoretically optimum mixture ratio is about 3, whereas the optimum value indicated by present tests lies somewhat below this figure. At a mixture ratio of 1.5 the experimental value of c* is 4.2% lower than the theoretical value, while at a mixture ratio of 3 the experimental value is 10% lower than the theoretical value.

These deviations of the experimental results from the theoretical calculations are an indication of the maximum improvement in c* which is possible. The differences may be ascribed to the fact that the oxidizer used in the tests had 2% to 5% water, whereas the calculations were made for pure RFNA; to incomplete combustion due to poor injection characteristics or insufficient burning volume; to a real effect of chamber pressure on the combustion process; and to heat loss through the chamber walls. Calculations based on experimental heat flow measurements indicate that the heat lost to the cooling fluid in a jet motor of the type shown in Fig 5 is equivalent to approximately ½% of the theoretical value of c* at a mixture ratio of 3 and a chamber pressure of 300 psi abs. It is believed that the addition of 2% of water to the acid at a mixture ratio of 3 will result in approximately a 1% decrease in c*.

The remaining difference between the theoretical and measured valuès of c* (about 8% at a mixture ratio of 3) means that the heat of combustion of the propellant is different from that corresponding to the theoretical equilibrium composition of the products of reaction given in Table VIII. This could be so for a variety of reasons; poor injection mixing, insufficient burning volume, or an effect of pressure or of water content in the acid on the equilibrium composition of the products of reaction.

On the other hand, the approximations involved in the theoretical calculations are such that the error in c* might approach in magnitude the above mentioned difference between the theoretical and measured values of c*.

A consideration of the equilibria between the products of reaction given in Table VIII shows that relatively large amounts of energy could be accounted for by small shifts in the composition of the products of reaction.

On the basis of the results so far discussed it can be concluded that in general c^* increases only slightly with chamber pressure and that at each chamber pressure c^* has a maximum at a mixture ratio which is lower than the stoichiometric value, assuming that in all cases the injection mixing is satisfactory and the combustion volume is adequate. (Cf. Fig. 50)

VIII. A STUDY OF THE EFFECT OF JET MOTOR SCALE ON THE PERFORMANCE PARAMETERS

One of the unanswered problems of jet motor design has been the effect of scale on the performance parameters C_F and c^*. Within the last year tests have been carried out by a number of investigators on jet motors of a wide range of sizes. All utilized red fuming nitric acid and aniline at a chamber pressure of approximately 300 psi abs. The values of L^* for these motors ranged from 45.5 to 73 in, that is, above the range where L^* may affect the performance parameters. (Cf Fig 52).

The smallest motor was designed to deliver 200 lb and the largest 6000 lb thrust. The results obtained by the various investigations are shown in Fig 53. All motors had a similar geometrical form with the exception of the Navy Bureau of Aeronautics Project motor which had a spherical combustion chamber and a rapidly divergent exhaust nozzle. Some of the motors were regeneratively cooled and others depended on the heat capacity of the materials of construction for their safe period of operation.

A study of the results plotted in Fig 53 shows that there is no scale effect on the characteristic velocity, c*, at a nominal mixture ratio of 1.5. The scatter in c* corresponds roughly to variations in the actual mixture ratio of the tests.

The values of C_F show considerable scatter, especially since it has been found that C_F is not appreciably affected by changes in mixture ratio. It is believed that the early GALCIT values of C_F were subject to error due to friction in the thrust measuring devices. The recent installation of the rotating piston hydraulic jack for measuring thrust should make the results described in the other parts of this report the most reliable. For this reason greater weight has been given to the results of the present authors in fairing the C_F curves.

The results available on C_F for motors delivering less than 500 lb thrust do not as yet conclusively prove that no scale effect exists in this range, and further experimental data is necessary.

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TABLE I

SUMMARY OF PERFORMANCE CHARACTERISTICS OF VARIOUS PROPELLANT COMBINATIONS

Propellant		Freezing Point of Fuel (^{o}F)	Mixture Ratio r	Performance at p_c = 300 psi,abs		
Oxidizer	Fuel			c^* (ft/sec)	C_F	c (ft/sec)
16% RFNA	Aniline	21.02	2.55 to 2.65	4490	1.37 (1)	6150
6½% RFNA	Aniline	21.02	2.62 to 2.70	4530	1.37	6200
White Fuming Nitric Acid	Aniline	21.02	2.50 to 2.70	4250 (2)	1.37	5815
"Mixed" Acid	Aniline	21.02	2.50 to 2.65	4250	1.37	5815
6½% RFNA	Furfuryl Alcohol	-23.18	1.59 to 1.65	4320	1.37	5915
6½% RFNA	Furfuryl Alcohol	-23.18	2.50 to 2.70	4400	1.37	6020
6½% RFNA	Aniline plus 20% Furfuryl Alcohol	- 1.0	1.35 to 1.41	4220	1.37	5780
6½% RFNA	Aniline plus 20% Furfuryl Alcohol	- 1.0	2.40 to 2.65	4550	1.37	6230
16% RFNA	Aniline plus 35% Furfuryl Alcohol	- 22	2.45 to 2.53	4550	1.37	6230

(1) This value is obtained from the average curve, Fig 48.

(2) See Fig 28. It is probable that a c* of 4400 ft/sec could be obtained at p_c = 300 psi with increased injector jet velocity.

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TABLE II

NOMENCLATURE

$AR = \frac{l_c}{d_c}$	Aspect ratio of cylindrical combustion chamber
$c = \frac{F}{W/g} = c^* \times C_F$	Effective exhaust velocity
$c^* = \frac{p_c f_t}{W/g}$	Characteristic velocity
C_d	Exhaust nozzle discharge coefficient
$C_F = \frac{F}{p_c f_t}$	Exhaust nozzle thrust coefficient
c_p	Specific heat of gas at constant pressure
c_v	Specific heat of gas at constant temperature
d_c	Diameter of cylindrical combustion chamber
d_e	Exhaust nozzle exit diameter
d_t	Exhaust nozzle throat diameter
$F = \frac{W}{g} \times c$	Thrust
f_e	Exhaust nozzle exit area
f_r	Exhaust nozzle throat area
g	Acceleration due to gravity
l_c	Length of cylindrical combustion chamber
$L^* = \frac{V_c}{f_t}$	Characteristic length of combustion chamber
p_c	Chamber pressure (absolute)
p_e	Exit pressure (absolute)
p_o	External pressure (absolute)
R	Gas constant
$r = \frac{W_o}{W_f}$	Propellant mixture ratio

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TABLE II (Cont)

q	Dynamic pressure of fluid flow
q	Heat flow per unit area
T_c	Temperature of gases in combustion chamber (absolute)
t	Time
V_c	Volume of combustion chamber, up to the throat of the exhaust nozzle
v	Velocity
V	Specific volume
W	Rate of propellant consumption (total)
W_f	Rate of fuel consumption
W_o	Rate of oxidizer consumption
$W_{sp} = \frac{W}{F} = \frac{g}{c}$	Specific propellant consumption
α	Half angle of exhaust nozzle expanding section
β	Half angle of exhaust nozzle entrance section
β	Direction of resultant momentum after injector jet impingement
δ	Weight density
$\epsilon = \frac{f_e}{f_t}$	Exhaust nozzle area ratio
Γ	$\left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{2(\gamma-1)}} \sqrt{\gamma}$
$\gamma = \frac{c_p}{c_v}$	Ratio of specific heats of gas
$\lambda = \frac{1}{2}(1+\cos\alpha)$	Exhaust nozzle divergence angle function
$\nu = \frac{\mu}{\rho}$	Kinematic viscosity
$\rho = \frac{\delta}{g}$	Mass density
σ	Specific gravity relative to water at 4° C. and 14.7 psi.
μ	Absolute viscosity

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TABLE III

VAPOR PRESSURE OF PURE NITRIC ACID (Cf Ref 9)

Temp. °F.	Pressure Lb/sq in abs
50.0	0.43
68.0	0.81
77.0	1.10
95.0	1.97
122.0	4.16
176.0	12.09
185.0	13.92
194.0	15.86

TABLE IV

ISOBARIC HEAT CAPACITY OF PURE

NITRIC ACID (LIQUID) (Cf Ref 12)

Temp. °F.	Isobaric Heat Capacity Btu/lb mole/°F
36.5	26.34
70.0	26.40
103.1	26.78
104.2	26.90

TABLE V

THERMAL PROPERTIES OF PURE NITRIC ACID

Heat of Formation (Liquid)	74.99×10^3 Btu/lb mole (Cf Ref 13)
Heat of Vaporization	13.50×10^3 Btu/lb mole (Cf Ref 9)

TABLE VI

ISOBARIC HEAT CAPACITY OF NITROGEN DIOXIDE (Cf Ref 12)

Temp. °F.	Solid Btu/lb-mole/°F.	Liquid Btu/lb-mole/°F.	Gas Btu/lb-mole/°F.
-423.4	2.03 *		
-369.4	8.70		
-279.4	14.51		
-189.4	18.36		
-99.4	21.92		
-9.4	25.63		
26.6		32.93	
44.6		33.28	
62.6		33.71	
80.6 to 152.6			149.4
152.6 to 217.4			114.0
217.4 to 302.0			54.0
302.0 to 388.4			18.2
388.4 to 487.4			17.8
487.4 to 536.0			25.8

* All values are per lb-mole of dinitrogen tetroxide (N_2O_4)

TABLE VII

THERMODYNAMIC PROPERTIES OF ANILINE

Enthalpy change for formation[a] at 77°F.	(Btu/lb mole) (Btu/lb)	13,210 (Cf Ref 17) 142.0
Entropy change for formation[a] at 77°F.	(Btu/lb mole°R) (Btu/lb °F)	-94.2 (Cf Ref 17) - 1.01
Enthalpy change for vaporization at 361.4°F.	(Btu/lb mole) (Btu/lb)	17,370 (Cf Ref 18) 186.6

Isobaric Heat Capacity (Cf Ref 19)

Temperature (°F)	Molar Heat Capacity (Btu/lb mole°F)	Specific Heat Capacity (Btu/lb °F)
	Solid	
-280	12.4	0.133
-260	13.3	0.143
-240	14.4	0.155
-220	15.4	0.165
-200	16.5	0.177
-180	17.7	0.190
-160	18.8	0.202
-140	20.1	0.216
-120	21.3	0.229
-100	22.7	0.244
- 80	24.0	0.258
- 60	25.4	0.273
- 40	26.8	0.288
- 20	28.2	0.303
	LIQUID	
40	44.7	0.480
60	45.1	0.485
80	45.9	0.493
100	46.9	0.504
120	47.9	0.515
140	48.8	0.524
160	49.4	0.531
180	49.9	0.536
200	50.4	0.542
220	51.3	0.551
240	52.8	0.567
260	55.2	0.593
280	60.3	0.648

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TABLE VIII

CALCULATED COMPOSITION OF THE PRODUCTS OF REACTION OF ANILINE AND RED FUMING NITRIC ACID AT EQUILIBRIUM (Cf Ref 3)

Pressure (lb/in.^{2})	300					600		
Weight Fraction Nitrogen Dioxide in Acid Phase	0.15	0.15	0.15	0.15	0.15	0.15	0.15	0.15
Weight Ratio Acid / Fuel	1	2	3	5	7	1	2	3
Composition of Equilibrium Mixture (mole fraction)								
Oxygen	--	--	0.0037	0.0898	0.1963	--	--	0.0027
Monatomic Oxygen	--	--	0.0020	0.0078	0.0026	--	--	0.0015
Triatomic Oxygen	--	--	--	--	--	--	--	--
Nitrogen	0.1122	0.1551	0.1919	0.2136	0.2246	0.1122	0.1551	0.1930
Nitric Oxide	--	--	0.0036	0.0171	0.0163	--	--	0.0032
Nitrogen Dioxide	--	--	--	--	0.0001	--	--	--
Monatomic Nitrogen	--	--	0.0002	0.0002	--	--	--	0.0002
Hydrogen	0.3446	0.2126	0.0632	0.0105	0.0016	0.3407	0.2126	0.0626
Water	0.0173	0.1594	0.2827	0.2919	0.2898	0.0211	0.1594	0.2881
Monatomic Hydrogen	--	0.0013	0.0114	0.0038	0.0003	--	0.0013	0.0092

(Cont on Next Page)

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TABLE VIII (cont)

Hydroxyl	--	0.0003	0.0316	0.0622	0.0338	--	0.0003	0.0279
Carbon Dioxide	0.0098	0.0550	0.1587	0.2431	0.2265	0.0100	0.0550	0.1622
Carbon Monoxide	0.3466	0.4163	0.2510	0.0601	0.0080	0.3422	0.4163	0.2494
Carbon	0.1695	--	--	--	--	0.1738	--	--
Reaction Temperature (°F.)	1950	3710	5065	4930	4320	2015	3710	5150
Exit Velocity (ft/sec)	5207	6514	7091	6688	6192	5658	6998	7694
Specific Impulse (lb.sec/lb)	161.9	202.6	220.5	207.95	192.54	175.9	217.6	239.2

GALCIT PROJECT NO. 1
TEST DATA SUMMARY
PROPELLANT CHARACTERISTICS TESTS
PIT B 7-7-43 THROUGH 9-23-43
TABLE IX a

RUN NO.	TIME EFF. sec	F lb	pc psi abs	f_t in²	w lb/sec	r	c ft/sec	C_F	c^* ft/sec	OXIDIZER	FUEL	L^* in	EXHAUST NOZZLE ϵ	NOTES
519	20.25	946.2	271.7	2.627	5.406	1.348	5636	1.326	4250	6½% RFNA	80% AN 20% F.A.	64.2	4.0	
520	19.85	922.7	262.7	2.610	5.256	1.407	5653	1.346	4200	"	"	"	"	
521	19.78	891.9	258.9	2.613	5.293	1.356	5426	1.318	4117	"	"	"	"	
522	20.07	768.8	240.3	2.607	4.809	1.396	5148	1.227	4196	"	"	"	"	
523	23.12	644.0	206.5	2.613	4.165	1.413	4979	1.193	4174	"	"	"	"	
524	30.45	669.4	210.5	2.607	3.862	1.826	5581	1.220	4575	"	"	"	"	
525	19.02	1050.4	310.4	2.613	5.569	2.516	6075	1.295	4691	"	"	"	"	
526	19.88	986.5	284.8	2.623	5.200	2.525	6109	1.321	4625	"	"	"	"	
527	20.76	1078.1	318.4	2.610	5.765	2.513	6022	1.297	4643	"	"	"	"	
528	30.5	821.4	235.4	2.610				1.337		"	"	"	"	
529	19.93	1082.3	295.1	2.623	5.760	2.495	6050	1.398	4328	16% RFNA	35% F.A. ANILINE	64.2	4.0	
530	20.46	982.4	272.8	2.606	5.369	2.489	5892	1.382	4263	"	"	"	"	
531	20.53	1155.5	336.1	2.600	6.015	2.357	6186	1.322	4679	"	"	"	"	
532	21.14	1092.9	300.0	2.682	5.680	2.434	6196	1.358	4563	"	"	"	"	
533	19.91	954.8	288.2	2.672	5.528	2.494	5562	1.239	4485	"	"	"	"	
534	20.94	1178.1	322.9	2.655	5.965	2.409	6360	1.374	4629	"	"	"	"	
535	FILM	DIDN'T	COME	OUT	— CAMERA	JAMMED				"	"	"	"	
536	19.63	1103.9	303.3	2.645	5.880	2.465	6045	1.376	4393	"	"	"	"	THRUST JACK PISTON ROTATED
537	20.51	1039.4	291.0	2.636	5.470	2.462	6119	1.355	4516	"	"	"	"	
538	19.54	871.0	262.2	2.646	5.119	2.487	5479	1.255	4366	"	"	"	"	
539	20.70	786.8	230.5	2.627	4.416	2.431	5737	1.299	4416	"	"	"	"	
540	20.26	688.6	204.6	2.619	3.888	2.535	5703	1.285	4438	"	"	"	"	
541	25.51	572.8	175.2	2.616	3.565	2.537	5174	1.250	4139	"	"	"	"	

GALCIT PROJECT NO. I
TEST DATA SUMMARY
PROPELLANT CHARACTERISTICS TESTS
PIT B 7-7-43 THROUGH 9-23-43
TABLE IX b

RUN NO.	TIME EFF. sec	F lbs	pc psi abs	f_t in²	w lb/sec	r	c ft/sec	C_F	c* ft/sec	OXIDIZER	FUEL	L* in	EXHAUST NOZZLE ε	NOTES
542	26.18	448.2	146.1	2.622	2.912	2.491	4956	1.170	4236	16% RFNA	35% F.A. ANILINE	64.2	4.0	
543	19.00	1087.1	311.3	2.627	5.764	2.462	6073	1.329	4570	"	"	"	"	
544	21.27	1178.1	327.4	2.602	5.755	2.459	6592	1.383	4766	"	"	"	"	
545	20.88	1068.4	299.2	2.599	5.573	2.509	6173	1.374	4493	"	"	45.5	4.0	
546	20.33	1063.7	303.6	2.538	5.341	2.523	6413	1.380	4647	"	"	"	"	
547	20.11	842.9	246.8	2.535	4.711	2.672	5761	1.347	4277	"	"	"	"	
548	21.43	1117.8	323.8	2.504	5.522	2.628	6518	1.379	4727	"	"	"	"	
549												"	"	NO DATA
550	19.84	887.8	262.8	2.502	4.748	2.605	6021	1.350	4460	16% RFNA	ANILINE	45.5	4.0	
551	20.25	988.4	287.8	2.470	5.106	2.573	6233	1.390	4484	"	"	"	"	
552	19.83	1066.3	311.6	2.479				1.380		"	"	"	"	
553	19.65	1063.8	310.2	2.427	5.494	2.531	6235	1.413	4413	"	"	"	"	
554	23.03	746.1	227.4	2.462	4.121	2.657	5830	1.333	4374	"	"	"	"	
555	19.64	1112.7	324.4	2.475	5.681	2.557	6307	1.386	4551	"	"	"	"	
556	19.77	1161.2	337.1	2.475	5.960	2.601	6274	1.392	4507	"	"	"	"	
557										6½% RFNA	ANILINE +20% F.A.	45.5	4.0	
558	19.07	866.9	245.6	2.639	4.801	2.580	5814	1.338	4345	"	"	"	"	
559	19.57	916.9	260.6	2.604	4.876	2.663	6055	1.351	4482	"	"	"	"	
560	21.31	950.9	270.0	2.630	5.051	2.562	6062	1.339	4527	"	"	"	"	
561	20.03	1039.8	289.7	2.600	5.364	2.605	6242	1.380	4523	"	"	"	"	
562	20.37	1049.7	292.4	2.593	5.417	2.580	6240	1.384	4509	"	"	"	"	
563	20.86	1093.4	304.6	2.592	5.627	2.577	6257	1.385	4518	"	"	"	"	
564	21.17	1112.4	315.9	2.577	5.859	2.488	6114	1.366	4476	"	"	"	"	

GALCIT PROJECT NO. 1
TEST DATA SUMMARY
PROPELLANT CHARACTERISTICS TESTS
PIT B 7-7-43 THROUGH 9-23-43
TABLE IX C

RUN NO.	TIME EFF sec	F lb	pc psi abs	f_t in²	w lb/sec	r	c ft/sec	C_F	c* ft/sec	OXIDIZER	FUEL	L* in	EXHAUST NOZZLE ε	NOTES
565	21.14	1137.0	319.8	2.567	5.789	2.475	6324	1.385	4566	6½% RFNA	ANILINE +20% F.A.	45.5	4.0	
566	26.21	375.1	127.8	2.539	3.032	2.378	3984	1.156	3446	"	"	"	"	
567	25.83	428.3	138.3	2.560	3.303	2.401	4175	1.210	3450	"	"	"	"	
568	26.11	374.9	132.1	2.553	3.078	2.447	3922	1.112	3527	"	"	"	"	
569	24.68	556.8	171.3	2.542	3.634	2.552	4934	1.279	3858	"	"	"	"	
570	21.14	1064.2	299.9	2.548	5.436	2.583	6304	1.393	4525	"	"	"	"	
571	20.87	1124.3	317.7	2.550	5.963	2.610	6071	1.388	4374	"	"	"	"	
572	21.32	1127.6	317.7	2.543	5.103	2.609	7115	1.396	5097	"	"	"	"	
573	21.03	1178.9	328.8	2.546	5.970	2.477	6359	1.408	4516	"	"	"	"	
574	30.57	1195.0	313.5	2.672	5.861	2.634	6565	1.427	4601	"	"	54.6	4.0	WATER COOLED MOTOR
575	31.35	1149.5	311.1	2.635	5.578	2.551	6636	1.402	4733	"	"	"	"	
576	30.54	1128.0	303.6	2.629	5.560	2.808	6533	1.413	4623	"	"	"	"	
577	41.05	659.3	185.2	2.624	3.685	2.678	5761	1.375	4245	"	"	"	"	
578	40.93	486.9	149.8	2.617	3.127	2.700	5014	1.242	4037	"	"	"	"	
579	35.68	846.1	237.3	2.622	4.511	2.612	6040	1.360	4441	"	"	"	"	
580	30.83	1175.6	319.0	2.622	5.877	2.581	6441	1.405	4584	"	"	"	"	
581	31.03	1216.8	334.6	2.620	6.114	2.575	6408	1.388	4617	"	"	"	"	
582		1082.1	299.1	2.624				1.379		6½% RFNA	ANILINE	54.6	4.0	
583	30.21	1079.4	300.9	2.620	5.716	2.629	6081	1.369	4442	"	"	"	"	
584										"	"	"	"	MOTOR FAILED
585	29.50	424.5			3.068	2.741	4455			"	"	45.5	4.0	
586	27.0	431.3	135.1	2.653	3.025	2.834	4591	1.203	3816	"	"	"	"	
587	21.50	547.2	163.4	2.652	3.398	2.714	5185	1.263	4105	"	"	"	"	

GALCIT PROJECT NO. I
TEST DATA SUMMARY
PROPELLANT CHARACTERISTICS TESTS
PIT B 7-7-43 THROUGH 9-23-43

TABLE IX d

RUN NO.	TIME EFF sec	F lb	pc psi abs	f_t in²	w lb/sec	r	c ft/sec	C_F	c* ft/sec	OXIDIZER	FUEL	L* in	EXHAUST NOZZLE ∈	NOTES
588	20.18	664.1	189.7	2.650	3.776	2.691	5663	1.321	4287	6½% RFNA	ANILINE	45.5	4.0	
589	19.70	818.9	223.8	2.720	4.453	2.714	5922	1.345	4403	"	"	"	"	
590	18.10	927.3	251.2	2.708	4.912	2.647	6079	1.363	4460	"	"	"	"	
591	16.10	1030.8	280.4	2.708	5.424	2.594	6119	1.357	4509	"	"	"	"	
592	15.90	1127.7	300.8	2.708	5.753	2.587	6312	1.384	4561	"	"	"	"	
593	16.01	1247.9	330.5	2.698	6.537	2.562	6147	1.399	4394	"	"	"	"	
594	16.50	1270.3	335.5	2.703	6.419	2.550	6372	1.401	4548	"	"	"	"	
595	20.40	794.6	225.3	2.691	4.783	1.614	5349	1.311	4080	6½% RFNA	FURFURYL ALCOHOL	45.5	4.0	
596	26.57	399.2	127.6	2.697	2.710	1.636	4743	1.160	4089	"	"	"	"	
597	21.69	485.1	148.5	2.691	3.130	1.650	4990	1.214	4110	"	"	"	"	
598	21.10	581.0	170.0	2.694	3.561	1.636	5254	1.269	4140	"	"	"	"	
599	19.18	683.5	198.3	2.699	3.898	1.632	5646	1.277	4421	"	"	"	"	
600	19.14	800.5	225.3	2.698	4.494	1.596	5736	1.317	4355	"	"	"	"	
601	18.14	896.1	252.2	2.698	5.311	1.589	5433	1.317	4125	"	"	"	"	
602	15.88	964.0	269.1	2.694	5.496	1.625	5648	1.330	4247	"	"	"	"	
603	16.67	1043.2	288.6	2.687	5.825	1.605	5767	1.345	4288	"	"	"	"	
604	17.47	1121.2	304.7	2.685	6.102	1.592	5917	1.370	4319	"	"	"	"	
605	15.43	1208.5	326.1	2.690	6.361	1.611	6118	1.378	4440	"	"	"	"	
606	21.00	683.0	195.7	2.688	3.912	1.650	5622	1.298	4331	"	"	"	"	
607		1076.8	273.0	2.665				1.480		6½% RFNA	FURFURYL ALCOHOL	45.5	4.0	
608	17.70	961.7	269.3	2.612	5.143	2.515	6021	1.367	4405	"	"	"	"	
609	21.18	859.7	248.5	2.542	4.668	2.544	5930	1.361	4357	"	"	"	"	
610	20.96	757.7	226.2	2.529	4.213	2.539	5791	1.324	4374	"	"	"	"	

GALCIT PROJECT NO. 1
TEST DATA SUMMARY
PROPELLANT CHARACTERISTICS TESTS
PIT B 7-7-43 THROUGH 9-23-43
TABLE IX e

RUN NO.	TIME EFF. sec	F lb.	pc psi abs	ft in²	w lb/sec	r	c ft/sec	C_F	c* ft/sec	OXIDIZER	FUEL	L* in	EXHAUST NOZZLE ε	NOTES
611	21.20	591.7	182.8	2.519	3.480	2.545	5475	1.285	4261	6½% RFNA	FURFURYL ALCOHOL	45.5	4.0	
612	25.10	511.1	165.1	2.522	3.269	2.564	5034	1.227	4103	"	"	"	"	
613	26.02	392.9	131.2	2.521	2.816	2.658	4493	1.188	3782	"	"	"	"	
614	16.66	1002.2	287.5	2.515	5.316	2.500	6071	1.386	4380	"	"	"	"	
615	17.01	1029.2	296.6	2.510	5.400	2.548	6137	1.382	4441	"	"	"	"	
616	15.75	1094.4	314.1	2.495	5.734	2.516	6146	1.396	4403	"	"	"	"	
617	18.15	696.2	411.5	1.173	3.613	2.711	6026	1.401	4301	"	"	100	3.5	
618	18.40	566.8	365.3	1.134	3.141	2.790	5811	1.368	4248	"	"	"	"	
619	21.48	496.7	323.9	1.131	2.646	2.728	6044	1.356	4457	"	"	"	"	
620	26.61	437.2	291.3	1.129	2.420	2.660	5817	1.329	4377	"	"	"	"	
621	26.23	367.2	249.3	1.122	2.111	2.666	5601	1.313	4266	"	"	"	"	
622	16.07	718.5	456.1	1.119	3.678	2.564	6290	1.408	4467	"	"	"	"	
623	15.80	774.0	490.2	1.113	3.986	2.562	6253	1.419	4407	"	"	"	"	
624	15.79	473.8	309.9	1.113	2.615	2.851	5834	1.374	4246	WFNA	ANILINE	100	3.5	LINES MADE FLEXIBLE TO REMOVE THRUST RESTRAINT
625	15.78	655.8	364.2	1.117	2.980	2.698	6164	1.402	4397	"	"	"	"	
626	15.90	750.7	414.5	1.131	3.319	2.607	6362	1.399	4548	"	"	"	"	
627	16.28	810.0	457.5	1.137	3.885	2.574	6222	1.443	4312	"	"	"	"	
628	16.47	882.6	486.4	1.147	3.998	2.535	6524	1.452	4493	"	"	"	"	
629	17.20	515.1	525.4	1.152	4.333	2.575	6559	1.458	4499	"	"	"	"	
630	20.62	446.7	320.9	1.157	2.743	2.585	6047	1.387	4360	"	"	"	"	
631	21.90	357.5	278.4	1.157	2.310	2.443	6227	1.387	4490	"	"	"	"	
632	21.68	932.7	234.8	1.162	2.272	2.549	5067	1.310	3868	"	"	"	"	
633	17.75	1085.4	245.8	2.687	5.218	2.629	5756	1.412	4076	"	"	45.5	4.0	

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GALCIT PROJECT NO. 1
TEST DATA SUMMARY
PROPELLANT CHARACTERISTICS TESTS
PIT B 7-7-43 THROUGH 9-23-43
TABLE IX f

RUN NO.	TIME EFF. sec	F lb	p_c psi, abs	f_t in^2	w lb/sec	r	c ft/sec	C_F	c^* ft/sec	OXIDIZER	FUEL	L^* in	EXHAUST NOZZLE ϵ	NOTES
634	17.04	1085.4	290.3	2.679	5.717	2.566	6113	1.396	4379	WFNA	ANILINE	45.5	4.0	
635														
636	21.31	960.9	264.2	2.692	5.473	2.617	5653	1.351	4184	MIXED ACID	ANILINE	54.6	4.0	
637	28.49	1069.3	286.9	2.684	5.893	2.618	5843	1.389	4207	"	"	"	"	
638	26.40	845.2	235.0	2.685	5.137	2.605	5298	1.339	3957	"	"	"	"	
639	26.35	748.2	214.0	2.688	4.399	2.565	5477	1.301	4210	"	"	"	"	
640	27.90	625.7	182.1	2.685	3.735	2.571	5394	1.280	4214	"	"	"	"	
641	25.71	942.9	261.7	2.687	5.324	2.521	5703	1.341	4253	"	"	"	"	
642	30.56	568.1	165.2	2.688	3.539	2.547	5169	1.279	4041	"	"	"	"	
643	33.19	419.2	129.6	2.684	2.832	2.537	4766	1.205	3955	"	"	"	"	
644	34.30	304.9	100.5	2.685	2.427	2.637	4045	1.130	3580	"	"	"	"	
645														
646	44.80	364.6	114.6	2.687	2.617	2.541	4486	1.184	3789	"	"	"	"	
647	26.61	1110.6	304.3	2.684	6.247	2.461	5726	1.360	4210	"	"	"	"	

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TABLE X

EFFECT OF LOW AMBIENT TEMPERATURE ON SPONTANEOUS IGNITION

Test No.	Oxidizer	Fuel	Propellant Temperature (° F)	Result
1	16% RFNA	Aniline	33	Good ignition
2	"	Aniline 15% F.A.	18	" "
3	"	"	18	" "
4	"	"	12	" "
5	"	"	5	" "
6	"	"	-1	" "
7	"	"	-10	Clogged lines; no ignition
8	"	Aniline 40% F.A.	-12	Good Ignition
9	"	"	-20	" "
10	"	"	-42	" "
11	"	"	-40	" "
12	"	Aniline 35% F.A.	-35	" "

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TABLE XI

CALCULATION OF THEORETICAL VALUES OF c* OF THE RFNA-ANILINE PROPELLANT

r	p_c psi abs	$\frac{p_c}{p_o}$	C theory (ft/sec)	γ theory	C_F theory	c* theory (ft/sec)
1	300	21.3	5207	1.272	1.401	3717
2	300	21.3	6514	1.261	1.403	4643
3	300	21.3	7091	1.221	1.414	5015
5	300	21.3	6688	1.206	1.418	4717
7	300	21.3	6192	1.210	1.417	4370
1	600	42.6	5658	1.269	1.500	3772
2	600	42.6	6998	1.261	1.503	4656
3	600	42.6	7694	1.220	1.520	5062

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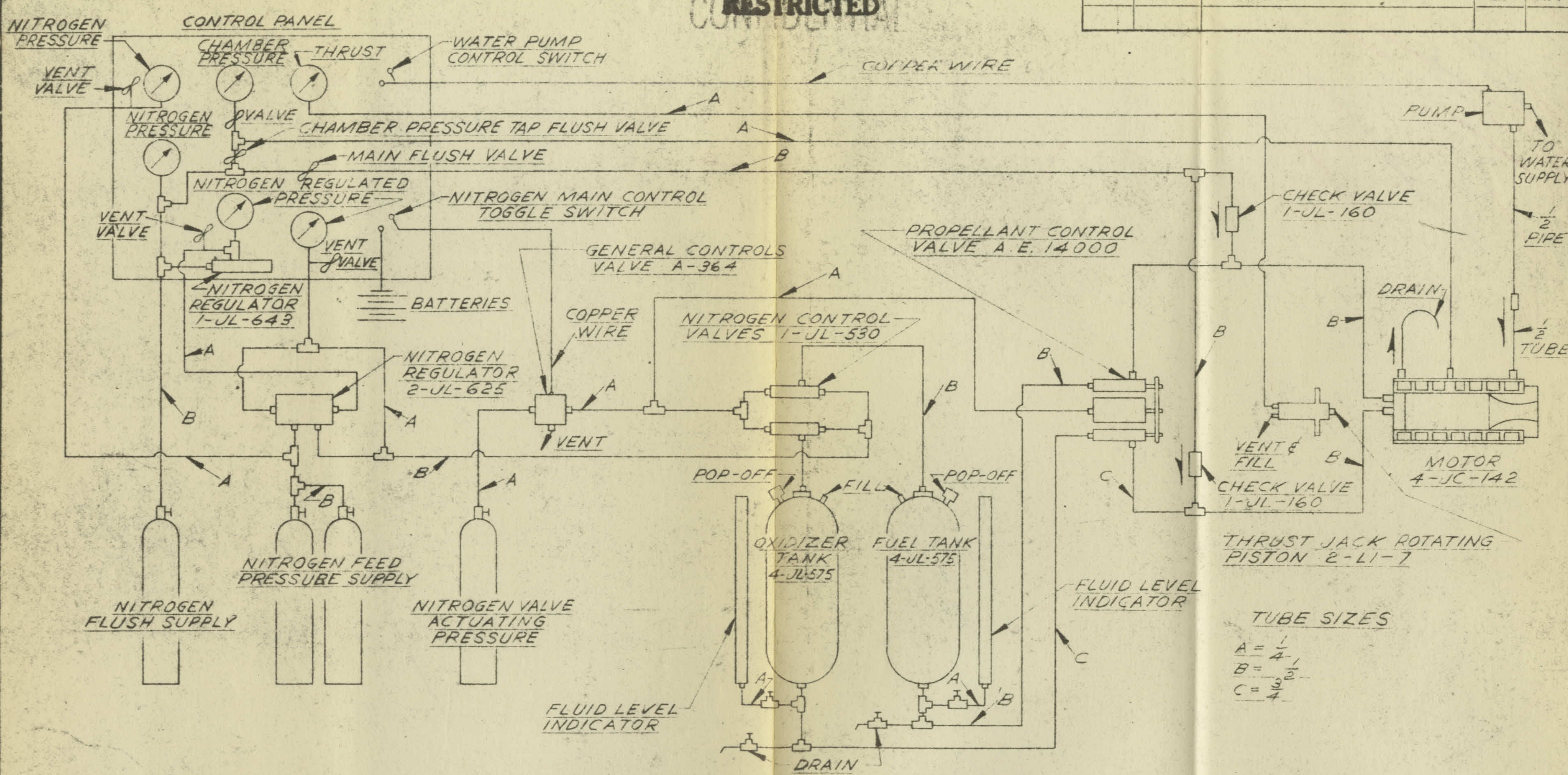
LET	DATE	CHANGE	BY	APP



NOTE:
1. ARROW ⟶ INDICATES DIRECTION OF FLOW.

