

SCHLIEREN OBSERVATION OF
SUPERSONIC DISCHARGE



E. L. PERRY,
L. W. A. RENSHAW

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MASSACHUSETTS INSTITUTE OF TECHNOLOGY
Department of Mechanical Engineering
Cambridge 39, Mass., U.S.A.

Room 1-202

September 24, 1946

Captain W. H. Buracker
Room 5-233
Massachusetts Institute of Technology
Cambridge 39, Massachusetts

Thesis Work of LT E. L. PERRY, USCG ←
LT L. W. A. RENSHAW, USCG
LCDR W. W. SIMONS, USN
LCDR J. S. BOWEN, USN

Dear Captain Buracker:

The thesis by Lieutenants E. L. Perry and L. W. A. Renshaw entitled "Schlieren Observation of Supersonic Discharge" presents pressure measurements and Schlieren photographs of supersonic streams discharging into an exhaust space under various conditions. The photographs show interesting detail which in general corresponds to analytical results. The most significant observation was a comparison of two supersonic streams alike in average conditions but differing in thickness of the boundary layer. The effect of boundary-layer thickness on the nature of the shock pattern is shown clearly.

The thesis by Lt. Comdr^s W. W. Simons and J. S. Bowen entitled "Investigation of the Condensation Shock in Air by Use of the Schlieren Method" presents pressure measurements and Schlieren photographs of the shock patterns when water vapor in air condenses to form a fog of liquid or solid particles. It has extended our knowledge of the conditions which control condensation and of the condensation shock which accompanies it.

From either of these theses a paper could be prepared which would be published in one of the journals of the professional societies.

Yours truly,

/s/ Joseph H. Keenan

Joseph H. Keenan

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SCHLITZEN OBSERVATION

OF

SUPERSONIC DISCHARGE

BY

Lieut. E. L. Perry
B.S., U.S.C.G. Academy
1941

and

Lieut L.W.A. Renshaw
B.S., U.S.C.G. Academy
1941

Submitted in partial fulfillment of the
requirements for the degree of
Master of Science
at the
Massachusetts Institute of Technology
1946

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RESEARCH AND DEVELOPMENT

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MASSACHUSETTS INSTITUTE OF TECHNOLOGY
77 Massachusetts Avenue
Cambridge, Massachusetts

September 15, 1946

Professor Joseph S. Newell
Secretary of the Faculty
Massachusetts Institute of Technology
77 Massachusetts Avenue
Cambridge, Massachusetts

Dear Professor Newell:

Herewith we submit our thesis entitled "Schlieren
Observation of Supersonic Discharge" in partial fulfillment of the
requirements for the Degree of Master of Science in Naval
Construction and Engineering at the Massachusetts Institute of
Technology.

AMERICAN INSTITUTE OF PHYSICS
77 Massachusetts Avenue
Cambridge, Massachusetts

October 13, 1944

Professor Joseph E. Smith
Secretary of the Faculty
Massachusetts Institute of Technology
77 Massachusetts Avenue
Cambridge, Massachusetts

Dear Professor Smith:

I am writing to you in regard to the

question of your possible election to the

position of the Department of Physics at MIT.

I am sure that your election to this

position

Very truly yours,
J. W. A. [Signature]
J. W. A. [Name]

J. W. A. [Signature]
J. W. A. [Name]

ACKNOWLEDGEMENT

With pleasure we acknowledge the help given us by Professor Joseph H. Keenan and Professor E. P. Neumann of the Mechanical Engineering Department. Thanks are also due to Dr. Joseph Kaye and Mr. Ferdinand Lustwerk. Mr. Lustwerk rendered invaluable assistance in developing laboratory equipment and technique.

Professor Joseph H. Keenan suggested the thesis topic.

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SUMMARY

This work was undertaken to observe the effect on the discharge phenomenon of a supersonic air stream due to a change in Mach Number and a change of boundary layer thickness at constant Mach Number. Two (2) two-dimensional nozzles were designed using the Prandtl Theory, one having a Mach Number of 1.85 and the other a Mach Number of 1.39. A third nozzle was formed by adding a length of straight tube to the profile of the first nozzle to bring the Mach Number down to 1.39 by friction. All nozzles were designed for the same flow per unit area in the exit.

A comparison of the discharge of the first and second nozzles should show the effect of Mach Number, whereas a comparison of the second and third nozzles should show the effect of boundary layer thickness. The comparisons were made by Schlieren photographs and pressure measurements by mercury manometers at a point one eighth ($1/8$) inch from the exit of the nozzle and in the discharge chamber. It is noted that the nozzles were mounted perpendicular to the knife edge in the apparatus.

The results of the first comparison are not too conclusive. Further study in this line is recommended. The second comparison shows that a thick boundary layer cannot support anything resembling a transverse shock whereas a thin boundary layer will. Pressure measurement revealed that even in the thinnest boundary layers we were able to obtain there was no abrupt rise in pressure in the exit of the nozzles - like that expected in frictionless flow - as the exhaust pressure was increased. It is pointed out that the pressure was measured at the wall at a point one eighth ($1/8$) inch from exit. The photographs show that as the exhaust pressure is increased, the

This work was undertaken to determine the effect on the frequency
 spectrum of a resonator all things being equal in each layer
 and a change of boundary layer thickness at constant film thickness.
 The (1) experimental results were compared with the theoretical theory
 one having a film thickness of 1.00 and the other a film thickness of 1.20.
 A third series was chosen for adding a layer of material to the
 middle of the first series to bring the film thickness down to 1.20 by
 folding. All series were designed for the same film but not were
 in the air.

A comparison of the thickness of the first and second series
 should show the effect of film thickness, however a comparison of the
 second and third series should show the effect of boundary layer
 thickness. The comparison was made by following frequencies and
 pressure measurements by means of a point on which (1) is
 taken from the side of the series and in the frequency spectrum. It is
 noted that the material was mounted perpendicular to the side edge
 in the experiment.

The results of the first comparison are not too conclusive,
 further study in this line is recommended. The second comparison
 shows that a thin boundary layer causes a slight frequency shift
 a thickness about shows a thin boundary layer will. Pressure
 measurements revealed that even in the thinnest boundary layers no
 new side to obtain there are no sharp rise in pressure in the air
 of the series - like that recorded in frequencies line - on the
 constant pressure was observed. It is pointed out that the pressure
 was measured at the well at a point on which (1) had been well.
 The frequency shift at the second pressure is towards, the

oblique shock tends to creep back from the exit. This is shown in Figures VIII, IX and X. The gradual rise in exit pressure shown by our measurements may be due to this creeping back of the oblique shock over the pressure tap. Figure I shows that there were slight discontinuities in the pressure curve for the high Mach Number discharge. The photographs in this region - Figures XI, XII and XIII - depict this instability in the flow.

It is recommended that further work of this nature be carried out with the nozzles mounted parallel to the knife edge of the Schlieren apparatus in order to observe more precisely the contribution of the boundary layer to the discharge phenomena.

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INTRODUCTION

The academic interest in the flow of fluids at supersonic velocities has recently become of practical importance due to the development of gas turbines etc. The theory of the manner in which a supersonic stream from a nozzle or tube adjusts itself to the pressure in the exhaust space is well developed.

This work proposes to investigate and observe by Schlieren methods of photography the manner in which such adjustments are accomplished and the effects of different Mach Numbers and different boundary layer thicknesses on the phenomena.

INTRODUCTION

The analysis of the data in this report is based on the results of a series of experiments conducted over a period of six months. The purpose of these experiments was to determine the effect of various factors on the rate of reaction between the two substances. The results show that the rate of reaction is directly proportional to the concentration of the reactants and inversely proportional to the temperature. The following table shows the results of the experiments:

Concentration of Reactants	Temperature (°C)	Rate of Reaction
1.0	20	0.5
2.0	20	1.0
3.0	20	1.5
4.0	20	2.0
5.0	20	2.5
1.0	30	0.8
1.0	40	1.2
1.0	50	1.8

PROCEDURE

Three (3) two-dimensional nozzles were designed. These were fitted with plane glass sides so that the flow in the exit and discharge chamber could be observed by a Schlieren apparatus. The first of these nozzles (designated Nozzle #1 and shown in Figure XXXVI, Appendix B) was designed to have as little boundary layer as possible and a Mach Number of 1.85. The second nozzle (designated Nozzle #2 and shown in Figure XXXVII, Appendix B) was designed for the same flow per unit area at the exit and a Mach Number of 1.39. A comparison of these two nozzles should show some effect of Mach Number change on the discharge. The boundary layer should be small in each since they are very short.

To compare the discharge at the same Mach Number and different boundary layer thicknesses a straight portion was added to the contour of Nozzle #1 to reduce the Mach Number by friction to the same value as that of Nozzle #2 - (1.39). It was anticipated that some adjustment of the length of the straight portion would have to be made to bring the Mach Number to 1.39. This was later found to be the case.

The laboratory procedure consisted of mounting the nozzles in the Schlieren apparatus and taking suction with a steam jet air ejector. Air at room temperature and atmospheric pressure was used as supply to all the nozzles. It is noted that in order to maintain the same flow per unit area for Nozzle #2, a specially designed reducing fitting shown in Figure XXXIX was used to reduce the inlet pressure to two thirds ($2/3$) atmosphere.

Starting with the lowest pressure we could obtain in the

Figure 1 shows the results of the experiments. The curves show that the rate of reaction is first order with respect to the concentration of the reactants. The rate constant, k , was determined from the slope of the straight line obtained from a plot of $\log \frac{a-x}{a}$ versus time. The value of k was found to be 0.0012 min^{-1} . The activation energy, E_a , was calculated from the Arrhenius equation, $\log k = \log A - \frac{E_a}{RT}$, using the values of k at different temperatures. The value of E_a was found to be 15.2 kcal/mole .

The results of the experiments are summarized in Table I. It is seen that the rate of reaction is first order with respect to the concentration of the reactants. The rate constant, k , was determined from the slope of the straight line obtained from a plot of $\log \frac{a-x}{a}$ versus time. The value of k was found to be 0.0012 min^{-1} . The activation energy, E_a , was calculated from the Arrhenius equation, $\log k = \log A - \frac{E_a}{RT}$, using the values of k at different temperatures. The value of E_a was found to be 15.2 kcal/mole .

The laboratory procedure consisted of measuring the concentration of the reactants at various times. The rate of reaction was determined from the slope of the straight line obtained from a plot of $\log \frac{a-x}{a}$ versus time. The value of k was found to be 0.0012 min^{-1} . The activation energy, E_a , was calculated from the Arrhenius equation, $\log k = \log A - \frac{E_a}{RT}$, using the values of k at different temperatures. The value of E_a was found to be 15.2 kcal/mole .

discharge the exhaust pressure was allowed to increase in steps until the pressure shocks were seen to move back into the nozzle. Readings of the exhaust pressure and the pressure one eighth ($1/8$) inch upstream from the exit plane were made by mercury manometer and recorded. Photographs were made at each step using the Edgerton Flash Unit described in Reference (1). Graphs of exit pressure vs exhaust pressure were plotted,

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RESULTS AND DISCUSSION OF RESULTS

The results of the experiment are shown in Figures I to XXXV.

