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By

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Measurements of the thrust produced by convergent-divergent nozzles at pressure ratios up to 20

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P. F. Ashwood, G. W. Crosse and Jean E. Goddard

SUMMARY

In continuation of a previous investigation, tests have been made on a series of convergent-divergent nozzles having an included divergence angle of 20° to obtain performance data at pressure ratios applicable to jet engines operating in the flight speed range M = 2 to M = 3.

A new test technique has been used which, by utilizing the ejector action of the jet from the test nozzle to reduce the exit pressure, enables the range of pressure ratios obtainable across the nozzle to be extended threefold. This technique, which has proved simple to apply and control, is described in detail.

The tests have shown that, as would be expected, the maximum thrust coefficient falls slightly with increasing design pressure ratio, the decrease amounting to about 1.2 per cent as between nozzles designed for pressure ratios of 10 and 25. This is shown to be in reasonable agreement with the calculated skin friction loss.

An empirical formula has been deduced for predicting the kink point, that is the lowest pressure ratio at which the shock system lies wholly beyond the nozzle exit plane. This formula supersedes that given in the report of the earlier tests which was based on a limited range of design and test pressure ratios and which could not confidently be extrapolated beyond the test limits.

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1.0 Introduction

A detailed investigation of the performance of fixed geometry convergent-divergent propelling nozzles and the influence on the thrust of design pressure ratio and divergence angle is given in Reference 1. Due to the limited test facilities then available the work had to be confined to pressure ratios below 9, a value which is below that reached on turbo-jet or ram jet engines flying at supersonic speeds in the range M = 2 to M = 3. It was therefore decided to extend the test range to cover pressure ratios up to 20. The present Memorandum describes this work.

2.0 Test equipment

2.1 Thrust rig

Since the maximum air supply pressure attainable at the rig was limited to 9 atmospheres, it was evident that pressure ratios across the nozzle in excess of 9 could only be obtained by reducing the pressure at the nozzle exit below that of the atmosphere. This could have been done either by means of exhaust pumps or by using a separately driven ejector, but both of these methods would have required the operation of additional plant and might have raised complicated control problems. It was therefore decided that an attempt would be made to use the ejector action of the jet from the test nozzle to reduce the back pressure. This scheme had the advantage of simplicity, but in the initial design stage it was not known how tractable it would prove in operation and whether the accuracy of the thrust measurement would be adversely affected. Experience has shown that neither of these fears were warranted.

Drawings of the thrust rig and an enlarged detail of the depression chamber are given in Figure 1. The general layout of the rig will be evident from this drawing.

Air from the plant compressor passed in turn through an orifice plate air meter, a control valve and thence to the duct to which the nozzle was attached. This duct was arranged with its axis horizontal, the air being led into the duct through a right angle bend. Two flexible joints were provided in the vertical supply pipe to render the nozzle and its attendant inlet duct free to move along a horizontal axis. Movement was restrained by means of a spring balance and since the entering air had no component of momentum along the line of action of the thrust, the force recorded by the balance was equal to the gross thrust of the jet, allowance being made for the pressure thrust on the rig due to the nozzle being enclosed in a region in which the pressure was below that of the atmosphere (see Appendix II for details of this correction).

Although the flexible joints in the supply pipe were initially sealed to prevent air leakage, it was found that at high air pressures the seals became stiff and imposed forces on the duct which affected the accuracy of the thrust measurement. To overcome this difficulty all the thrust measurements were made with the seals removed. Some leakage then occurred but the only effect of this was to reduce the maximum pressure that could be obtained at the nozzle. When mass flow measurements were required both flexible joints were removed and replaced by a rigid length of ducting. With the rig in this condition the thrust could not of course be measured.

The design of the depression chamber, mixing duct and diffuser were based on the results of tests at N.G.T.E. on ejectors for use as test plant, emphasis being given to simplicity of construction. A conical shape was chosen for the mixing duct entry section for this reason to ease manufacture. It was known that the axial position of the test nozzle relative to the mixing tube entry would exert an appreciable effect on the ejector performance and a compromise had to be reached between achieving the required depression whilst minimising the influence of the mixing duct entry on the distribution of static pressure in the plane of the nozzle exit. It was finally decided to place the nozzle exit and the commencement of the mixing duct contraction in the same plane and the rig was assembled in this way with the nozzle designed for a pressure ratio of 25. However, the alignment was not maintained and as the design pressure ratio was reduced by cutting back the divergent portion of the nozzle so the axial distance between the exit and the mixing duct contraction increased. The maximum distance was 2.46 in. and this occurred with the nozzle having a design pressure ratio of 10.

In order to avoid introducing other forces which might have disturbed the accuracy of the thrust measurements no attempt was made to seal the gap around the air supply pipe where it entered the depression chamber. Leakage at this point was not serious and its only effect was to reduce the maximum pressure ratio obtainable by reducing the depression. To help keep leakage to a minimum the gap was kept as small as practicable (0.050 in.) and a long sealing length was provided. The whole assembly was carefully aligned to ensure concentricity and the supply pipe held in the required position by four ball bearings mounted on adjustable supports.

2.2 Test nozzle

The profile of the test nozzle was geometrically similar to that of the Group I nozzles of Reference 1 and consisted of a circular arc entry blending into a conical divergent portion of 20° included angle. The nominal throat diameter was 2 in., that is twice the value used in the tests described in Reference 1.

As originally constructed the nozzle had an area ratio of 3.32, that is the value corresponding (for $\gamma = 1.40$) to a pressure ratio of 25. It was cut back successively to give design pressure ratios of 20, 15 and 10.

The internal profile of the nozzle was carefully machined and polished to eliminate major surface irregularities.

For comparison a conical convergent nozzle having an included angle of 20° was also tested.

Drawings of the two nozzles are given in Figure 2.

3.0 Test procedure

As explained in Section 2.1, no attempt was made to present all leakage from the flexible joints in the air supply duct and therefore the air mass flow and thrust measurements had to be made separately.

In the present investigation only one mass flow calibration was made and this was done with the nozzle set to a design pressure ratio of 10. It was reasoned that the addition of a greater length of divergence would not affect the flow since under the conditions of test the throat was always choked.

For each setting of the air supply pressure measurements were taken of the total pressure and temperature at entry to the nozzle, the static pressure in the depression chamber and either the air mass flow or the thrust. This was done for a range of supply pressure varying from about 1.5 atmospheres to the maximum obtainable.

At the conclusion of the tests a check was made with the nozzle blanked off to determine if the balance reading was effected by the air pressure in the inlet duct. A small effect was observed which increased almost linearly from zero, at an air supply pressure of $2\frac{1}{2}$ atmospheres, to about $3\frac{1}{2}$ lb, that is about 1 per cent of the thrust balance reading, at a pressure of 7 atmospheres. A calibration curve was obtained and a correction applied to all the thrust measurements.

The reason for this sensitivity to inlet pressure is obscure, but it is thought likely that it was due to a horizontal movement of the inlet ducting above the uppermost flexible joint. Since the axial position of the nozzle was kept fixed to within ± 0.001 in., any movement of the top ducting would have the effect of tilting the axis of the duct containing the flexible joints away from the vertical and the horizontal component of the weight of the pipework would therefore be recorded on the balance.

4.0 Performance of depression chamber

In addition to the main programme of nozzle tests a brief investigation was made to determine the effect on the performance obtainable from the depression chamber of simple changes in the geometry of some of its components. The design pressure ratio of the nozzle used for these tests was 10 and it was set in an axial position such that its exit plane was 2.46 in. upstream from the commencement of the contraction to the mixing tube. The temperature of the air supply varied over the range 30 to 35°C.

The first test was made with a 6 in. diameter mixing tube 36 in. long with no outlet diffuser fitted. It was found that with a supply pressure of 6 atmospheres the pressure inside the chamber was reduced to 4.7 lb/sq.in. abs., giving a pressure ratio across the nozzle of 18.5. A diffuser having an area ratio of 4 was then fitted to the mixing tube and this reduced the pressure in the chamber by a further 1.2 lb/sq.in. and enabled a pressure ratio of 25 to be achieved.