—	—	PART
ENG. EXAM.	PROD. EXAM.	TOOL

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MATERIAL	FINISH	HEAT TREAT

GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY

DRAFTSMAN	CHECKED	APPROVED	ENGINEER
IMUS	[illegible]	[illegible]	WBP
9/14/43	9-17-43	9-17-43	9-18-43

TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED

SCALE	WEIGHT

NAME: SCHEMATIC DIAGRAM-1000# WATER COOLED MOTOR TEST CIRCUIT

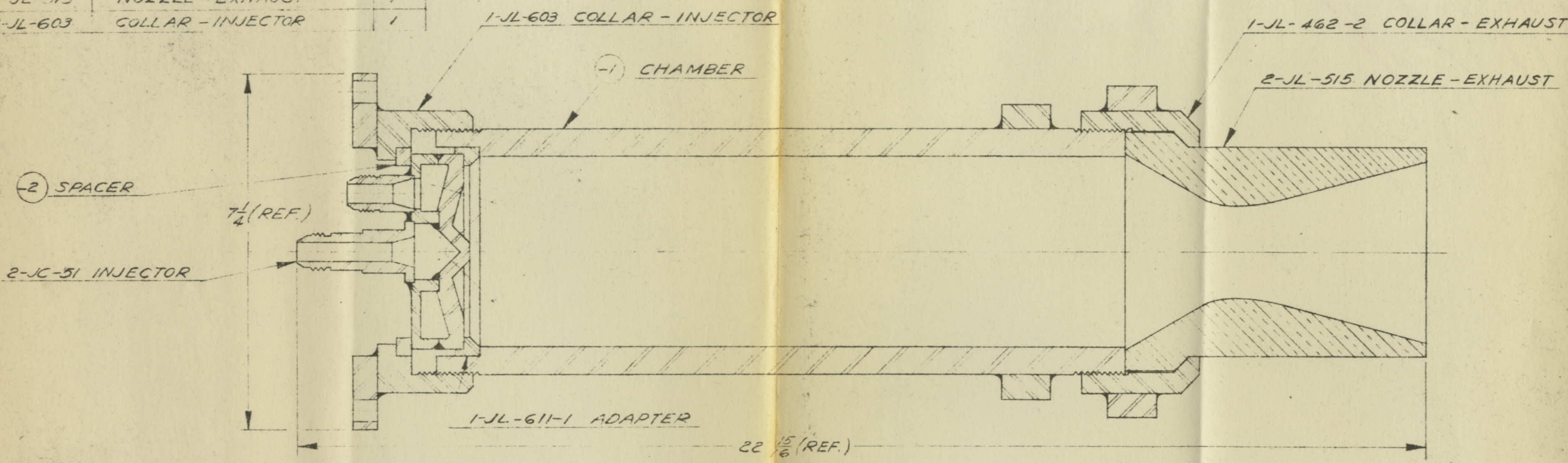
DRAWING NO. 2-JL-662

LATEST CHG. LET.

RESTRICTED

LET	DATE	CHANGE	BY	APP

2-JL-657	ASSEMBLY	
-1	CHAMBER	1
-2	SPACER	1
1-JL-611-1	ADAPTER	1
2-JC-51	INJECTOR	1
1-JL-462-2	COLLAR - EXHAUST	1
2-JL-515	NOZZLE - EXHAUST	1
1-JL-603	COLLAR - INJECTOR	1



SECTIONAL VIEW

OVERALL CHAMBER LENGTH = 14 IN.

$\ell_c = 13\frac{7}{16}$ IN.

$d_c = 3\frac{7}{8}$ IN.

$A.R = 3.47$

$d_c/d_t = 2.09$

$f_t = 2.69\ IN^2$

$\epsilon = 3.98$

$L^* = 64.2$ IN.

$L_{COMB.} = 14\frac{15}{16}$ IN.

LAST DASH NO. -2

	Morton	8 PART				ROBERTSON	[illegible]	E.G. [illegible]	WBP	TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED	
ENG. EXAM.	8-28-43 PROD. EXAM.	TOOL	MATERIAL	FINISH	HEAT TREAT	8-16-43 DRAFTSMAN	8-19-43 CHECKED	8-20-43 APPROVED	9-20-43 ENGINEER	HALF SCALE	65# EST. WEIGHT

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ASSEMBLY
1000 # MOTOR $L^* = 69$ IN
NAME

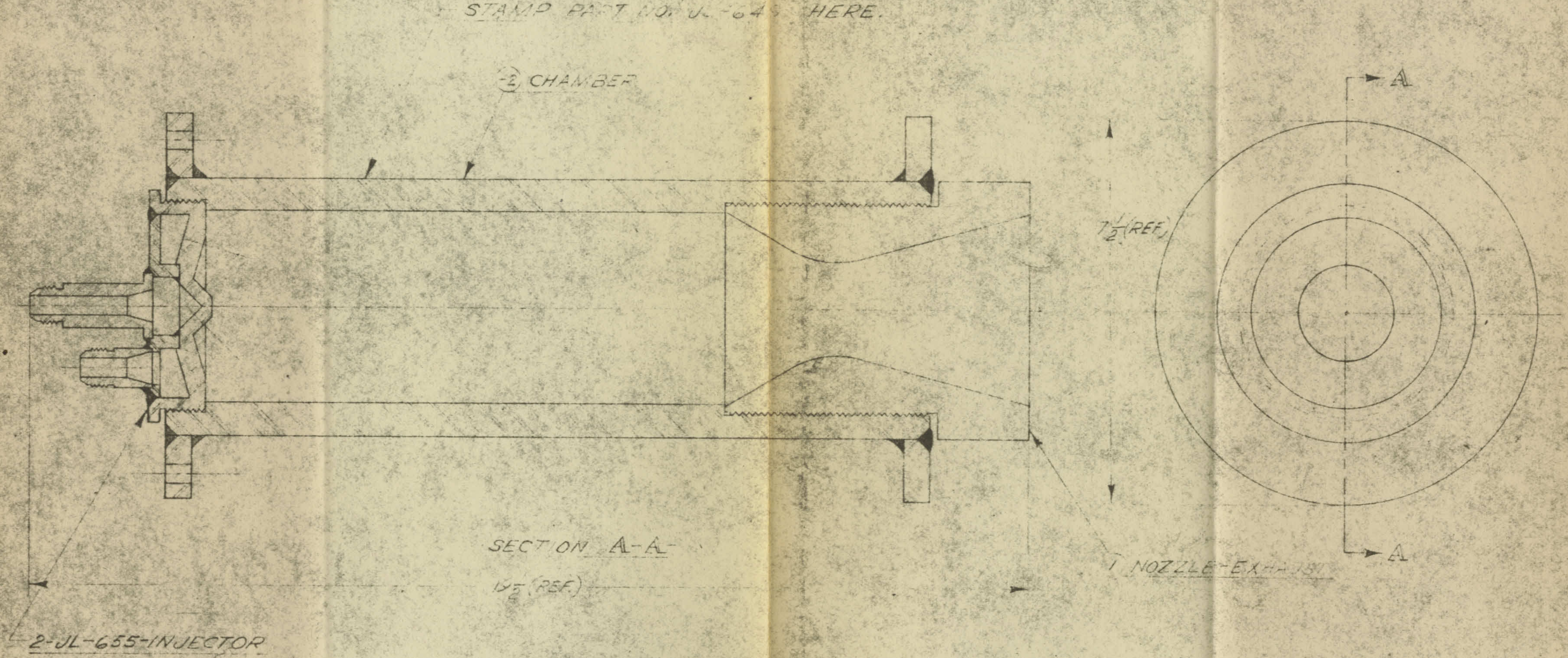
2-JL-657
DRAWING NO.

LATEST CHG. LET.

RESTRICTED

PART	DESCRIPTION	REQ.
2-JL-649	ASSEMBLY	—
-1	NOZZLE-EXHAUST	1
-2	CHAMBER	1
2-JL-655	INJECTOR	1

LET	DATE	CHANGE	BY	APP
A	[illegible]	$\ell_{comb} = 11\frac{2}{8}$ WAS $11\frac{3}{8}$	[illegible]	E.C.



LAST DASH NO. -2

$F = 1000$ LBS.
$L^* = 45.5$ IN.
I.D. $= 3.728$ IN.
A.R. $= 2.72$
$d_t = 1.850$ IN.
$\ell_{COMB} = 11\frac{2}{8}$ IN.

		2 PART
ENG. EXAM.	PROD. EXAM.	TOOL

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GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY

DRAFTSMAN	CHECKED	APPROVED	ENGINEER
JMOS	[illegible]	E.C. CROFOOT	W.P.
8/3/43	8-7-43	8-7-43	8-9-43

TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED

SCALE	WEIGHT
HALF	42# EST.

NAME: ASSEMBLY - 1000# MOTOR THREADED NOZZLE & INJECTOR

DRAWING NO. 2-JL-649

LATEST CHG. LET. A

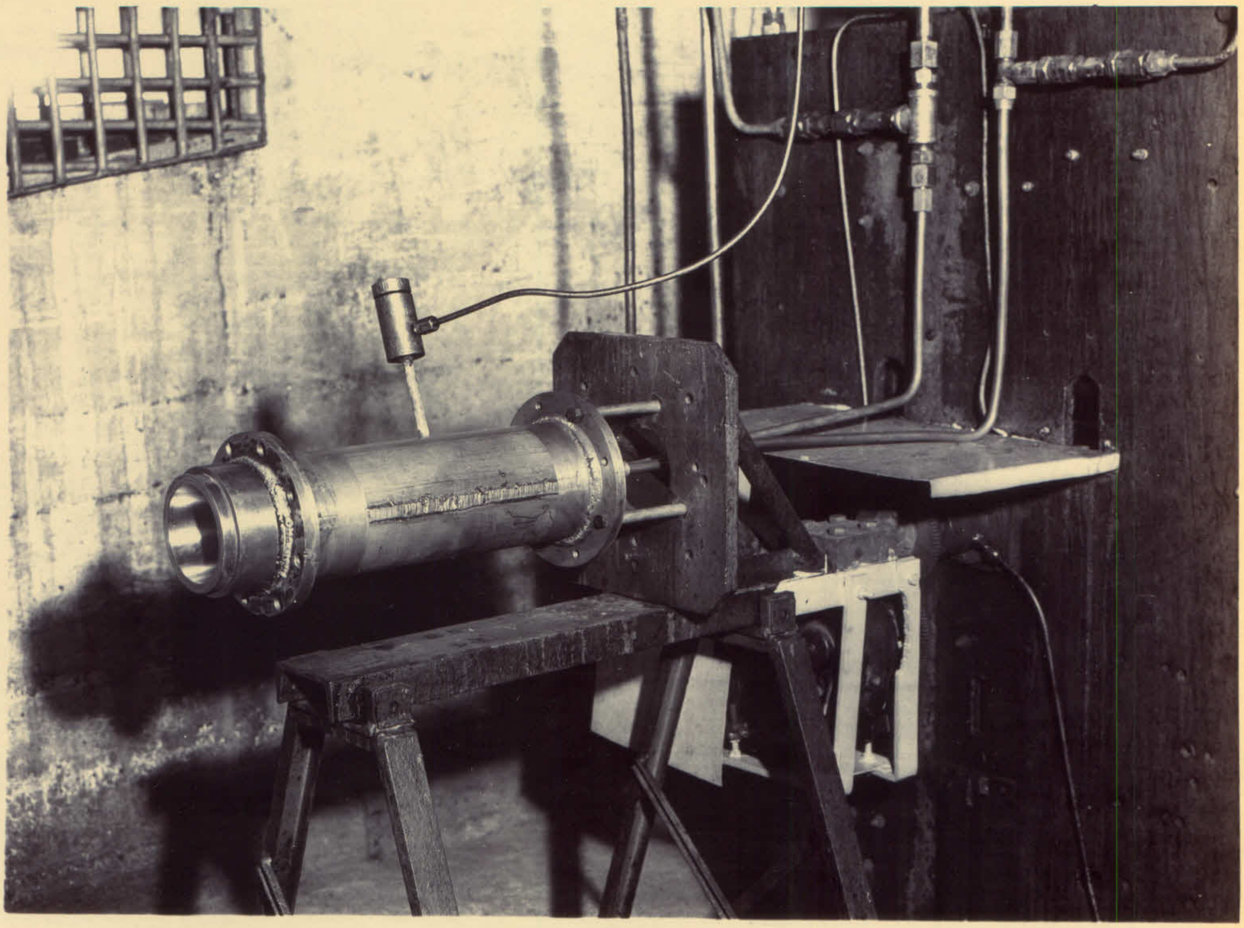

FIGURE 4. THE MOTOR OF FIGURE 3 INSTALLED ON THE TEST STAND.

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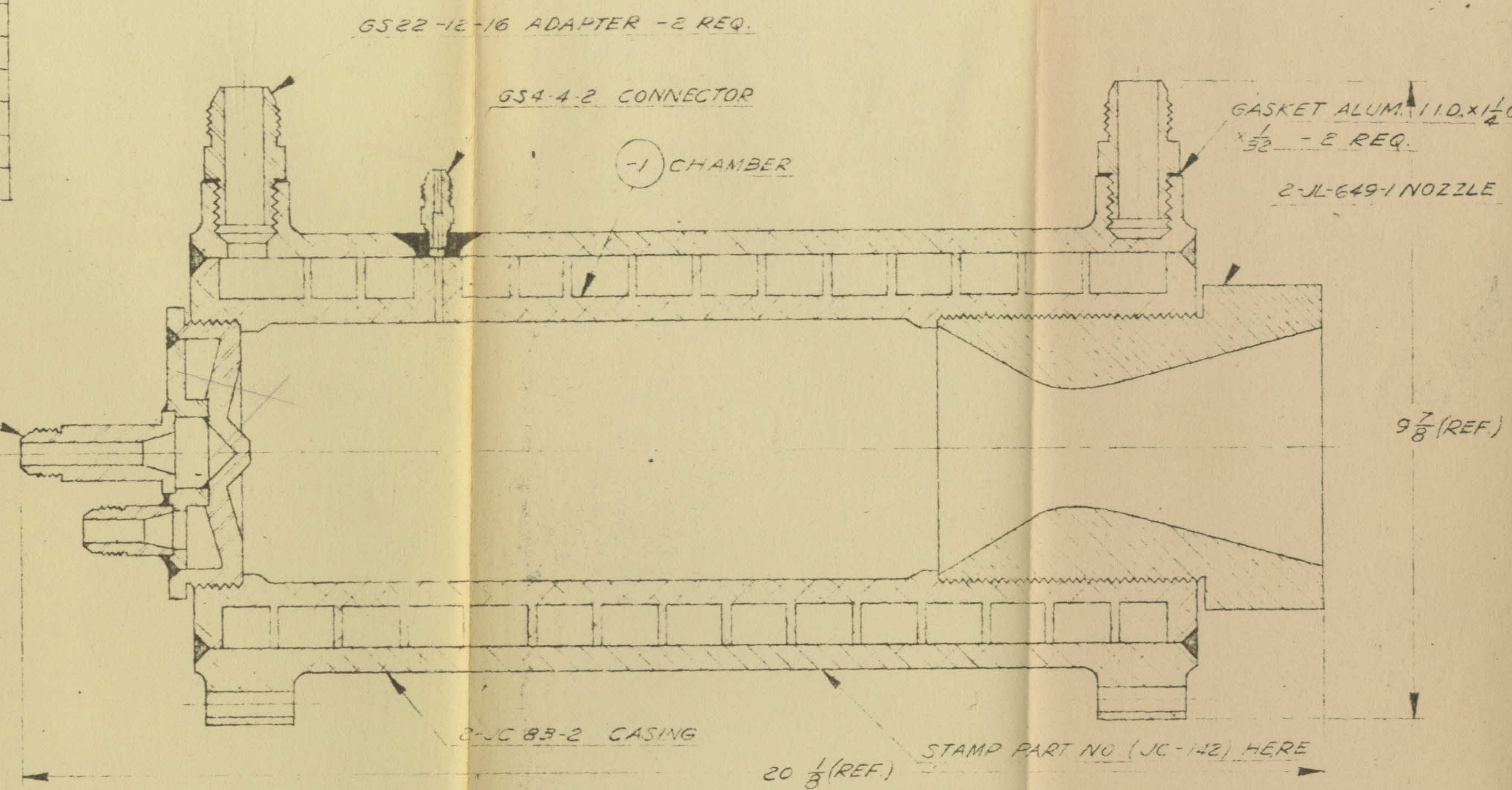
PART	DESCRIPTION	RE
2-JC-142	ASSEMBLY	
-1	CHAMBER	1
2-JL-655	INJECTOR	1
2-JL-649-1	NOZZLE	1
2-JC-83-2	CASING	1
GS 22-12-16	ADAPTER	2
GS 4-4-2	CONNECTOR	1
GASKET	ALUM. 1 I.D. × 1 1/4 O.D. × 1/32	2

LET	DATE	CHANGE	BY	APP



SECTIONAL VIEW

$L^* = 54.6$ IN.
$F = 1000^{\#}$
$AR = 2.69$
$d_c = 1.850$ IN
I.D. = 4 IN
$L_{COMB} = 12.188$ IN

3. -1 CHAMBER & -2 CASING ARE TO BE ASSEMBLED AS A SHRINK FIT WITH VARIATION FROM DRY ICE TO BOILING WATER TEMP'S
2. PROCEDURE FOR PRESSURE TAP ASSEMBLY
(a) ALIGN HOLE IN CASING WITH WIDE PORTION OF HELICAL FIN
(b) PUDDLE HOLE WITH WELD
(c) FACE OFF WELDED SURFACE
(d) DRILL 1/8 (.125) THRU ONE SIDE AT ℄
(e) DRILL 21/64 (.328) - 13/32 DEEP & TAP 1/8 N.P.

NOTE: 1. WELD AS SHOWN

LAST DASH NO. -1

ENG. EXAM.	PROD. EXAM.	PART	MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	SCALE	WEIGHT
[illegible]	[illegible]	10				PIBERTSON	[illegible]	E.C. CROFUT	[illegible]	HALF	50# EST.
9-7-43	9-7-43	TOOL				9-1-43	9-7-43	9-7-43	9-8-43		

TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED

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NAME: MOTOR #35 ASSEMBLY - REGENERATIVE

DRAWING NO.: 2-JC-142

LATEST CHG. LET.

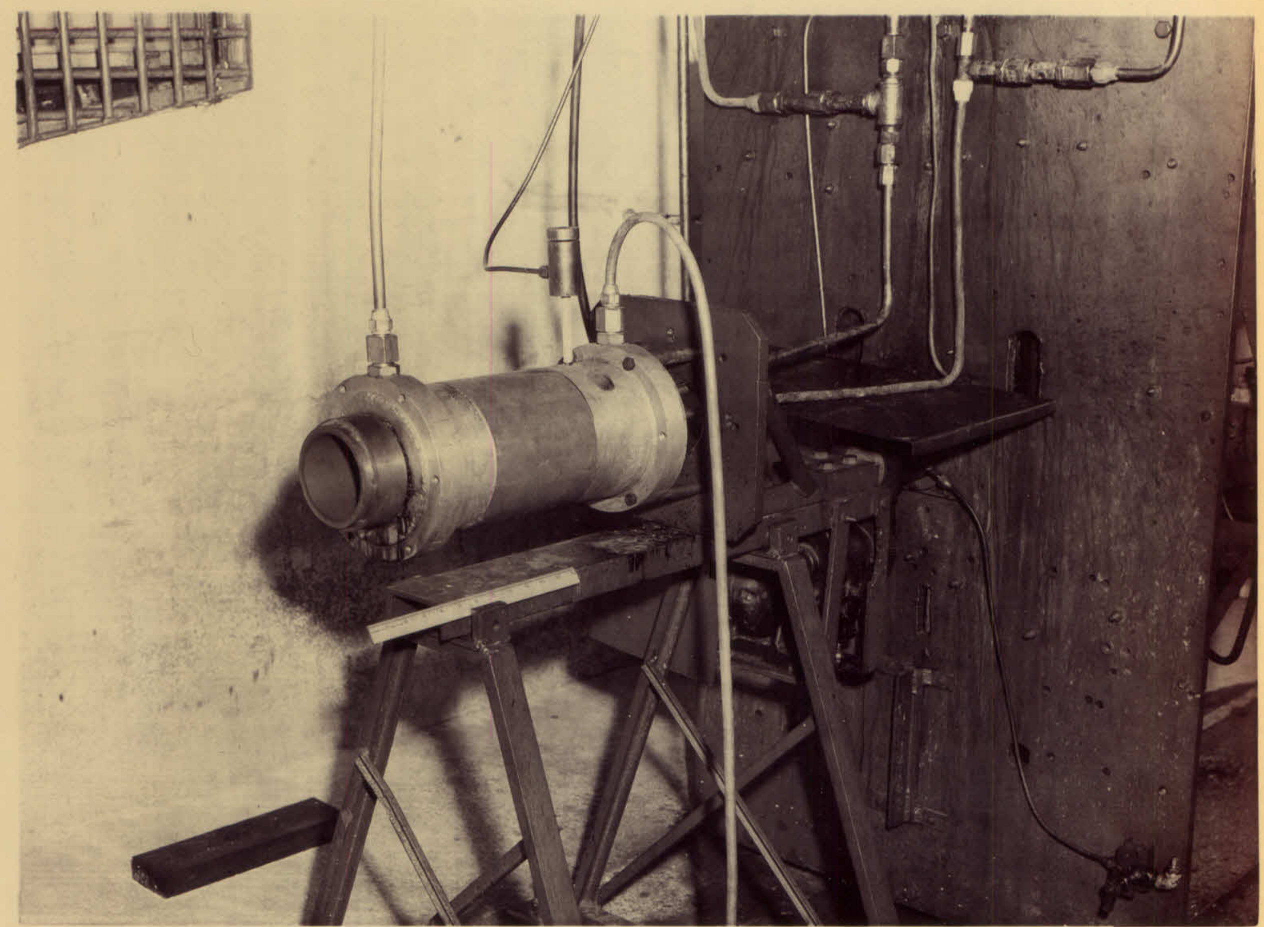


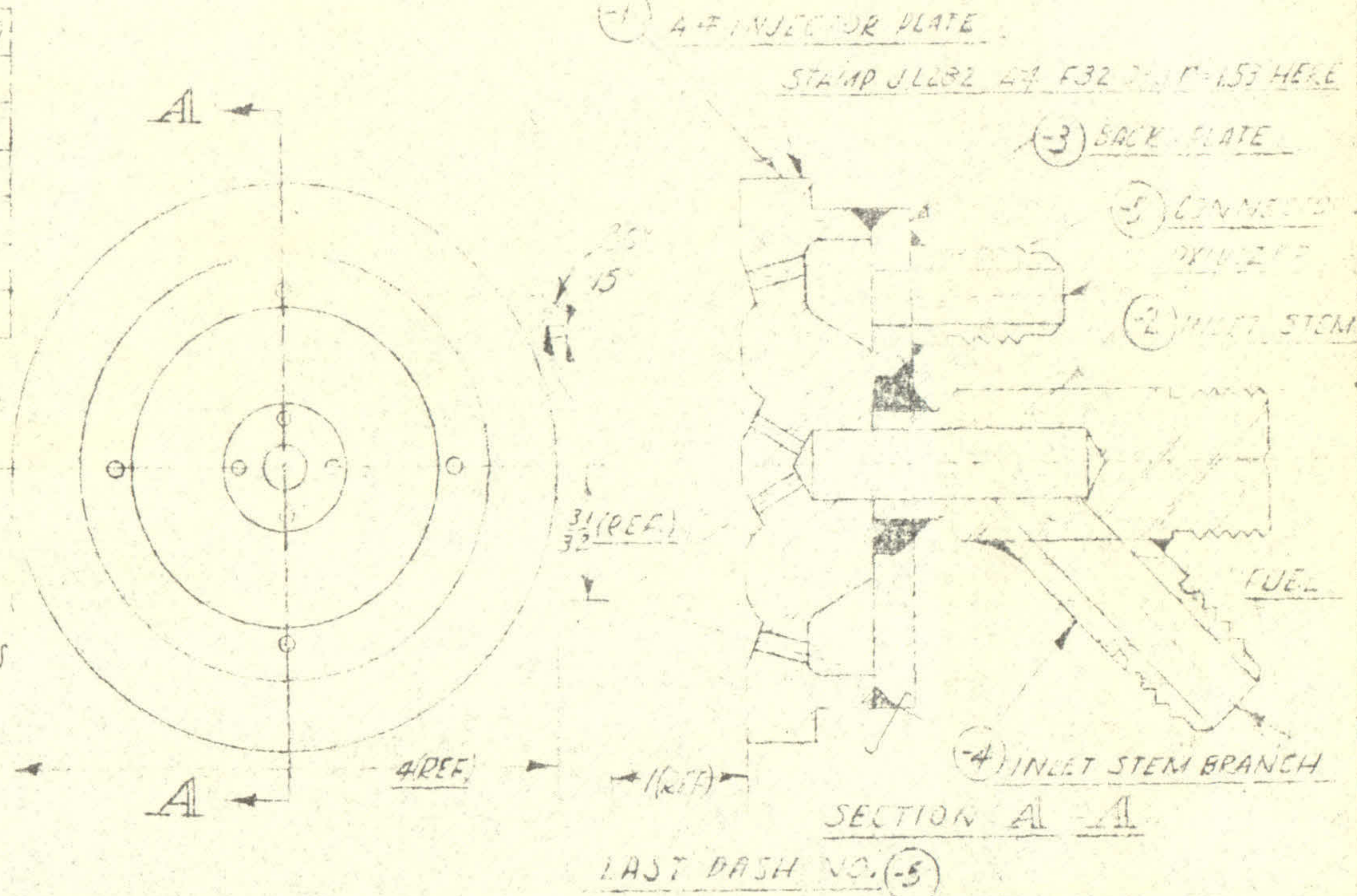
FIGURE 6. THE MOTOR OF FIGURE 5 INSTALLED ON THE TEST STAND.

RESTRICTED

CONFIDENTIAL RESTRICTED J 10-9-43 REDRAWN KING

PART	DESCRIPTION	REQ
1-JL-282	ASSEMBLY	✓
-1	INJECTOR PLATE	1
-2	INLET STEM	1
-3	BACK PLATE	1
-4	INLET STEM BRANCH	1
-5	CONNECTOR	1

F = 1900#
r = 1.53
β = 5°7'
P_c = 113 psi @ 3.66 #/sec.
c_c = 1.58
c_r = 1.02
OXIDIZER HOLE #30 (.128) 4 PLACES
FUEL HOLE #32 (.116) 4 PLACES



NOTE: WELD AS SHOWN

LAST DASH NO. (-5)

STAINLESS STEEL (REF.)			KING				TOLERANCES ± .010 OR ... UNLESS OTHERWISE NOTED	
			10-9-43	10-30-43	10-30-43		3/4	3.25#A
MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	SCALE	WEIGHT

GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY

ASSEMBLY
INJECTOR A-4 TYPE
NAME

1-JL-282
DRAWING NO.

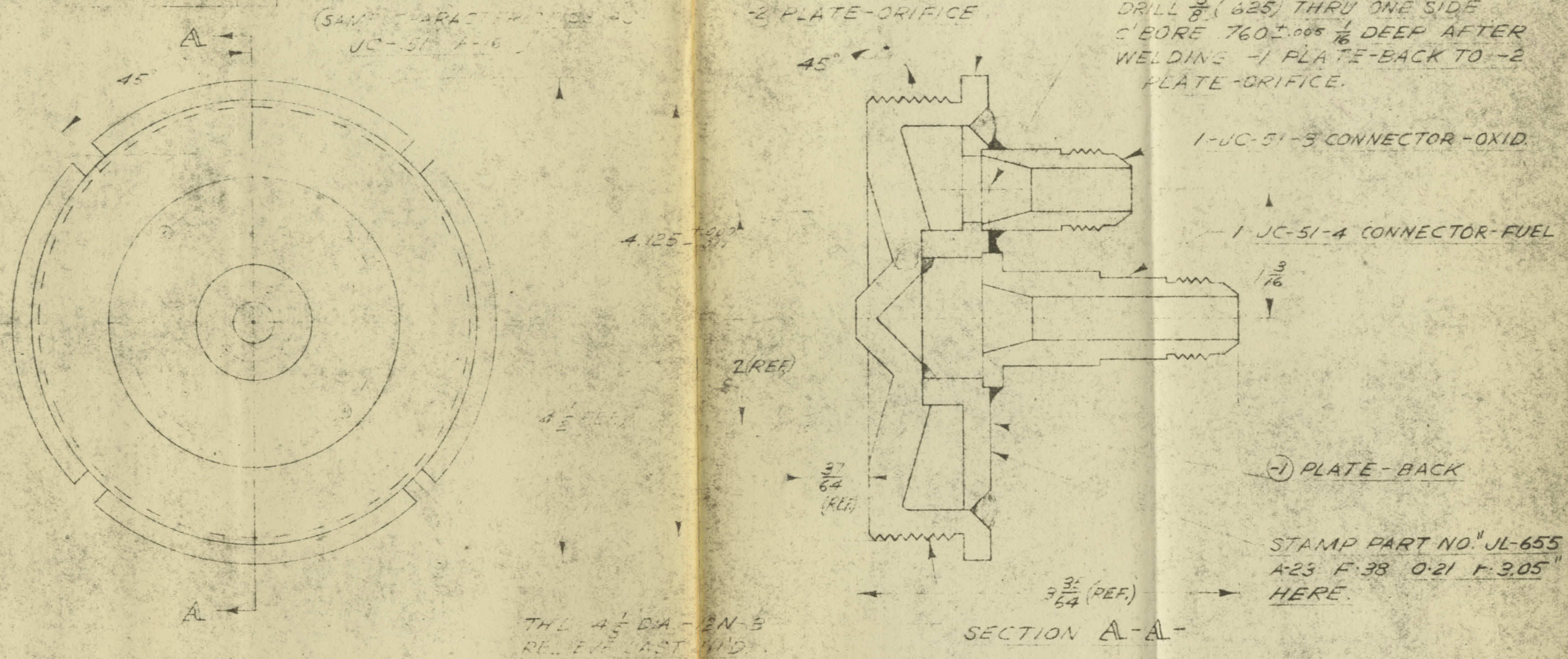
CONFIDENTIAL RESTRICTED

RESTRICTED

LET	DATE	CHANGE	BY	APP
A	8/14/43	CHAMFER FIRST THD 45° WA- 10°	IMUS	B.C.