A comparison of Figure I and Figure II would indicate that a phenomenon more closely approaching a theoretical transverse shock is found in flow at higher Mach Number. The break in the pressure curve for Nozzle #1 (Mach Number 1.85) at an exhaust pressure of about 240 mm Hg. is much more pronounced than for any in the curve for Nozzle #2 (Mach Number 1.39). Examination of Figures X, XI and XII shows some instability of the discharge at the instant the shock occurs at the exit of the nozzle for the higher Mach Number. No such instability was observed at the lower Mach Number (1.39). Figures XXV, XXVI and XXVII show, however, what appears to be a transverse shock at the lower Mach Number. It is believed that the comparatively smooth pressure curve for Nozzle #2 is caused by the length of the shock. Apparently the flow separates from the tube wall near the exit and the shock passes smoothly up the nozzle as the exhaust pressure increases; whereas at the higher Mach Number the shock is much shorter and the flow less stable. We were unable to stop the shock in the exit of this nozzle.

Under all conditions the pressure in the stream adjusted itself to a lower exhaust pressure by the expansion wedges expected from the Meyer Theory of flow around a corner. This is shown in Figures IV, XXII and XXVIII. Small and moderate adjustments to a higher exhaust pressure were made in all cases by the medium of an oblique shock. There was a tendency for the oblique shock to creep back into the nozzle as the exhaust pressure increased. It is observed that this tendency became very pronounced in the case

of the thick boundary layer. It is possible that the gradual rise in the observed exit pressure as the exhaust pressure is increased is due to the oblique shock creeping back over the pressure tap which is located one eighth ($1/8$) inch from the exit. In that event the observed pressures are probably not the true pressures in the center of the stream at exit.

A comparison of Figure II and III show a marked similarity in the pressure relations of the two discharges at the same Mach Number (1.39) but different boundary layer thicknesses. It is noted that the curve for Nozzle #3 with a thick boundary layer is displaced to the right by about 15 mm Hg. on the exhaust pressure scale.

The mechanism by which the pressure in the stream adjusts itself to a considerably higher exhaust pressure is shown in Figures XXXII to XXXV and Figures XXIV to XXVII to be somewhat different in these two cases. In the case of the thick boundary layer Figures XXXII to XXXV show that nothing resembling a transverse shock occurs. Instead, the boundary layer, which is subsonic, appears to increase in area while the supersonic stream decreases in area; thus the pressure rises to that of the exhaust chamber. The oblique shocks which are set up and reflect downstream appear to originate at the point where contraction of the supersonic stream begins. It is possible that this apparent enlargement of the boundary layer cross section is actually a flow separation from the wall. The observation that this phenomenon occurs only in the case with thick boundary layer supports the former assumption, however.

It is recommended that further work on this point be carried

of the thick boundary layer. It is possible that the boundary layer
is the thickest and contains the highest velocity in the boundary
layer. In the region of the boundary layer, the velocity is
higher than in the region of the boundary layer. In the region
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The boundary layer is the region of the flow where the velocity
is higher than in the region of the boundary layer. It is
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the highest velocity in the boundary layer. In the region of
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the highest velocity in the boundary layer. In the region of
the boundary layer, the velocity is higher than in the region
of the boundary layer.

out with the nozzle mounted parallel to the knife edge of the Schlieren apparatus so that a better idea of what is going on in the boundary layer may be obtained.

In the case of the thin boundary layer no such separation or enlargement is observed. What appears to be a transverse shock with perhaps a little separation is shown in Figures XXiv to XXVII.

Investigation of the effect of Mach Number on the discharge with thick boundary layer is also recommended. It would be interesting to make observations at a Mach Number of 1.39 and with a boundary layer intermediate in thickness between the two cases used in this work.

As is noted in Appendix A the length of straight tube necessary to reduce the Mach Number of Nozzle 1 to that of Nozzle 2 was calculated to be 10.35 inches. Actual experiment revealed that this value should be 6.02 inches and the length was accordingly reduced to that value.

Due to extremely low temperatures of the stream it was practically impossible to prevent the condensation of moisture on the outside surfaces of the glass plates. This resulted in smudges similar to those shown in Figures IX, X, XXVIII and XXX.

and that the results obtained are in the same order of magnitude as those obtained by other authors. The present paper is devoted to the study of the effect of the thickness of the boundary layer on the rate of reaction. It is shown that the rate of reaction is independent of the thickness of the boundary layer for a certain range of thicknesses. The results are compared with those obtained by other authors. The present paper is devoted to the study of the effect of the thickness of the boundary layer on the rate of reaction. It is shown that the rate of reaction is independent of the thickness of the boundary layer for a certain range of thicknesses. The results are compared with those obtained by other authors.

FIGURE I

NOZZLE 1

P_e = Exhaust Pressure

P_g = Pressure In Exit

Nozzle Inlet Pressure 761.2 mm Hg.

Inlet Temp. 85° F.

July 19, 1946

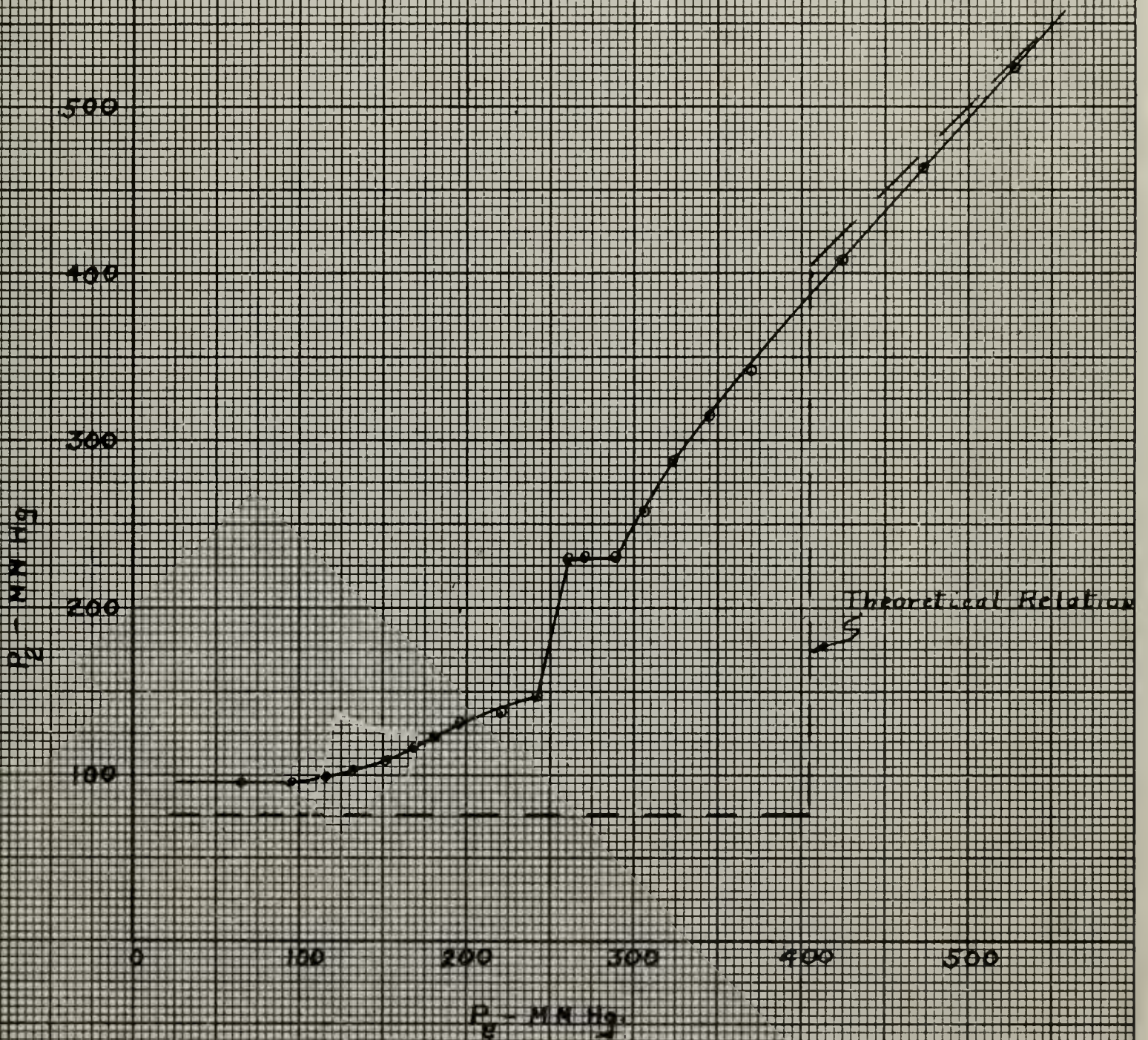


FIGURE II

NOZZLE 2

P_e = EXHAUST PRESSURE

P_2 = PRESSURE IN EXIT.

NOZZLE INLET PRESSURE 502 mm Hg.