A test was then made with a 4 in. diameter mixing tube 24 in. long fitted with a diffuser having an area ratio of 4. To accommodate the 4 in. mixing tube diameter, the original conical entry section was extended by the addition of a short adaptor piece, an expedient which had the effect of increasing the distance between the nozzle exit plane and the commencement of the mixing section. With this arrangement a considerably greater depression could be obtained at low supply pressures. However, it was found that instead of decreasing continuously, the depression chamber pressure passed through a minimum value of 2.5 lb/sq.in.abs. at an inlet pressure of 4.1 atmospheres and any further increase of the supply pressure beyond this point caused the chamber pressure to rise. As a result the maximum overall pressure ratio obtainable across the nozzle was not as great as that reached with the 6 in. diameter mixing tube and diffuser.

An additional test was made using the 4 in. diameter mixing tube, the nozzle being moved downstream so that its exit plane was in line with the mixing tube entry. This change improved the performance at low supply pressures but at the expense of the maximum pressure ratio obtainable.

Even had this not been so, such a configuration would not have been adopted since the close proximity of the mixing tube to the nozzle exit would doubtless have disturbed the jet flow and introduced inaccuracies into the test measurements.

The performance obtained with the various ejector configurations tested is given in Figure 3.

When the convergent nozzle was tested in conjunction with the 6 in. diameter mixing tube the ejector action was found to be slightly less effective, but more seriously it was observed that above a critical inlet pressure (about $5\frac{1}{2}$ atmospheres) an instability occurred which made all the manometers and the thrust indicator fluctuate wildly and at the same time caused loud explosion-like reports. If the inlet pressure was increased above this critical value the reports occurred more frequently until at the highest pressure the noise took the form of a continuous loud buzzing. Intermittent reverse flow into the outlet end of the mixing tube was observed and it was found that the instability could be reduced by restricting the outlet with the hands. It was thought that the instability was due to the nozzle exit being too far upstream (the exit plane was 2.86 in. from the mixing tube convergence) and so a test was made with the nozzle moved closer to the mixing tube. This made little difference and testing was therefore confined to the region in which instability was not evident. This limited the maximum pressure ratio obtainable across the nozzle to approximately 13.

5.0 Thrust measurement check

To check that the rig was functioning satisfactorily, thrust measurements were made both with the mixing tube and its entry section removed (i.e. with the nozzle discharging to atmosphere) and with the tube in position. The design pressure ratio of the nozzle used for these tests was 25. The maximum pressure ratio that could be achieved with the nozzle discharging to atmosphere was 6.3 but with the 6 in. diameter mixing tube fitted this was extended to 19.

The results are shown in Figure 4, in which the non-dimensional thrust coefficient, C_F , is plotted against the pressure ratio across the nozzle. It will be seen that the experimental points from the two tests lie on the same curve and on this evidence it was concluded that the thrust measuring technique and the procedure adopted for reducing the test data were satisfactory.

6.0 Discussion of results

6.1 Mass flow characteristics

A curve showing the variation of discharge coefficient with pressure ratio is shown in Figure 5. The plotted points were obtained with the depression chamber assembled in three different configurations as noted on the graph.

A constant mean value of 0.985 best fits the test results, all but two of the points having a scatter less than ±0.7 per cent.

The mean value of discharge coefficient for the convergent nozzle was found to be 0.978.

6.2 Thrust characteristics

After all the necessary corrections had been applied to the test results, values of the non-dimensional thrust, F/A_2P_{1t}, and the non-dimensional thrust coefficient, C_F, were calculated. It is shown in Reference 1 that for conditions where the shock system lies beyond the exit plane a linear relationship exists between F/A_2P_{1t} and the ratio of ambient to inlet total pressure, P_a/P_{1t}, the limiting condition (i.e. the largest value of P_a/P_{1t}) to which the linearity extends being termed the kink point.

The best straight line was drawn through the experimental points and values picked off this line were used to calculate mean values of C_F. The results are shown in terms of the non-dimensional thrust coefficient, C_F, in Figure 6 and in terms of the specific thrust coefficient, C_T, in Figure 7. On both these graphs the points were plotted from the actual test measurements whilst the curves were drawn through mean values calculated as has just been described.

The results are as expected, the thrust coefficient curve for each nozzle passing through a maximum value at a pressure ratio which coincides as closely as can be determined with the design value. The coefficients decrease more rapidly when the supply pressure is reduced below the design value (conditions of over-expansion) than they do when the pressure is increased above design (conditions of under-expansion). This agrees with previous work and it confirms the view that in order to maintain high efficiency over the widest possible range of operation the design pressure ratio should be chosen to be slightly below the maximum pressure at which it is desired to operate the nozzle.

The curves of Figures 6 and 7 show a definite tendency for the design point thrust coefficients to decrease as the design pressure ratio increases. This result, although contradicting that of Reference 1, is as expected since the nozzle length increases continuously with design pressure ratio and the losses due to skin friction will therefore do likewise. It is thought that whilst the explanation of the opposite effect which was observed in the tests reported in Reference 1 could be that it was due to the influence of Reynolds number, it is more likely that it was the result of inaccuracies arising from the necessity for obtaining the design point performance of the two nozzles having the highest design pressure ratios by extrapolation owing to the limited maximum test pressure ratio then available.

In order to investigate the magnitude of the losses due to skin friction calculations were made using the method of Reference 2 to determine the maximum thrust coefficients for several nozzles having a constant included divergence angle of 20° and with design pressure ratios ranging between 10 and 30. In these calculations the flow up to the throat was assumed to be isentropic but in the diverging portion a friction factor of 0.002 was used. This value is that appropriate to the mean of the Reynolds numbers between the throat and exit planes. For the range of conditions covered by the tests these mean values (based on the nozzle diameter) varied from about 2.3×10^6 to 4.5×10^6.

The results are shown plotted in Figure 8, the full line showing the effect of friction alone and the dotted line the combined effects of friction and non-axial exit flow. The latter effect was calculated assuming the exit streamlines to radiate from the apex of the conical divergent section and the exit velocity to be constant over a spherical surface centred on the apex and intersecting the nozzle exit. On this basis the divergence coefficient for a nozzle having an angle of 20° is 0.992.

The experimentally determined maximum values of C_T are also shown plotted in Figure 8 and it will be seen that most of the points fall between the two lines.

6.3 Determination of kink points

The kink point pressure ratios were determined directly from the non-dimensional thrust graphs. These values, together with the slopes of the linear part of these graphs, which on the basis of one-dimensional theory should be numerically equal to the area ratio of the nozzle, are tabulated below.

Design pressure ratio	Area ratio	Slope of non-dim. thrust graph	Kink point pressure ratio
10	1.931	1.86	3.88
15	2.614	2.38	6.12
20	2.899	2.76	7.00
25	3.321	3.12	9.30

The results are shown graphically in Figures 9 and 10. It will be seen from Figure 9 that the empirical rule given in Reference 1 for predicting the kink point, which was based on tests on nozzles having design pressure ratios between 4 and 12, becomes progressively less accurate as the nozzle design pressure ratio increases. A rule which fits all the available data more closely is:-

$$\text{Kink point pressure ratio} = 1 + \tfrac{1}{3}\,(\text{design pressure ratio}).$$

Reference 3 reports tests made to determine the kink points of a series of two-dimensional nozzles from measurements of the wall static pressure. The mean curve obtained from these tests has been superimposed on Figure 9 and it will be seen that the values are slightly higher than those for axi-symmetric nozzles obtained from thrust measurements. This is almost certainly due to the different experimental techniques used (there is evidence in unpublished tests to show that the kink points of axi-symmetric nozzles determined from static pressure measurements agree with the mean curve of Reference 3) and a rational explanation can be advanced as follows.

In the tests described in Reference 3, the kink points were obtained by extrapolating to the exit plane a curve linking the applied pressure ratio to the 'shock position', determined from pressures measured by means of static taps at the nozzle wall. The 'shock position' thus deduced was thus the furthermost upstream position at which the influence of the shock system could be detected at the wall. This technique assumes that when the shock system is located beyond the nozzle exit it has no influence on the pressure at the wall just inside the nozzle. If, however, this assumption was incorrect, and there is some evidence from recent shadowgraph pictures taken at N.G.T.E. to suggest that it is, then the extrapolation should not be continued as far as the exit plane. The discrepancy between the kink points obtained from pressure measurements and from thrust could then be resolved, at least qualitatively.