PART	DESCRIPTION	REQ
2-JL-655	[illegible]	1
-1	PLATE-BACK	1
-2	PLATE-ORIFICE	1
2-JC-51-3	CONNECTOR-OXIDIZ.	1
2-JC-51-4	CONNECTOR-FUEL	1

F = 0.0[illegible]55 [illegible] + 0°35'
r = 3.05 q_0 = 20#/in²@4.07#/SEC.
OXIDIZER HOLES-#[illegible] 4 REQ
FUEL HOLES-#38(.1015) 4 REQ
σ_0 = 1.53
σ_f = 1.02
(SAME CHARACTERISTICS AS JC-51 A-16)



LAST DASH NO. -2

NOTE:
1. WELD AS SHOWN BEFORE MACH. THDS.
2. BREAK SHARP EDGES.

		PART
ENG. EXAM.	PROD. EXAM.	TOOL

GALCIT PROJECT No. 1
RESTRICTED
PROJECT No. MX121
This document contains information affecting the national defense of the United States within the meaning of the Espionage act, U. S. C. 50; 31 and 32. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

MATERIAL	FINISH	HEAT TREAT
	MACH	

GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY

DRAFTSMAN	CHECKED	APPROVED	ENGINEER
IMUS	[illegible]	E.C. CROFUT	W.B. P[illegible]
8/5/43	8-9-43	8-10-43	8-11-43

NAME: ASSEMBLY 1000# INJECTOR "A-23"

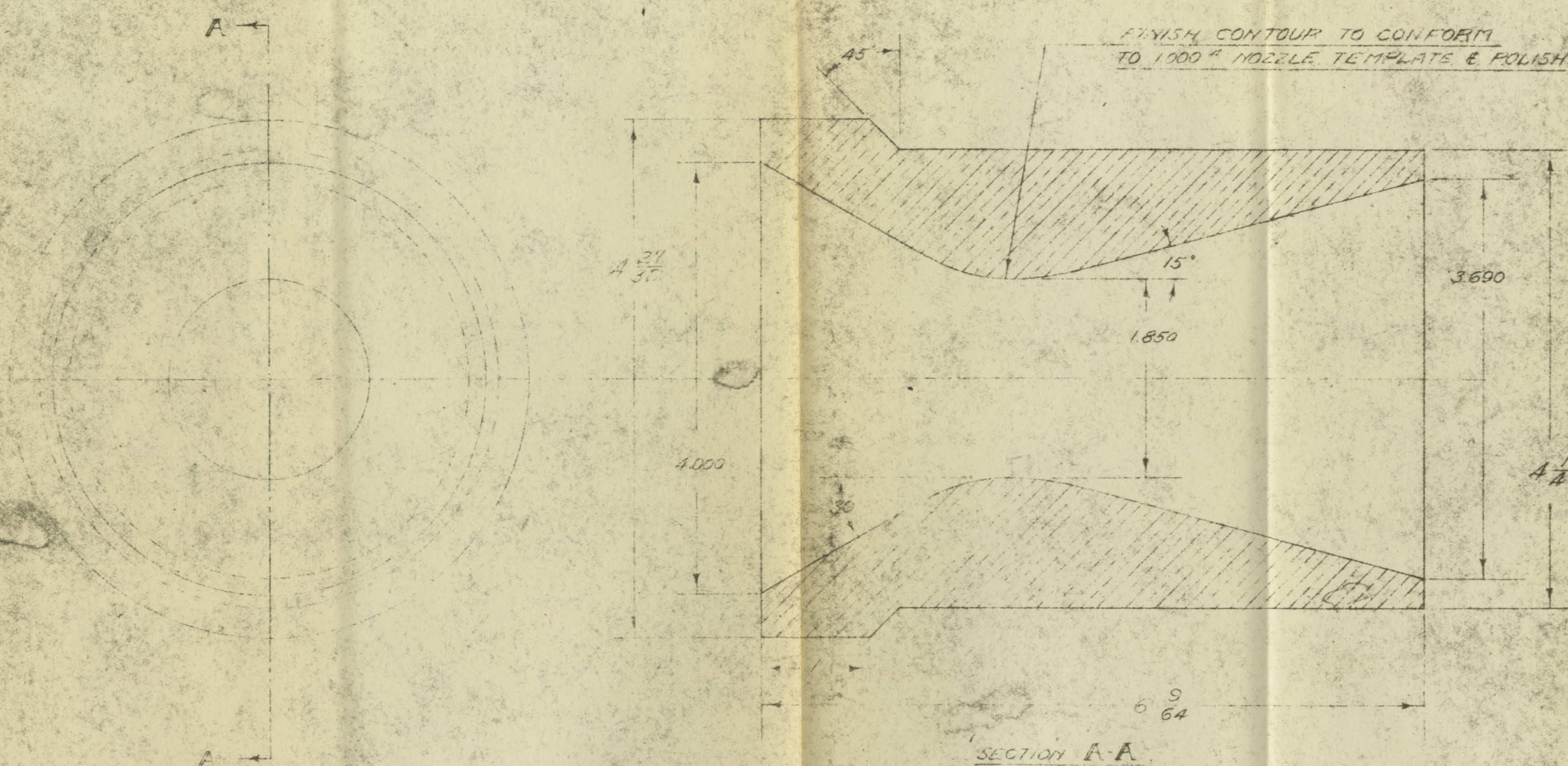
TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED

SCALE	WEIGHT
FULL	3.2# EST.

DRAWING NO.: 2-JL-655

LATEST CHG. LET.: A

LET	DATE	CHANGE	BY	APP
A	1/26/43	CHROME PLATE ADDED	IMUS	EGC
B	2/24/43	WEIGHT ADDED	KEINATH	EGC



NOTE: INSIDE CONTOUR DIMENS. AS GIVEN ARE DIMENS. AFTER CHROME PLATING.

MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	SCALE	WEIGHT
COPPER	MACH & CHROME PLATE INSIDE CONTOUR		Barkhit	E.C.	E.C.	WBP	FULL	19# ACT.

TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED

GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY

NAME: NOZZLE - EXHAUST 1000# THRUST

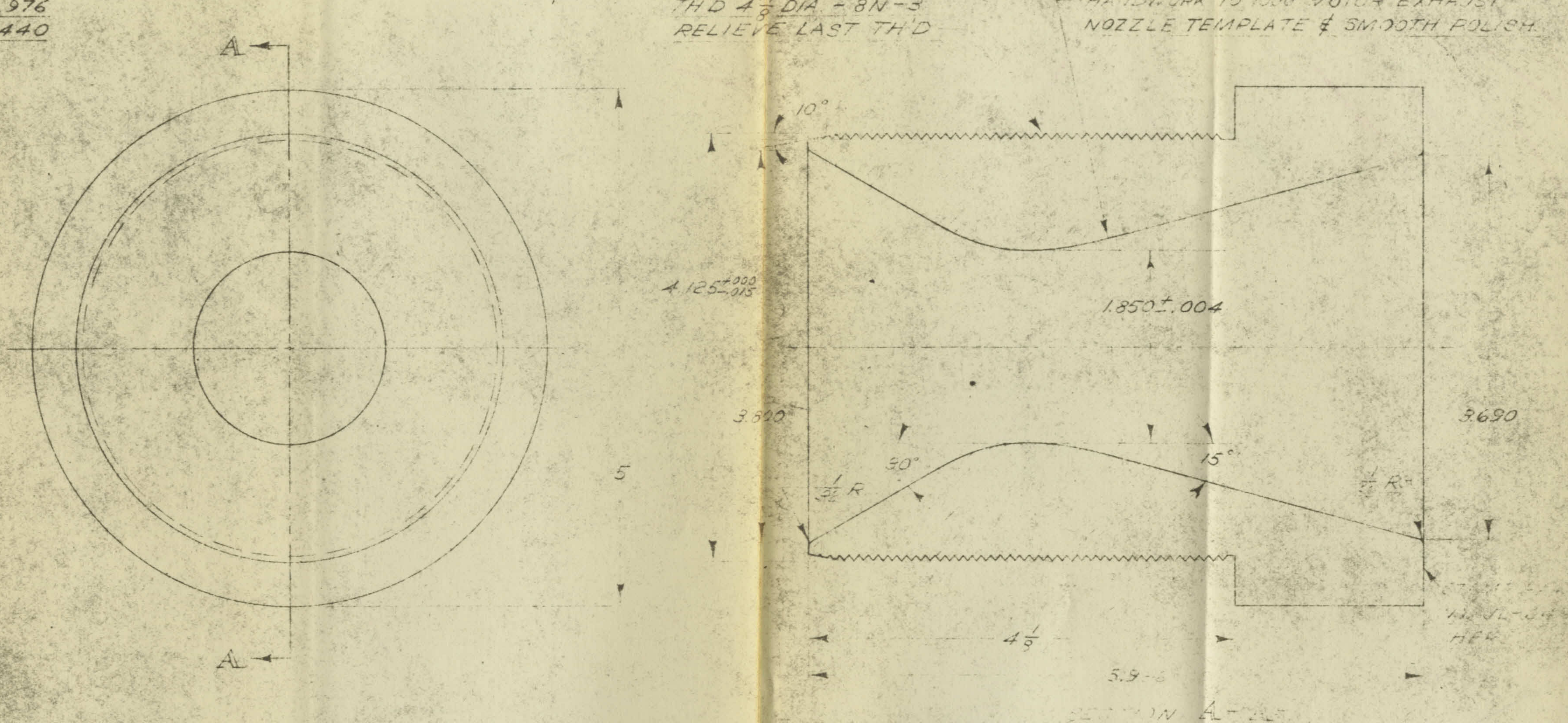
DRAWING NO. 2-JL-515

LATEST CHG. LET. B

LET	DATE	CHANGE	BY	APP

RESTRICTED

$F = 1000$ LBS.
$p_c = 300$ LBS./IN^2
L^* VOLUME = 11.8 IN^3
$f_t = 2.683$ IN^2
$\epsilon = 3.976$
$K = .440$



NOTE:
1. HARD CHROME PLATE BOTH ENDS & INSIDE CONTOUR .008 THICK.
2. DIMENSIONS TO BE MET AFTER PLATING.
3. BREAK SHARP EDGES.

—	—	PART	COPPER ROD 5 DIA. × $5\frac{3}{32}$	MACH. & NOTED		IMUS				TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED
ENG. EXAM.	PROD. EXAM.	TOOL				8/6/43	8/5/43	8-5-43	8-5-43	FULL / 16# EST
			MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	SCALE / WEIGHT

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PROJECT No. MX121

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CALIFORNIA INSTITUTE OF TECHNOLOGY

NOZZLE EXHAUST - 1000# MOTOR
THREADED NOZZLE & INJECTOR

NAME

2-JL-649-1

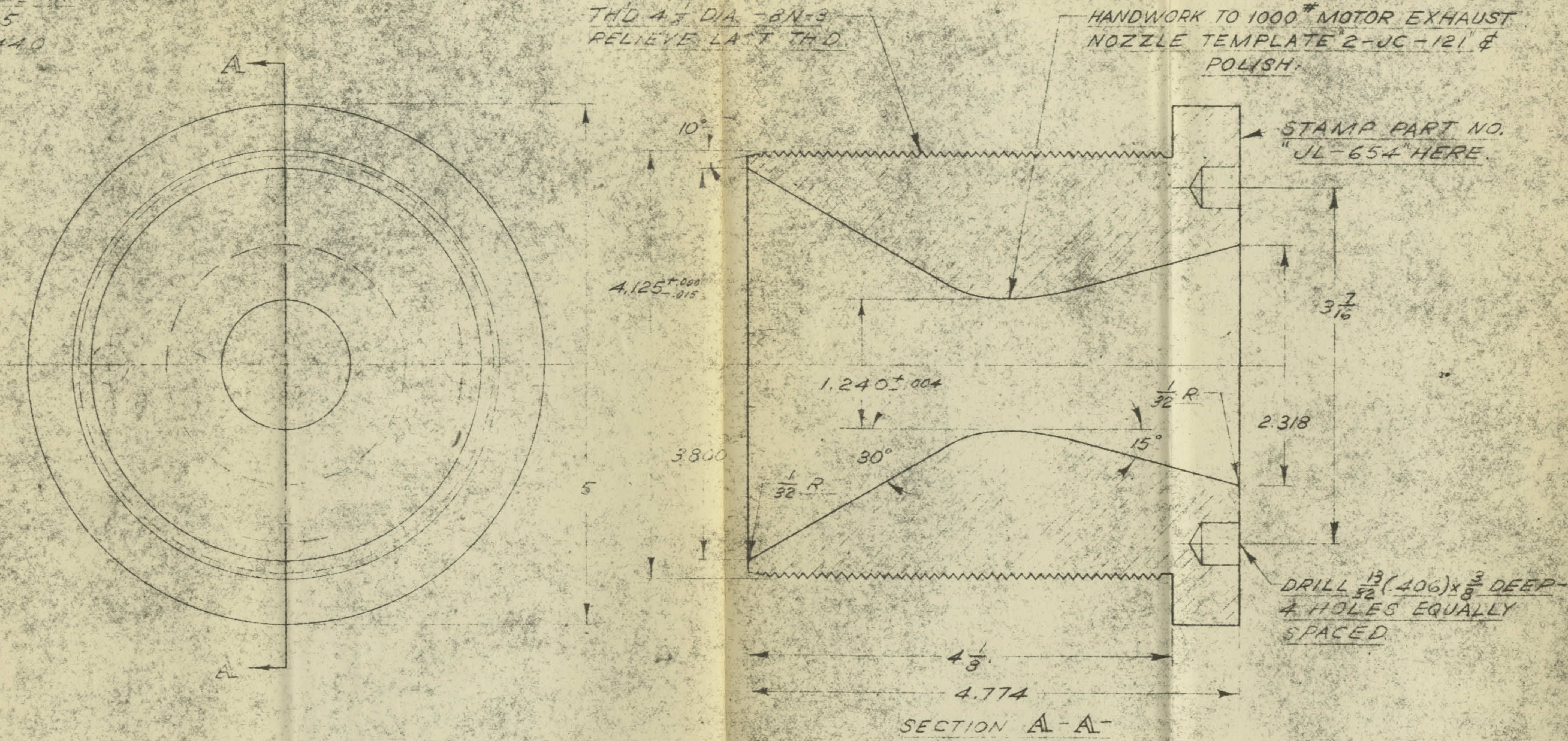
DRAWING NO.

LATEST CHG. LET.

CONFIDENTIAL RESTRICTED

LET	DATE	CHANGE	BY	APP

F = 1000 LBS
p_c = 550 LBS./IN.^{2}
L* VOLUME = 11.8 IN.^{3}
f_t = [illegible] IN.^{2}
ϵ = 5.5
K = .440



3. BREAK SHARP EDGES.
2. DIMENSIONS TO BE MET AFTER PLATING.
1. HARD CHROME PLATE BOTH ENDS & INSIDE CONTOUR .008 THICK.

NOTE:

		PART	COPPER ROD 5 1/4 x 4 1/2	MACH. & NOTED		IMUS	Barney	[illegible]	WBP	TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED	
ENG. EXAM.	PROD. EXAM.	TOOL				8/4/43	8/5/43	8-5-43	8-5-43	FULL	15# EST.
			MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	SCALE	WEIGHT

GALCIT PROJECT NO. 1
CONFIDENTIAL RESTRICTED
PROJECT No. MX121

GUGGENHEIM AERONAUTICAL LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY

NOZZLE - EXHAUST
1000# MOTOR - THREADED TYPE
NAME

2-JL-654
DRAWING NO.

LATEST CHG. LET.

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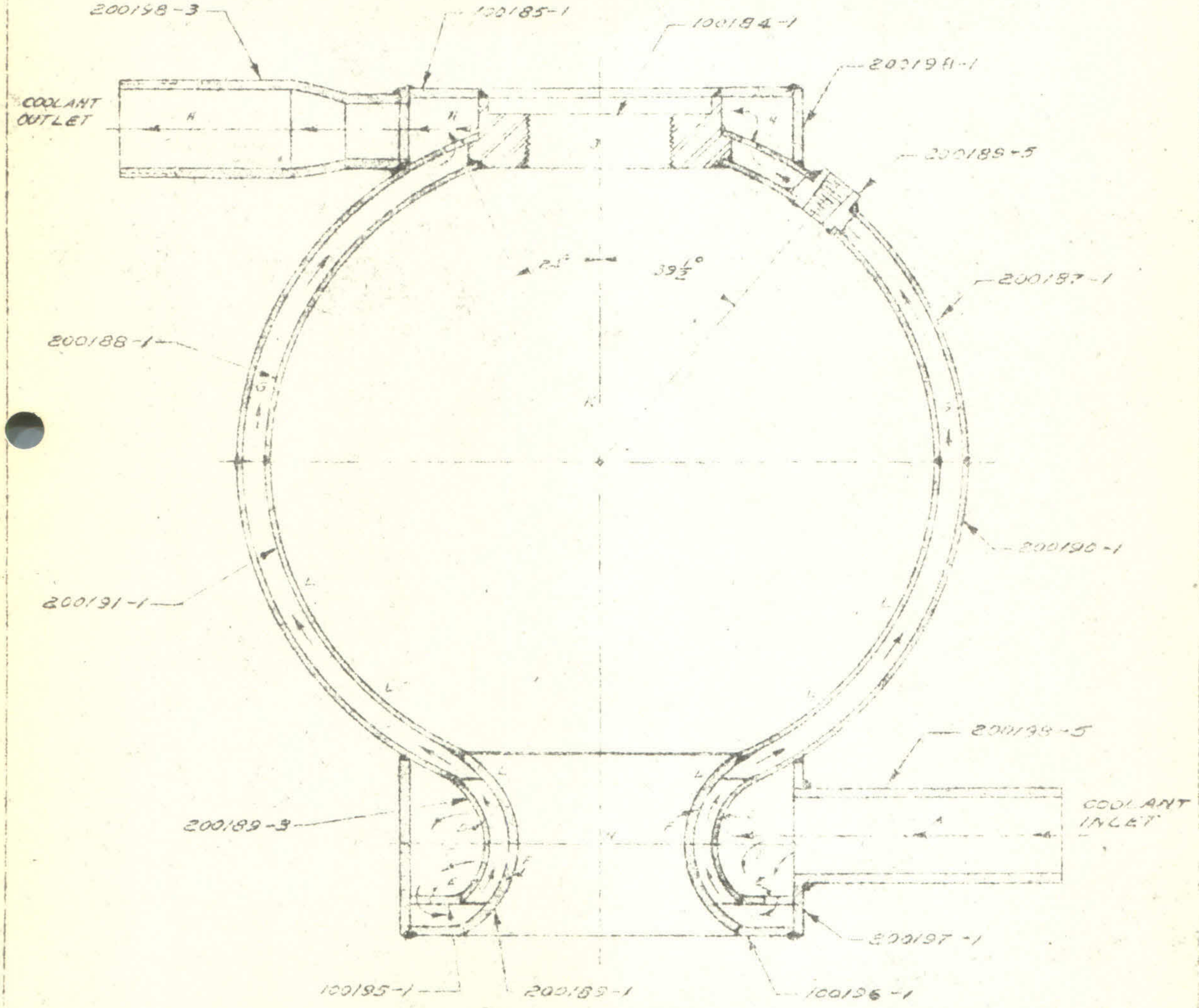
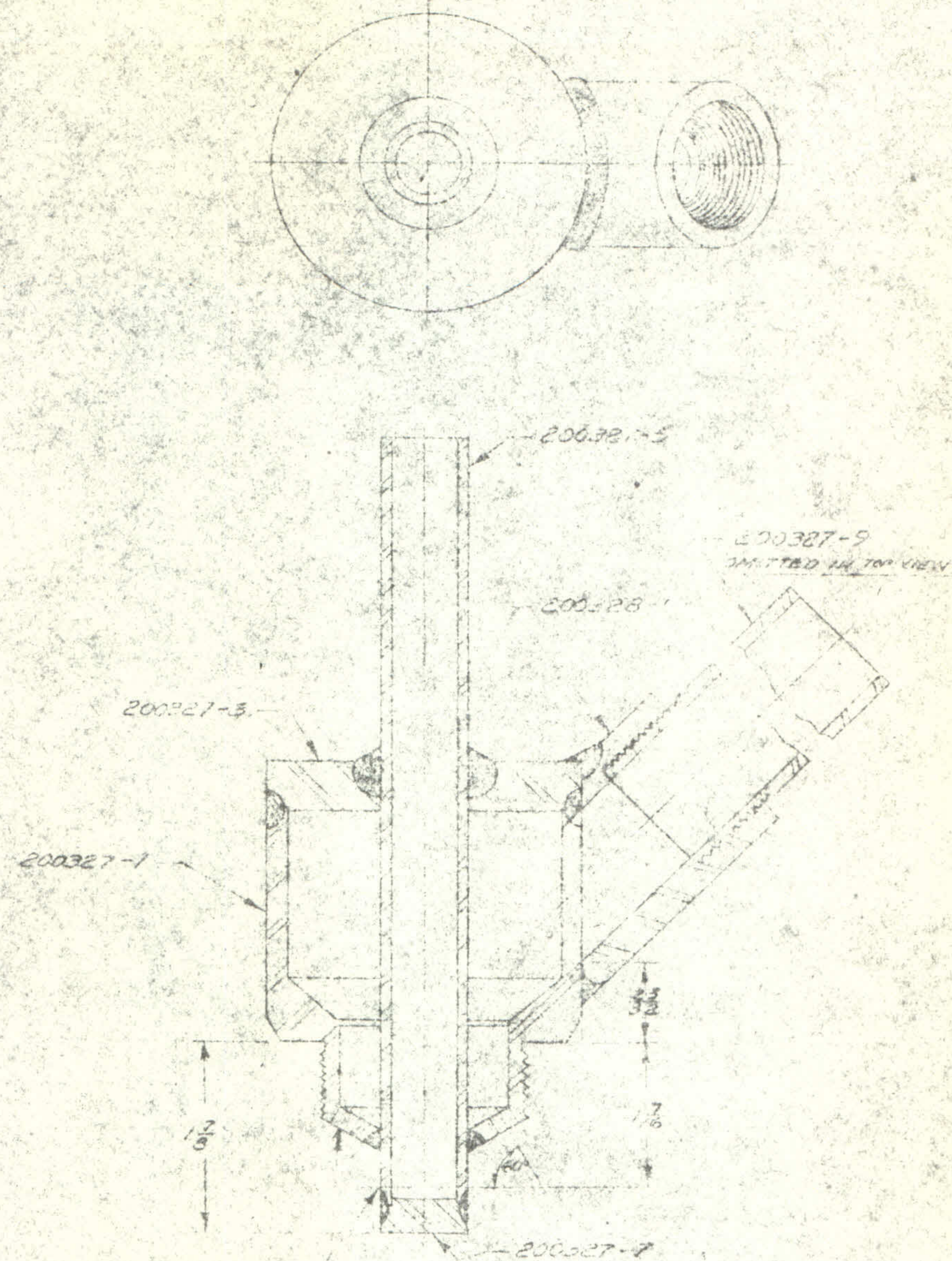


RESTRICTED

18-8 S.S.	POLISH INSIDE WELDS		R.S.W. 5-43	R.C.S.	R.C.S.		TOL.
MATERIAL	FINISH		DRAWN	ENGR			
U.S.N. ENG. EXP. STA. ANNAPOLIS, M			ASSEMBLY -				

RESTRICTED



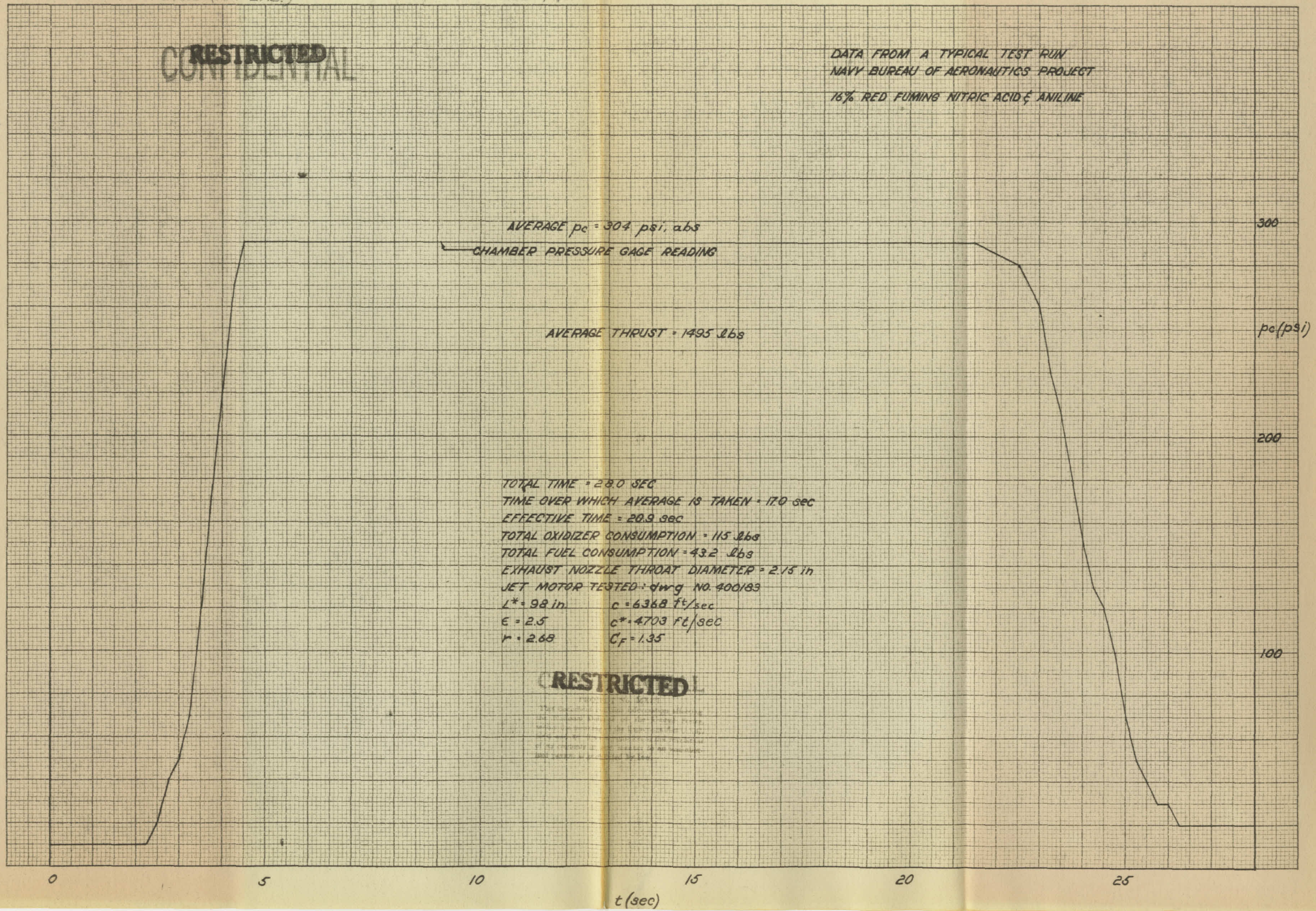
SEE DETAIL DRAWING NO. 200327
FOR DETAIL OF DRILLED HOLES.
SIZE OF DRILLS TO BE DETERMINED.

18-8 S.S.	MACH		R.S.H. / 3-29-43			T.L. / TEST-B3350
MATERIAL	FINISH	QUAN.	DRAFTSMAN	ENG.	CHECKED / APPROVED	SCALE-
USN ENG. EXP. STA. ANNAPOLIS, MD. AERO PROJECT			ASSEMBLY INJECTOR DUAL MOTOR, MK. I, MOD. II			200329
			NAME			DRAWING NO.

TRACED | 3-31-43

GALCIT PROJECT NO. 1
REPORT NO. 20 (AIR LAB.)