Inlet Temp 85°F

July 16, 1946

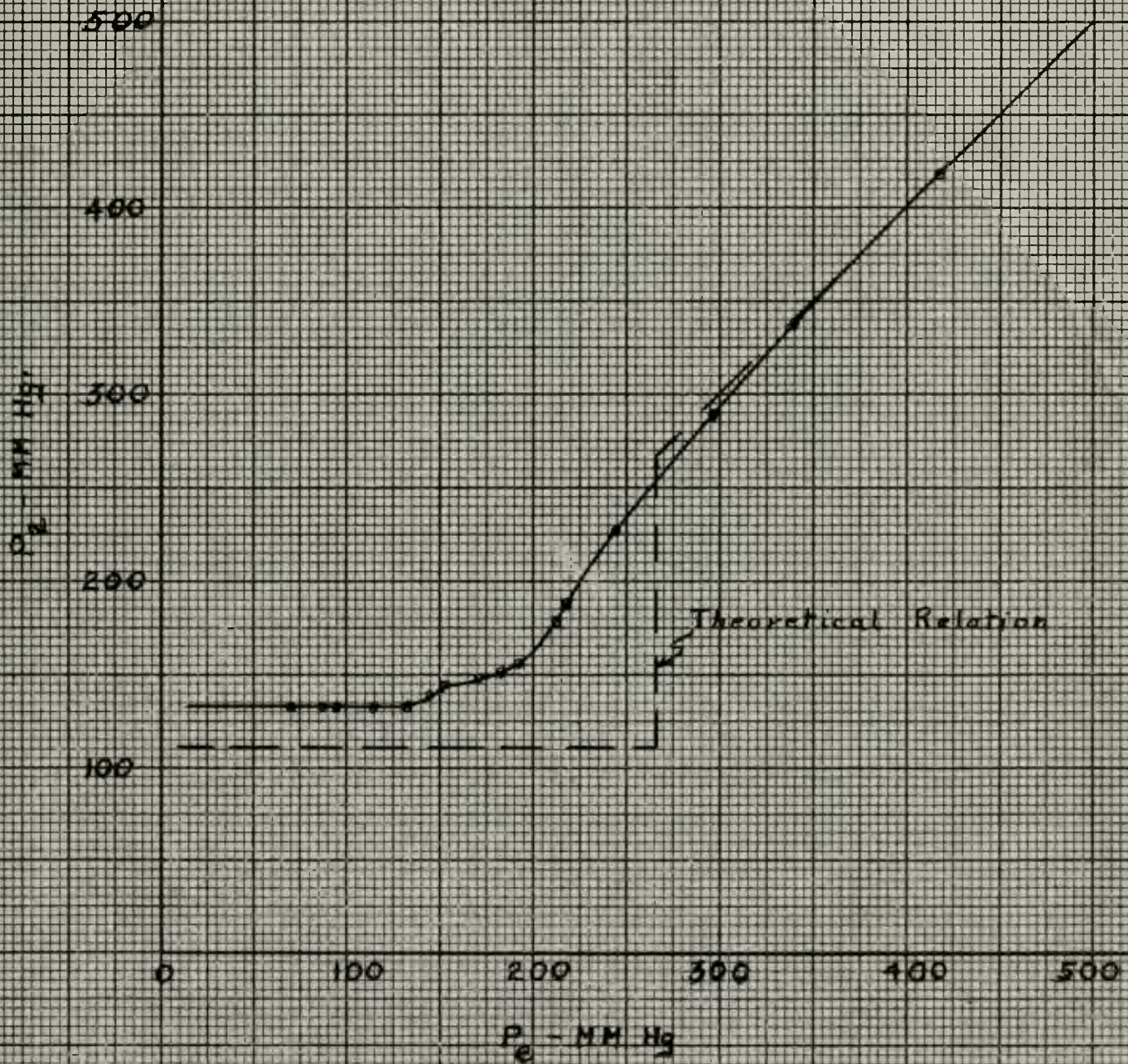


FIGURE III

NOZZLE 3

P_e = Exhaust Pressure mm Hg

P_g = Pressure in Exit "

Nozzle Inlet Pressure 769.4 mm Hg

Tube Inlet Pressure 83.4 "

July 22, 1946

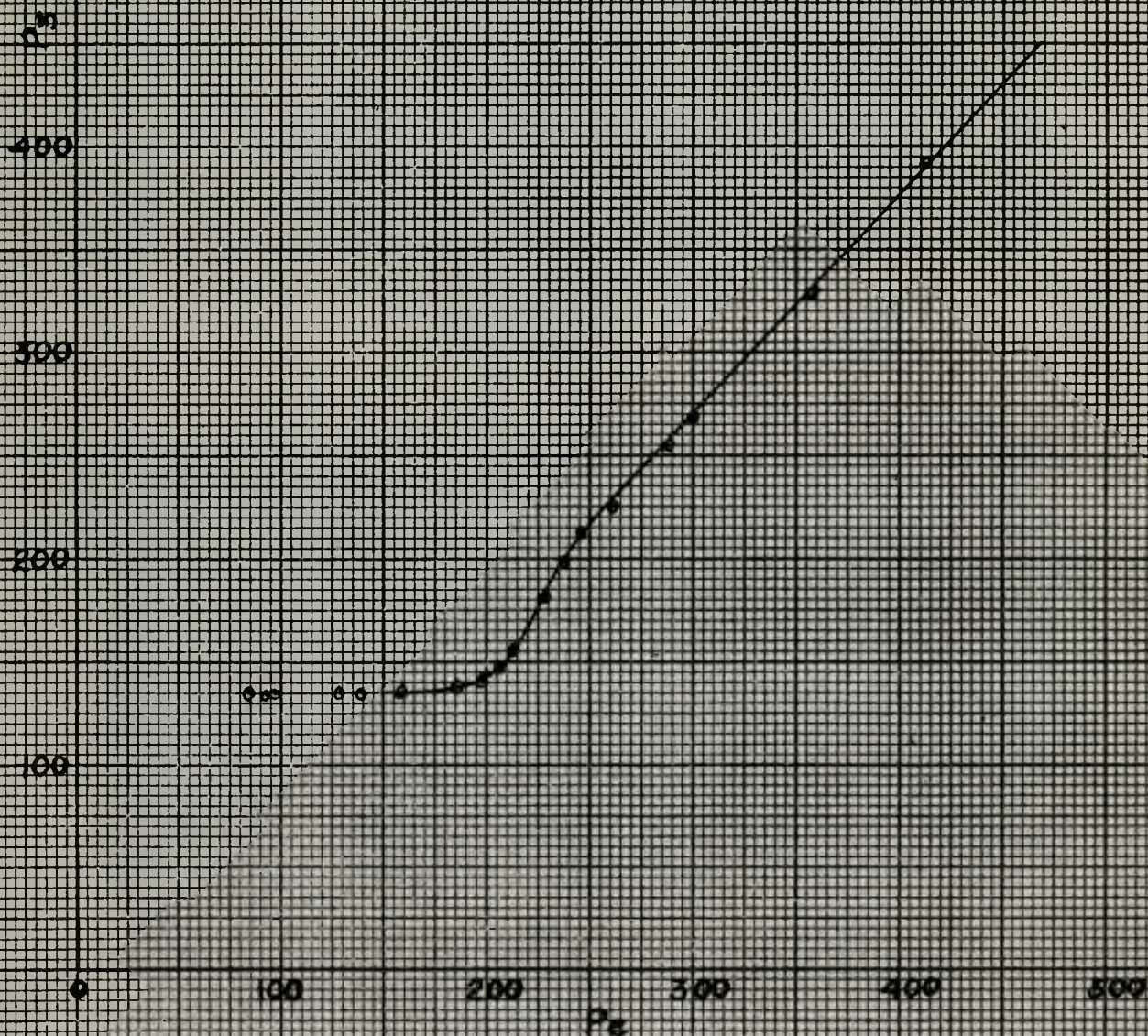




FIGURE IV

Nozzle #1

$P_0 = 74$
 $P_2 = 95$

Flash



FIGURE V

Nozzle #1

$P_0 = 95$
 $P_2 = 95$

Flash



FIGURE VI

Nozzle #1

$P_0 = 111$
 $P_2 = 99$

Flash



FIGURE IV

Nozzle #1

$P_0 = 74$
 $P_2 = 95$

Flash



FIGURE V

Nozzle #1

$P_0 = 95$
 $P_2 = 95$

Flash



FIGURE VI

Nozzle #1

$P_0 = 111$
 $P_2 = 99$

Flash

TABLE I
TABLE II
TABLE III
TABLE IV

TABLE I
TABLE II
TABLE III
TABLE IV

TABLE I
TABLE II
TABLE III
TABLE IV

TABLE I
TABLE II
TABLE III
TABLE IV

TABLE I
TABLE II
TABLE III
TABLE IV



FIGURE VII

Nozzle #1

P_0 - 139
 P_2 - 107

Flash



FIGURE VIII

Nozzle #1

P_0 - 171
 P_2 - 117

Flash

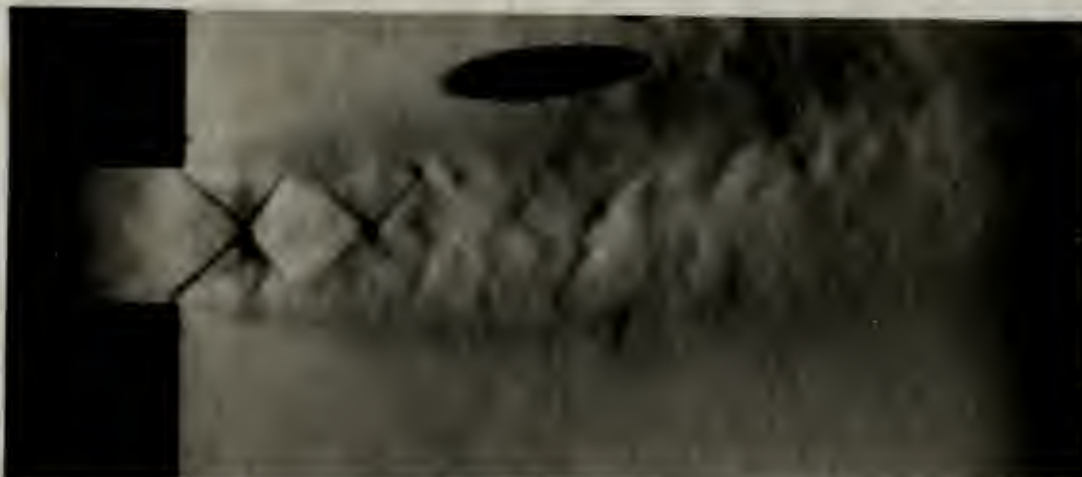


FIGURE IX

Nozzle #1

P_0 - 194
 P_2 - 131

Flash

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Handwritten notes in a rectangular box at the bottom of the page, containing several lines of cursive text.

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FIGURE I

Nozzle #1

P_0 - 212
 P_2 - 140

Flash



FIGURE XI

Nozzle #1

P_0 - 246
 P_2 - 169

Flash



FIGURE XII

Nozzle #1

P_0 - 271
 P_2 - 230

Flash

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FIGURE XIII

Nozzle #1

P_0 - 123
 P_2 - 288

Flesh



FIGURE XIV

Nozzle #1

P_0 - 92
 P_2 - 95

1/80 Sec.



FIGURE XV

Nozzle #2

P_0 - 73.59
 P_2 - 133.5
 F_1 - 500.5

1/80 Sec.

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Handwritten text on a rectangular piece of paper, possibly a note or a small document, with some illegible markings.

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FIGURE XVI

Nozzle #2

P_e - 96.5
 P_2 - 133.5
 P_1 - 508.5

1/80 Sec.



FIGURE XVII

Nozzle #2

P_e - 132
 P_2 - 133
 P_1 - 504

1/80 Sec.