Examination of the results published in Reference 3 has shown that in order to make the two values of kink point pressure ratio agree numerically, the influence of the exit shock system would have to extend into the nozzle by an amount which increased progressively with the exit Mach number. The distances required would be of the order of 0.04 in. at a Mach number of 1.5 and 0.12 in. at a Mach number of 2. As yet no experimental data is available to enable this hypothesis to be checked.

Figure 10 shows the slopes of the non-dimensional thrust graphs plotted against the nozzle area ratio. It will be seen that with increase of design pressure ratio the measured slope departs progressively from the value predicted by one-dimensional theory. This suggests that the effective exit area is less than the actual area, a reduction which could be due to a thickening of the boundary layer in the diverging portion of the nozzle. Calculation shows that the rate of growth of boundary layer thickness necessary to account for the observed thrust curve slopes is approximately 0.01 in. per in. of nozzle length. This value has been compared with those calculated using an empirical formula quoted in Reference 4. This states that for design Mach numbers below 2.5:-

$$\text{Mean rate of boundary layer growth} = 0.29/Re^{1/5}$$

where Re is the Reynolds number based on the nozzle design exit Mach number and distance from throat to run-out position.

The calculated values, although of the right order, do not agree exactly with those required to give the measured thrust curve slopes. The present work, however, was done with conically divergent nozzles and this, combined with the fact that humid air was used for all the tests, resulted in the supersonic part of the flow being far from shock-free. It is possible, therefore, that the effect of these shocks was to cause the boundary layer to behave differently from what it would do in a correctly shaped nozzle expanding dry air.

In Reference 3 it is shown that for two-dimensional nozzles, the ratio of the inlet total to the exit static pressures is 0.92 of the value which would obtain if the expansion were isentropic. If we assume that this ratio also applies to axi-symmetric nozzles, then the area ratio of the "equivalent" nozzle can be calculated using conventional one-dimensional relationships. The values thus obtained are given in the table below and it will be seen that they agree very closely with the measured slopes of the non-dimensional thrust curves.

Design pressure ratio	Design pressure ratio of "equivalent" nozzle	Area ratio of "equivalent" nozzle	Slope of non-dim. thrust graph
10	9.2	1.85	1.86
15	13.8	2.33	2.38
20	18.4	2.75	2.76
25	23.0	3.16	3.12

7.0 Conclusions

A simple apparatus has been evolved to enable nozzles to be tested to a pressure ratio equal to about three times that of the air supply available. The apparatus, which utilises the ejector action of the jet from the nozzle to lower the nozzle exit pressure, has proved easy to control and the accuracy of thrust measurement compared with that obtained with a free jet test has been unimpaired.

It has been confirmed that the peak thrust coefficient of a divergent nozzle occurs at a pressure ratio which is identical with the isentropic design value to within the limits of experimental accuracy.

The maximum thrust coefficient falls slightly with increasing design pressure ratio, the decrease amounting to about 1.2 per cent as between nozzles designed for pressure ratios of 10 and 25. This effect is as expected and can be accounted for quantitatively by the increased skin friction due to the greater length of the diverging portion of the nozzle.

As empirical formula for predicting the kink point (i.e. the lowest pressure ratio for which the shock system lies wholly beyond the nozzle exit plane) has been deduced; this is:-

$$\text{Kink point pressure ratio} = 1 + \tfrac{1}{3}\,(\text{design pressure ratio}).$$

This formula supersedes that given in Reference 1 which was based on a limited range of design and test pressure ratios.

Discrepancies have been observed between kink points determined from thrust measurements and from wall static pressures. Although all the thrust data was confined to tests on axi-symmetric nozzles whilst the majority of the pressure data was obtained from two-dimensional ones, there is evidence to suggest that the discrepancy is due to experimental technique rather than to a fundamental difference in the behaviour of axi-symmetric and two-dimensional nozzles. A hypothesis has been advanced which explains the difference qualitatively, but there is as yet no experimental data available to enable this to be checked.

The slopes of the non-dimensional thrust graphs - $(F/A_2P_{1\,t})$ plotted against $(P_a/P_{1\,t})$ - have been determined and compared with the nozzle exit/throat area ratios. Although one-dimensional theory states that the two values should be numerically equal, the test results show that the measured slope becomes progressively less than the area ratio as the design pressure ratio increases. It is suggested that this is due to the influence of boundary layer thickness, an idea supported by the fact that the order of magnitude required to explain the difference is similar to that calculated using an empirical formula obtained from tests in supersonic wind tunnel nozzles. However, the use in the present tests of humid air and conically divergent nozzles undoubtedly influences the flow in the supersonic flow region and this may well account for the discrepancy between the two values.

The slopes of the non-dimensional thrust graphs can be predicted exactly using the concept of an "equivalent" nozzle which has a design pressure ratio less than that of the nozzle being considered. The two design pressure ratios can be related by a factor obtained from pressure distribution tests on two-dimensional nozzles.

REFERENCES

No.	Author(s)	Title, etc.
1	P. F. Ashwood D. G. Higgins	The influence of design pressure ratio and divergence angle on the thrust of convergent-divergent propelling nozzles. CP. 325, March 1955.
2	W. R. Thomson	The thermodynamics of frictional resisted adiabatic flow of gases through ducts of constant and varying cross section. CP. 158, September 1952.
3	P. F. Ashwood G. W. Crosse	The influence of pressure ratio and divergence angle on the shock position in two-dimensional, over-expanded convergent-divergent nozzles. CP. 327, August 1956.
4	E. W. E. Rogers B. M. Davis	A note on turbulent boundary-layer allowances in supersonic nozzle design. A.R.C. 18,490, June 1956.

APPENDIX I

Notation

A	Flow area within nozzle
A_s	Area of inlet duct at seal into depression chamber
C_D	Discharge coefficient
C_F	Non-dimensional thrust coefficient
C_T	Specific thrust coefficient
F	Thrust
g	Gravitational acceleration
P_a	Ambient pressure
P_c	Static pressure within depression chamber
P_s	Static pressure
P_t	Total pressure
T_t	Total temperature
V	Air velocity
W	Air mass flow
γ	Specific heat ratio

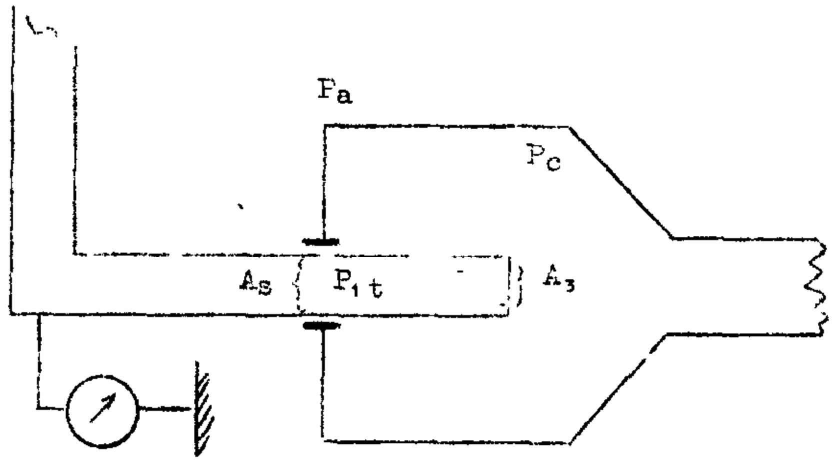
Subscripts $_{1}$, $_{2}$ and $_{3}$ are used to denote conditions at the nozzle entry, throat and exit respectively.