PAGE 82
FIG 14



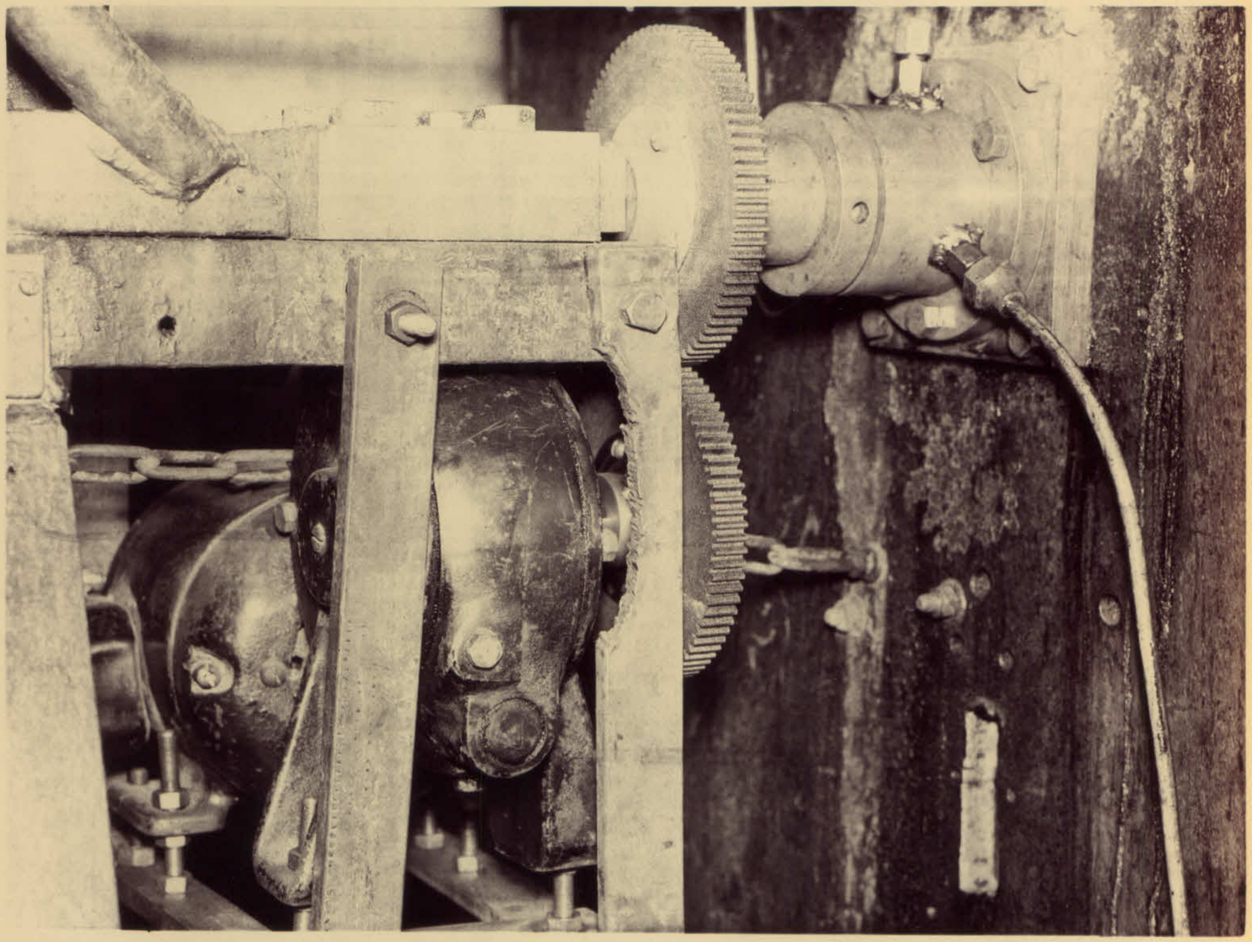
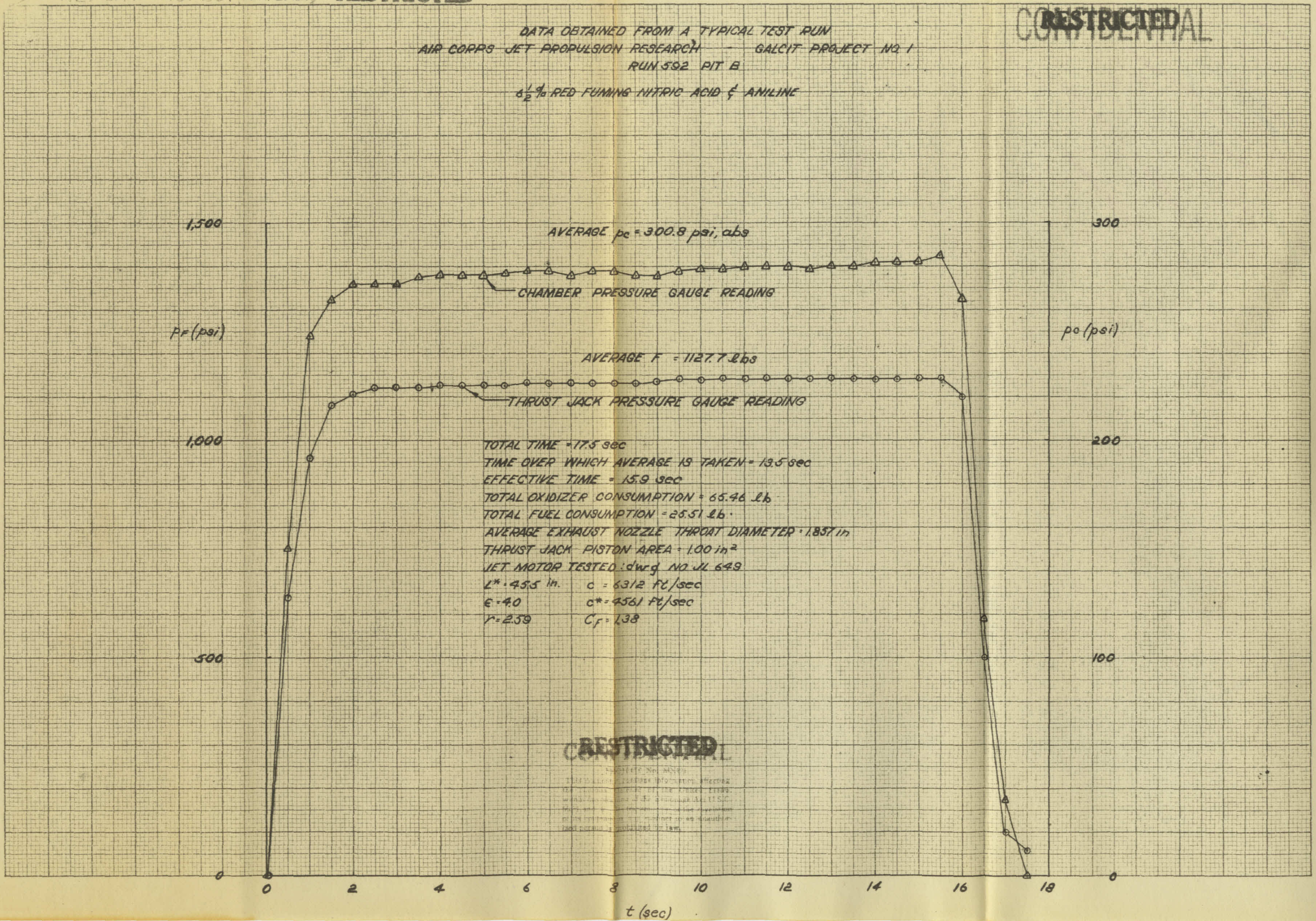
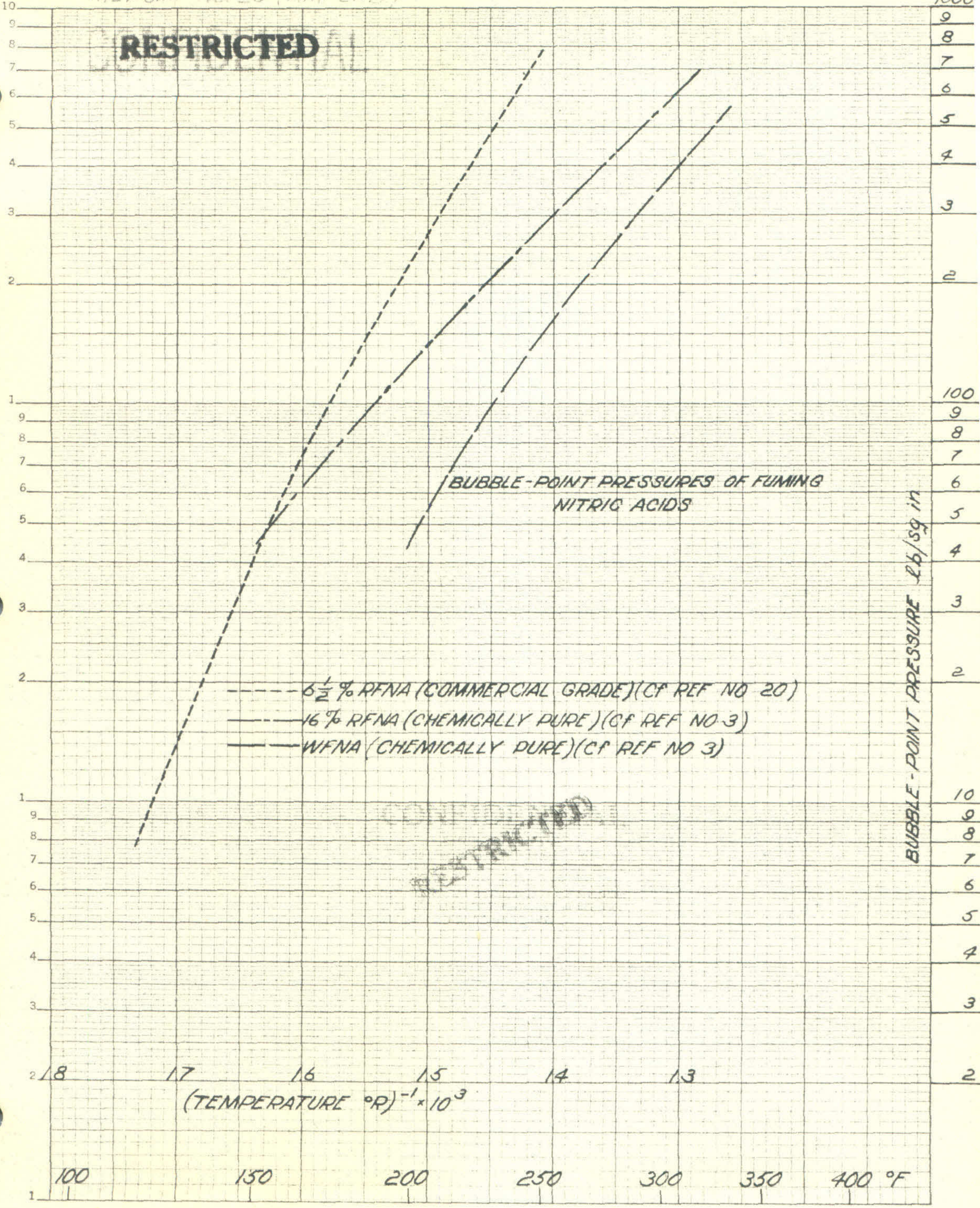


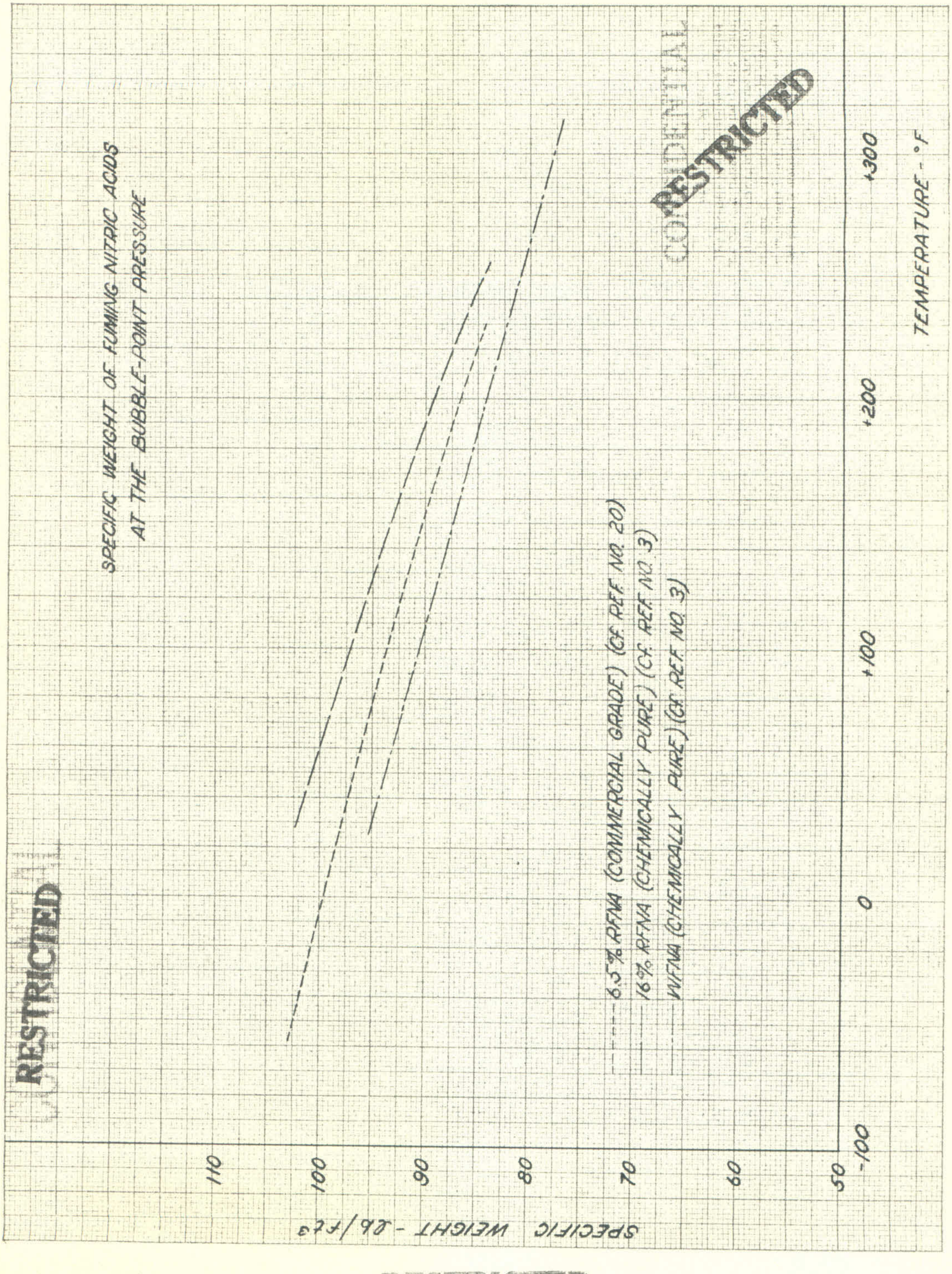
FIGURE 15. VIEW OF HYDRAULIC THRUST JACK WITH ROTATING PISTON SHOWING ELECTRIC MOTOR AND GEARS USED TO ROTATE THE PISTON.


RESTRICTED
DATA OBTAINED FROM A TYPICAL TEST RUN
AIR CORPS JET PROPULSION RESEARCH - GALCIT PROJECT NO. 1
RUN 592 PIT B
$6\frac{1}{2}\%$ RED FUMING NITRIC ACID & ANILINE
AVERAGE pc = 300.8 psi, abs
CHAMBER PRESSURE GAUGE READING
AVERAGE F = 1127.7 lbs
THRUST JACK PRESSURE GAUGE READING
TOTAL TIME = 17.5 sec
TIME OVER WHICH AVERAGE IS TAKEN = 13.5 sec
EFFECTIVE TIME = 15.9 sec
TOTAL OXIDIZER CONSUMPTION = 65.46 lb
TOTAL FUEL CONSUMPTION = 25.51 lb
AVERAGE EXHAUST NOZZLE THROAT DIAMETER = 1.857 in
THRUST JACK PISTON AREA = 1.00 in²
JET MOTOR TESTED: dwg NO JL 649
L* = 45.5 in.
c = 6312 ft/sec
ε = 4.0
c* = 4561 ft/sec
r = 2.59
Cf = 1.38
Pf (psi)
pc (psi)
0
500
1,000
1,500
0
100
200
300
0
2
4
6
8
10
12
14
16
18
t (sec)
RESTRICTED

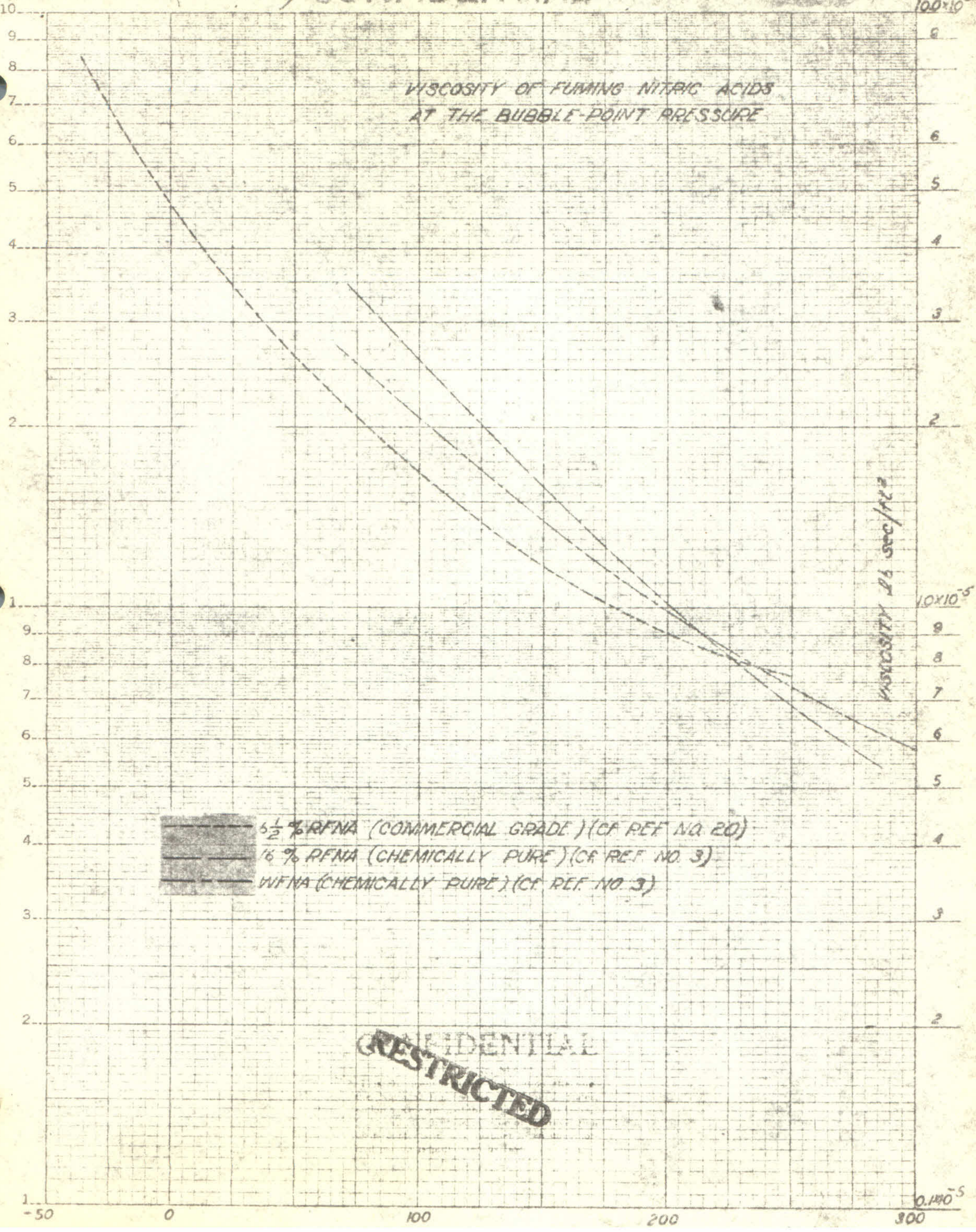
RESTRICTED



RESTRICTED


SPECIFIC WEIGHT OF FUMING NITRIC ACIDS
AT THE BUBBLE-POINT PRESSURE
6.5% RFNA (COMMERCIAL GRADE) (CF. REF. NO. 20)
16% RFNA (CHEMICALLY PURE) (CF. REF. NO. 3)
WFNA (CHEMICALLY PURE) (CF. REF. NO. 3)
SPECIFIC WEIGHT - lb/ft³
110
100
90
80
70
60
50
-100
0
+100
+200
+300
TEMPERATURE - °F
RESTRICTED
RESTRICTED
CONFIDENTIAL

CONFIDENTIAL RESTRICTED



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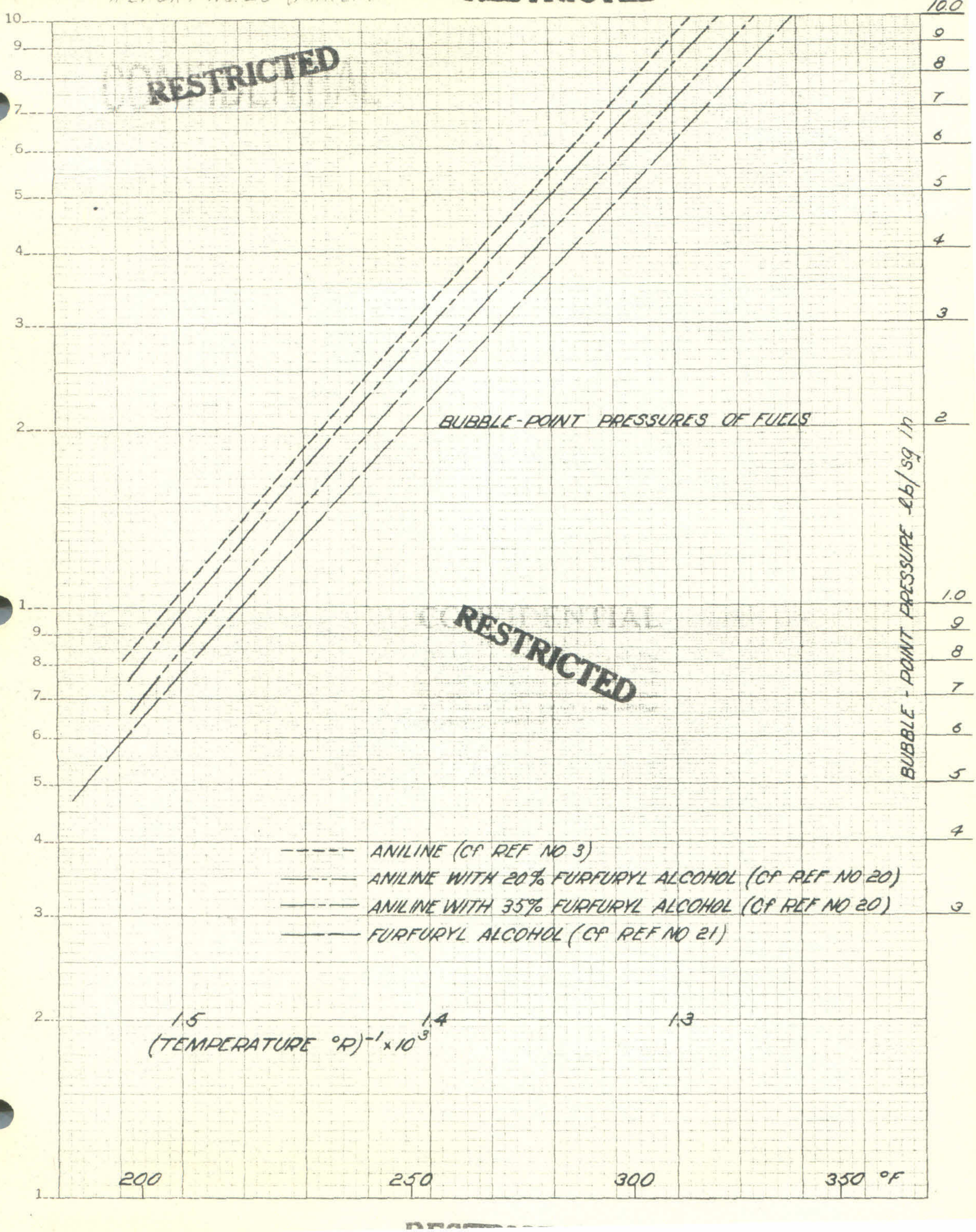
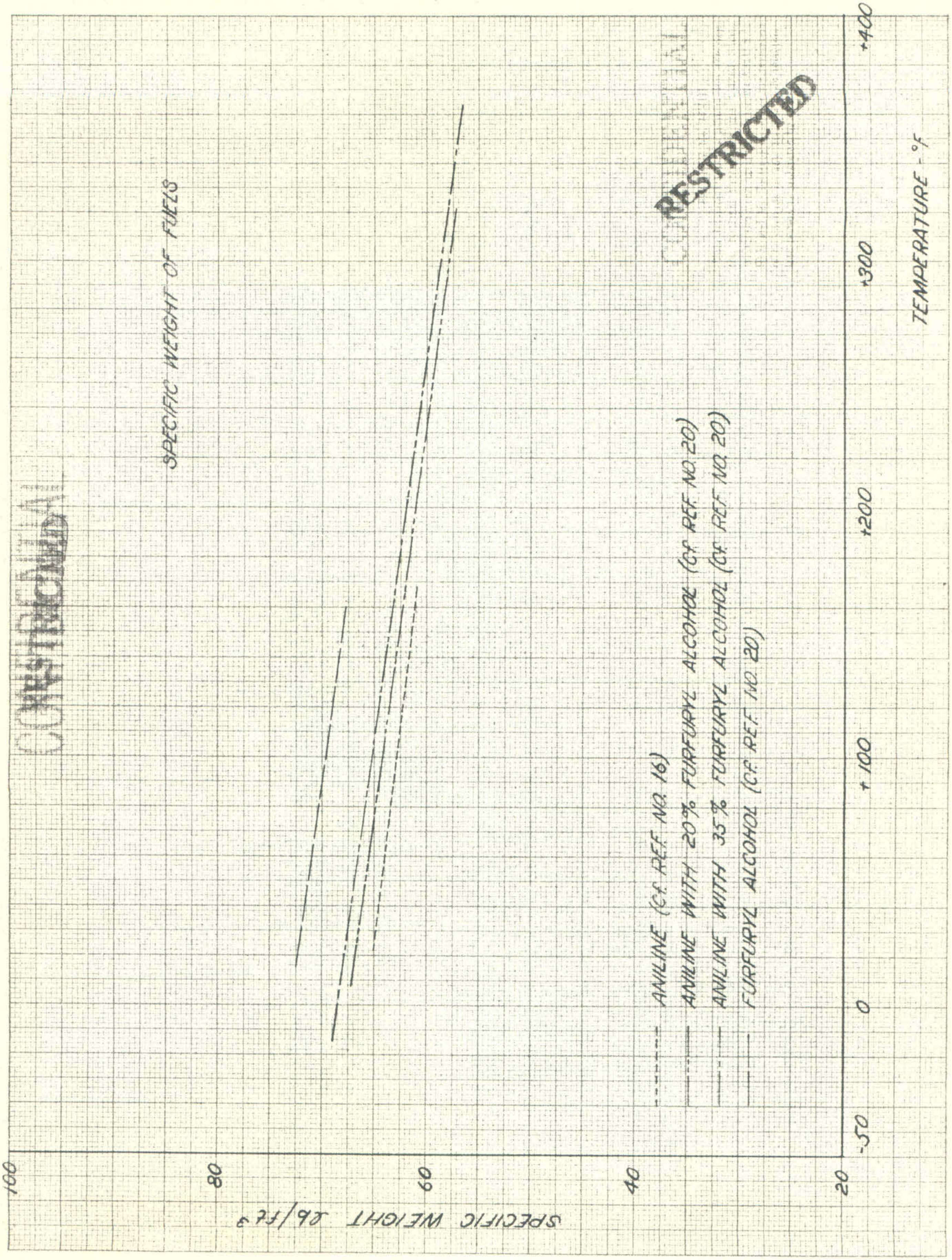
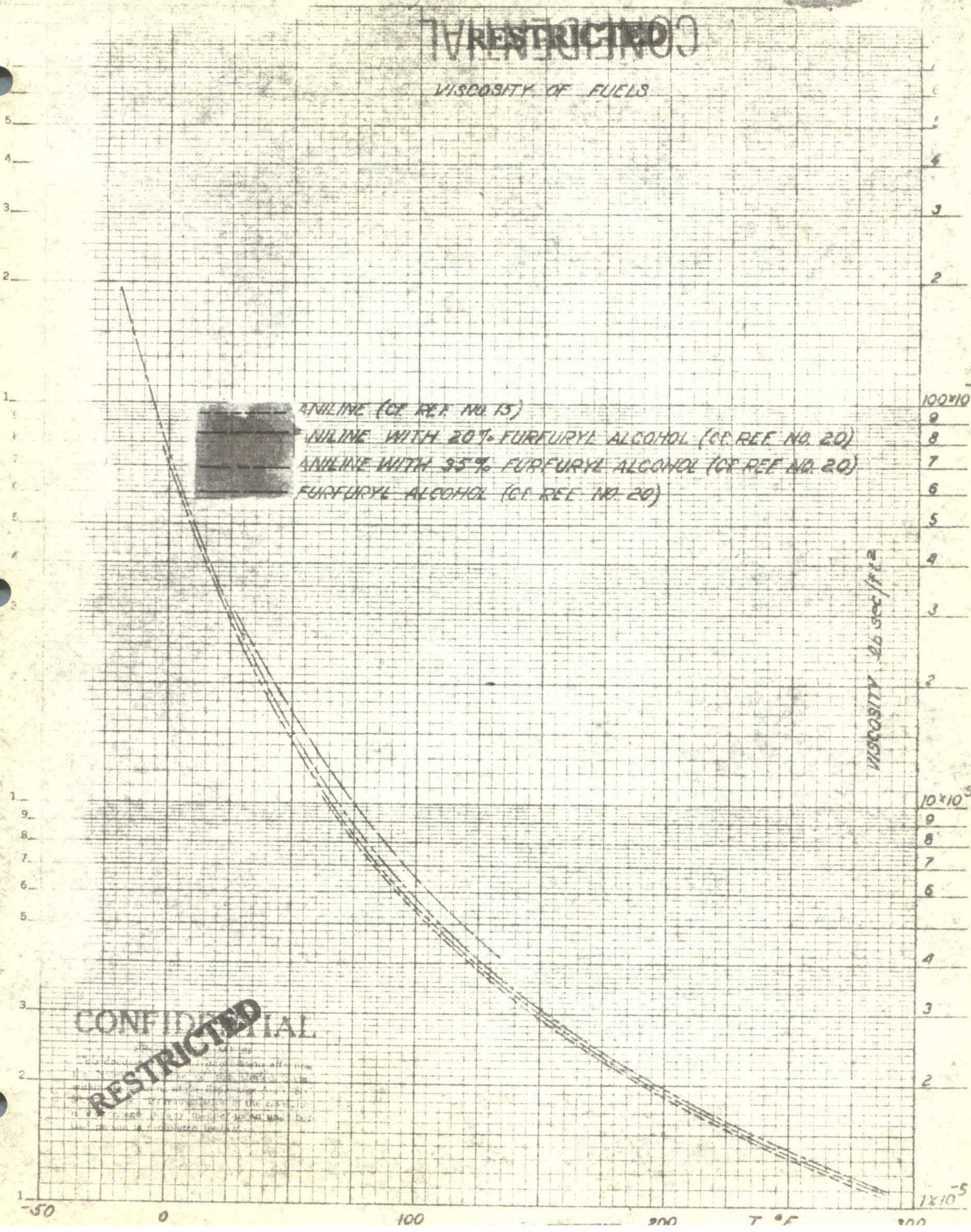

RESTRICTED
BUBBLE-POINT PRESSURES OF FUELS
RESTRICTED
ANILINE (Cf REF NO 3)
ANILINE WITH 20% FURFURYL ALCOHOL (Cf REF NO 20)
ANILINE WITH 35% FURFURYL ALCOHOL (Cf REF NO 20)
FURFURYL ALCOHOL (Cf REF NO 21)
BUBBLE-POINT PRESSURE lb/sq in
(TEMPERATURE °R)$^{-1}$ x 10^3
1.5
1.4
1.3
200
250
300
350 °F
10.0
1.0