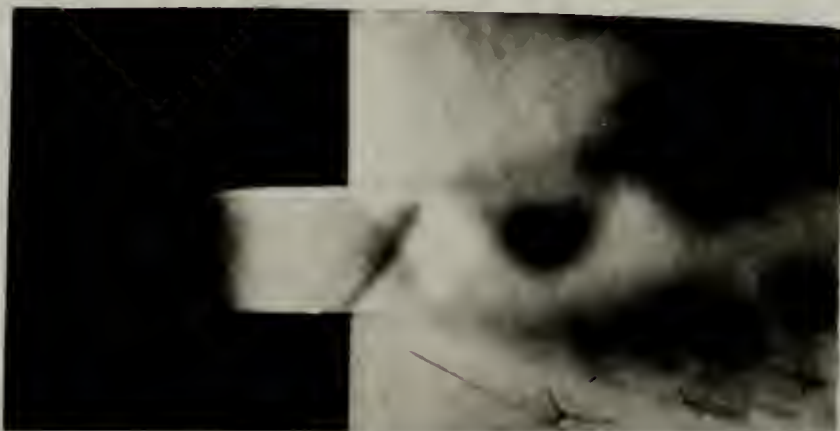


FIGURE XVIII

Nozzle #2

P_e - 144
 P_2 - 138
 P_1 - 502

1/80 Sec.

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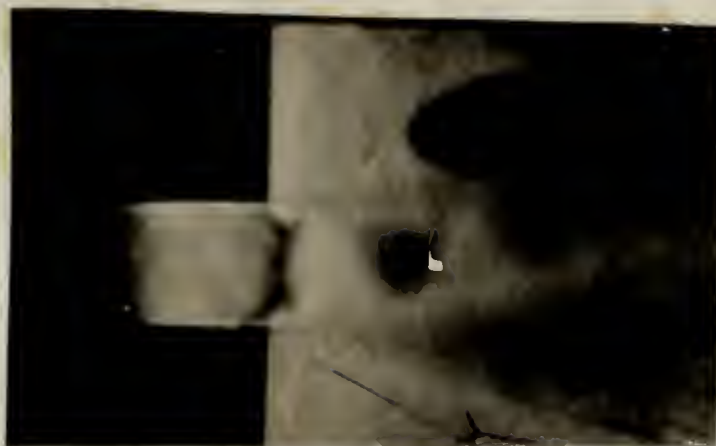


FIGURE XIX

Nozzle #2

P_0 - 192
 P_2 - 155
 P_1 - 504

1/80 Sec.

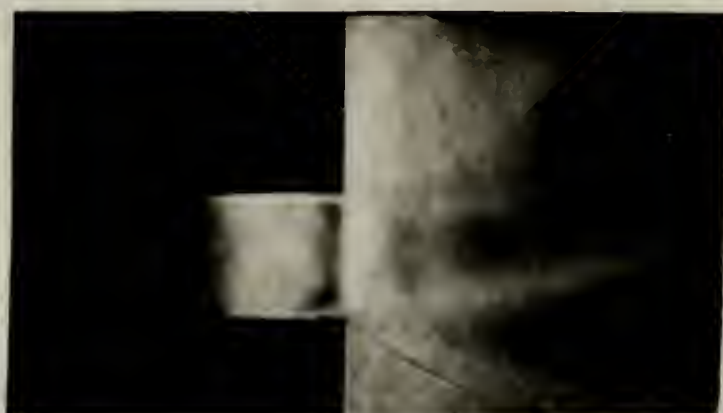


FIGURE XX

Nozzle #2

P_0 - 217
 P_2 - 187
 P_1 - 502

1/80 Sec.



FIGURE XXI

Nozzle #2

P_0 - 235
 P_2 - 217
 P_1 - 501

1/80 Sec.



FIGURE 101

Section 101

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FIGURE 102

Section 102

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FIGURE 103

Section 103

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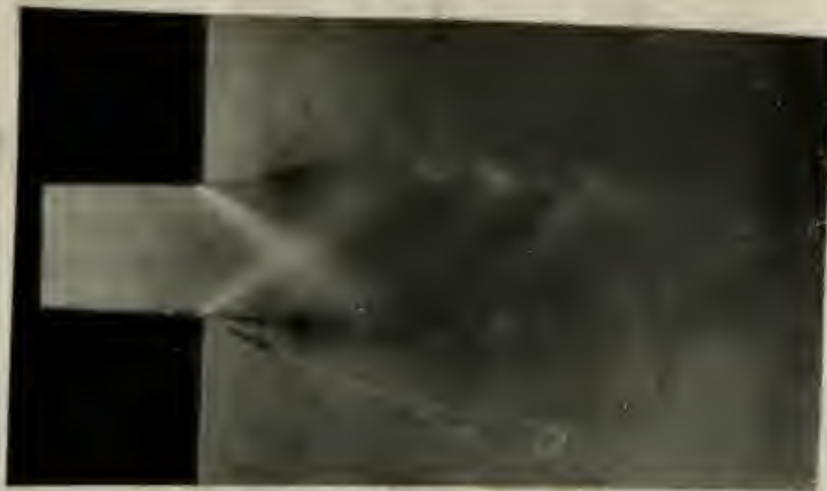


FIGURE XXII

Nozzle #2

P_e - 98
 P_2 - 130
 F_1 - 509

Flash



FIGURE XXIII

Nozzle #2

P_e - 131
 P_2 - 131
 F_1 - 508

Flash



FIGURE XXIV

Nozzle #2

P_e - 156
 P_2 - 145
 F_1 - 508

Flash



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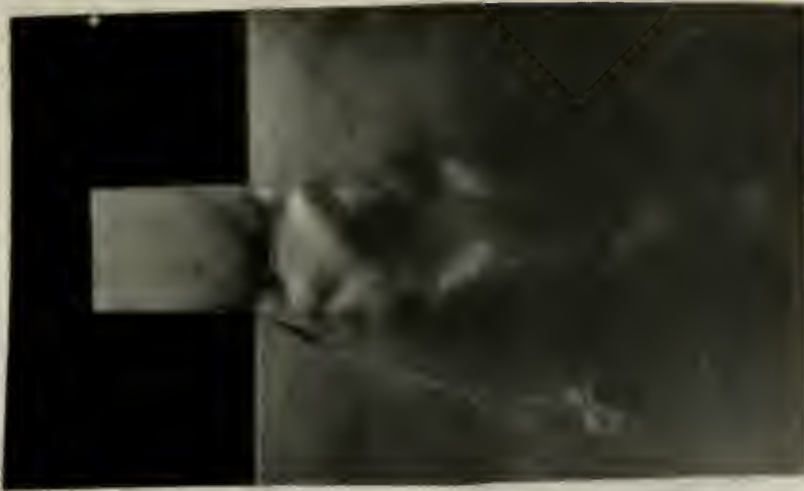


FIGURE XXV

Nozzle #2

P₀ - 186
P₂ - 148
P₁ - 506

Flash

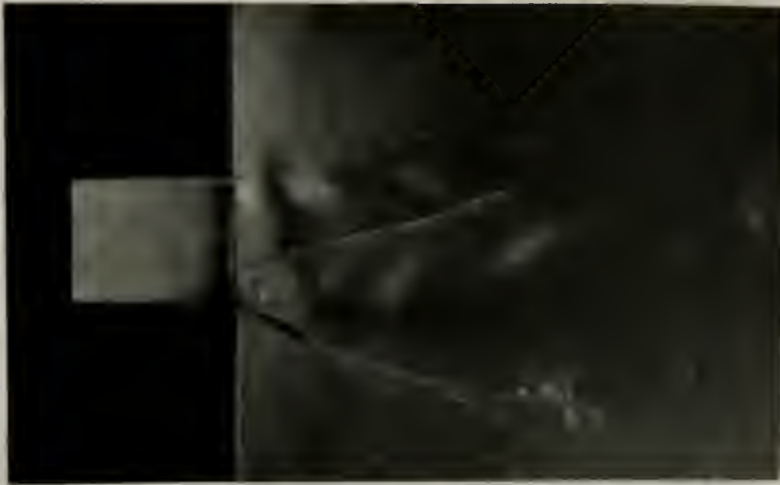


FIGURE XXVI

Nozzle #2

P₀ - 219
P₂ - 191
P₁ - 504

Flash



FIGURE XXVII

Nozzle #2

P₀ - 235
P₂ - 215
P₁ - 504

Flash



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FIGURE XXVIII

Nozzle #3

P_0 - 80.4
 P_3 - 136.4

Flash



FIGURE XXIX

Nozzle #3

P_0 - 113.4
 P_3 - 136.4

Flash

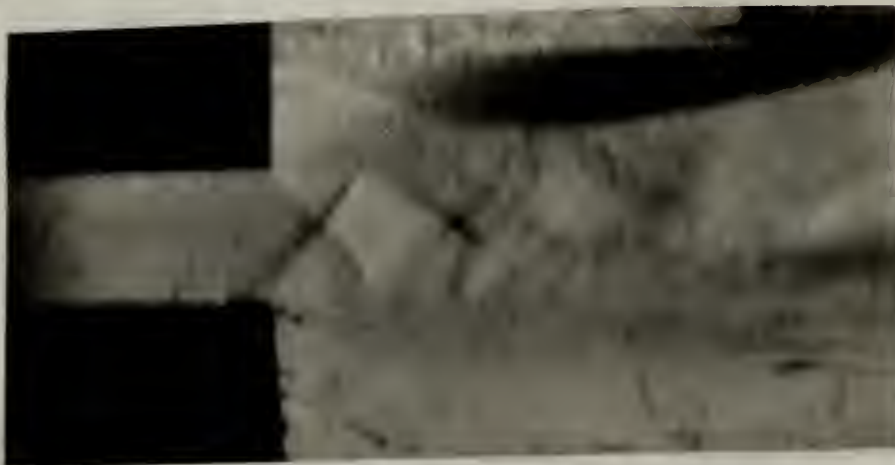


FIGURE XXX

Nozzle #3

P_0 - 172.4
 P_3 - 136.4

Flash



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1.00 - 1/2
1.00 - 1/2

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1.00 - 1/2



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1.00 - 1/2



FIGURE XXXI

Nozzle #3

$P_e - 197.3$
 $P_3 - 141.4$

Flash



FIGURE XXXII

Nozzle #3

$P_e - 213.4$
 $P_3 - 156.4$

Flash

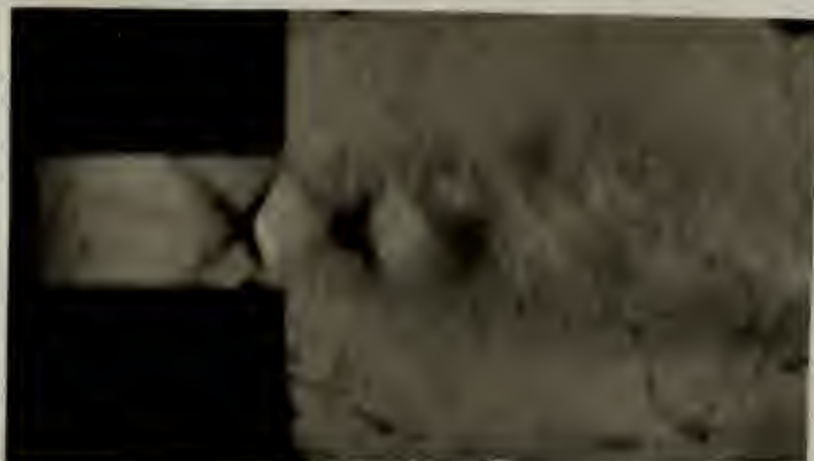


FIGURE XXXIII

Nozzle #3

$P_e - 238.4$
 $P_3 - 198.4$

Flash



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1847 - 1848
1848 - 1849

1847 - 1848

1847 - 1848



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1849 - 1850

1848 - 1849

1848 - 1849



1849

1849 - 1850
1850 - 1851

1849 - 1850

1849 - 1850



FIGURE XXXIV

Nozzle #3

P_e - 261.4
 P_3 - 225.4

Flash



FIGURE XXXV

Nozzle #3

P_e - 300.4
 P_3 - 267.4

Flash

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CONCLUSIONS AND RECOMMENDATIONS