Derived parameters

$F/A_2P_{1\,t}$	Non-dimensional thrust
$F/W\sqrt{T_{1\,t}}$	Specific thrust

APPENDIX II

Derivation of correction to indicated thrust



Using the notation given in Appendix I and with reference to the above diagram

$$\text{Reading on thrust balance} = \frac{WV_3}{g} + (P_{3\,s} - P_a)A_3 - (P_a - P_c)(A_s - A_3)$$

$$= \frac{WV_3}{g} + (P_{3\,s} - P_c)A_3 - (P_a - P_c)A_s \quad \ldots \ldots \quad (1)$$

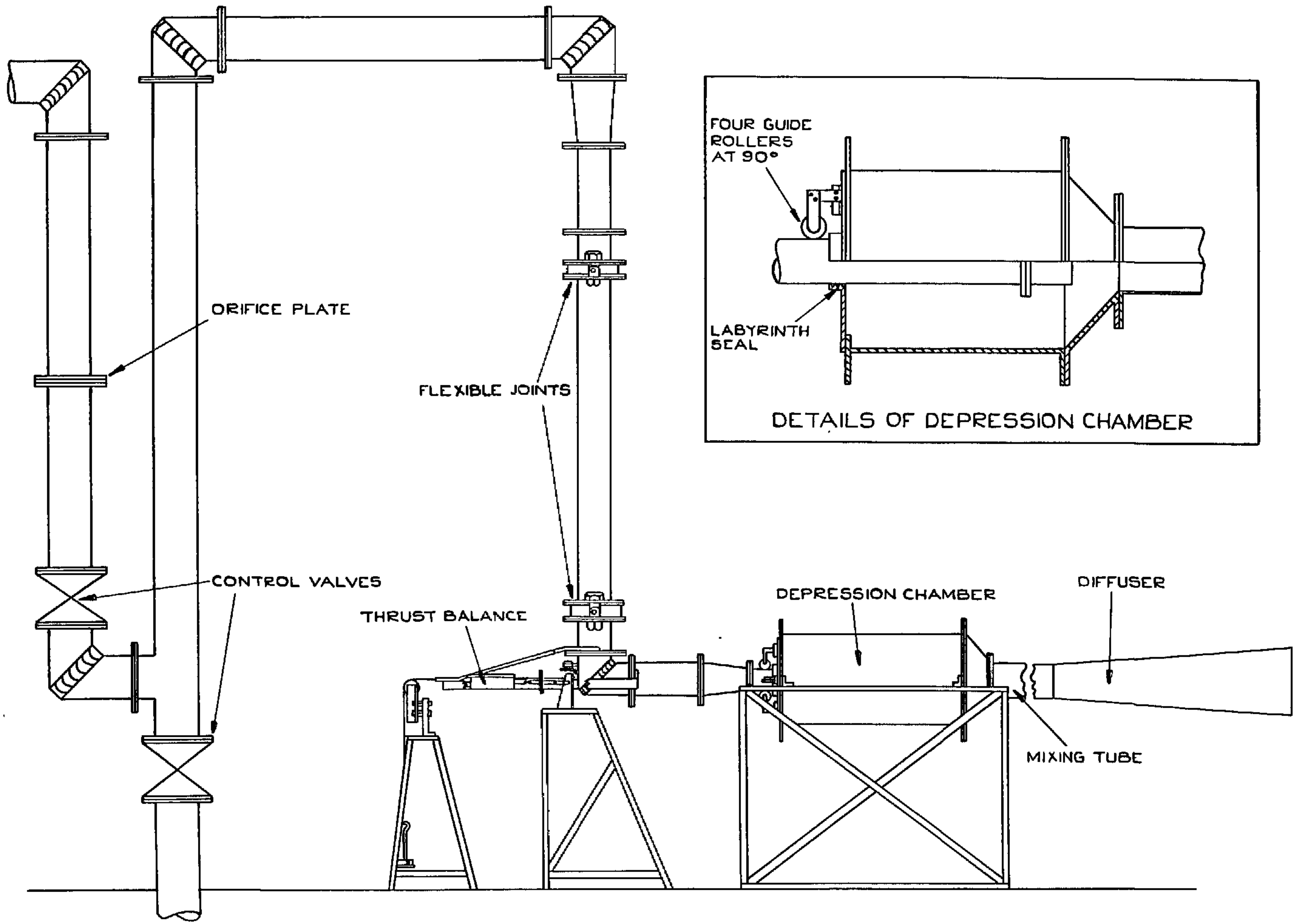
$$\text{Now, nozzle gross thrust} = \frac{WV_3}{g} + (P_{3\,s} - P_c)A_3 \quad \ldots \ldots \ldots \ldots \quad (2)$$

Hence, from Equations (1) and (2),

$$\text{Nozzle gross thrust} = \text{Reading on thrust balance} + (P_a - P_c)A_s$$

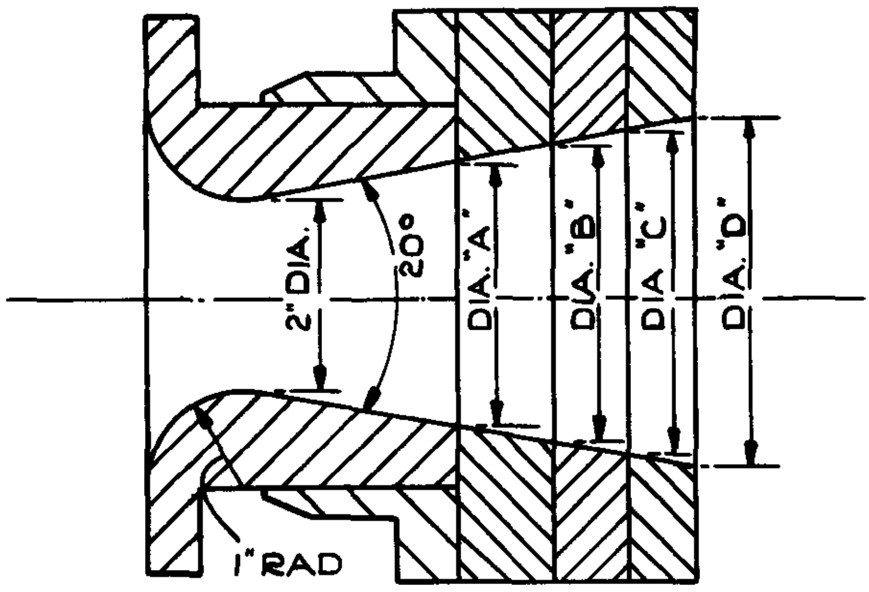
$$\ldots \ldots \ldots \ldots \quad (3)$$