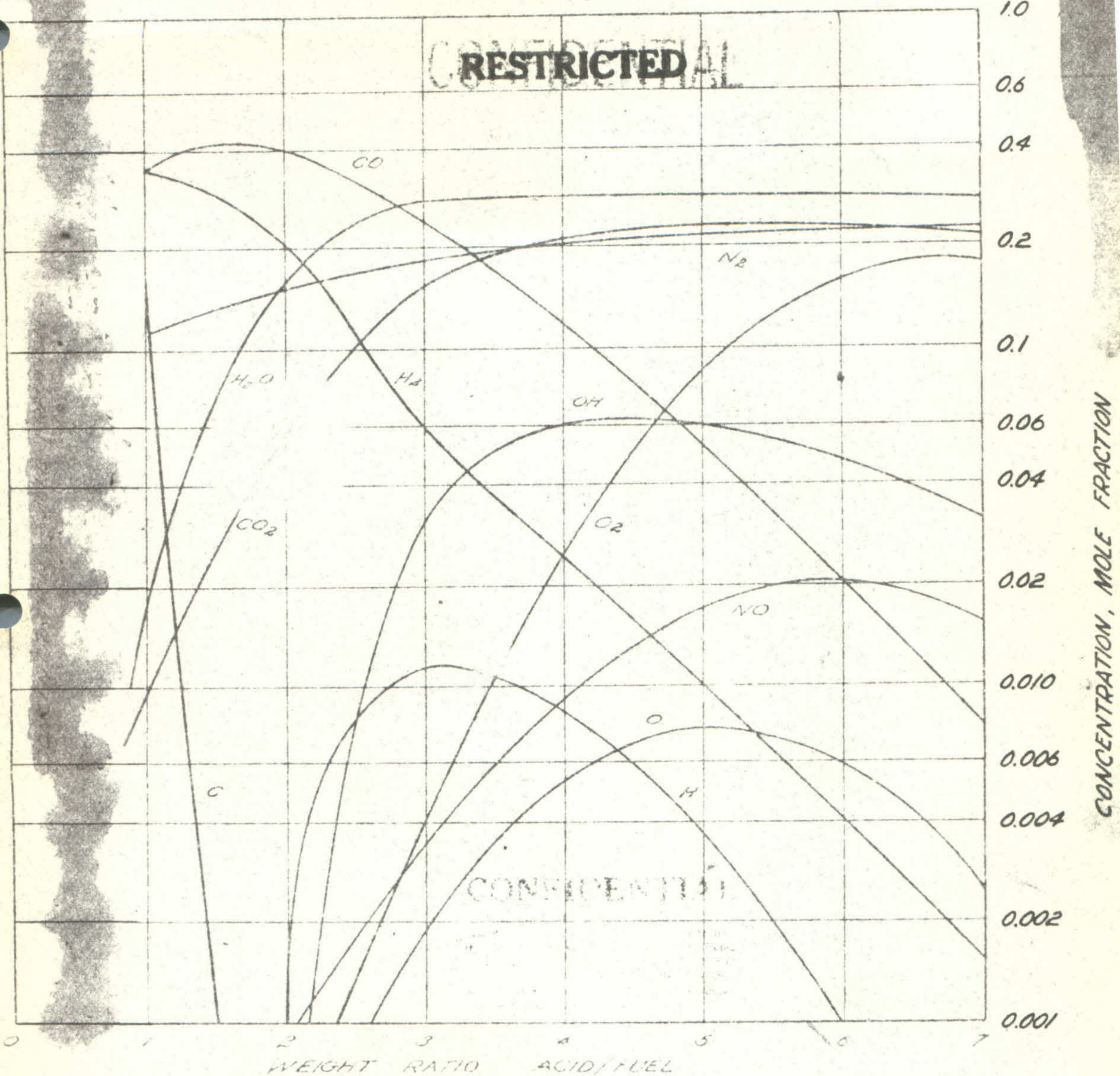
 RESTRICTED



RESTRICTED


RESTRICTED
VISCOSITY OF FUELS
ANILINE (CF. REF. NO. 13)
ANILINE WITH 20% FURFURYL ALCOHOL (CF. REF. NO. 20)
ANILINE WITH 35% FURFURYL ALCOHOL (CF. REF. NO. 20)
FURFURYL ALCOHOL (CF. REF. NO. 20)
VISCOSITY lb sec/ft²
100×10⁻⁵
10×10⁻⁵
1×10⁻⁵
-50
0
100
200
300
T °F
CONFIDENTIAL
RESTRICTED

RESTRICTED



CALCULATED EQUILIBRIUM COMPOSITION OF THE REACTION PRODUCTS OF ANILINE AND RED FUMING NITRIC ACID. (Cf REF 3)

RESTRICTED

RESTRICTED

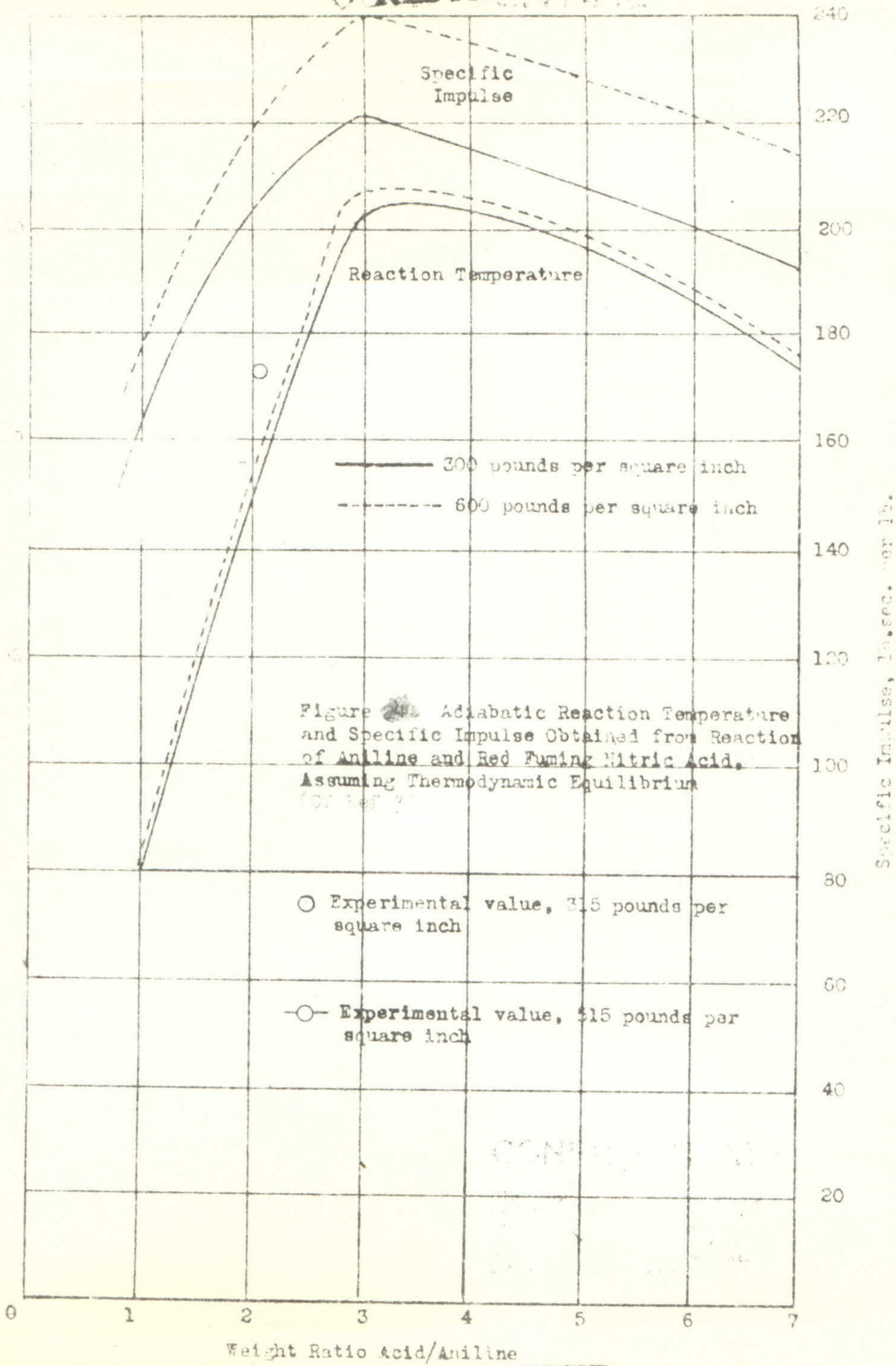
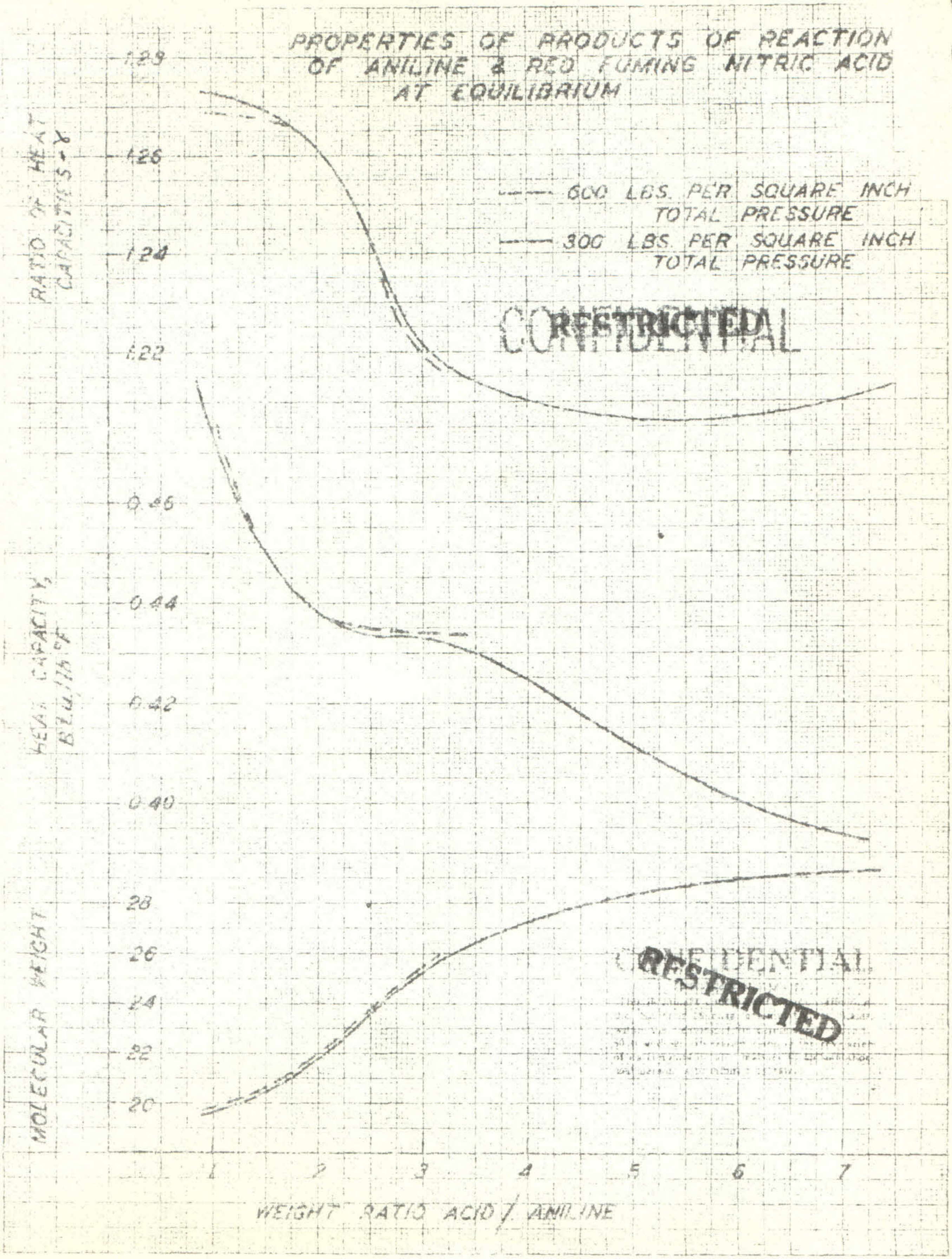
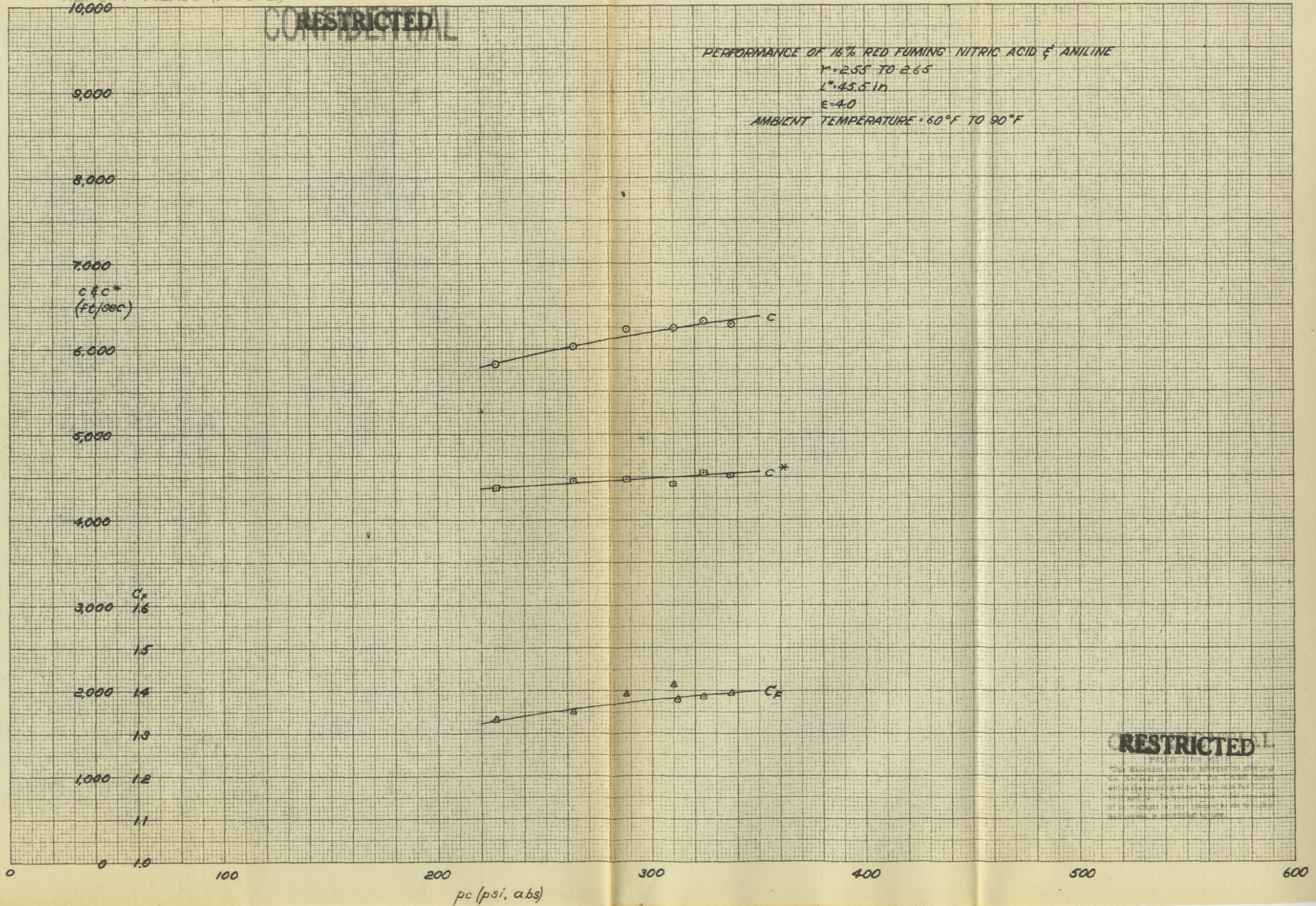
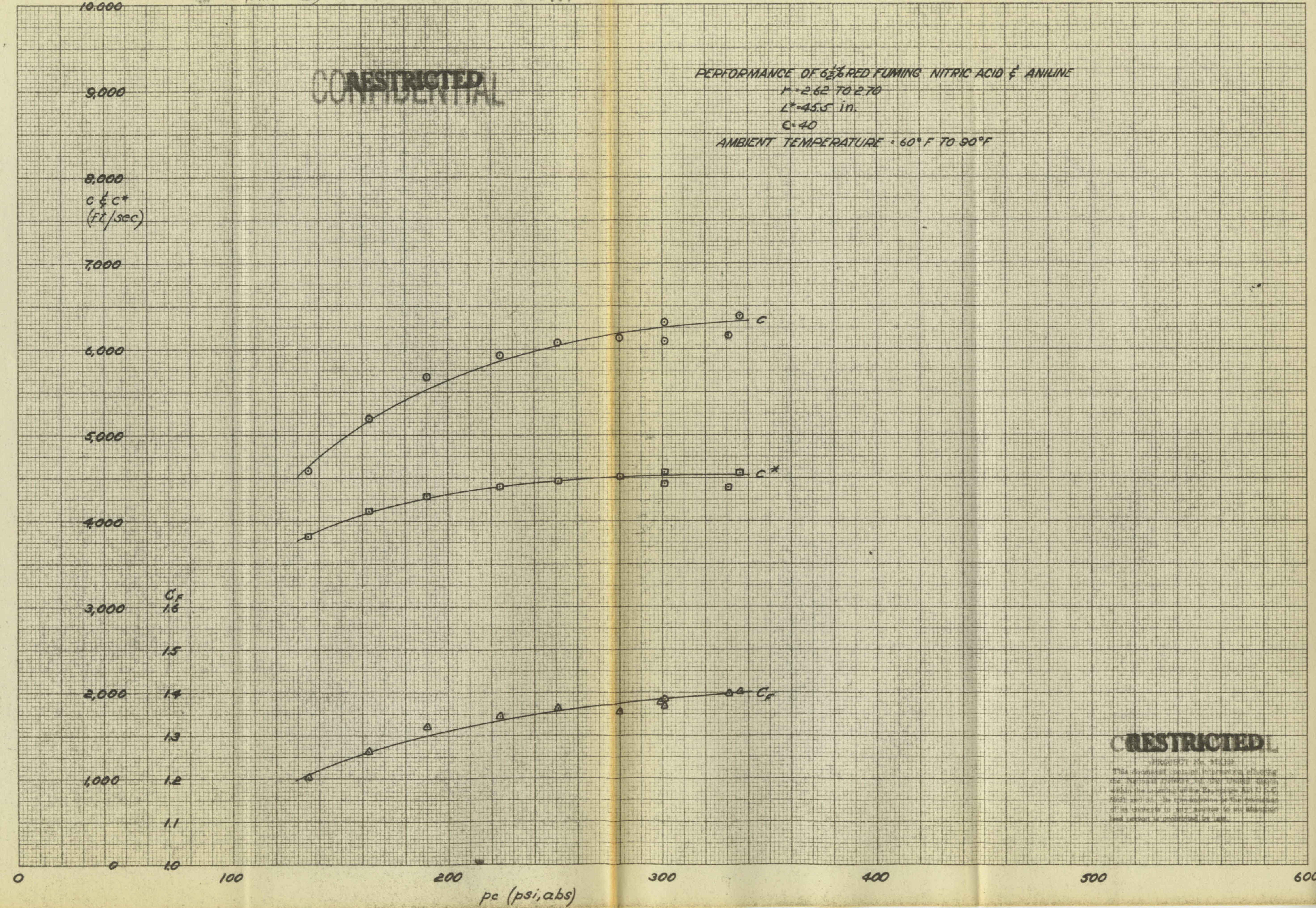


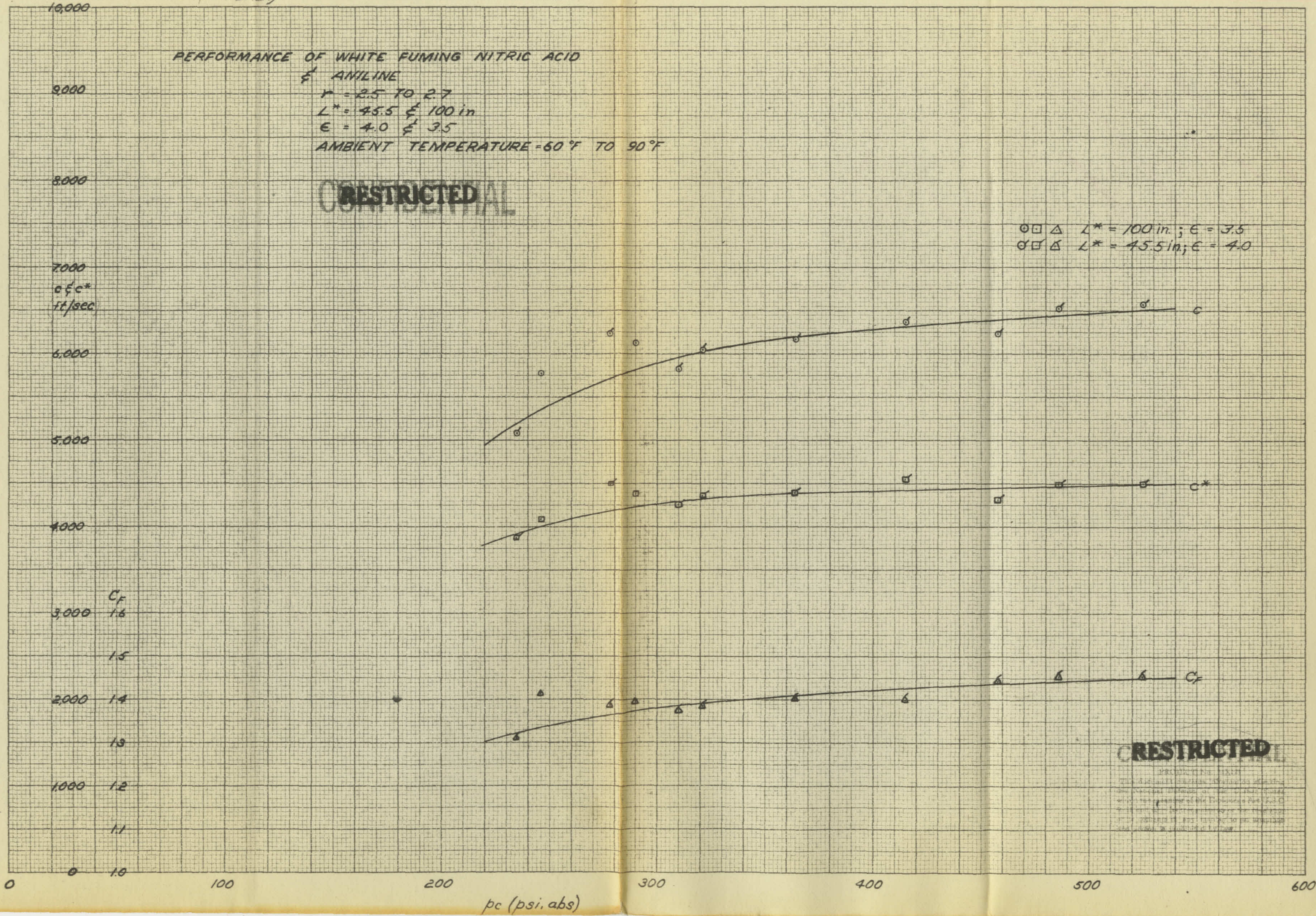
Figure Adiabatic Reaction Temperature and Specific Impulse Obtained from Reaction of Aniline and Red Fuming Nitric Acid. Assuming Thermodynamic Equilibrium


PROPERTIES OF PRODUCTS OF REACTION
OF ANILINE & RED FUMING NITRIC ACID
AT EQUILIBRIUM
RATIO OF HEAT CAPACITIES - γ
1.28
1.26
1.24
1.22
HEAT CAPACITY, B.T.U./LB °F
0.46
0.44
0.42
0.40
MOLECULAR WEIGHT
28
26
24
22
20
1
2
3
4
5
6
7
WEIGHT RATIO ACID / ANILINE
600 LBS. PER SQUARE INCH TOTAL PRESSURE
300 LBS. PER SQUARE INCH TOTAL PRESSURE
CONFIDENTIAL
RESTRICTED
CONFIDENTIAL
RESTRICTED


RESTRICTED
PERFORMANCE OF 16% RED FUMING NITRIC ACID & ANILINE
r = 2.55 TO 2.65
L* = 45.5 in
ε = 4.0
AMBIENT TEMPERATURE = 60°F TO 90°F
c & c*
(ft/sec)
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
C_F
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
C_F
0
100
200
300
400
500
600
pc (psi, abs)
RESTRICTED


RESTRICTED
PERFORMANCE OF 6½% RED FUMING NITRIC ACID & ANILINE
r = 2.62 TO 2.70
L* = 45.5 in.
ε = 40
AMBIENT TEMPERATURE = 60° F TO 90°F
c & c* (ft/sec)
C_F
C
C*
C_F
pc (psi, abs)
RESTRICTED

RESTRICTED


PERFORMANCE OF WHITE FUMING NITRIC ACID
& ANILINE
r = 2.5 TO 2.7
L* = 45.5 & 100 in
ε = 4.0 & 3.5
AMBIENT TEMPERATURE = 60°F TO 90°F
RESTRICTED
L* = 100 in. ; ε = 3.5
L* = 45.5 in; ε = 4.0
c & c*
ft/sec
10,000
9,000
8,000
7,000
6,000
5,000
4000
3,000
2,000
1,000
0
CF
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
CF
0
100
200
300
400
500
600
pc (psi. abs)
RESTRICTED

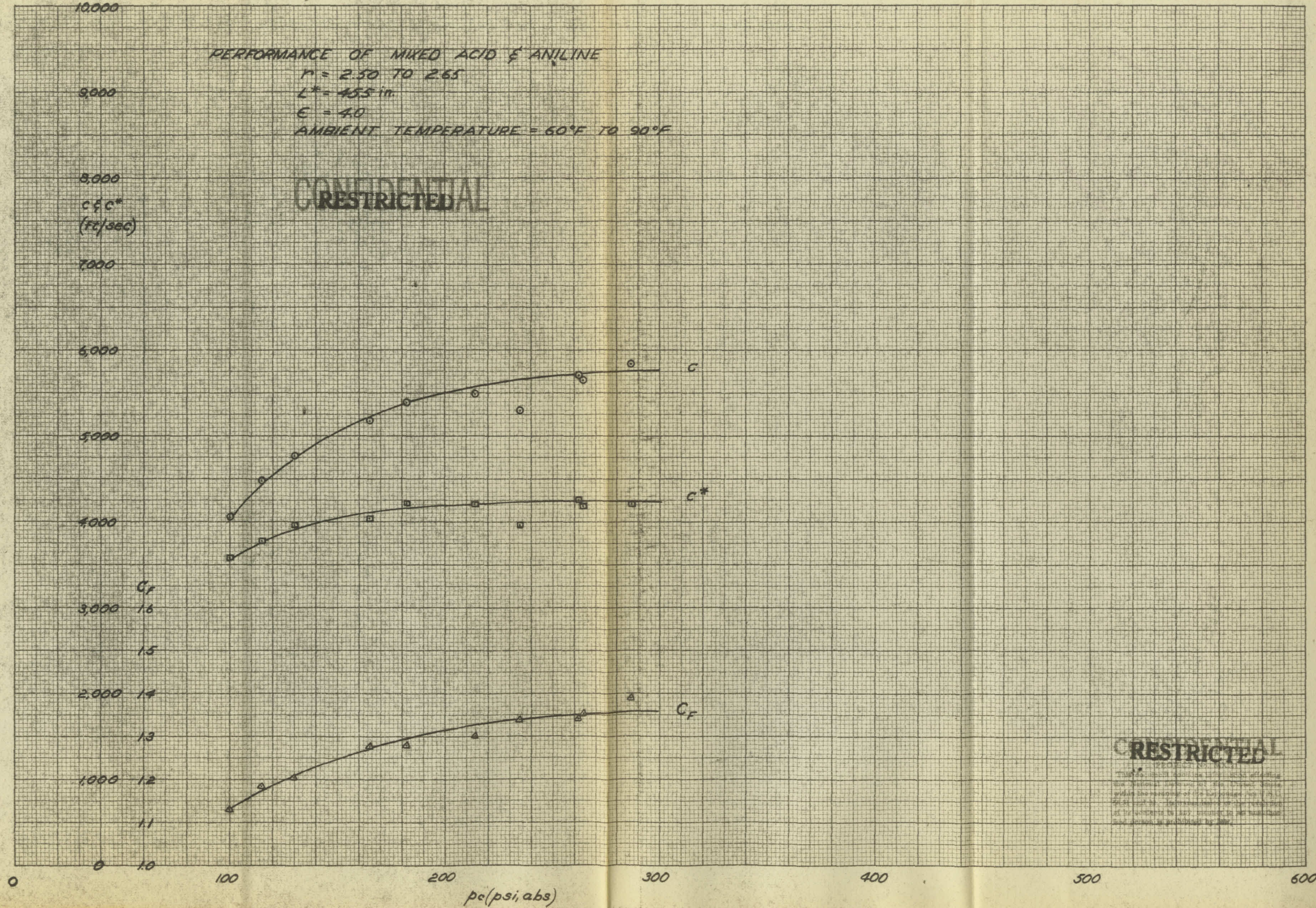
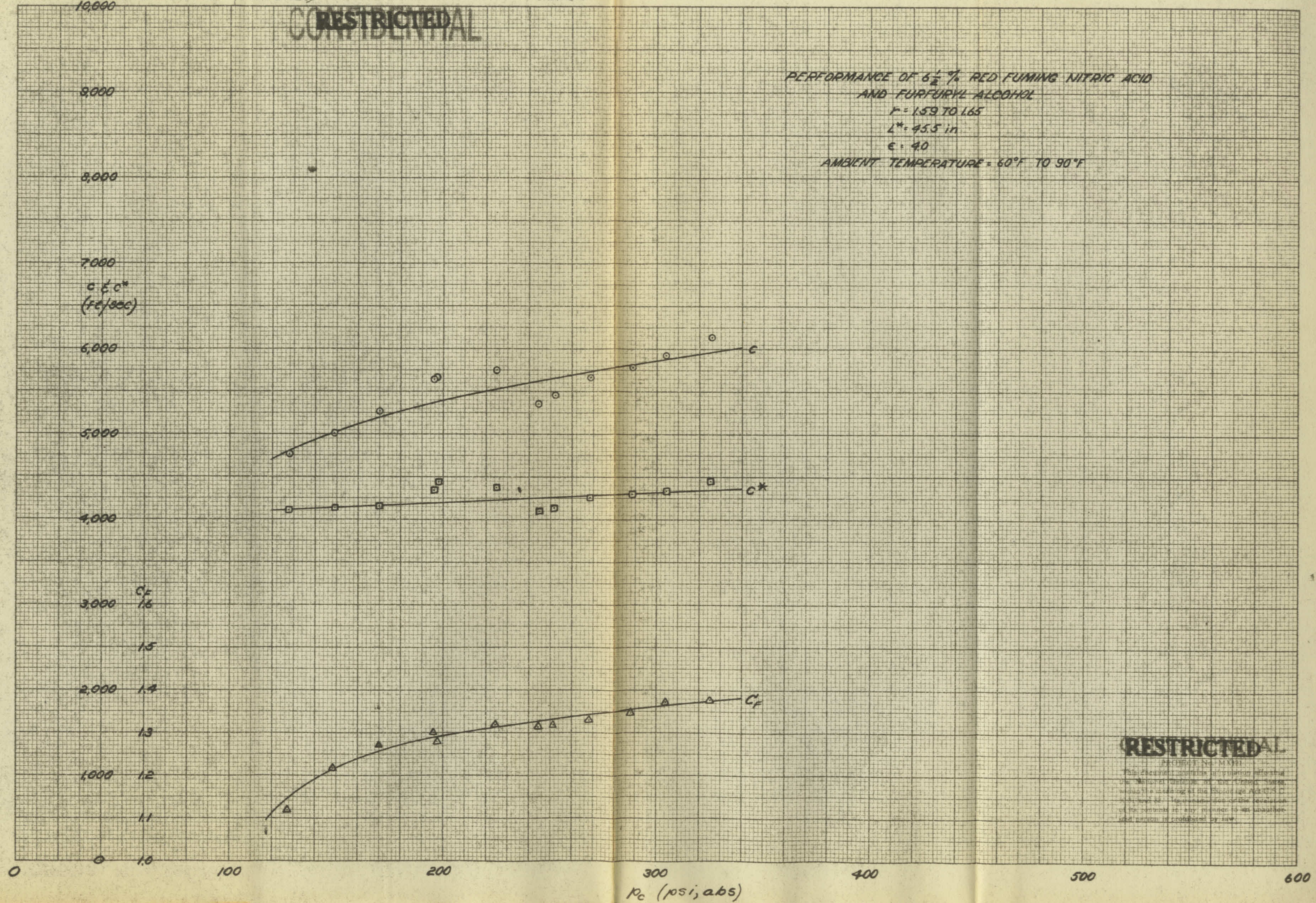
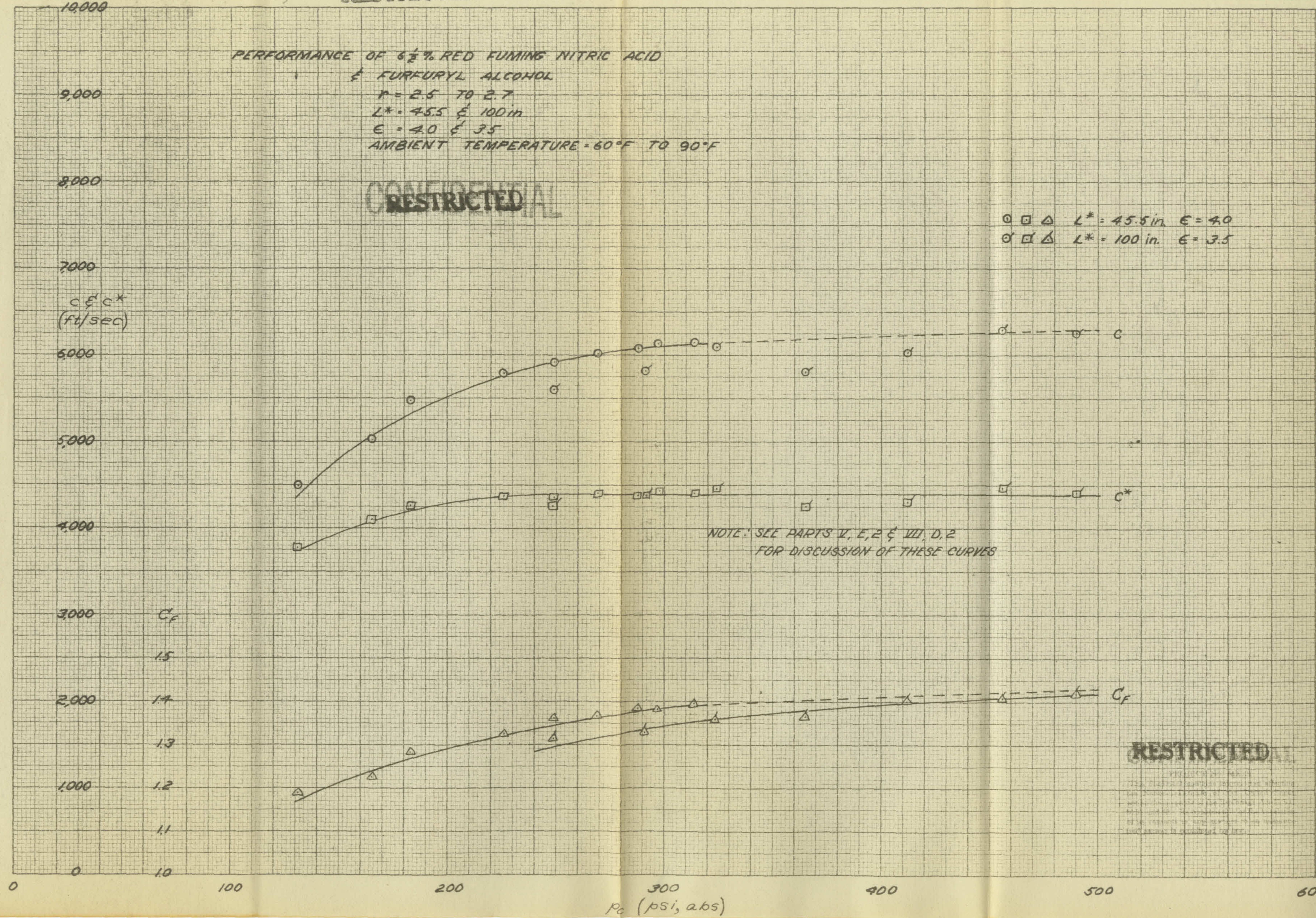
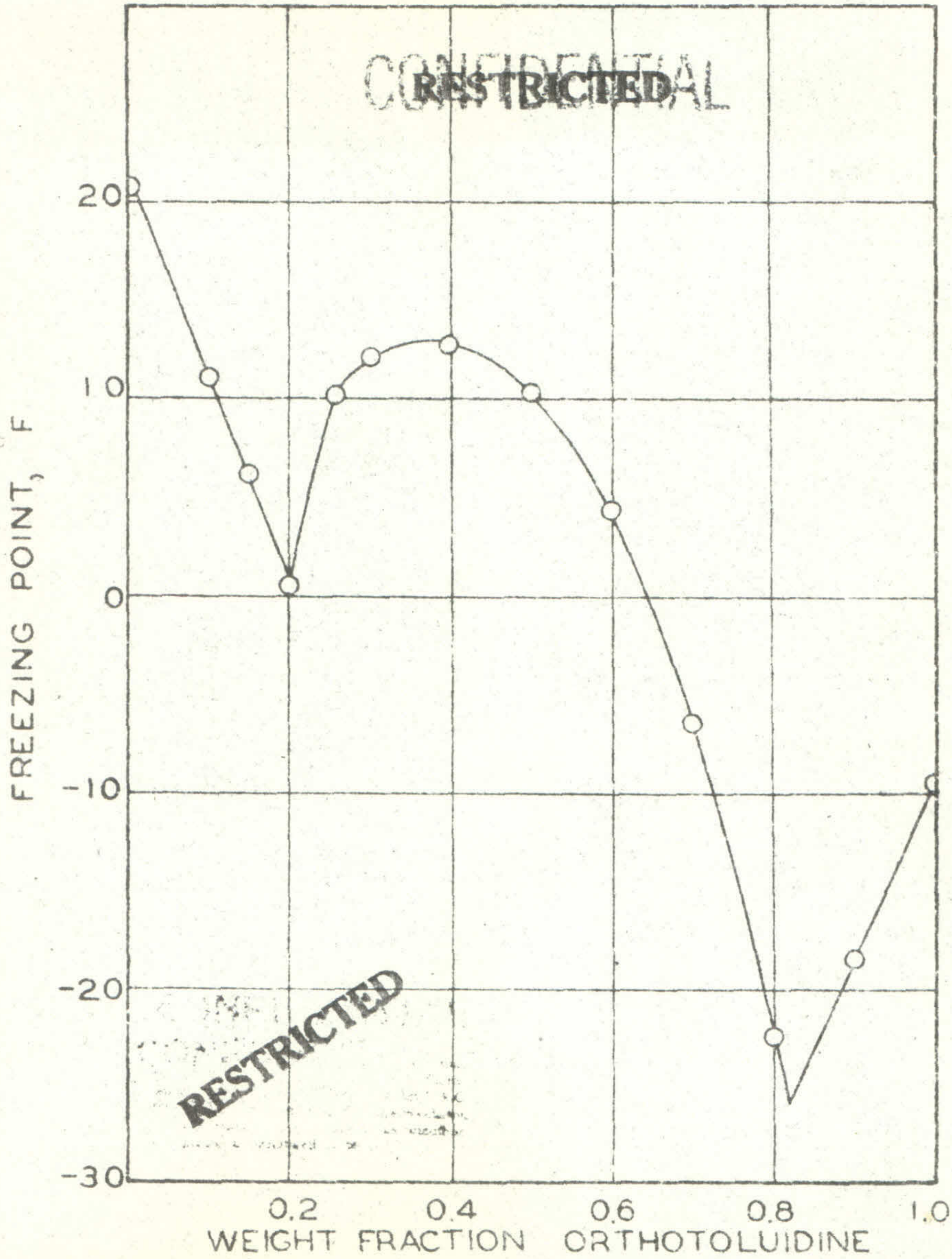