1. Regardless of Mach Number, in supersonic flow the rise of pressure in the exit plane of a practical nozzle is not sudden (in accordance with the theoretical relation) but occurs slowly over a considerable range of exhaust chamber pressure.
2. With thick boundary layer the flow will not support anything resembling a transverse shock.
3. Thickness of boundary layer has the controlling influence on the mechanism by which a supersonic stream adjusts itself to the pressure in the exhaust chamber.
4. It is recommended that further work in this line be carried out with the nozzles mounted parallel to the knife edge of the Schlieren apparatus under the following conditions:
 - (a) Use nozzles #1 and #3 of Appendix B.
 - (b) Use a nozzle with a tube approximately three (3) inches long at a Mach Number of about 1.39 at exit.
 - (c) Use a nozzle with a tube approximately six (6) inches long at a Mach Number of about 1.85 at exit.
5. It is also recommended that investigations of the effect of flow per unit area at the same Mach Number upon the discharge phenomena be carried out.

1. Description of the system, in accordance with the view of pressure in the class of a practical model in the matter (in accordance with the theoretical relation) but occurs almost over a considerable range of values of pressure.

2. With this pressure law the law will not support against receiving a pressure shock.

3. The theory of pressure law and the controlling influence of the medium by which a pressure shock is adjusted itself to the pressure in the medium.

4. It is recommended that further work in this line be carried out with the model mounted parallel to the edge of the substrate apparatus under the following conditions:

- (a) The velocity of the shock is 100 ft/sec.
- (b) The shock is a wave with a wave length of about 1.5 ft.
- (c) The shock is a wave with a wave length of about 1.5 ft.
- (d) The shock is a wave with a wave length of about 1.5 ft.

5. It is also recommended that investigations of the effect of the pressure law on the wave shock law when the discharge pressure is varied out.

APPENDIX

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APPENDIX A -- DETAILS OF PROCEDURE

Reference (3) illustrates that a good shockless nozzle may be designed by the use of the Prandtl Theory; therefore it was decided to use this method as the basis of the nozzle design. The nozzle design was merely a reproduction of the work of Reference (3) but using different area ratios. A theoretical pressure ratio of .10 was chosen for the basic nozzle (Figure XXXVI, Appendix B) with an angle of divergence of $14^{\circ} 15'$. The theoretical Mach number at the exit of this nozzle is 2.152 based on $k = 1.400$. The area ratio is 1.9307. A velocity coefficient of .95 was assumed and the actual Mach number calculated to be 1.85 with a pressure ratio of .124.

It was desired to investigate the effect of Mach number with approximately constant boundary layer thickness on the discharge phenomena. To accomplish this a second nozzle (Figure XXXVII, Appendix B) was designed with an area ratio of 1.287. In order to maintain the same flow per unit area at the exit of the two nozzles the inlet pressure in this nozzle was reduced to two thirds ($2/3$) of an atmosphere by a specially designed adjustable fitting (Figure XXXIX, Appendix B). With an assumed velocity coefficient of .95, r was calculated as .275 and the Mach number at exit as 1.39. In order to keep both nozzles the same length the angle of divergence was reduced to six (6) degrees. It was believed that any differences that might be caused by this change of angle of divergence would be less than those caused by a change in length which would affect the boundary layer.

In order to observe the effect of boundary layer thickness

on the discharge phenomena for the same Mach number at exit, it was decided to add to the basic nozzle (Nozzle 1) a straight constant area section of such length as to reduce the Mach number of Nozzle #1 (1.85) to the Mach number of Nozzle #2 (1.39). To eliminate the possibility of shock formation at the junction of the nozzle and tube it was decided to fabricate another nozzle with the straight portion integral with the nozzle itself (Figure XXXVIII). By use of data obtained from Reference (6) the length of tube necessary was calculated to be 10.35 inches. This figure was regarded as highly approximate due to the use of a two dimensional tube instead of the circular section upon which the data of Reference (6) is based.

Provision was made for pressure measurement by mercury manometer at a point one eighth (1/8) inch from the nozzle or tube exit and in the discharge chamber of all nozzles by a .020 inch diameter hole in the steel contour.

All pictures were taken with the axis of the nozzle perpendicular to the knife edge of the Schlieren apparatus described adequately in Reference (1).

The pictures designated "Flash" were made by using the Edgerton Flash Unit also described in Reference (1). This gave an exposure time of approximately $.5 \times 10^{-6}$ seconds. A few pictures were taken using a steady light source and an exposure time of 1/80 seconds, to show the difference in detail of pictures obtained by the use of the two different methods.

on the discharge phenomena for the case of a constant current, it was
decided to use in the present work (see Table I) a constant current
and sections of such length as to reduce the error of about 1%
(1.5%) to the same order of magnitude as in the case of a constant
potential of about 100 V. The function of the current and the
It was decided to fabricate another series of similar sections
integral with the series (see Table I), of size of 1 cm
obtained from reference (5) the length of the sections was increased
to be 10.5 cm. This figure was chosen in order to give approximately the
of the size of a few dimensions, the length of the sections was chosen
which the size of reference (5) is based.
Provision was made for pressure measurement by means of a
of a point one eighth (1/8) inch from the center of the end and
in the discharge chamber of all sections by a 100 inch diameter hole
in the steel container.
All sections were taken with the axis of the needle perpendicular
to the axis of the hollow electrode (see Table I).
in reference (1).
The pressure distribution within the sections was measured by means of the
This point also described in reference (1). This was an exposure
size of approximately 2×10^{-6} cm. The sections were taken
using a sharp light source and an exposure time of 1/80 second.
So also the differences in detail of pressure obtained by the use of
the two different sections.

APPENDIX B---EXPERIMENTAL DATA

TABLE I

19 JULY, 1946

PRESSURE READINGS, NOZZLE I

P_e = Exhaust Chamber Pressure, mm. Hg.

P_2 = Pressure in Exit of Nozzle, mm. Hg.

P_a = Nozzle Inlet Pressure, Atmospheric

T_1 = Inlet Temperature, Degrees F.

P_e	P_2	P_a	T_1
74	95	761.2	85
95	95		
116	99		
133	104		
153	109		
169	116		
181	122		
196	131		
220	136		
241	197		
260	228		
287	230		
324	289		
370	341		
424	407		
473	464		
527	520		

EXHIBIT B - FINANCIAL DATA

TABLE I

FOR THE YEAR 1966

IN MILLIONS OF DOLLARS

1	2	3	4
100	100	100	100
101	101	101	101
102	102	102	102
103	103	103	103
104	104	104	104
105	105	105	105
106	106	106	106
107	107	107	107
108	108	108	108
109	109	109	109
110	110	110	110
111	111	111	111
112	112	112	112
113	113	113	113
114	114	114	114
115	115	115	115
116	116	116	116
117	117	117	117
118	118	118	118
119	119	119	119
120	120	120	120
121	121	121	121
122	122	122	122
123	123	123	123
124	124	124	124
125	125	125	125
126	126	126	126
127	127	127	127
128	128	128	128
129	129	129	129
130	130	130	130
131	131	131	131
132	132	132	132
133	133	133	133
134	134	134	134
135	135	135	135
136	136	136	136
137	137	137	137
138	138	138	138
139	139	139	139
140	140	140	140
141	141	141	141
142	142	142	142
143	143	143	143
144	144	144	144
145	145	145	145
146	146	146	146
147	147	147	147
148	148	148	148
149	149	149	149
150	150	150	150

TABLE II

PRESSURE READINGS, NOZZLE # 2

13 JULY, 1946

P_e = Exhaust Chamber Pressure, mm. Hg.

P_2 = Pressure in Exit of Nozzle, mm. Hg.

P_1 = Nozzle Inlet Pressure, mm. Hg.

P_a = Atmospheric Pressure, mm. Hg.

T_1 = Nozzle Inlet Temperature, Degrees F.

P_e	P_2	P_1	P_a	T_1
71	133	502	761.5	85
86	133	502		
94	133	502		
113	133	502		
131	133	502		
144	138	502		
151	143	502		
171	146	502		
183	150	502		
192	155	502		
211	178	502		
217	188	502		
244	227	502		
296	288	502		
340	338	502		
418	417	502		

TABLE II

1912, 1913, 1914

PERCENTAGE OF ...

- 1. ...
- 2. ...
- 3. ...
- 4. ...
- 5. ...