FIG. I

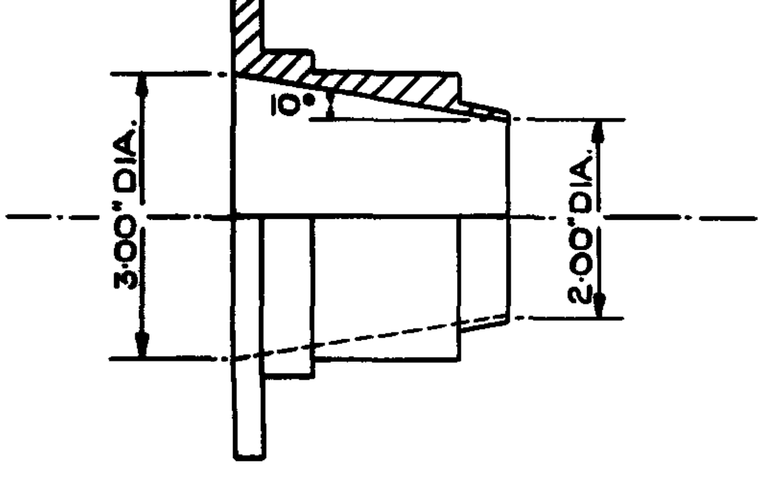


THRUST RIG AND DEPRESSION CHAMBER

FIG 2

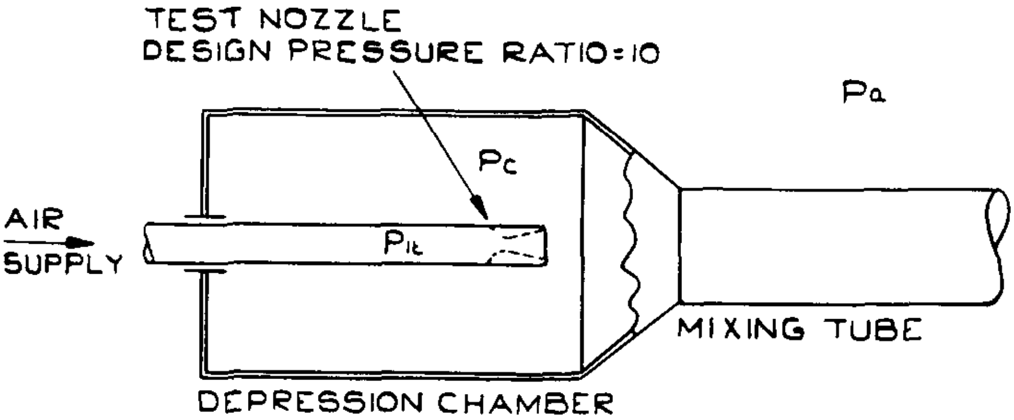


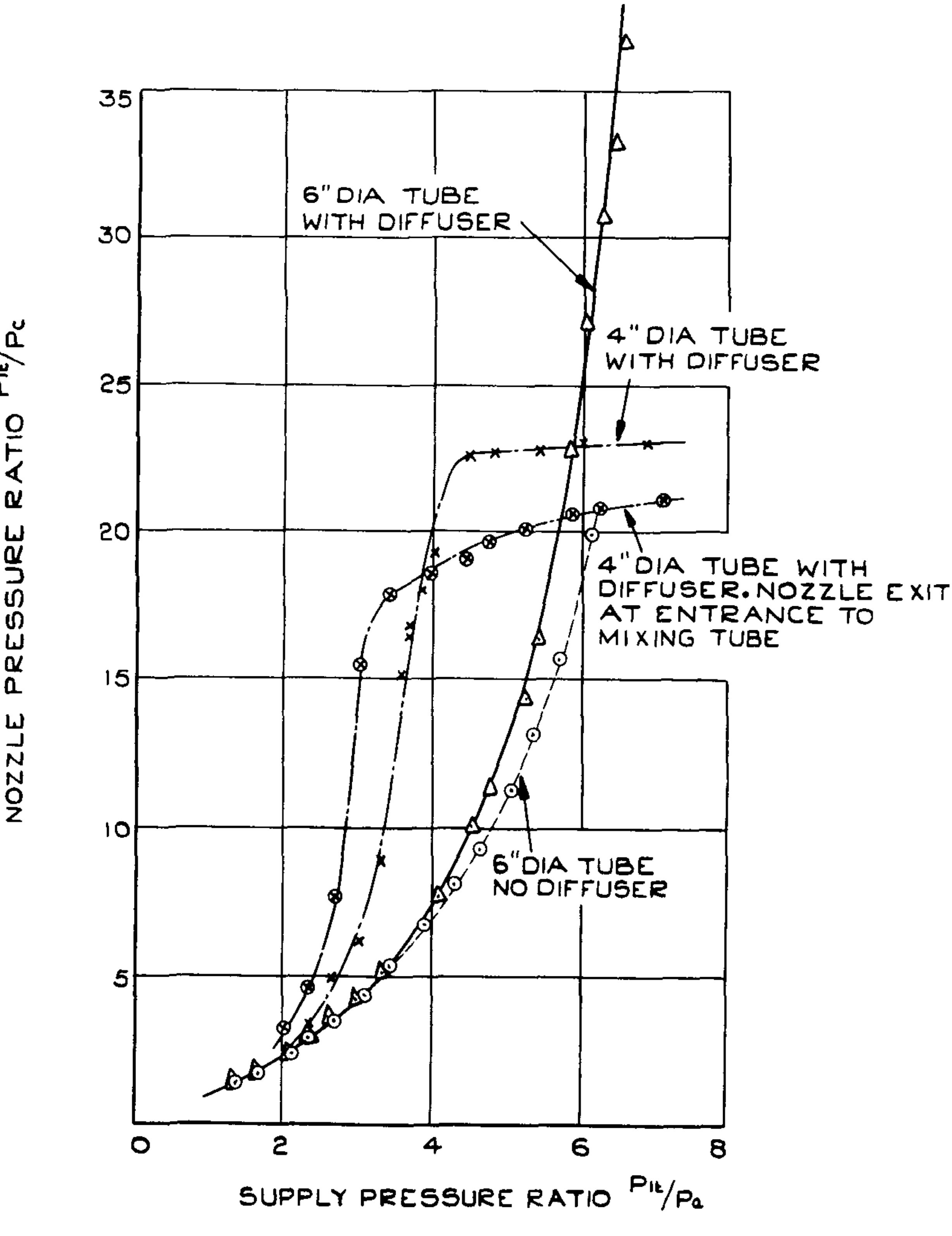
SECTION	EXIT DIA. (INS.)	AREA RATIO	DESIGN PRE RATIO	DESIGN EXIT MACH No
A	2·780	1·932	10	2·161
B	3·124	2·440	15	2·415
C	3·408	2·904	20	2·599
D	3·648	3·327	25	2·746