PERFORMANCE OF MIXED ACID & ANILINE
r = 2.50 TO 2.65
L* = 45.5 in.
ϵ = 4.0
AMBIENT TEMPERATURE = 60°F TO 90°F
c & c*
(ft/sec)
C_F
10,000
9,000
8,000
7000
6,000
5,000
4,000
3,000
2,000
1,000
0
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
C_F
0
100
200
300
400
500
600
pc(psi, abs)

FIG. 30


CONFIDENTIAL RESTRICTED
PERFORMANCE OF 6½ % RED FUMING NITRIC ACID
AND FURFURYL ALCOHOL
r = 1.59 TO 1.65
L* = 45.5 in
ε = 4.0
AMBIENT TEMPERATURE = 60°F TO 90°F
c & c*
(ft/sec)
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
C_F
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
C_F
0
100
200
300
400
500
600
p_c (psi, abs)
RESTRICTED

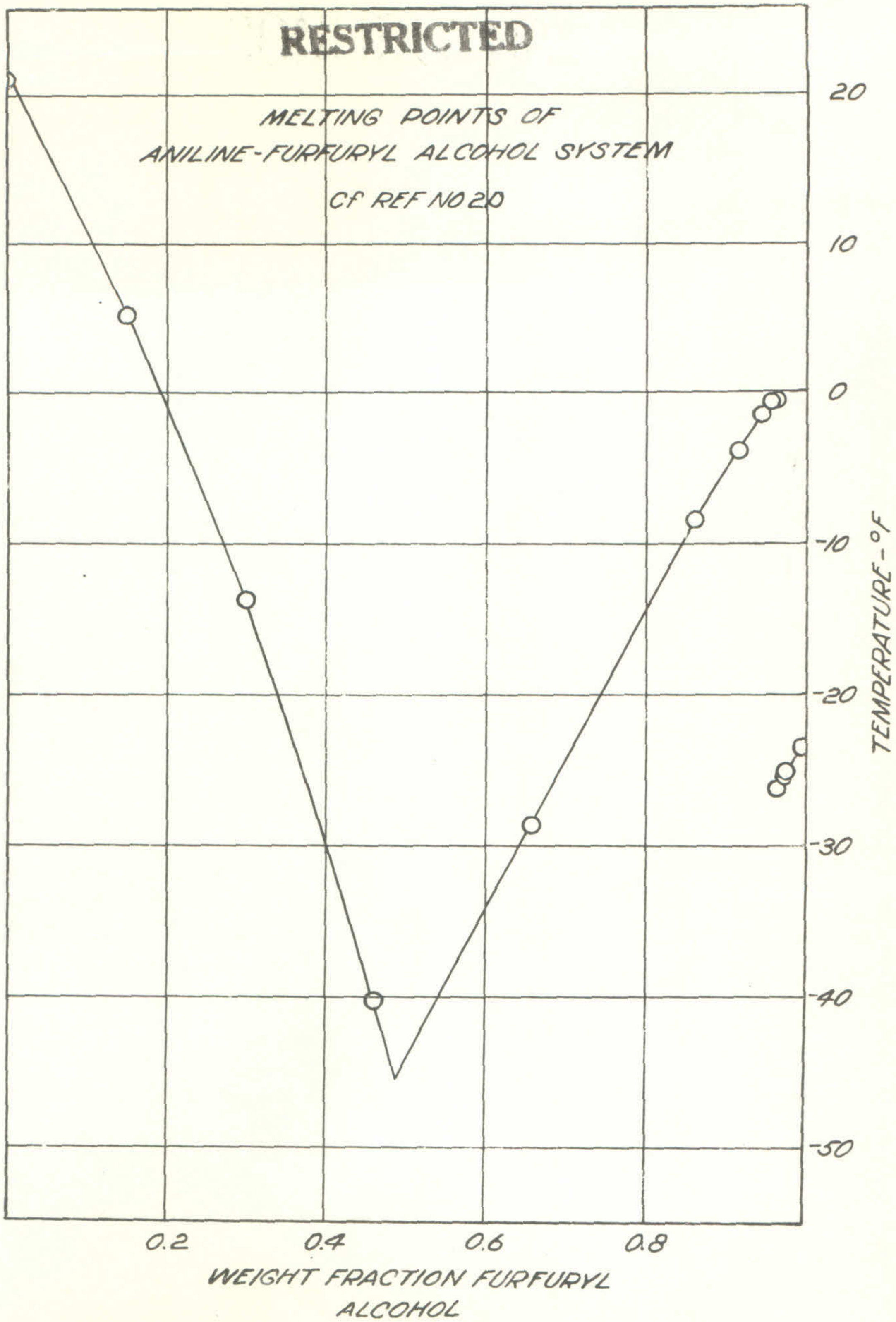

PERFORMANCE OF 6½% RED FUMING NITRIC ACID
& FURFURYL ALCOHOL
r = 2.5 TO 2.7
L* = 45.5 & 100 in
ε = 4.0 & 3.5
AMBIENT TEMPERATURE = 60°F TO 90°F
RESTRICTED
L* = 45.5 in. ε = 4.0
L* = 100 in. ε = 3.5
c & c*
(ft/sec)
10,000
9,000
8,000
7000
6,000
5,000
4,000
3000
2,000
1,000
0
C_F
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
C_F
NOTE: SEE PARTS V, E, 2 & VIII, D, 2
FOR DISCUSSION OF THESE CURVES
0
100
200
300
400
500
600
P_c (psi, abs)
RESTRICTED

RESTRICTED



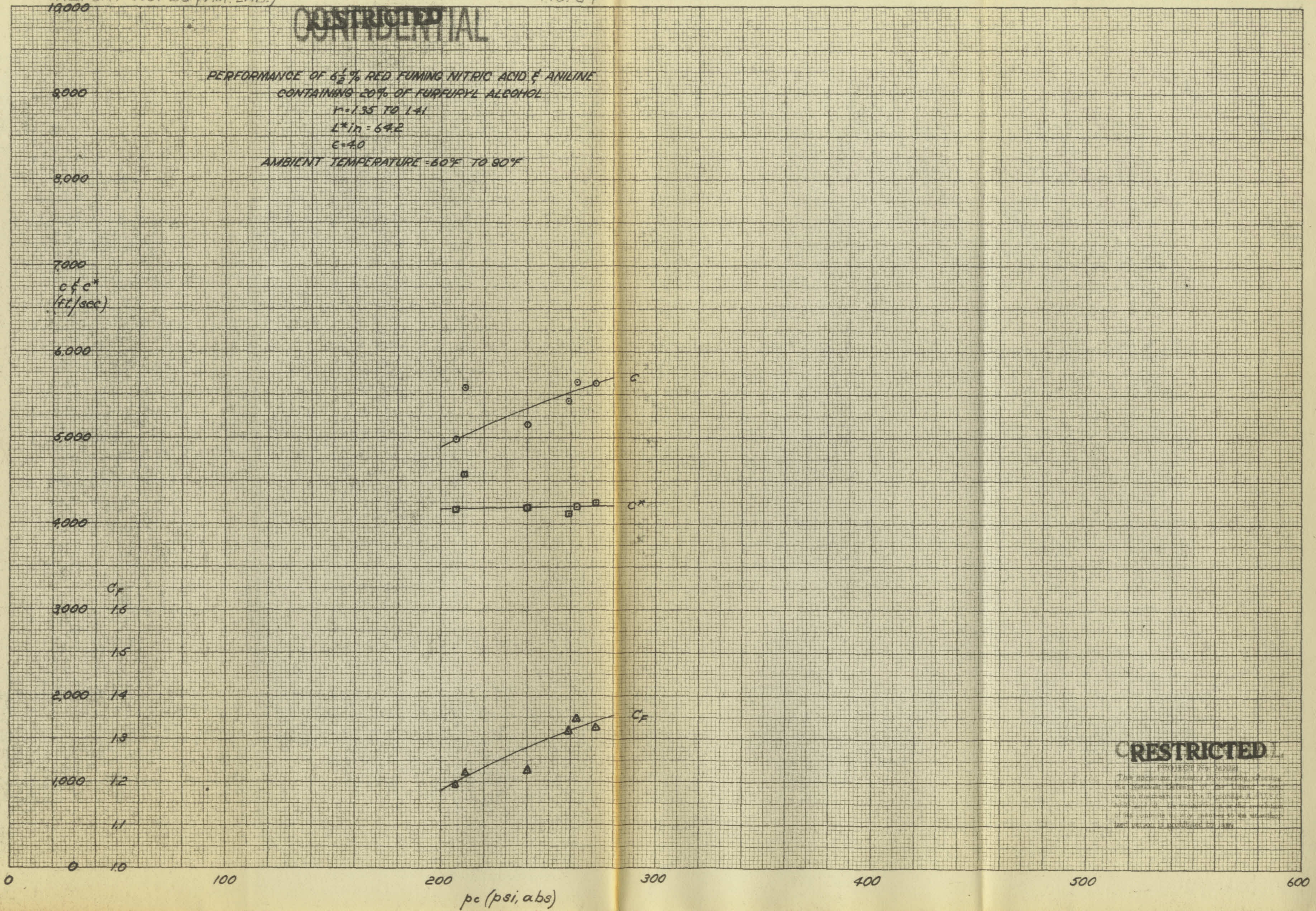
Freezing Points of the Aniline-Orthotoluidine System.
(CC Ref 3)

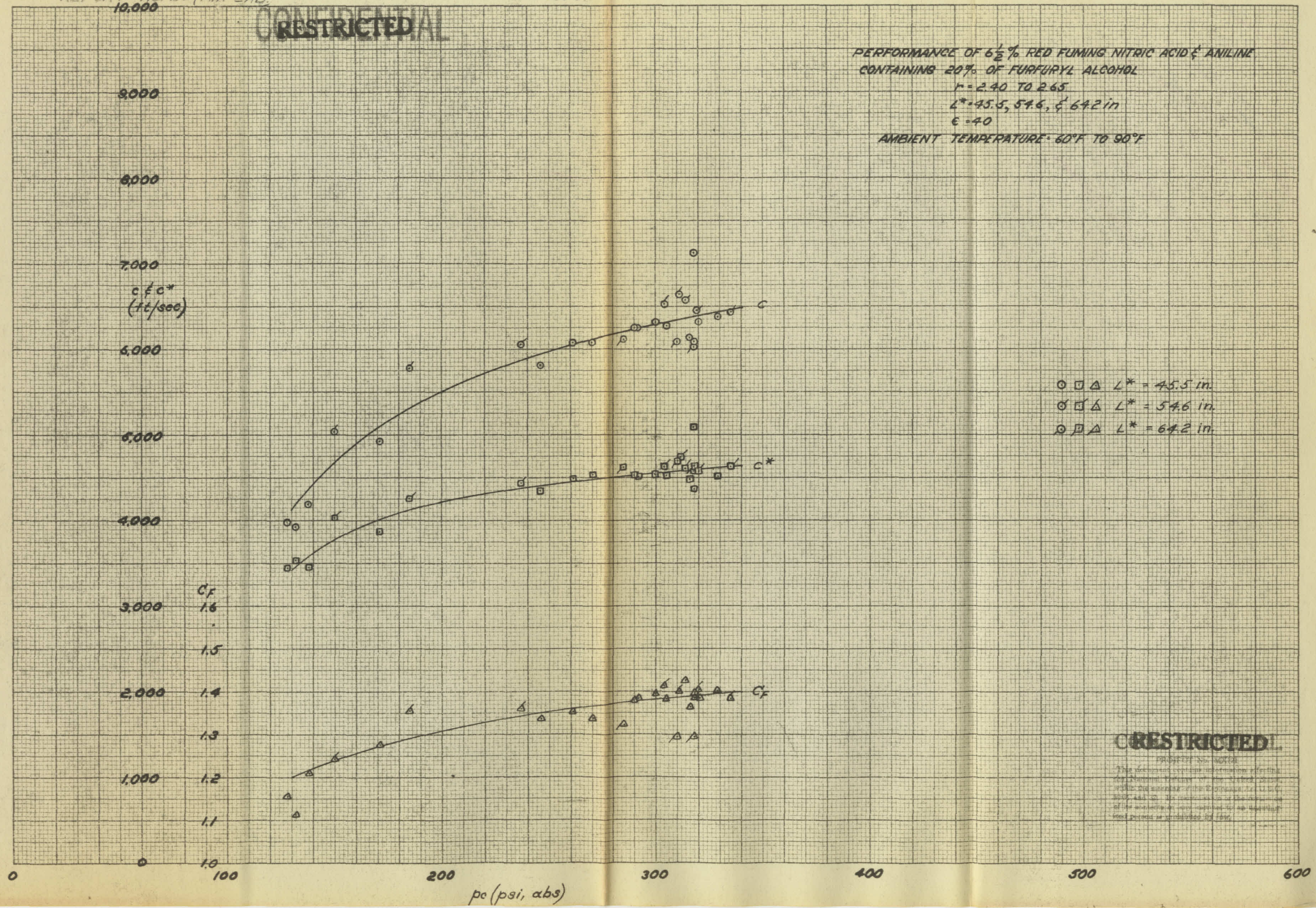
RESTRICTED

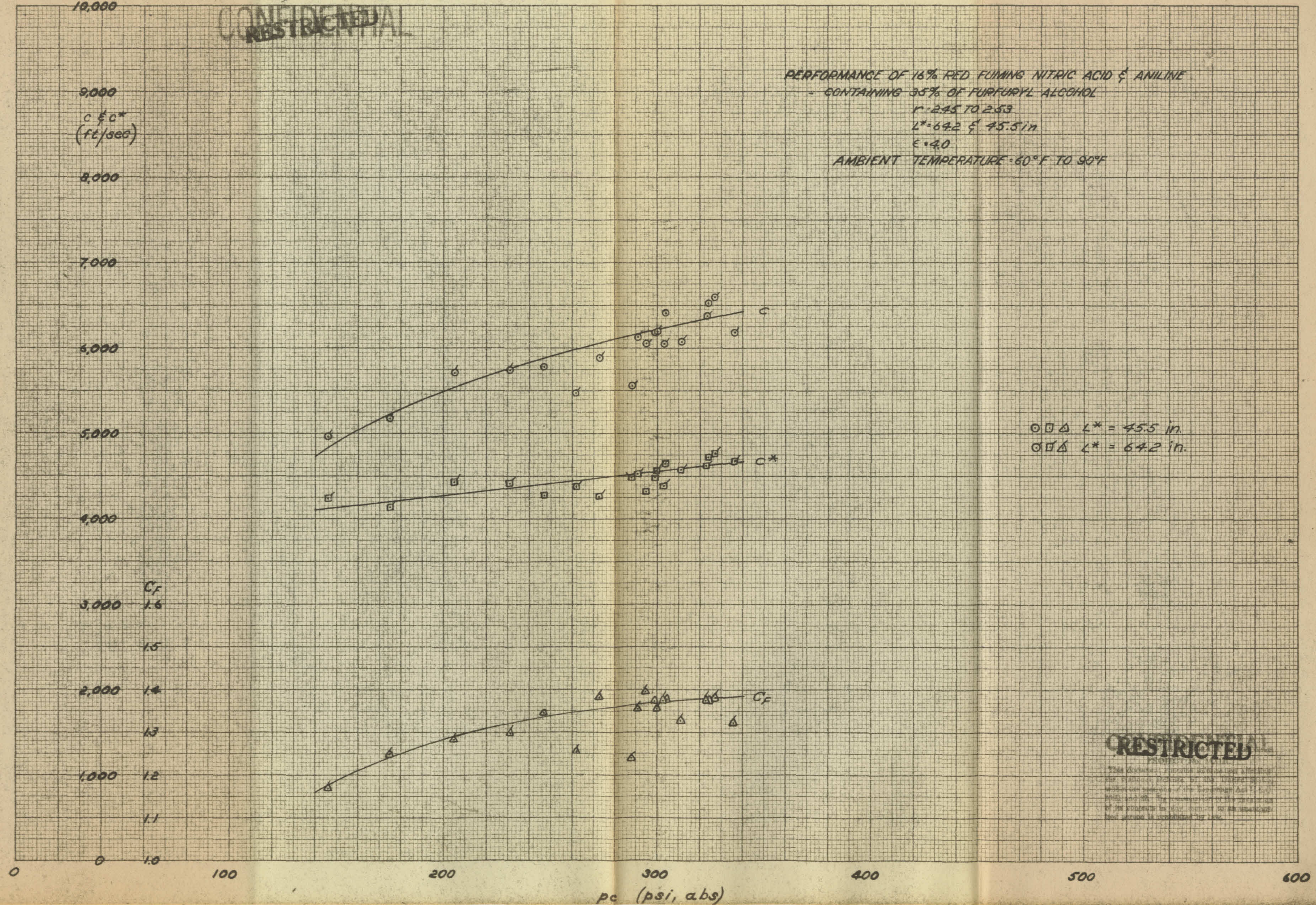


RESTRICTED

RESTRICTED


PERFORMANCE OF 6½% RED FUMING NITRIC ACID & ANILINE
CONTAINING 20% OF FURFURYL ALCOHOL
r = 1.35 TO 1.41
L*in = 64.2
ε = 4.0
AMBIENT TEMPERATURE = 60°F TO 90°F
c & c*
(ft/sec)
C_F
c
c*
C_F
pc (psi, abs)
0
100
200
300
400
500
600
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3000
2,000
1,000
1.6
1.5
1.4
1.3
1.2
1.1
1.0


PERFORMANCE OF 6½% RED FUMING NITRIC ACID & ANILINE
CONTAINING 20% OF FURFURYL ALCOHOL
r = 2.40 TO 2.65
L* = 45.5, 54.6, & 64.2 in
ε = 40
AMBIENT TEMPERATURE = 60°F TO 90°F
c & c* (ft/sec)
C_F
C
C*
C_F
L* = 45.5 in.
L* = 54.6 in.
L* = 64.2 in.
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
1.6
1.5
1.4
1.3
1.2
1.1
1.0
0
100
200
300
400
500
600
pc (psi, abs)


PERFORMANCE OF 16% RED FUMING NITRIC ACID & ANILINE - CONTAINING 35% OF FURFURYL ALCOHOL
r = 2.45 TO 2.53
L* = 64.2 & 45.5 in
ε = 4.0
AMBIENT TEMPERATURE = 60°F TO 90°F
L* = 45.5 in.
L* = 64.2 in.
c & c* (ft/sec)
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
C_F
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
C_F
0
100
200
300
400
500
600
pc (psi, abs)

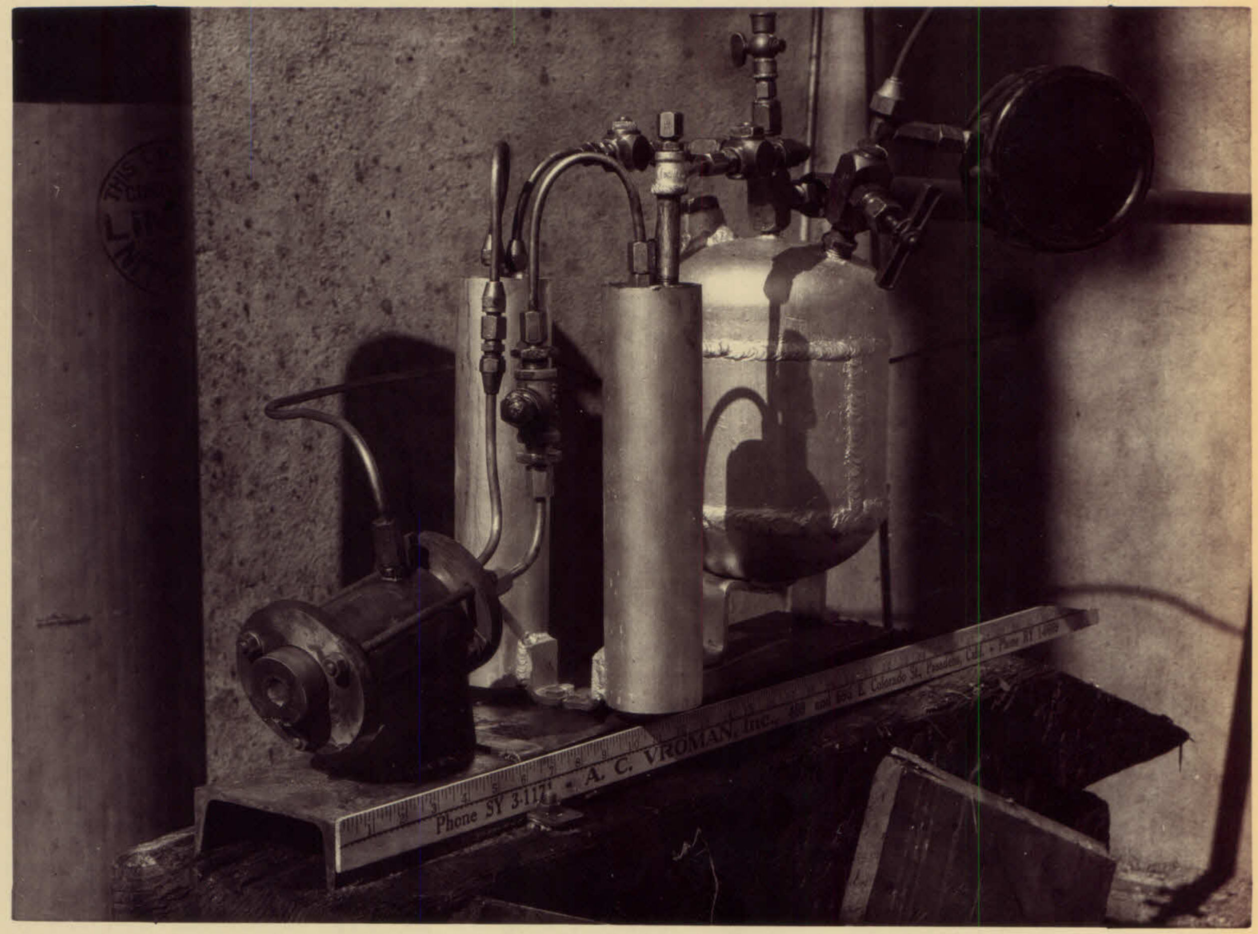


FIGURE 37. VIEW OF 50 LB. THRUST TEST MOTOR.

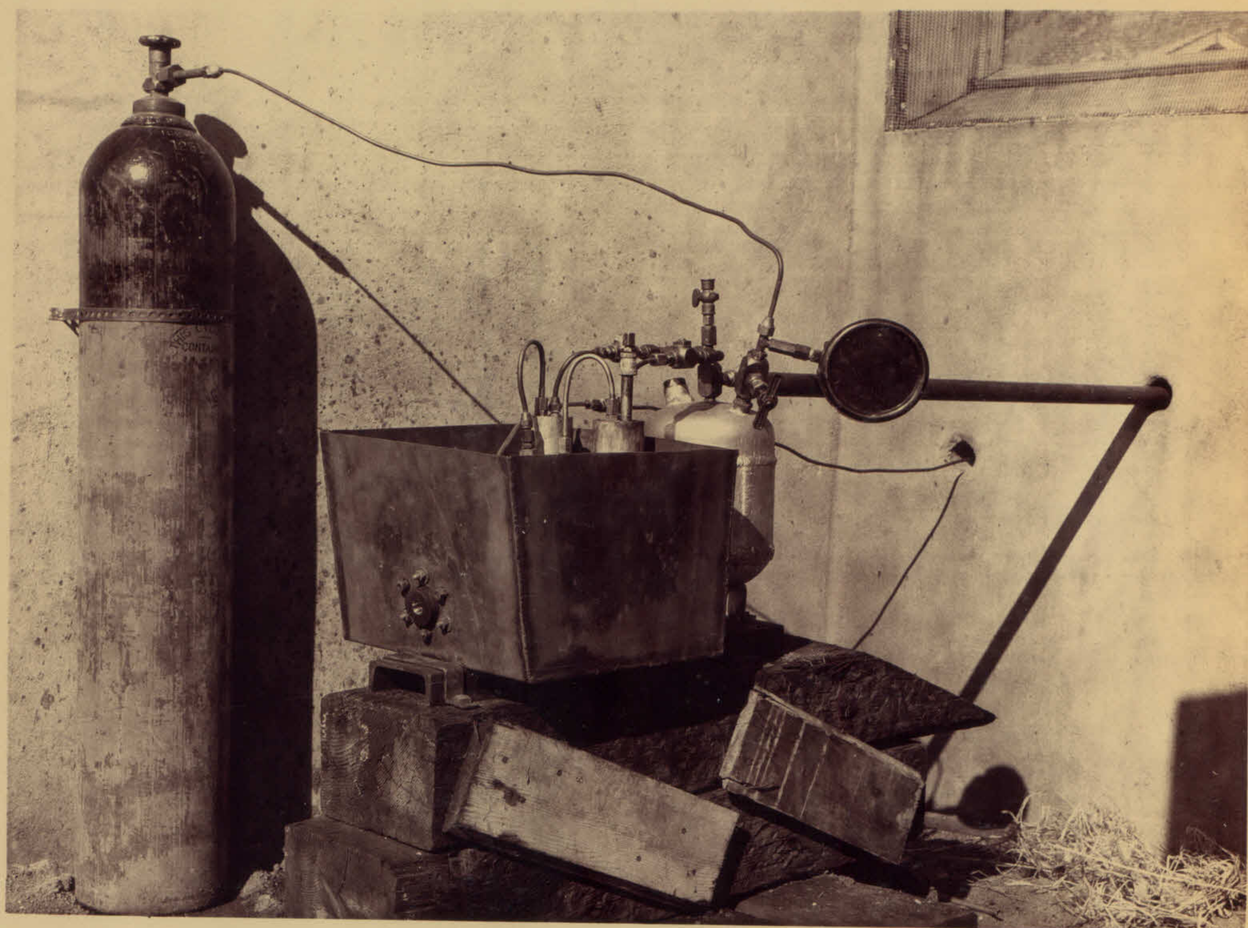


FIGURE 38. THE 50 LB. THRUST TEST MOTOR ENCLOSED IN THE FREEZING TANK.