1912	1913	1914	1915	1916
100	115	120	125	130
100	117	122	127	132
100	119	124	129	134
100	121	126	131	136
100	123	128	133	138
100	125	130	135	140
100	127	132	137	142
100	129	134	139	144
100	131	136	141	146
100	133	138	143	148
100	135	140	145	150
100	137	142	147	152
100	139	144	149	154
100	141	146	151	156
100	143	148	153	158
100	145	150	155	160
100	147	152	157	162
100	149	154	159	164
100	151	156	161	166
100	153	158	163	168
100	155	160	165	170
100	157	162	167	172
100	159	164	169	174
100	161	166	171	176
100	163	168	173	178
100	165	170	175	180
100	167	172	177	182
100	169	174	179	184
100	171	176	181	186
100	173	178	183	188
100	175	180	185	190
100	177	182	187	192
100	179	184	189	194
100	181	186	191	196
100	183	188	193	198
100	185	190	195	200
100	187	192	197	202
100	189	194	199	204
100	191	196	201	206
100	193	198	203	208
100	195	200	205	210
100	197	202	207	212
100	199	204	209	214
100	201	206	211	216
100	203	208	213	218
100	205	210	215	220
100	207	212	217	222
100	209	214	219	224
100	211	216	221	226
100	213	218	223	228
100	215	220	225	230
100	217	222	227	232
100	219	224	229	234
100	221	226	231	236
100	223	228	233	238
100	225	230	235	240
100	227	232	237	242
100	229	234	239	244
100	231	236	241	246
100	233	238	243	248
100	235	240	245	250
100	237	242	247	252
100	239	244	249	254
100	241	246	251	256
100	243	248	253	258
100	245	250	255	260
100	247	252	257	262
100	249	254	259	264
100	251	256	261	266
100	253	258	263	268
100	255	260	265	270
100	257	262	267	272
100	259	264	269	274
100	261	266	271	276
100	263	268	273	278
100	265	270	275	280
100	267	272	277	282
100	269	274	279	284
100	271	276	281	286
100	273	278	283	288
100	275	280	285	290
100	277	282	287	292
100	279	284	289	294
100	281	286	291	296
100	283	288	293	298
100	285	290	295	300
100	287	292	297	302
100	289	294	299	304
100	291	296	301	306
100	293	298	303	308
100	295	300	305	310
100	297	302	307	312
100	299	304	309	314
100	301	306	311	316
100	303	308	313	318
100	305	310	315	320
100	307	312	317	322
100	309	314	319	324
100	311	316	321	326
100	313	318	323	328
100	315	320	325	330
100	317	322	327	332
100	319	324	329	334
100	321	326	331	336
100	323	328	333	338
100	325	330	335	340
100	327	332	337	342
100	329	334	339	344
100	331	336	341	346
100	333	338	343	348
100	335	340	345	350
100	337	342	347	352
100	339	344	349	354
100	341	346	351	356
100	343	348	353	358
100	345	350	355	360
100	347	352	357	362
100	349	354	359	364
100	351	356	361	366
100	353	358	363	368
100	355	360	365	370
100	357	362	367	372
100	359	364	369	374
100	361	366	371	376
100	363	368	373	378
100	365	370	375	380
100	367	372	377	382
100	369	374	379	384
100	371	376	381	386
100	373	378	383	388
100	375	380	385	390
100	377	382	387	392
100	379	384	389	394
100	381	386	391	396
100	383	388	393	398
100	385	390	395	400
100	387	392	397	402
100	389	394	399	404
100	391	396	401	406
100	393	398	403	408
100	395	400	405	410
100	397	402	407	412
100	399	404	409	414
100	401	406	411	416
100	403	408	413	418
100	405	410	415	420
100	407	412	417	422
100	409	414	419	424
100	411	416	421	426
100	413	418	423	428
100	415	420	425	430
100	417	422	427	432
100	419	424	429	434
100	421	426	431	436
100	423	428	433	438
100	425	430	435	440
100	427	432	437	442
100	429	434	439	444
100	431	436	441	446
100	433	438	443	448
100	435	440	445	450
100	437	442	447	452
100	439	444	449	454
100	441	446	451	456
100	443	448	453	458
100	445	450	455	460
100	447	452	457	462
100	449	454	459	464
100	451	456	461	466
100	453	458	463	468
100	455	460	465	470
100	457	462	467	472
100	459	464	469	474
100	461	466	471	476
100	463	468	473	478
100	465	470	475	480
100	467	472	477	482
100	469	474	479	484
100	471	476	481	486
100	473	478	483	488
100	475	480	485	490
100	477	482	487	492
100	479	484	489	494
100	481	486	491	496
100	483	488	493	498
100	485	490	495	500
100	487	492	497	502
100	489	494	499	504
100	491	496	501	506
100	493	498	503	508
100	495	500	505	510
100	497	502	507	512
100	499	504	509	514
100	501	506	511	516
100	503	508	513	518
100	505	510	515	520
100	507	512	517	522
100	509	514	519	524
100	511	516	521	526
100	513	518	523	528
100	515	520	525	530
100	517	522	527	532
100	519	524	529	534
100	521	526	531	536
100	523	528	533	538
100	525	530	535	540
100	527	532	537	542
100	529	534	539	544
100	531	536	541	546
100	533	538	543	548
100	535	540	545	550
100	537	542	547	552
100	539	544	549	554
100	541	546	551	556
100	543	548	553	558
100	545	550	555	560
100	547	552	557	562
100	549	554	559	564
100	551	556	561	566
100	553	558	563	568
100	555	560	565	570
100	557	562	567	572
100	559	564	569	574
100	561	566	571	576
100	563	568	573	578
100	565	570	575	580
100	567	572	577	582
100	569	574	579	584
100	571	576	581	586
100	573	578	583	588
100	575	580	585	590
100	577	582	587	592
100	579	584	589	594
100	581	586	591	596
100	583	588	593	598
100	585	590	595	600
100	587	592	597	602
100	589	594	599	604
100	591	596	601	606
100	593	598	603	608
100	595	600	605	610
100	597	602	607	612
100	599	604	609	614
100	601	606	611	616
100	603	608	613	618
100	605	610	615	620
100	607	612	617	622
100	609	614	619	624
100	611	616	621	626
100	613	618	623	628
100	615	620	625	630
100	617	622	627	632
100	619	624	629	634
100	621	626	631	636
100	623	628	633	638
100	625	630	635	640
100	627	632	637	642
100	629	634	639	644
100	631	636	641	646
100	633	638	643	648
100	635	640	645	650
100	637	642	647	652
100	639	644	649	654
100	641	646	651	656
100	643	648	653	658
100	645	650	655	660
100	647	652	657	662
100	649	654	659	664

TABLE III

PRESSURE READINGS, NOZZLE #3

22 JULY, 1946

P_e = Exhaust Chamber Pressure, mm. Hg.

P_3 = Pressure at Exit of Tube, mm. Hg.

P_2 = Pressure at Tube Inlet (Nozzle Exit), mm. Hg.

P_a = Nozzle Inlet Pressure, Atmospheric

T_1 = Inlet Temperature, Degrees F.

P_e	P_3	P_2	P_a	T_1
80.4	135.4	93.4	764.4	85
97	135			
113	135			
129	135			
140	135			
158	135			
185	137			
197	141			
206	148			
213	156			
228	181			
238	198			
246	213			
261	225			
288	255			
300	267			
358	329			

TABLE III

NO. 1000, 1940

PERCENTAGE OF ...

- 1. ...
- 2. ...
- 3. ...
- 4. ...
- 5. ...

1	2	3	4	5
100	100	100	100	100
95	95	95	95	95
90	90	90	90	90
85	85	85	85	85
80	80	80	80	80
75	75	75	75	75
70	70	70	70	70
65	65	65	65	65
60	60	60	60	60
55	55	55	55	55
50	50	50	50	50
45	45	45	45	45
40	40	40	40	40
35	35	35	35	35
30	30	30	30	30
25	25	25	25	25
20	20	20	20	20
15	15	15	15	15
10	10	10	10	10
5	5	5	5	5
0	0	0	0	0

TABLE IV
NOZZLE CHARACTERISTICS

- A_t = Cross-sectional Area at Throat
 A_e = Cross-sectional Area at Exit
 r_t = Theoretical Ratio of Exit Pressure to Inlet Pressure
 r_a = Actual Ratio of Exit Pressure to Inlet Pressure
 M_t = Theoretical Mach Number at Exit (Frictionless Flow)
 M_a = Actual Mach Number at Exit
 C_v = Assumed Velocity Coefficient
 w = Flow in Pounds per Second
 G = Flow per Unit Area at Exit, Pounds per Squarefoot per Second
 T_1 = Inlet Temperature, Degrees F.
 P_1 = Inlet Pressure, Atmospheres

<u>NOTATION</u>	<u>NOZZLE # 1</u>	<u>NOZZLE # 2</u>	<u>NOZZLE # 3</u>
A_t/A_e	1.9307	1.2870	1.9307
r_t	.1000	.2200	.1000
r_a	.1240	.2750	.1833
M_t	2.1520	1.6180	2.1520
M_a	1.8500	1.3900	1.3900
C_v	.95	.95	
P_1	1.0000	.6667	1.0000
T_1	85	85	85
w	.0676	.0676	.0676
G	25.2000	25.2000	25.2000