PROFILES OF TEST NOZZLES

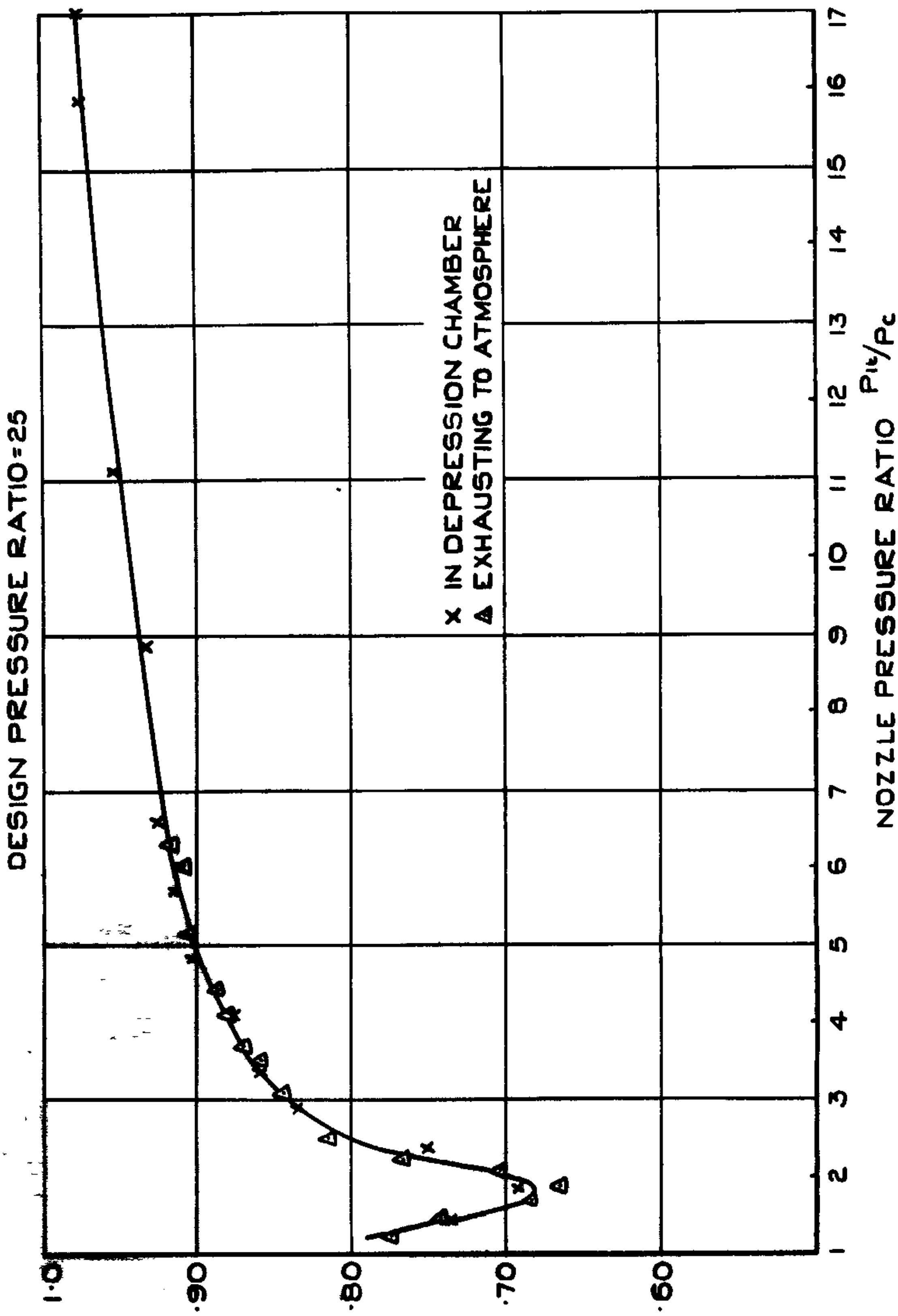
FIG 3





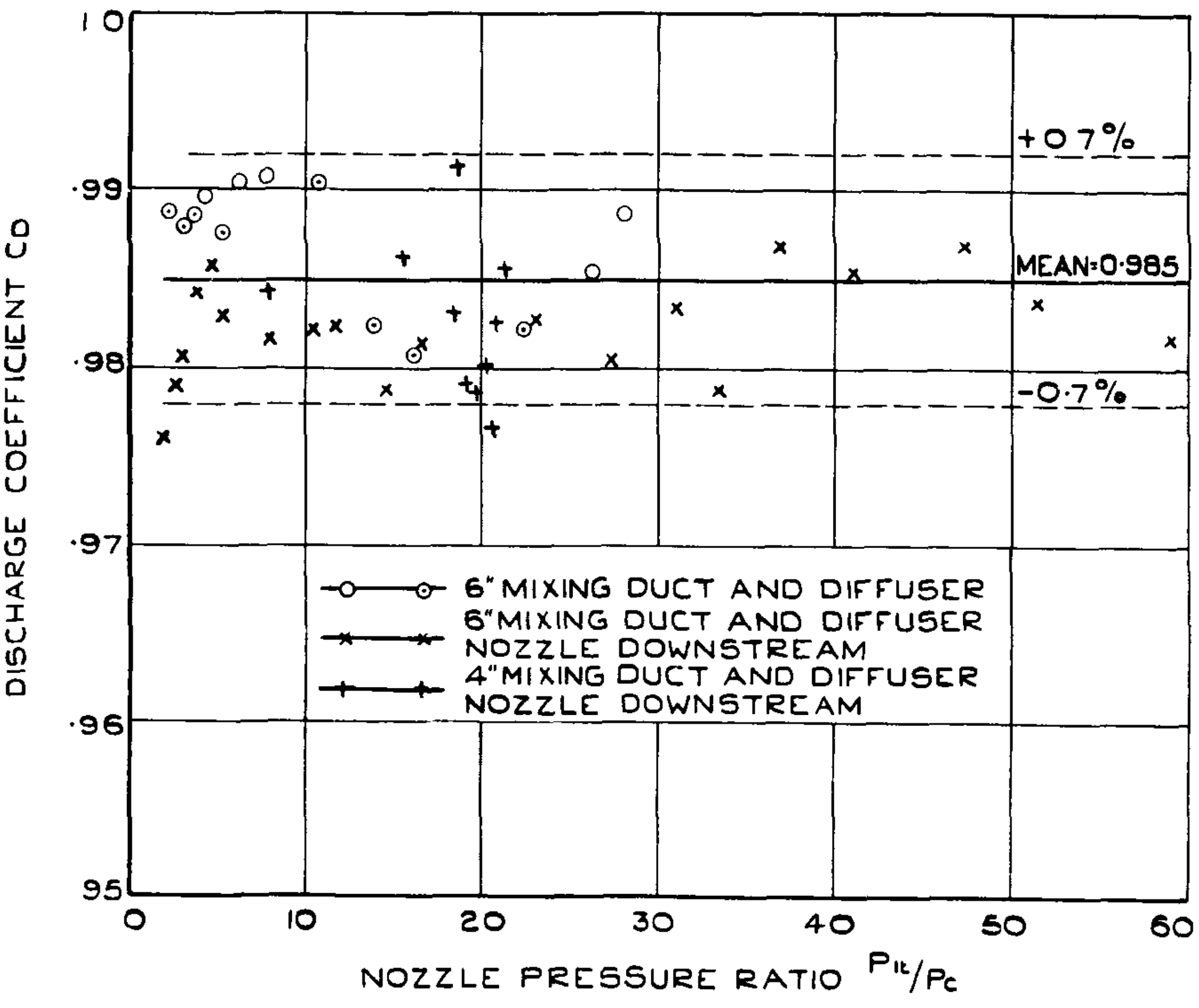
PERFORMANCE OF DEPRESSION CHAMBER

FIG. 4



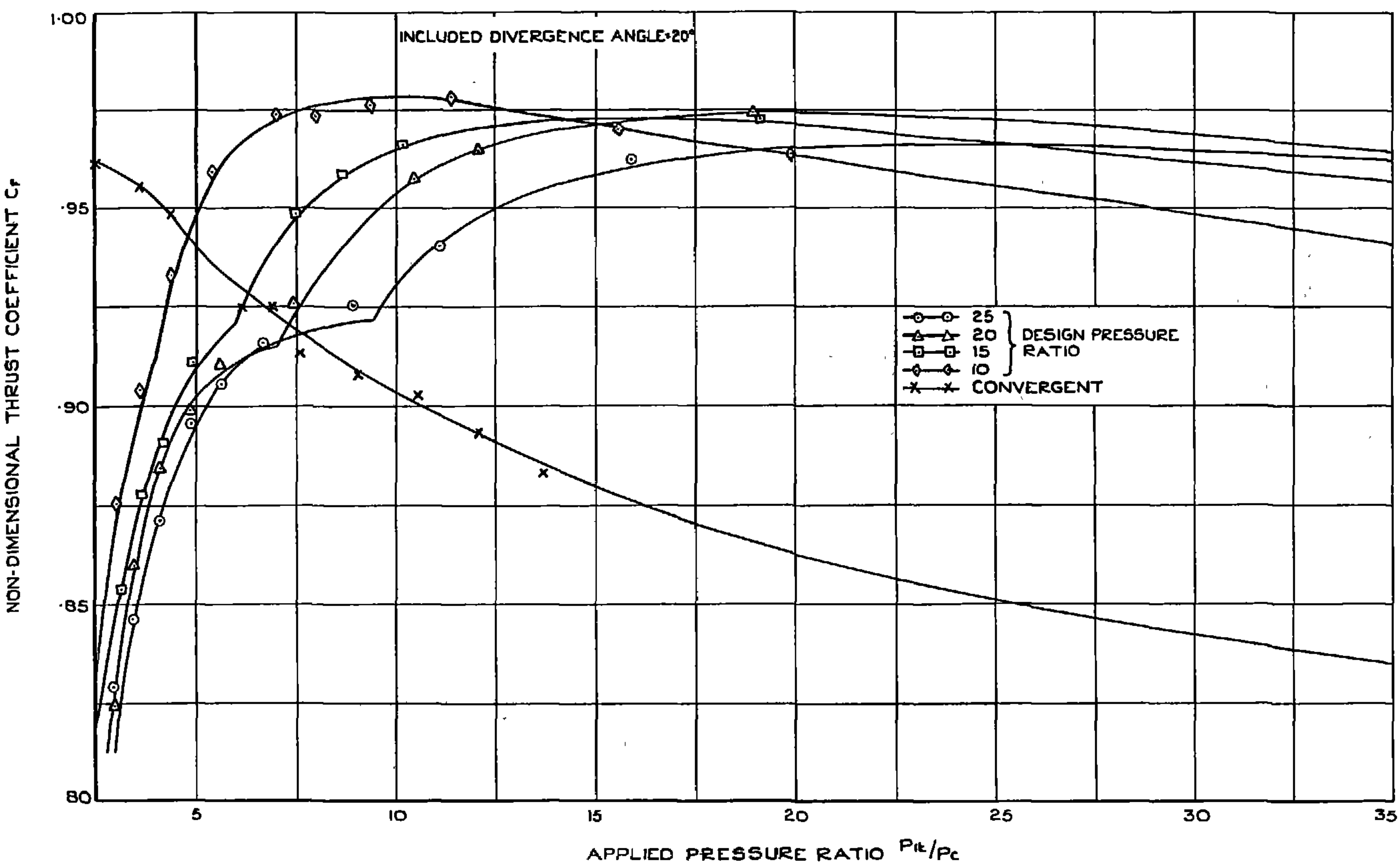
THRUST MEASUREMENT CHECK

FIG. 5