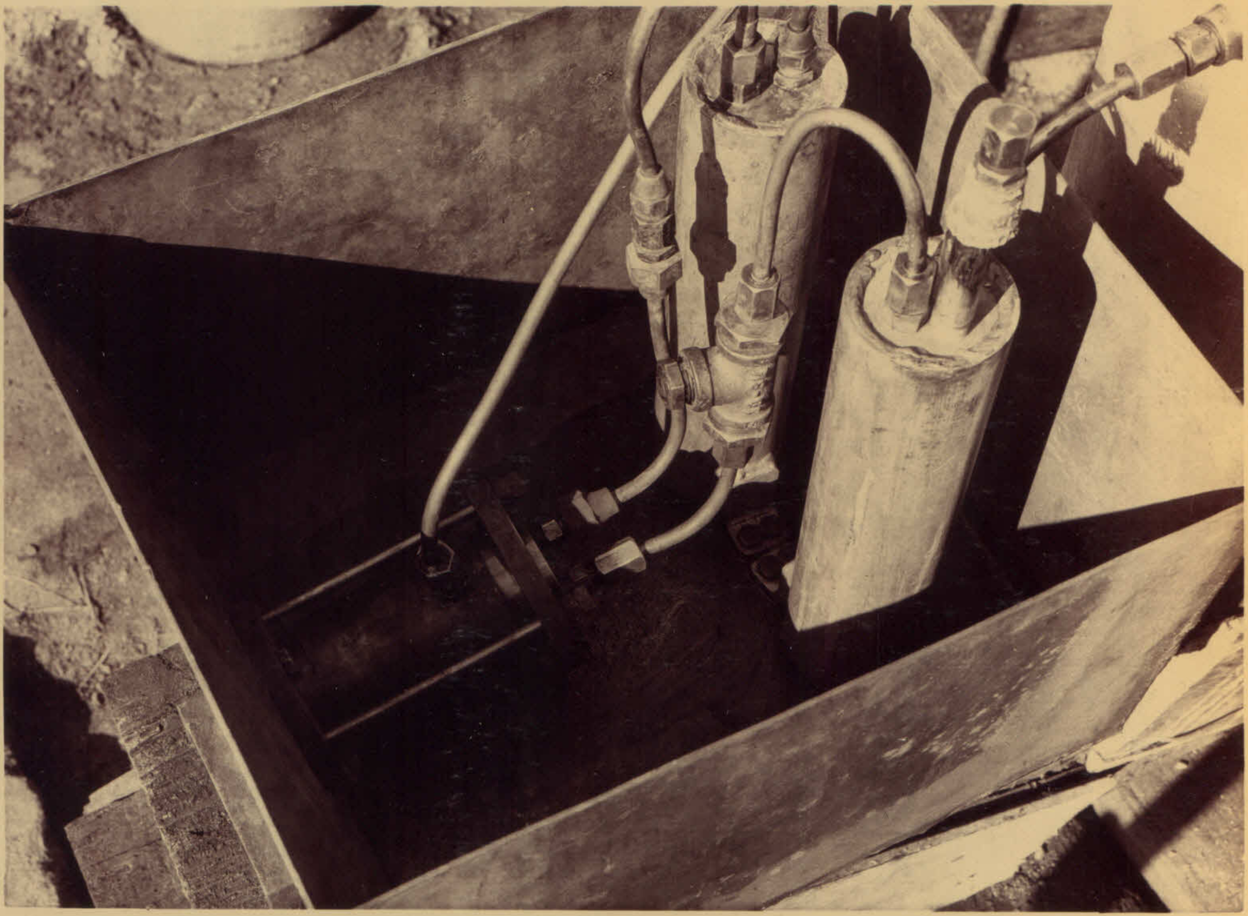


FIGURE 39. TOP VIEW OF THE 50 LB. THRUST TEST MOTOR IN THE FREEZING TANK.

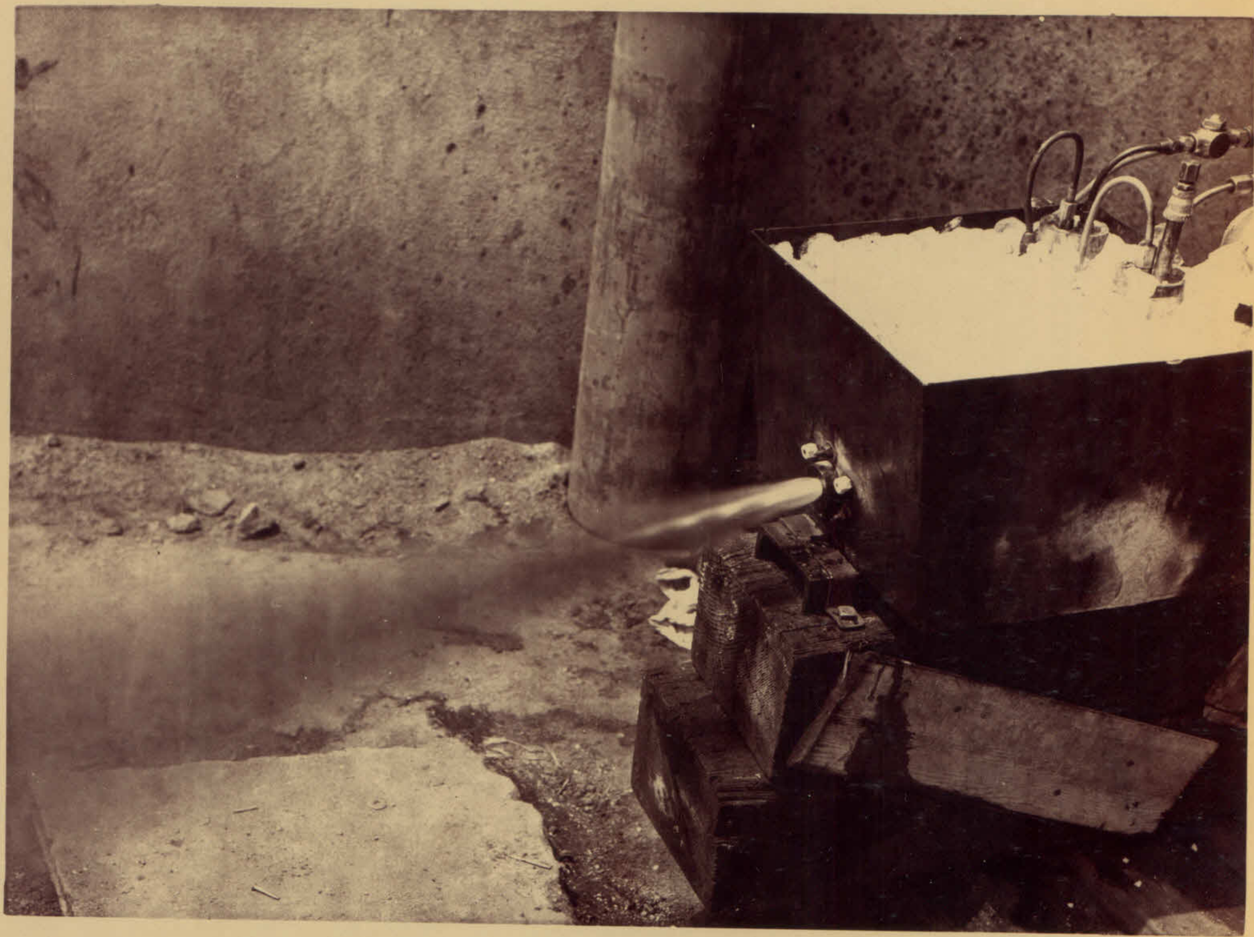
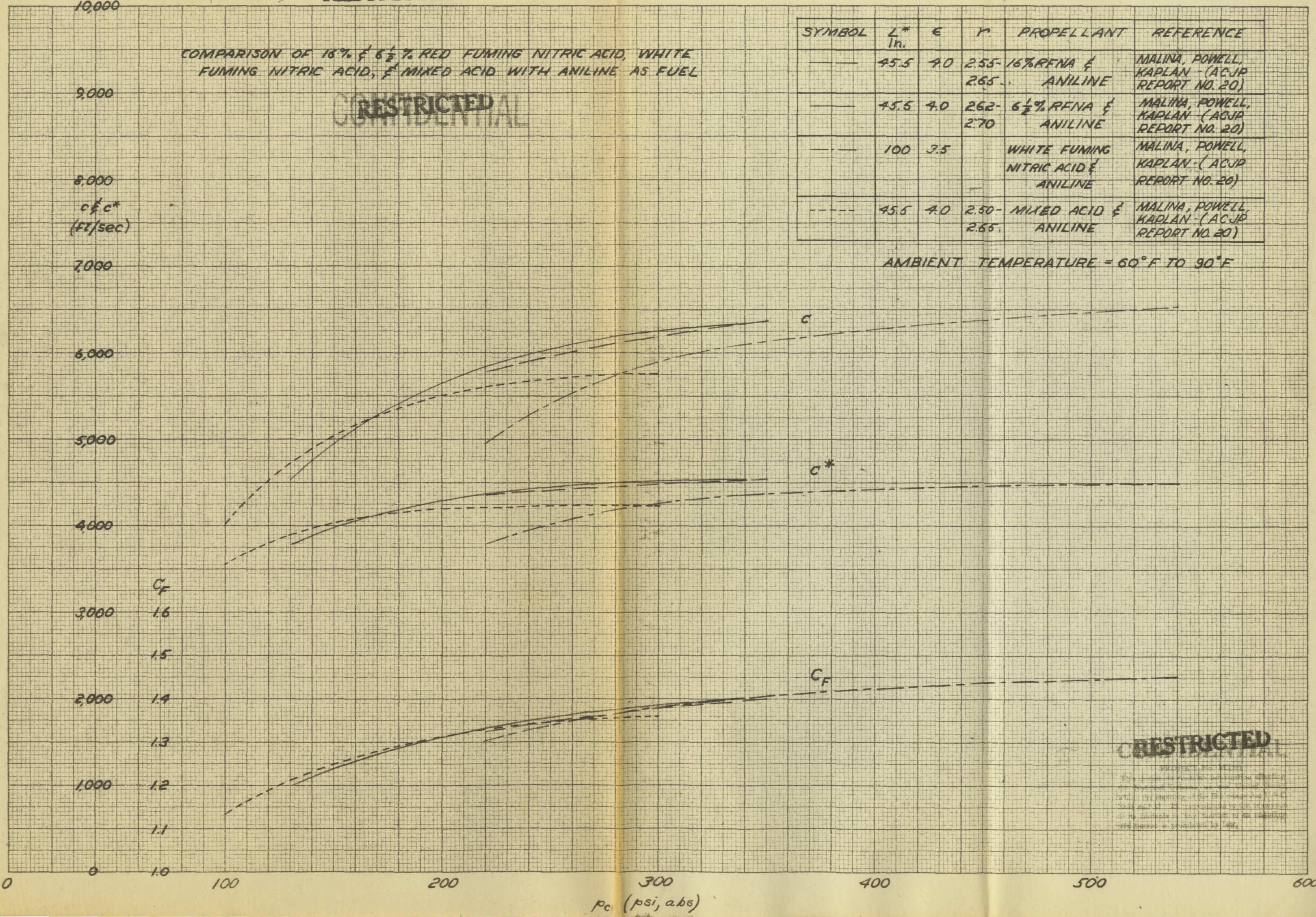
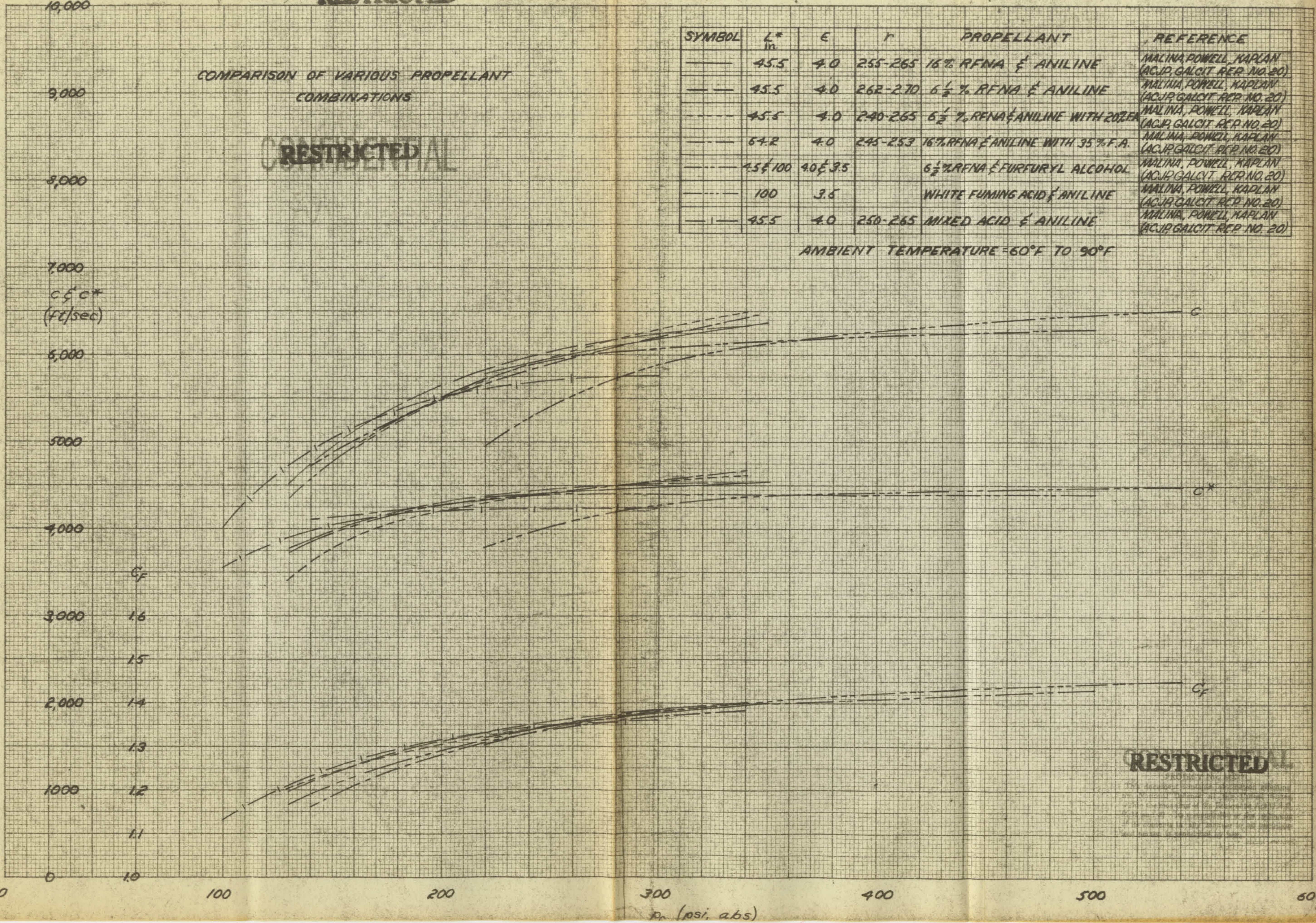


FIGURE 40. LOW TEMPERATURE IGNITION TEST WITH 50 LB. THRUST TEST MOTOR.


COMPARISON OF 16% & 6 1/2 % RED FUMING NITRIC ACID, WHITE FUMING NITRIC ACID, & MIXED ACID WITH ANILINE AS FUEL
RESTRICTED
SYMBOL | L* in. | ε | r | PROPELLANT | REFERENCE
45.5 | 4.0 | 2.55-2.65 | 16% RFNA & ANILINE | MALINA, POWELL, KAPLAN - (ACJP REPORT NO. 20)
45.5 | 4.0 | 2.62-2.70 | 6 1/2 % RFNA & ANILINE | MALINA, POWELL, KAPLAN - (ACJP REPORT NO. 20)
100 | 3.5 | | WHITE FUMING NITRIC ACID & ANILINE | MALINA, POWELL, KAPLAN - (ACJP REPORT NO. 20)
45.5 | 4.0 | 2.50-2.65 | MIXED ACID & ANILINE | MALINA, POWELL, KAPLAN - (ACJP REPORT NO. 20)
AMBIENT TEMPERATURE = 60°F TO 90°F
c & c* (ft/sec)
10,000
9,000
8,000
7,000
6,000
5,000
4000
3,000
2,000
1,000
0
C_F
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c
c*
C_F
0
100
200
300
400
500
600
pc (psi, abs)
RESTRICTED

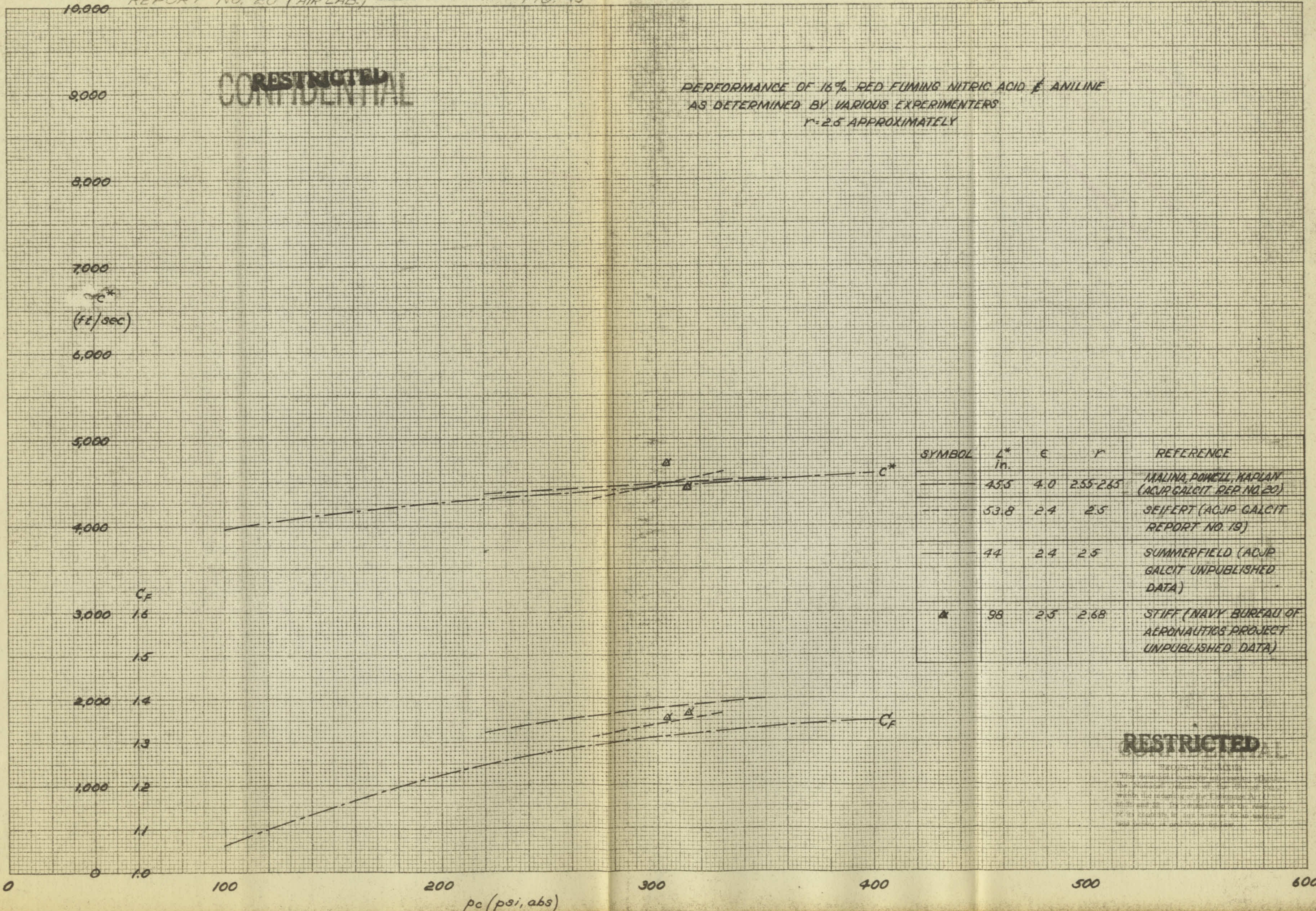
RESTRICTED



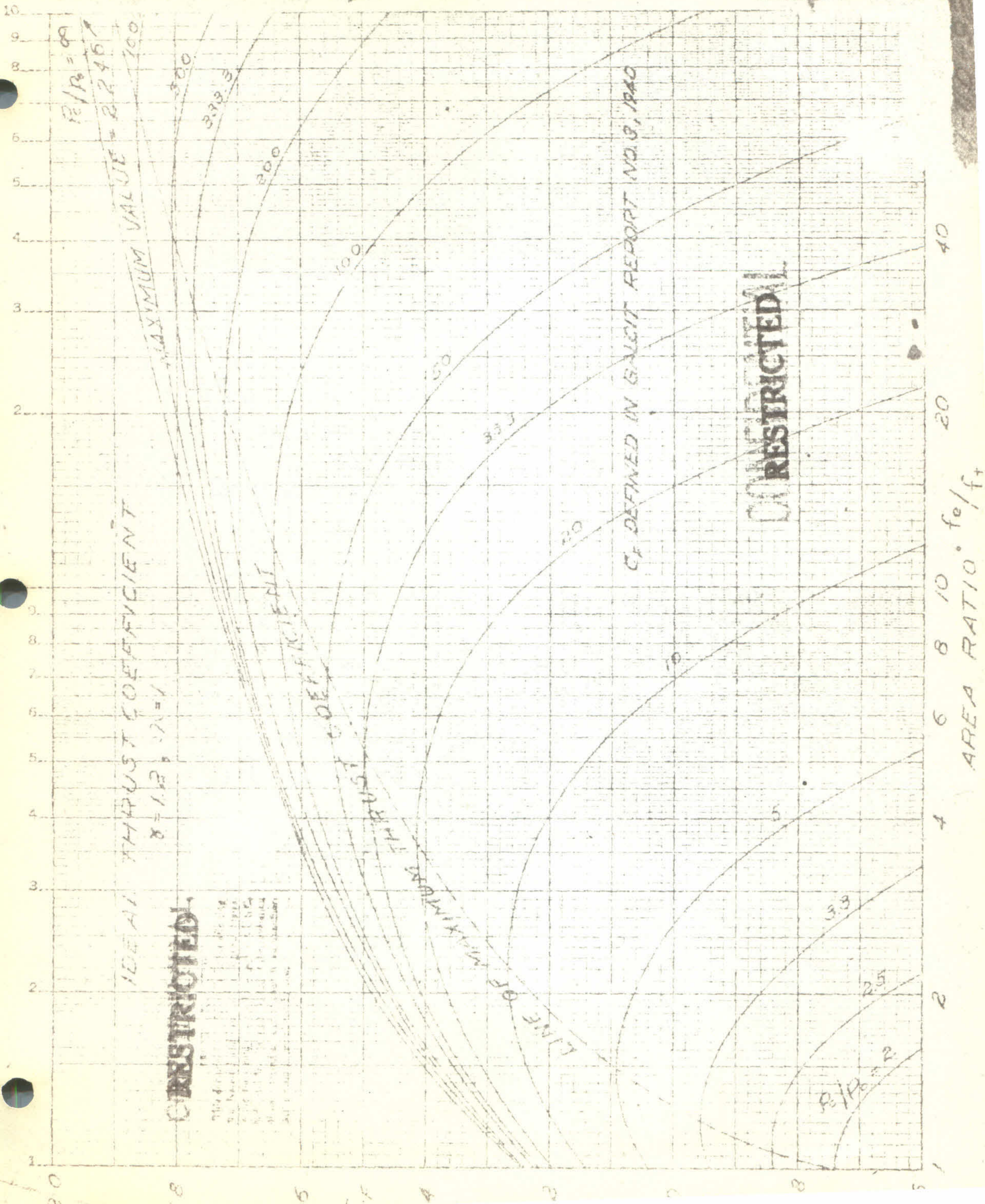
SYMBOL	L^* in.	ϵ	r	PROPELLANT	REFERENCE
———	45.5	4.0	2.55-2.65	16% RFNA & ANILINE	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)
— —	45.5	4.0	2.62-2.70	$6\frac{1}{2}$% RFNA & ANILINE	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)
- - - - -	45.5	4.0	2.40-2.65	$6\frac{1}{2}$% RFNA & ANILINE WITH 20% F.A.	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)
— - —	64.2	4.0	2.45-2.53	16% RFNA & ANILINE WITH 35% F.A.	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)
— - - —	45.5 & 100	4.0 & 3.5		$6\frac{1}{2}$% RFNA & FURFURYL ALCOHOL	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)
— - - - —	100	3.5		WHITE FUMING ACID & ANILINE	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)
—I—	45.5	4.0	2.50-2.65	MIXED ACID & ANILINE	MALINA, POWELL, KAPLAN (ACJP, GALCIT REP. NO. 20)

AMBIENT TEMPERATURE = 60°F TO 90°F

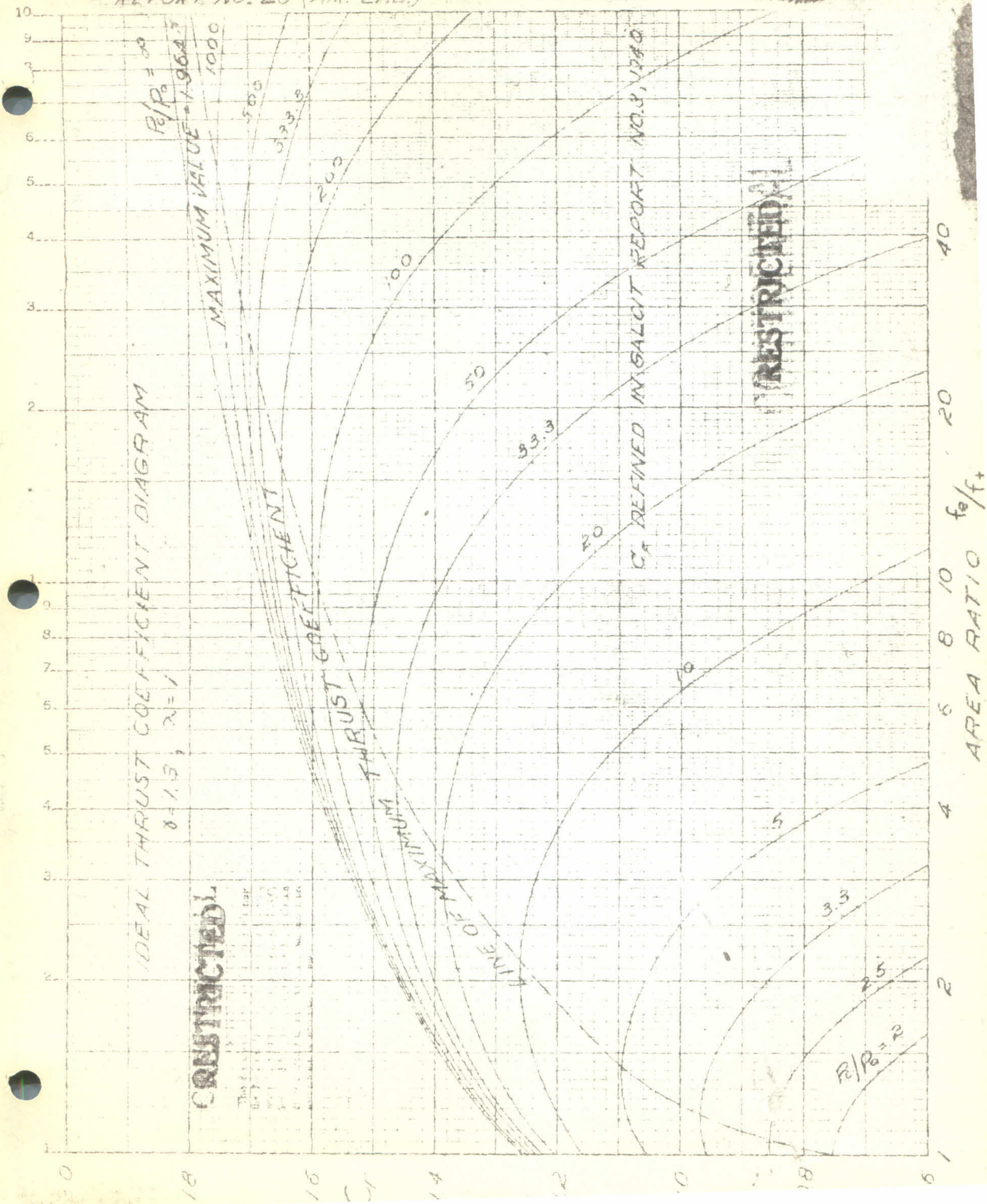
RESTRICTED


CONFIDENTIAL
RESTRICTED
PERFORMANCE OF 16% RED FUMING NITRIC ACID & ANILINE
AS DETERMINED BY VARIOUS EXPERIMENTERS
r = 2.5 APPROXIMATELY
10,000
9,000
8,000
7,000
c*
(ft/sec)
6,000
5,000
4,000
3,000
2,000
1,000
0
C_F
1.6
1.5
1.4
1.3
1.2
1.1
1.0
c*
C_F
0
100
200
300
400
500
600
pc (psi, abs)
SYMBOL
L* in.
ϵ
r
REFERENCE
45.5
4.0
2.55-2.65
MALINA, POWELL, KAPLAN (ACJP GALCIT REP. NO. 20)
53.8
2.4
2.5
SEIFERT (ACJP GALCIT REPORT NO. 19)
44
2.4
2.5
SUMMERFIELD (ACJP GALCIT UNPUBLISHED DATA)
98
2.5
2.68
STIFF (NAVY BUREAU OF AERONAUTICS PROJECT UNPUBLISHED DATA)
RESTRICTED