FIGURE XXXVI

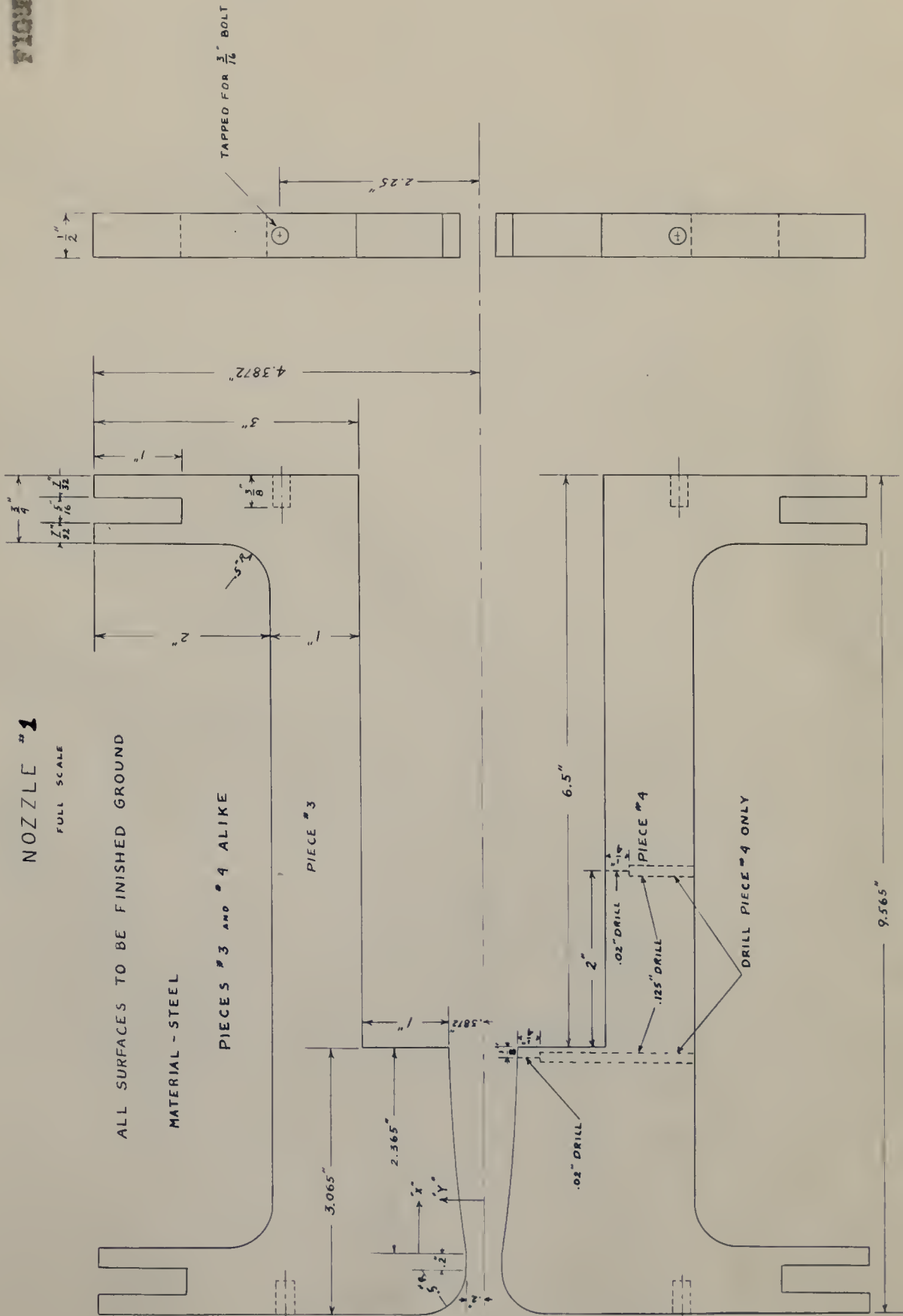
NOZZLE #1

FULL SCALE

ALL SURFACES TO BE FINISHED GRIND

MATERIAL - STEEL

PIECES #3 AND #4 ALIKE



OFFSETS NOZZLE #1

"X"	"Y"
0	.2000
.87	.3100
.95	.3200
1.00	.3250
1.10	.3360
1.20	.3470
1.30	.3550
1.40	.3635
1.50	.3690
1.60	.3740
1.70	.3790
1.80	.3830
1.90	.3850
2.00	.3860
2.10	.3865
2.20	.3870
2.24	.3872
2.365	.3872

STRAIGHT SLOPE

CURVED PORT

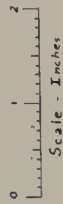
ST. HORZ. PORT: O

Scale - Inches

FIGURE XXXVII

OFFSETS NOZZLE 2

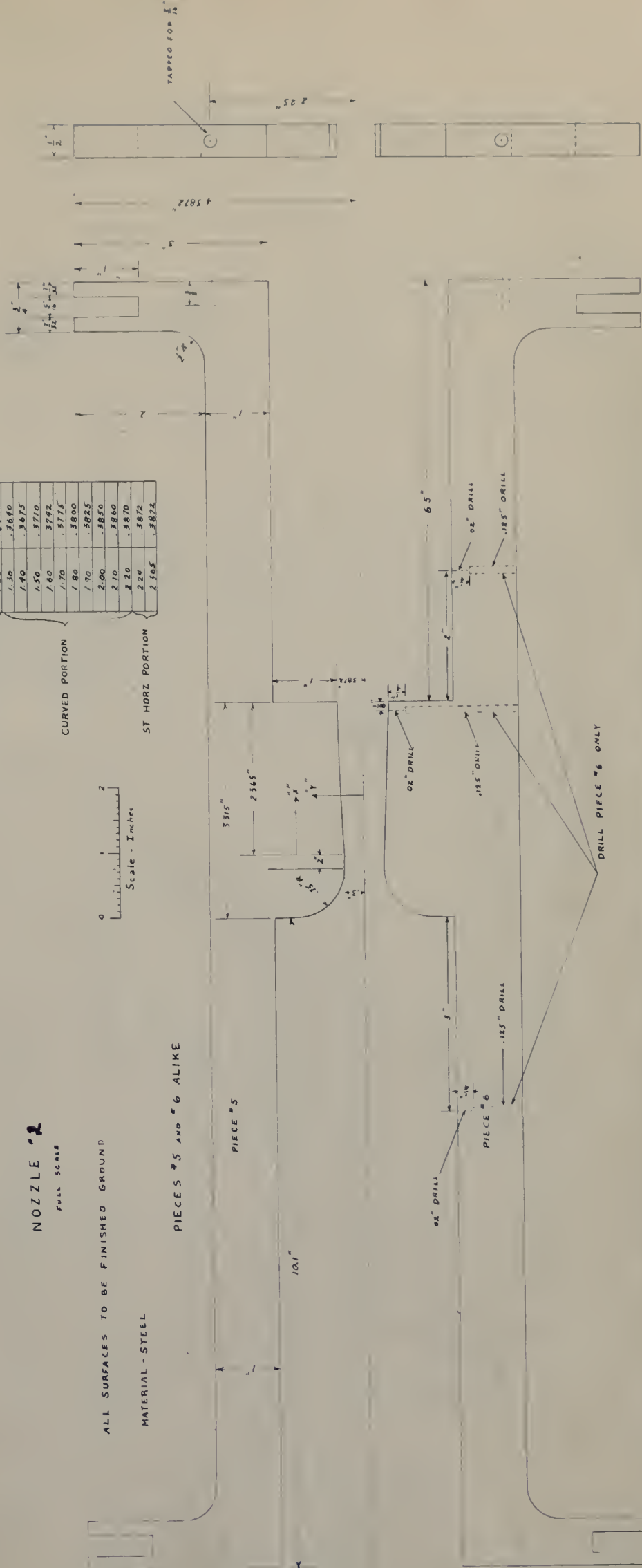
	X"	Y"
STRAIGHT SLOPING	0	.3000
	1.00	.3350
	1.70	.3550
	1.70	.3600
	1.70	.3670
CURVED PORTION	1.90	.3675
	1.90	.3710
	1.60	.3792
	1.70	.3775
	1.80	.3800
ST HORZ PORTION	1.90	.3825
	2.00	.3850
	2.10	.3860
	2.20	.3870
	2.24	.3872
	2.265	.3872



NOZZLE #2
FULL SCALE

ALL SURFACES TO BE FINISHED GROUND
MATERIAL - STEEL

PIECES #5 AND #6 ALIKE



19.915"

FIGURE XXVIII

OFFSETS NOZZLE "3"

X"	Y"
0	.2000
.87	.3100
.95	.3200
1.00	.3300
1.10	.3400
1.20	.3500
1.30	.3600
1.40	.3700
1.50	.3800
1.60	.3900
1.70	.4000
1.80	.4100
1.90	.4200
2.00	.4300
2.10	.4400
2.20	.4500
2.24	.4572
2.265	.4671