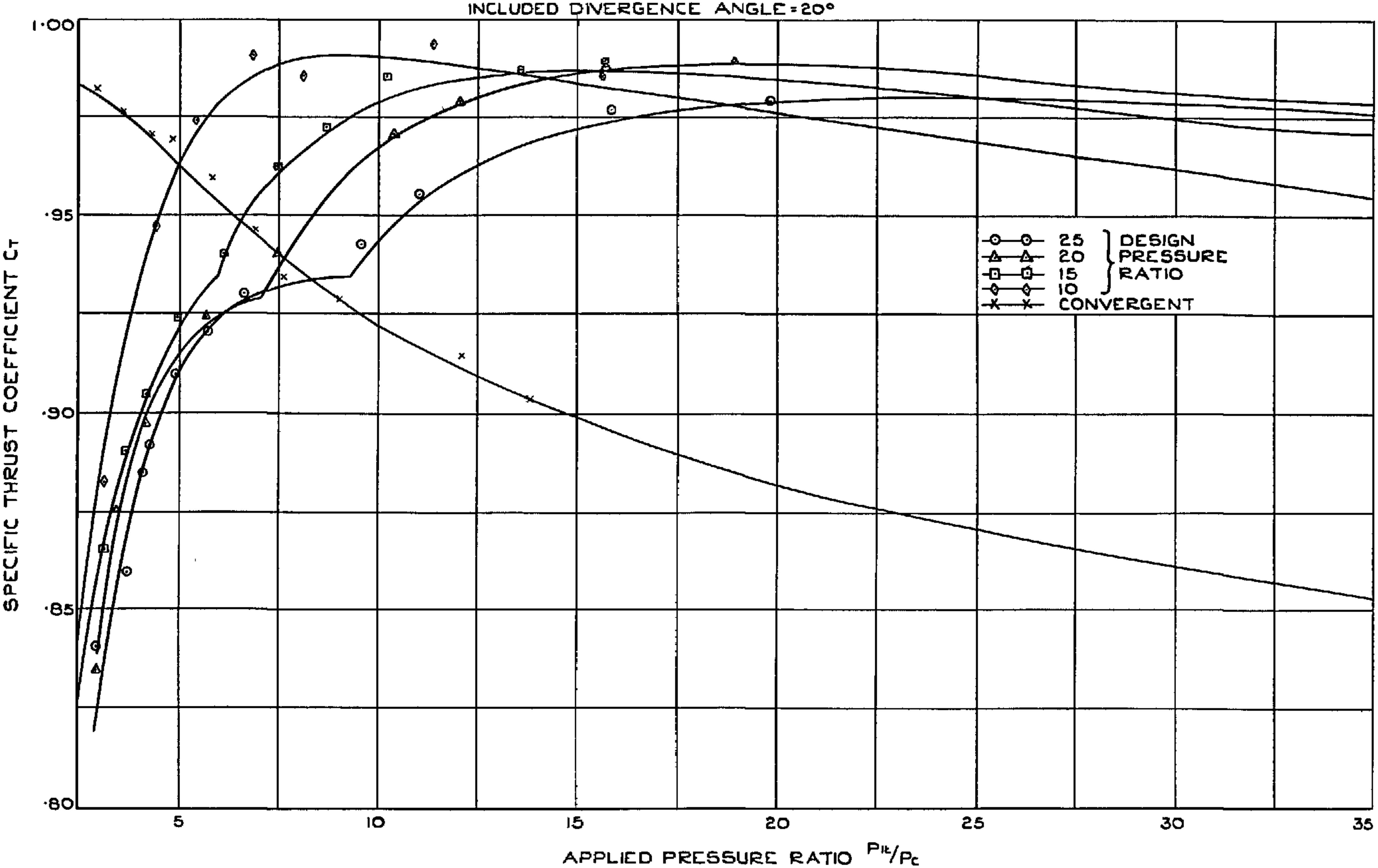
NOZZLE DISCHARGE COEFFICIENT

FIG. 6



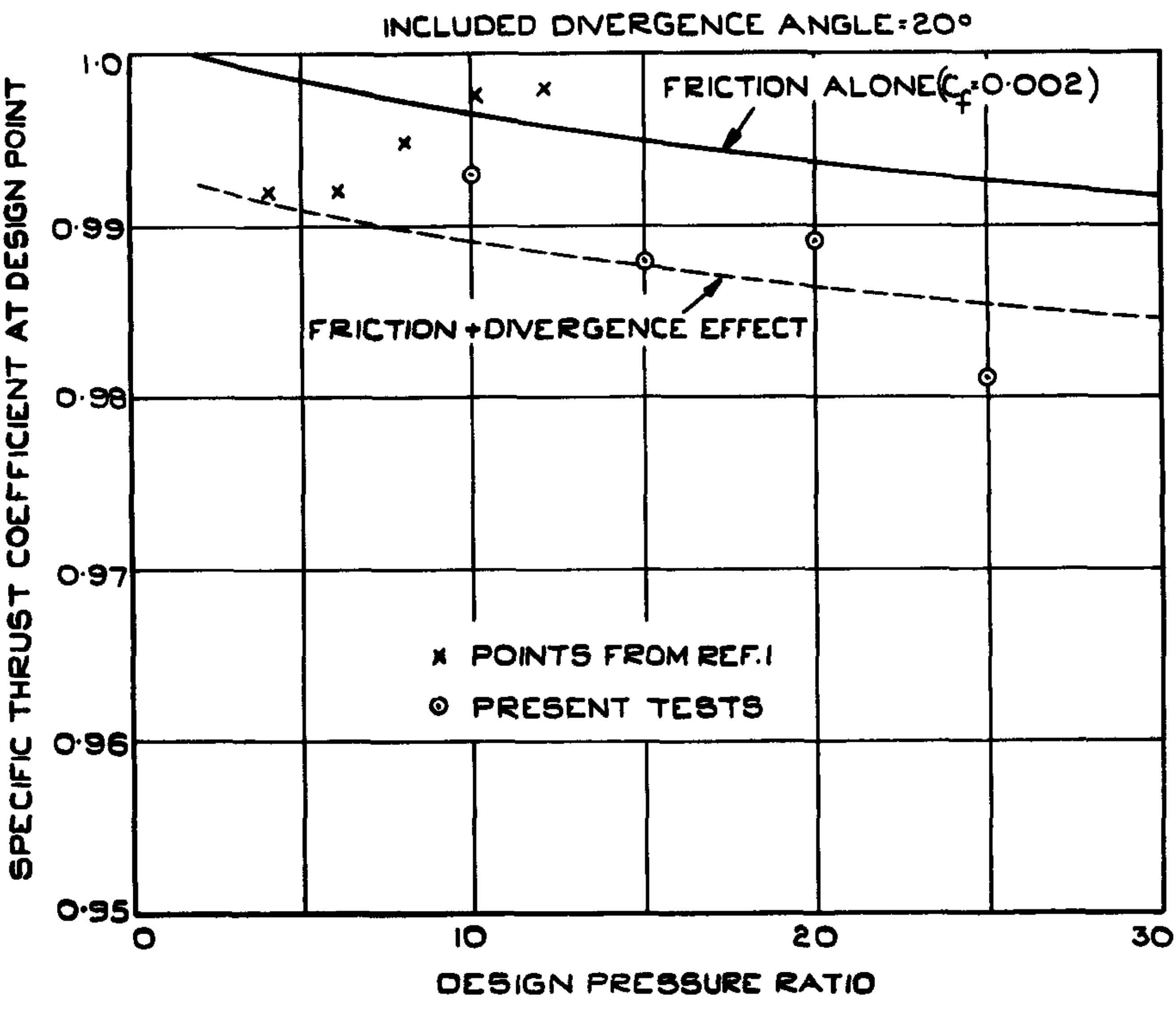
NON-DIMENSIONAL THRUST COEFFICIENTS

FIG. 7



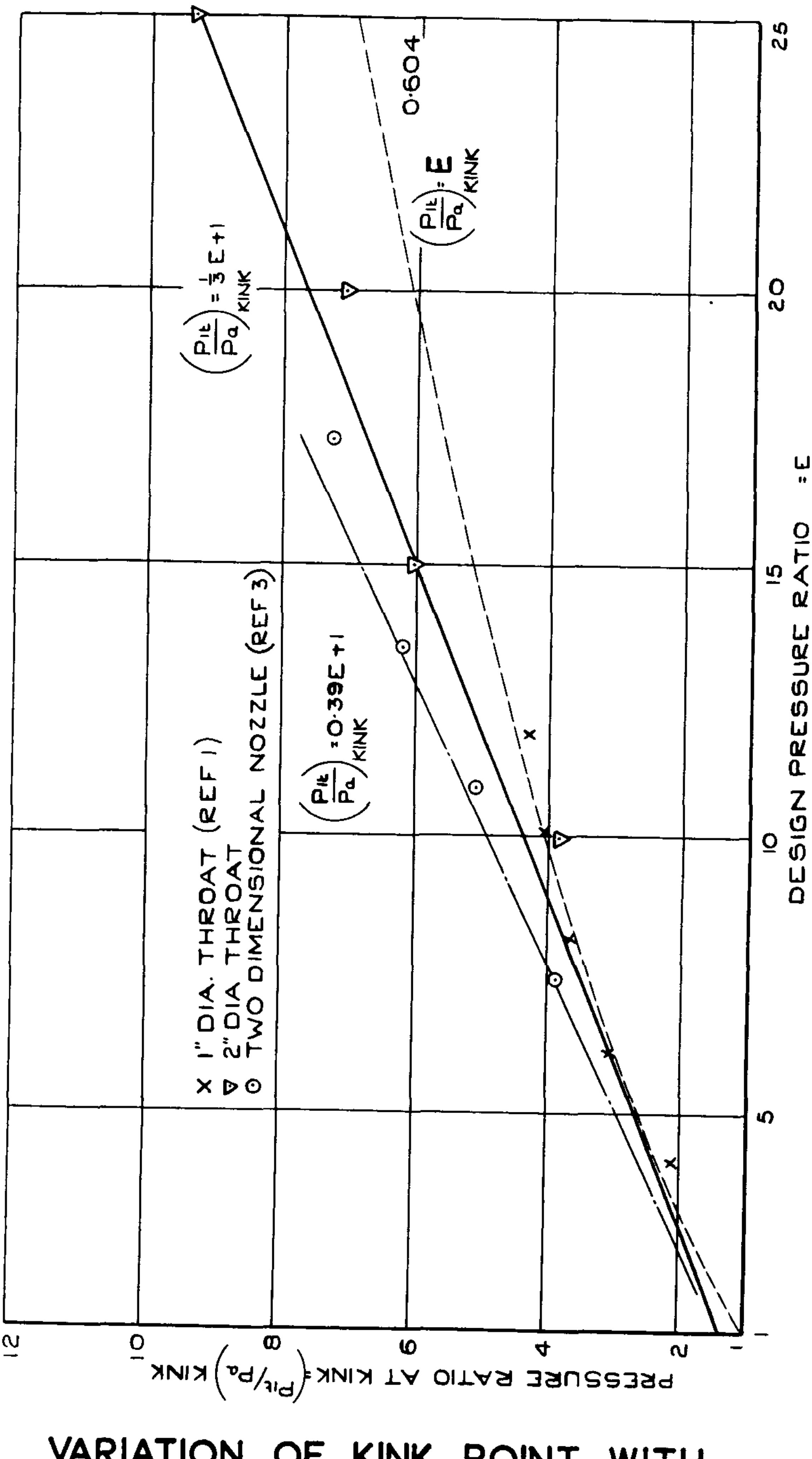
SPECIFIC THRUST COEFFICIENTS

FIG. 8