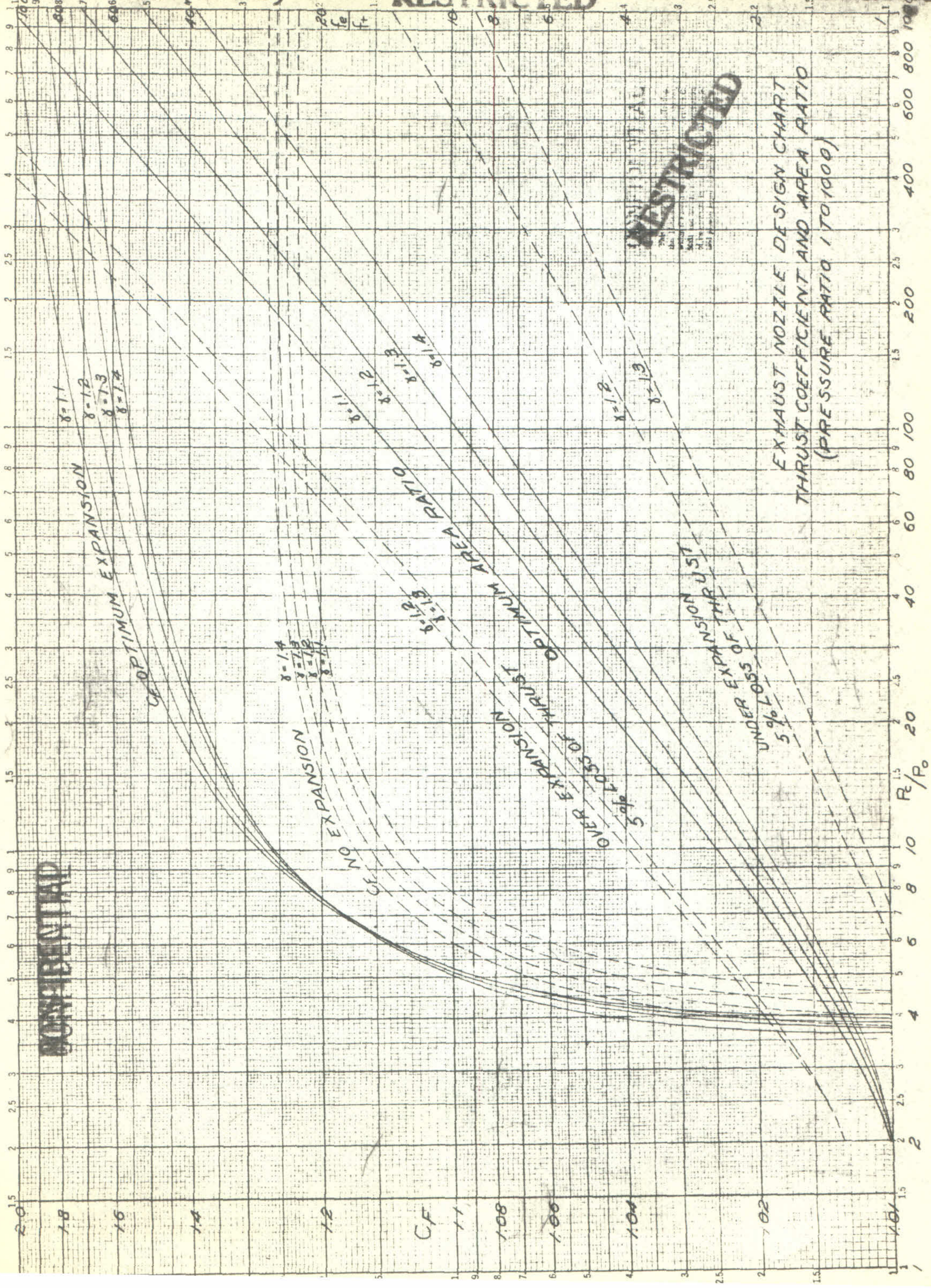
RESTRICTED


IDEAL THRUST COEFFICIENT
γ = 1.2, η = 1
$p_c/p_o = \infty$
MAXIMUM VALUE = 2.246
1000
333.3
200
100
50
33.3
20
10
5
3.3
2.5
$p_c/p_o = 2$
LINE OF MAXIMUM THRUST COEFFICIENT
C_F DEFINED IN GALCIT REPORT NO. 3, 1940
AREA RATIO · f_e/f_t
RESTRICTED
RESTRICTED
RESTRICTED

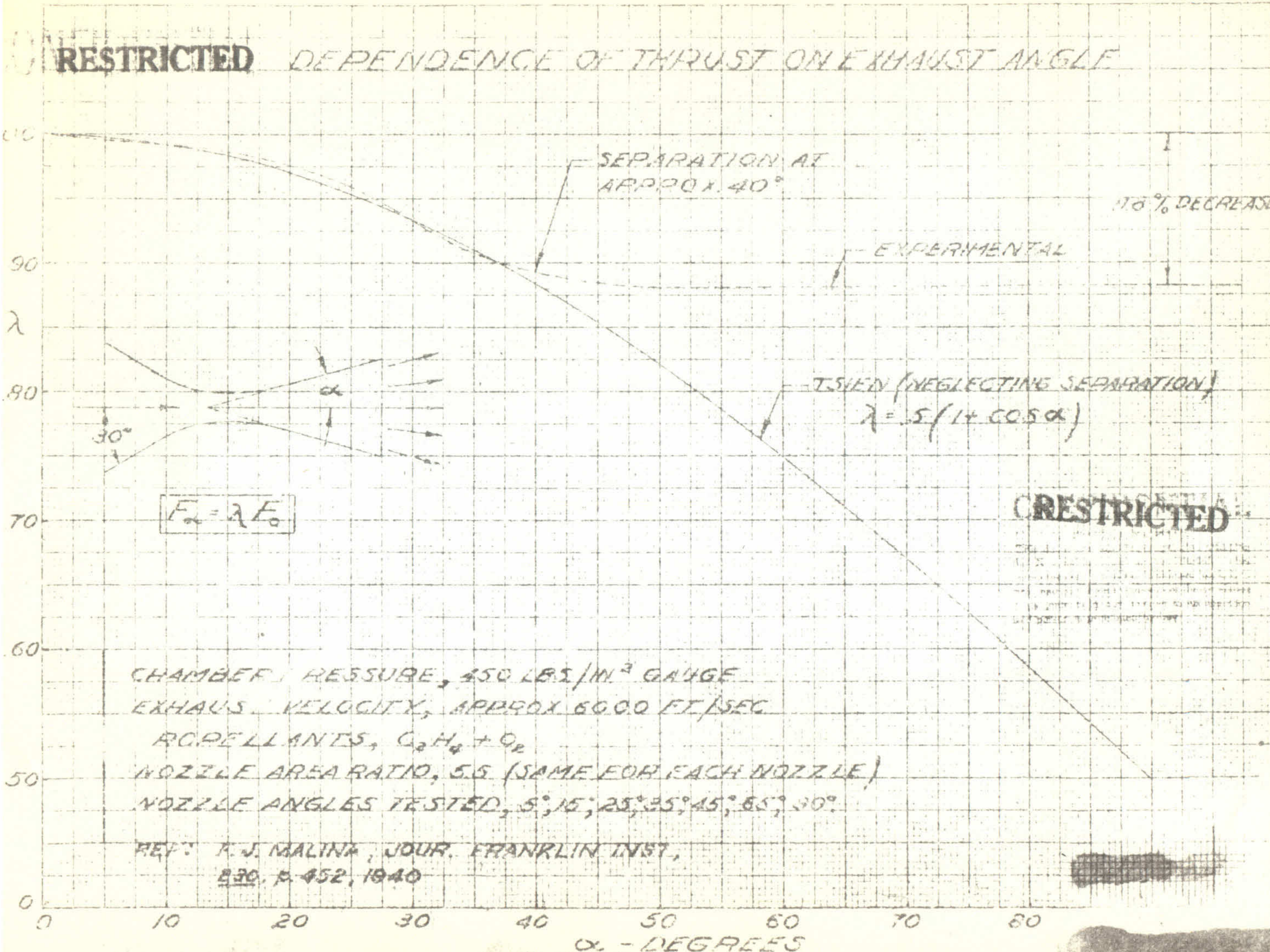
RESTRICTED


IDEAL THRUST COEFFICIENT DIAGRAM
γ = 1.3, λ = 1
Pc/Po = ∞
MAXIMUM VALUE = 1.9624
1000
500
333.3
200
100
50
33.3
20
10
5
3.3
2.5
Pc/Po = 2
LINE OF MAXIMUM THRUST COEFFICIENT
CF DEFINED IN GALCIT REPORT NO. 3, 1940
AREA RATIO fe/ft
RESTRICTED
RESTRICTED

RESTRICTED


EXHAUST NOZZLE DESIGN CHART
THRUST COEFFICIENT AND AREA RATIO
(PRESSURE RATIO 1 TO 1000)
C_F OPTIMUM EXPANSION
C_F NO EXPANSION
OPTIMUM AREA RATIO
OVER EXPANSION 5% LOSS OF THRUST
UNDER EXPANSION 5% LOSS OF THRUST
f_e/f_t
C_F
P_c/P_o
RESTRICTED
RESTRICTED
CONFIDENTIAL

RESTRICTED



RESTRICTED

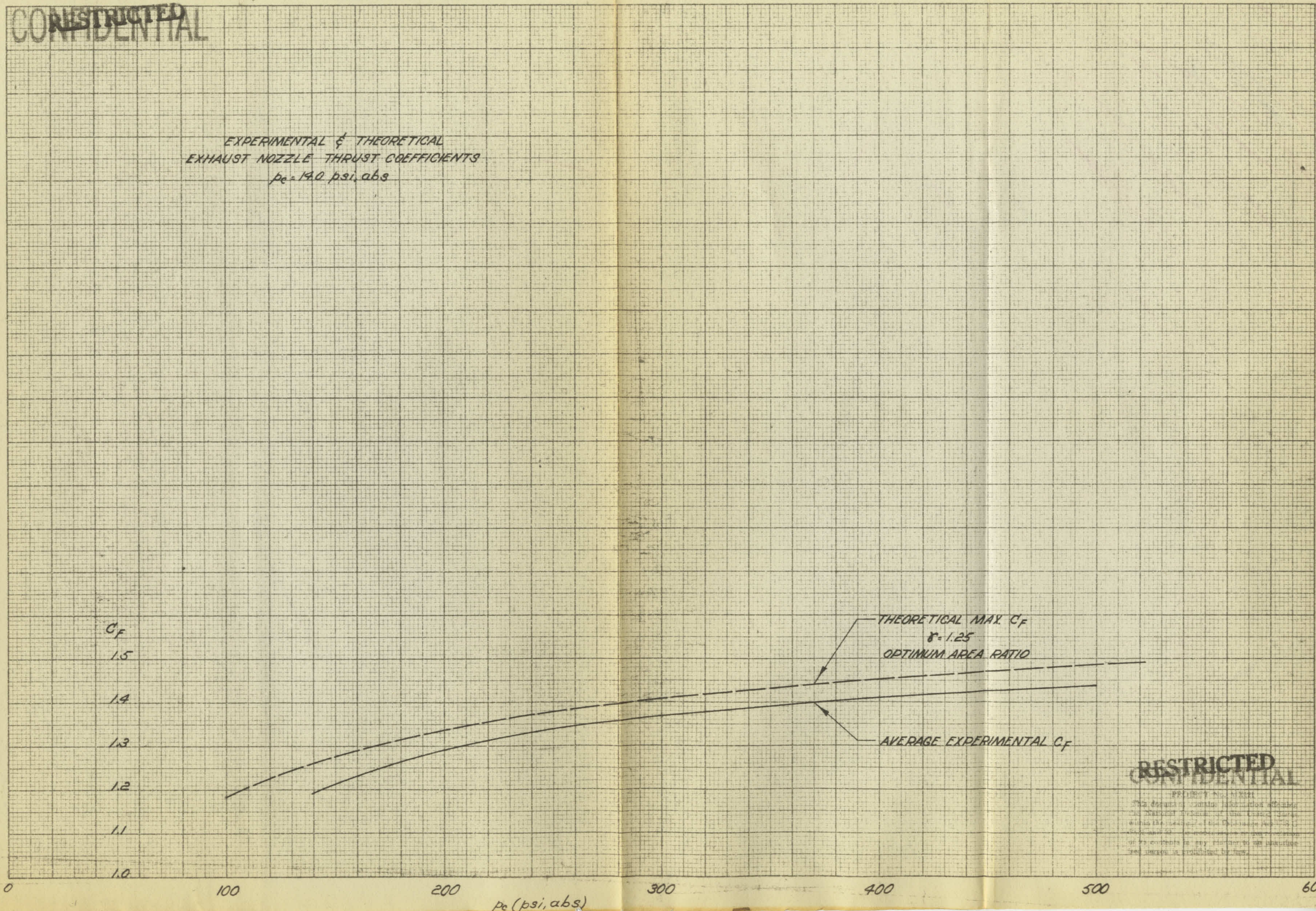
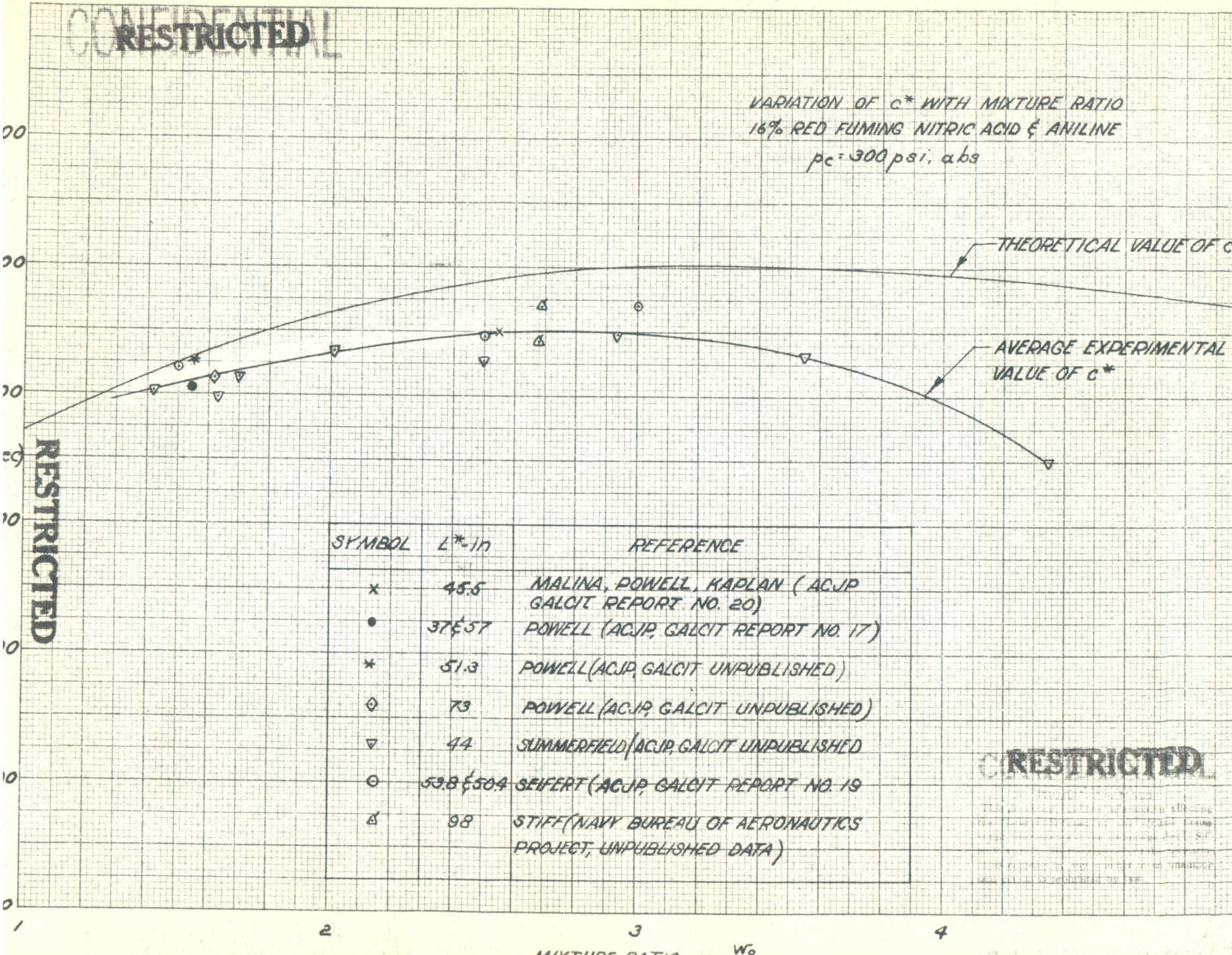

CONFIDENTIAL
RESTRICTED
EXPERIMENTAL & THEORETICAL
EXHAUST NOZZLE THRUST COEFFICIENTS
p_e = 14.0 psi, abs
C_F
1.5
1.4
1.3
1.2
1.1
1.0
0
100
200
300
400
500
600
p_c (psi, abs)
THEORETICAL MAX. C_F
γ = 1.25
OPTIMUM AREA RATIO
AVERAGE EXPERIMENTAL C_F
RESTRICTED
CONFIDENTIAL

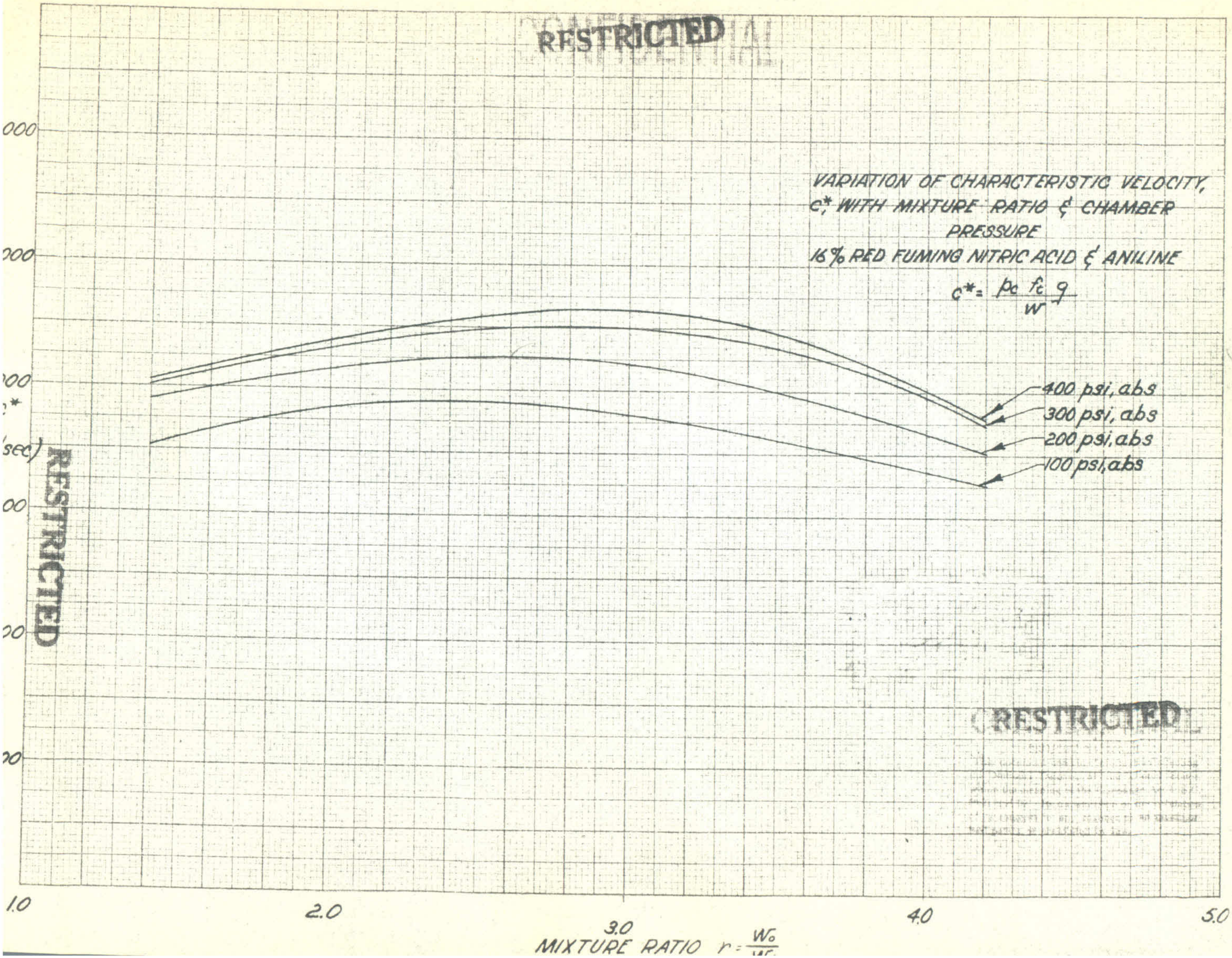
FIG. 49

RESTRICTED



RESTRICTED

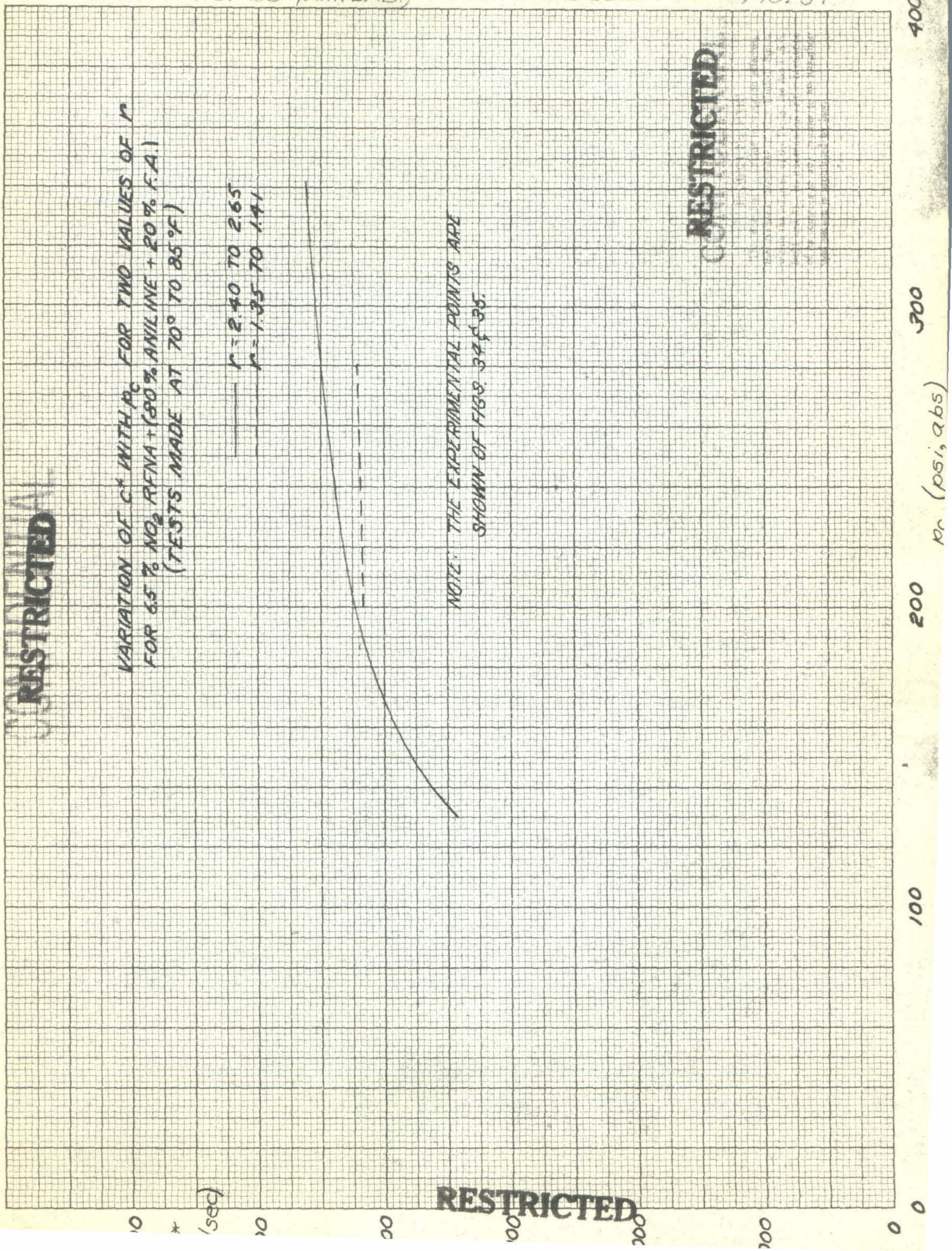
RESTRICTED



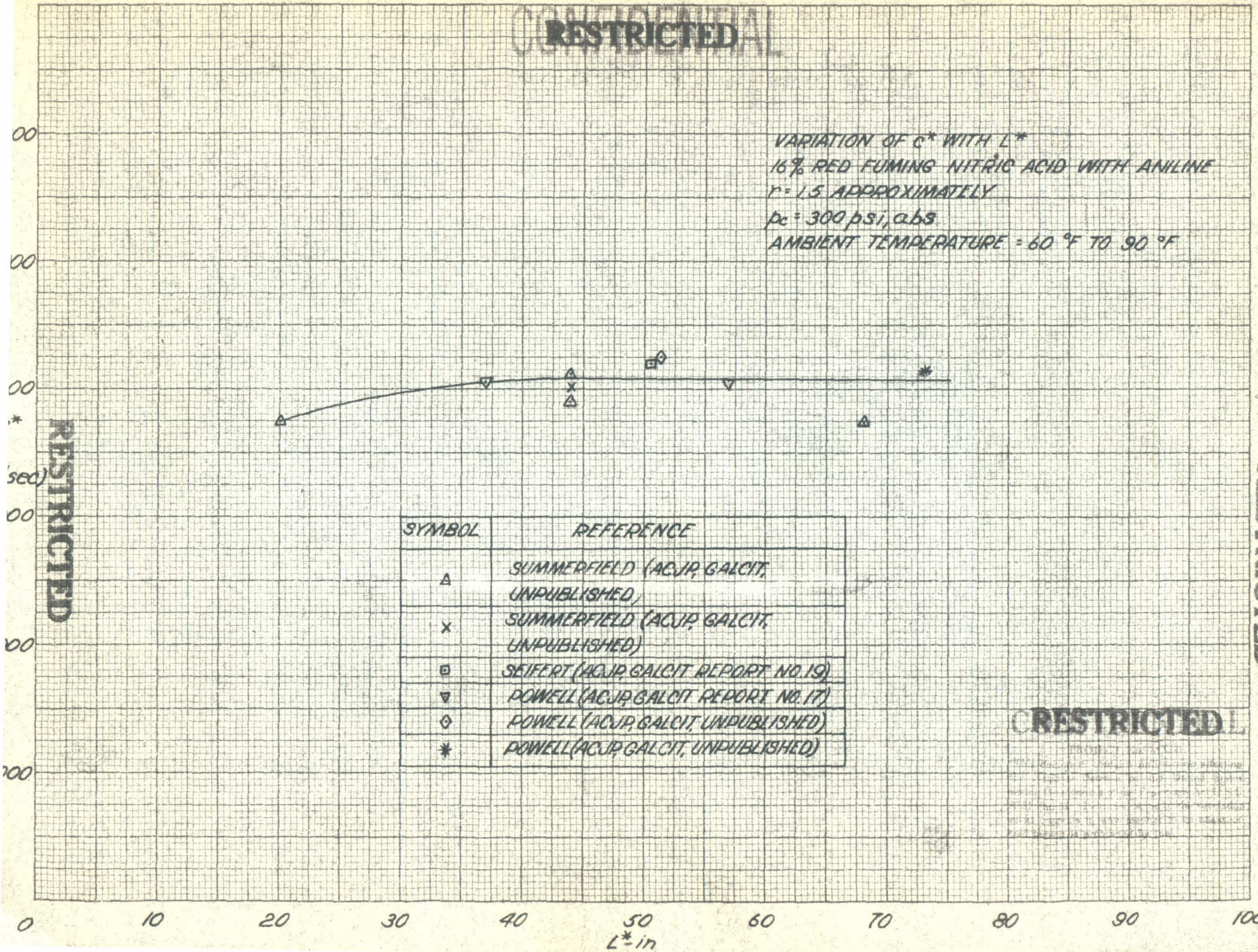
RESTRICTED

RESTRICTED

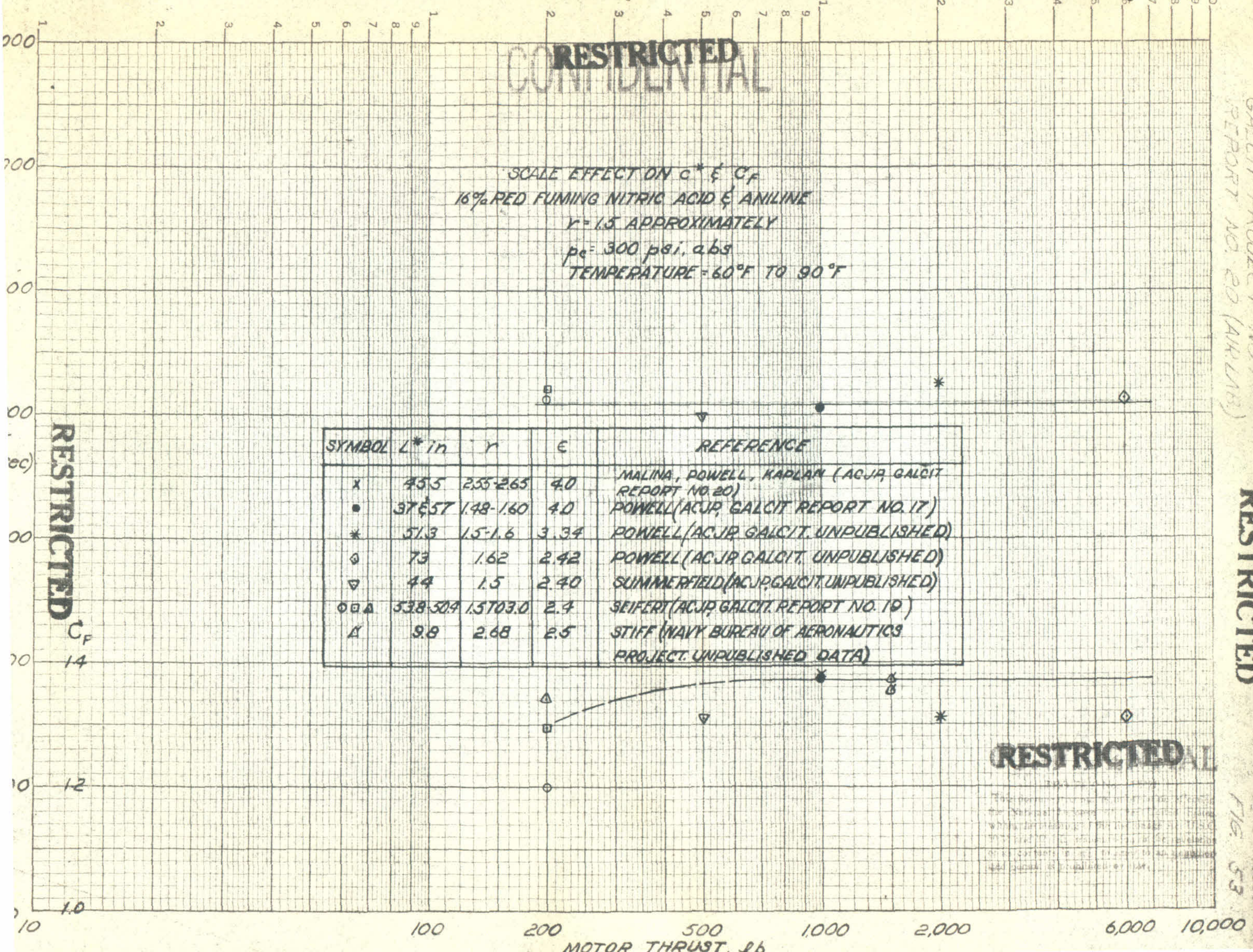
FIG. 51



RESTRICTED


RESTRICTED
VARIATION OF C* WITH L*
16% RED FUMING NITRIC ACID WITH ANILINE
r = 1.5 APPROXIMATELY
pc = 300 psi, abs
AMBIENT TEMPERATURE = 60 °F TO 90 °F
SYMBOL
REFERENCE
△ SUMMERFIELD (ACJP, GALCIT, UNPUBLISHED)
× SUMMERFIELD (ACJP, GALCIT, UNPUBLISHED)
⊡ SEIFERT (ACJP, GALCIT REPORT NO. 19)
▽ POWELL (ACJP, GALCIT REPORT NO. 17)
◇ POWELL (ACJP, GALCIT, UNPUBLISHED)
✱ POWELL (ACJP, GALCIT, UNPUBLISHED)
0
10
20
30
40
50
60
70
80
90
100
L* in
sec)
RESTRICTED
RESTRICTED
RESTRICTED

RESTRICTED



SYMBOL	L^* in	r	ε	REFERENCE
×	45.5	2.55-2.65	4.0	MALINA, POWELL, KAPLAN (ACJP, GALCIT REPORT NO. 20)
●	37 & 57	1.48-1.60	4.0	POWELL (ACJP, GALCIT REPORT NO. 17)
✳	51.3	1.5-1.6	3.34	POWELL (ACJP, GALCIT, UNPUBLISHED)
◇	73	1.62	2.42	POWELL (ACJP, GALCIT, UNPUBLISHED)
▽	44	1.5	2.40	SUMMERFIELD (ACJP, GALCIT, UNPUBLISHED)
○ □ △	33.8-50.4	1.5 TO 3.0	2.4	SEIFERT (ACJP, GALCIT REPORT NO. 19)
△	9.8	2.68	2.5	STIFF (NAVY BUREAU OF AERONAUTICS PROJECT, UNPUBLISHED DATA)

FIG. 53

RESTRICTED