STRAIGHT SLOPING

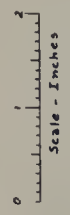
CURVED PORTION

ST HORIZ PORTION

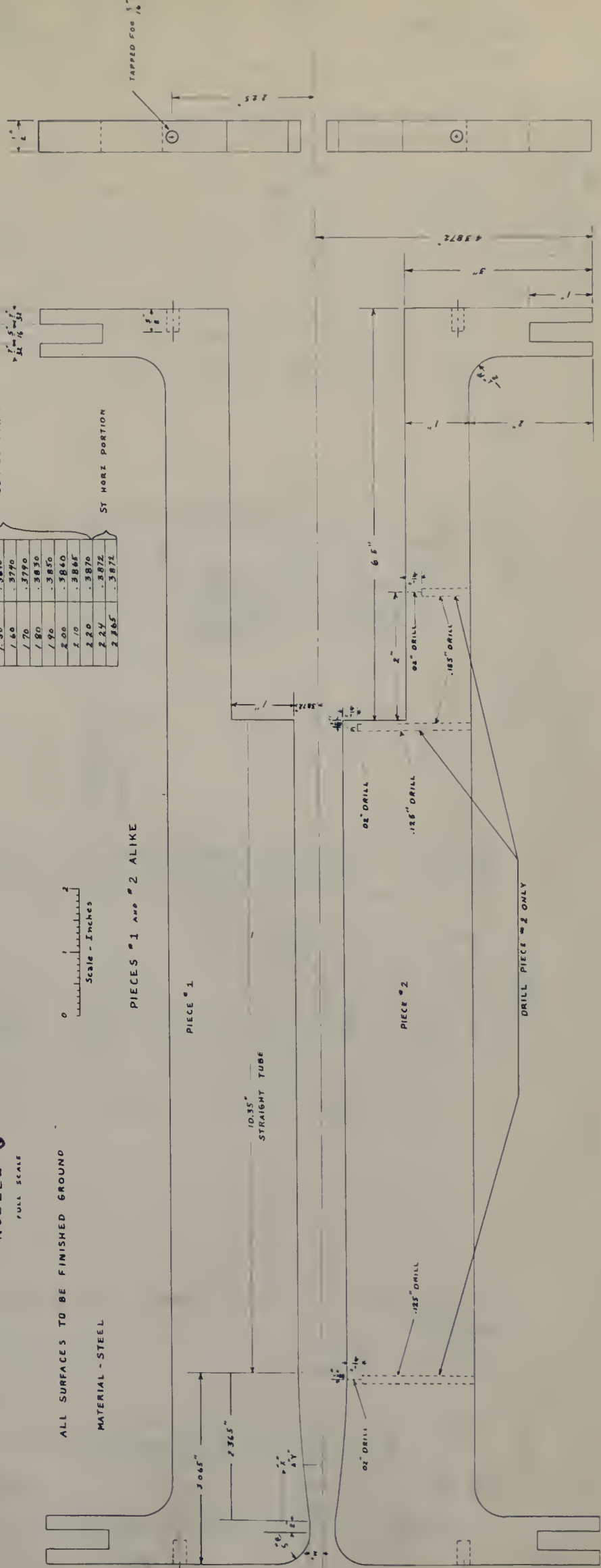
NOZZLE "3" FULL SCALE

ALL SURFACES TO BE FINISHED GROUND

MATERIAL - STEEL

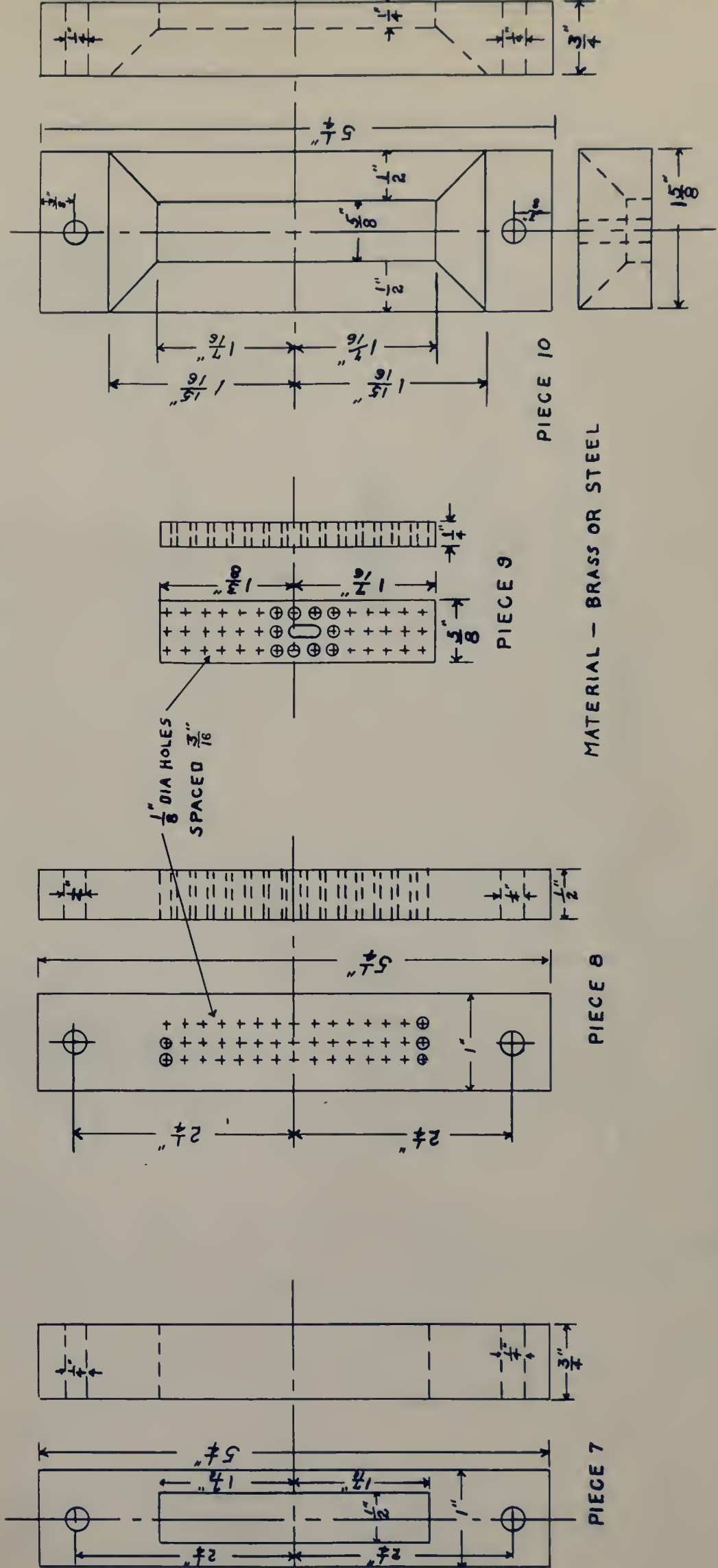


PIECES # 1 AND # 2 ALIKE



TAPPED FOR 1/8"

FIGURE XXXIX
REDUCING FITTING



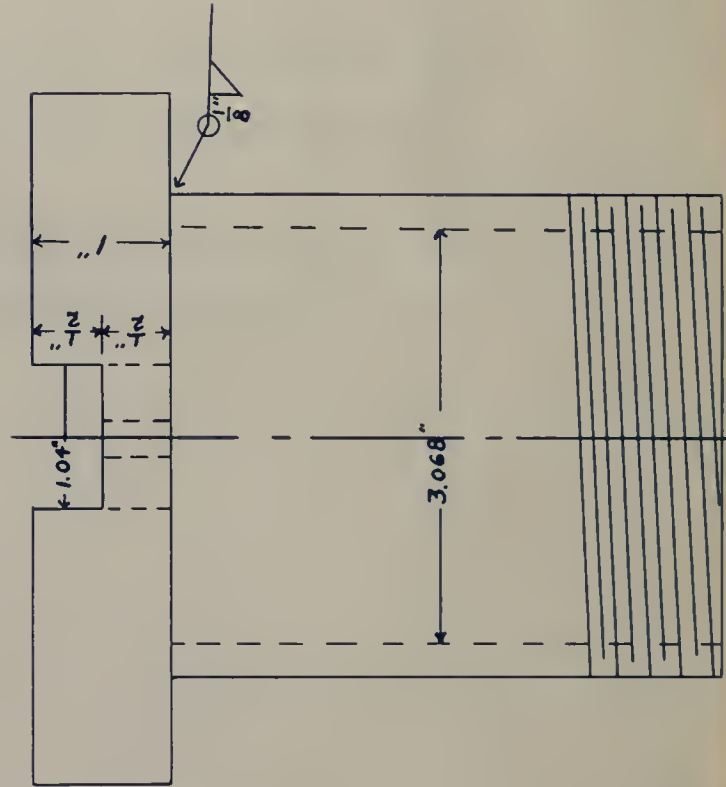
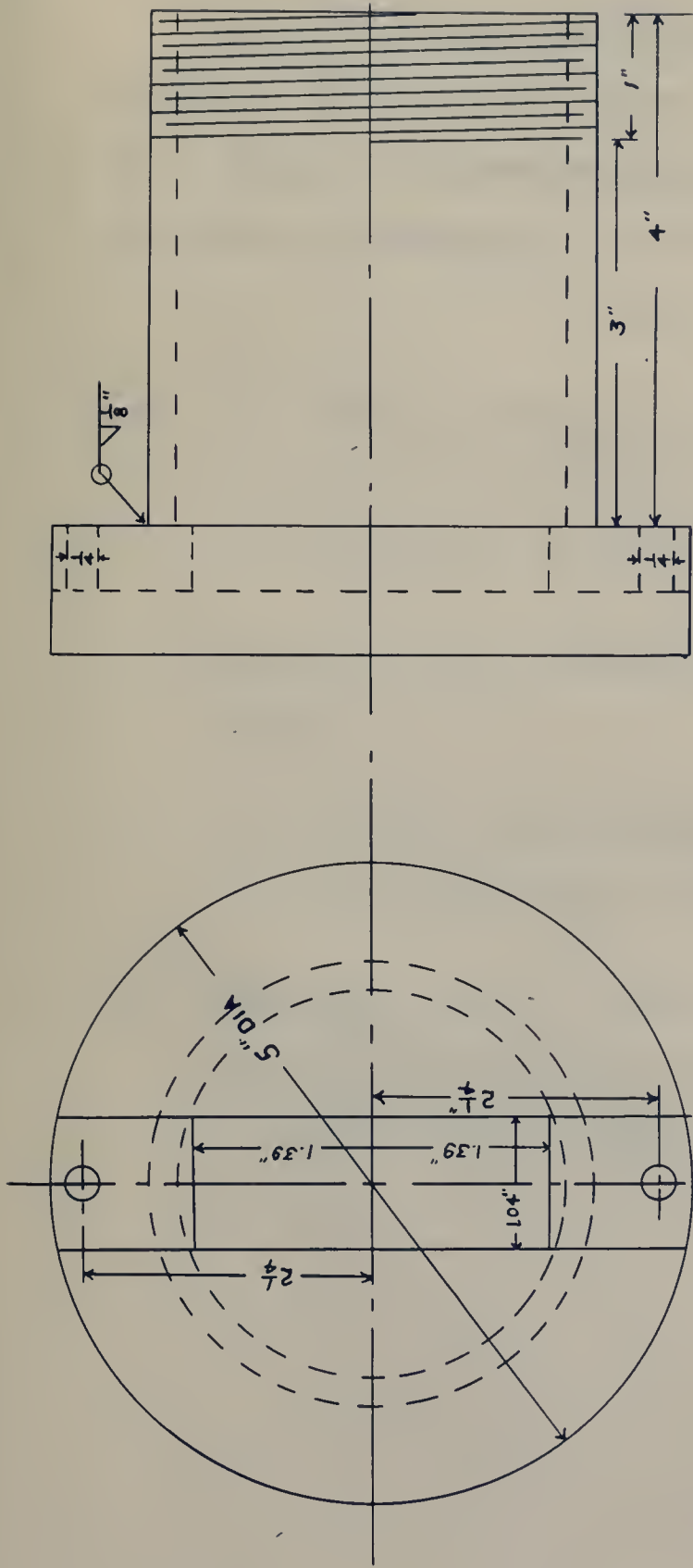
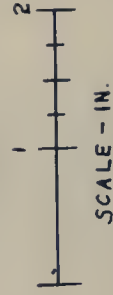


FIGURE XXXX

EXHAUST FITTING
MATERIAL- STEEL



APPENDIX C -- LOCATION OF ORIGINAL DATA

All of the original nozzle design calculations, photographic negatives, and the nozzle profiles are in the possession of Mr. E. P. Neumann of the Mechanical Engineering Department, Massachusetts Institute of Technology.

APPENDIX B -- LIST OF ORIGINAL DATA

All of the original source material, including photographs, negatives, and the source profiles are in the possession of Dr. E. P. Wehman of the Mechanical Engineering Department, Massachusetts Institute of Technology.

APPENDIX D -- BIBLIOGRAPHY

1. L.A. DeFrate, "Investigation of Supersonic Flow in Nozzles and Tubes by the Schlieren Method", Master's Thesis, M.I.T., 1943.
2. Lieut. F.M. Huron, U.S.N., and Lieut. W.P. Nelson, U.S.N., "Investigation of Supersonic Flow in Nozzles and Tubes", Master's Thesis, M.I.T., 1944.
3. Lt. Comdr. J.W. Dolan, Jr., U.S.N. and Lt. Comdr. R.V. Laney, U.S.N., "Experimental Study of Supersonic Flow in the Prandtl Nozzle", Master's Thesis, M.I.T., 1944.
4. Joseph H. Keenan, Thermodynamics, John Wiley and Sons, New York, 1941.
5. Dr. A. Stodola, Steam and Gas Turbines, Vol. II, translated by Dr. Louis C. Lowenstein, McGraw Hill Book Co., Inc., New York.
6. J.H. Keenan and E.P. Neumann, "Friction in Pipes at Supersonic and Subsonic Velocities", National Advisory Committee for Aeronautics, Technical Note # 963.

APPENDIX - BIBLIOGRAPHY

1. L.A. Huxley, "Investigation of Secondary Flow in Channels and Pipes to the Bottomward Method", Master's Thesis, M.I.T., 1947.
2. L.A. Huxley, "Investigation of Secondary Flow in Channels and Pipes", Master's Thesis, M.I.T., 1947.
3. L.A. Huxley, "Investigation of Secondary Flow in Channels and Pipes", Master's Thesis, M.I.T., 1947.
4. Joseph E. Lemmon, "Investigation of Secondary Flow in Channels and Pipes", M.S. Thesis, M.I.T., 1947.
5. L.A. Huxley, "Investigation of Secondary Flow in Channels and Pipes", Master's Thesis, M.I.T., 1947.
6. L.A. Huxley, "Investigation of Secondary Flow in Channels and Pipes", Master's Thesis, M.I.T., 1947.

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Thesis

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Perry

Schlieren observation of super-
sonic discharge.

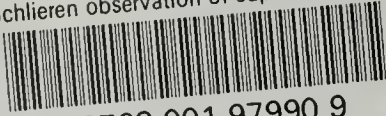
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