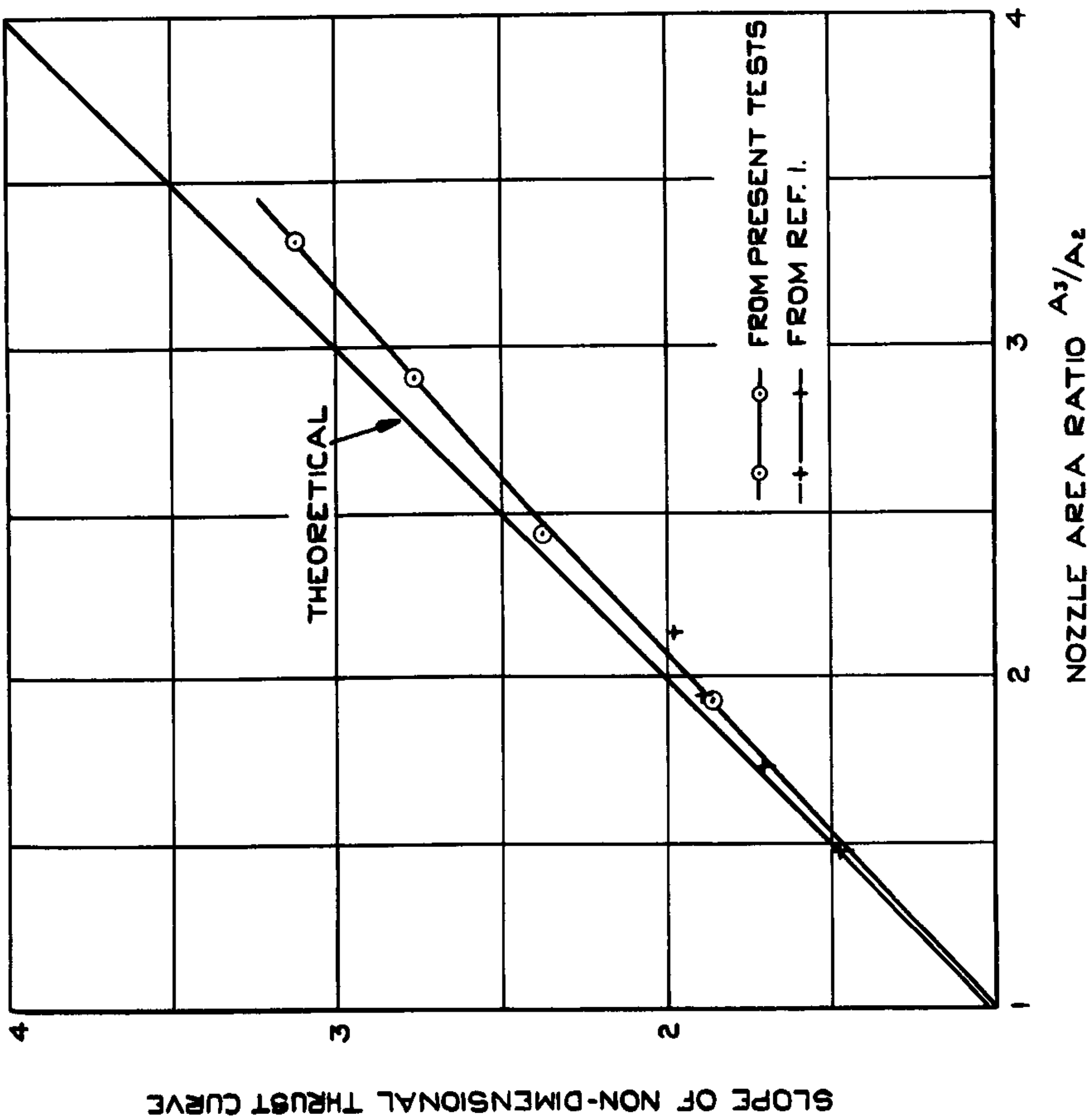
EFFECT OF FRICTION ON DESIGN POINT THRUST COEFFICIENT

<u>FIG. 9</u>



<u>VARIATION OF KINK POINT WITH</u>
<u>DESIGN PRESSURE RATIO</u>

FIG. 10



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