C.P. No. 693

C.P. No. 693

LIBRARY
ROYAL AIRCRAFT ...
BEDFORD

MINISTRY OF AVIATION

AERONAUTICAL RESEARCH COUNCIL

CURRENT PAPERS

Tunnel Tests on a Double Cascade to Determine the Interaction between the Rotor and the Nozzles of a Supersonic Turbine

By

B.S. Stratford and G.E. Sansome

LONDON: HER MAJESTY'S STATIONERY OFFICE

1963

PRICE 4s 6d NET

U.D.C. No. 621.438-253.5:533.6.011.5

C.P. No. 693

Tunnel tests on a double cascade to determine the interaction between the rotor and the nozzles of a supersonic turbine

- By -
B. S. Stratford and G. E. Sansome

August, 1962

SUMMARY

Experimental confirmation has been required that in a supersonic turbine the leading edges of the rotor governs the rotor incidence and, hence, the gas exit angle from the nozzles. Evidence has also been required that, once the rotor incidence has been allowed for, there is no adverse effect of the rotors on the nozzle flow, even when the rotors have a large turning angle.

The present test cascade represented the stationary configuration of a turbine of 2.5 nozzle Mach number and 74° swirl angle, the rotors being designed to operate at 1.9 relative Mach number and to provide a turning angle of 140°. In the tests, fully supersonic flow could be established through the system, but the losses were fairly high and an increase in loss of about 25% would have caused choking in the rotor.

The flow in the nozzles appeared well behaved. According to the Schlieren photographs the individual shock waves from the leading edges of the rotors were fairly well attenuated before reaching the nozzle exit, with the result that, for a given exit angle, the flow in the nozzles appeared to be unaffected by the presence of the rotors. The gas exit angle from the nozzles and hence the incidence on the rotors were determined as expected by conditions downstream, in this instance the leading edge of the rotor blades. Considerable secondary flow occurred in and just ahead of the rotor blades at the ends of the span. The rotor incidence agreed with simple theory in which an allowance is made for boundary layer growth on the convex surface of the blade, but not for losses through the bow shocks, nor for the secondary flow at the ends of the span.

In a turbine rotor with energy extraction considerable flare might be needed to prevent choking, while the three-dimensional effects from the flare and from the centrifugal field could somewhat modify the interaction between the nozzles and the rotors.

Replaces N.G.T.E. Memo. No. M.359 - A.R.C. 24,027.

CONTENTS

1. Introduction

Considerable theoretical and experimental work has been carried out on the mode of operation of cascades for supersonic compressors and turbines, as for example, in References 1 to 9. One of the features characteristic of such cascades is that the incidence on to a downstream blade row is determined by the thickness of the leading edges of that blade row, provided the axial component of velocity is subsonic. As a result the gas angle and static pressure at the exit from, say, a nozzle cascade is determined by the leading edge geometry of the succeeding rotor cascade. The theoretical arguments for this behaviour in two dimensional flow seem conclusive, but the evidence from experiment is sparse. Consequently as part of the N.G.T.E. programme on supersonic turbines[7,8,9] tunnel tests have been made on a double cascade containing a row of nozzle blades of Mach number 2.5 and a stationary row of rotor blades.

Initially the double tunnel cascade had been planned because in the turbine tests of Reference 7 the rotor was found to prevent the full expansion of the nozzle flow. In some instances at least the behaviour could be explained by incidence effects resulting from the non-zero thickness of the rotor leading edges, but there seemed the possibility that other interaction effects could be present, including choking of a subsequent blade row. It was therefore required to demonstrate that a fully expanded supersonic flow could in fact be obtained through two successive cascades having large turning angles. If the flow were fully supersonic in the double cascade the effects of leading edge thickness could then be investigated.

2. Apparatus

Figures 1 and 2 show the double cascade tunnel, which consisted of the nozzle cascade tunnel tested in Reference 8 (where all details will be found) modified to take a row of rotor blades just downstream. The nozzle blades were 1.56 times turbine scale, whereas the rotors were spare blades from the turbine itself, so the axial distance between the blade rows was made $1\frac{1}{4}$ times turbine scale.

The test programme included three designs of rotor blades, being the three designs tested in the turbine first-stage rotor and described in Reference 7. The profiles at the leading edge are shown in Figure 3. (As indicated in Figure 3a it is only the thickness 't' on the upper, or convex, side of the leading edge that enters into the theoretical caclulation of incidence.) The leading edges in the first and second designs were chamfered on the convex surface of the blade. Inspection figures for the second design showed that 70% of the blades had a leading edge thickness within 0.002 in. of the mean for the cascade. For the third design the leading edges of the first design were cut back on the concave surface to give a nominally zero value for the thickness 't'; from inspection the mean blockage effective in causing incidence was probably about $\frac{1}{3}$%.

The following builds were tested:

Test 1	1st turbine blade design;	3% tip clearance*;	face AB/ at 78°
Test 2	2nd turbine blade design;	no tip clearance;	face AB at 75°
Test 3	2nd turbine blade design;	no tip clearance;	face AB at 78°
Test 4	2nd turbine blade design;	3% tip clearance;	face AB at 78°
Test 5	3rd turbine blade design;	3% tip clearance;	face AB at 75°
Test 6	3rd turbine blade design;	no tip clearance;	face AB at 75°
Test 7	3rd turbine blade design;	3% tip clearance;	face AB at 78°
Test 8	3rd turbine blade design;	no tip clearance;	face AB at 78°

*The tip clearance indicated in the programme was confined to the rotor blades.

/The face AB, shown in Figure 2, formed the control for the gas angle at the nozzle outlet in the previous tests of Reference 8.

Figure 1 shows the static tappings, which consisted of the previous row at the nozzle exit augmented at one end by three additional positions, numbered 24 to 26, and several further positions at the rotor mid-chord and downstream of the rotor exit. No attempt was made to traverse the flow by yawmeter or pitot tube because of the rapid variations in flow angle and total pressure.

3. Results and discussion

3.1 General behaviour

The composite Schlieren photograph for Test 3 shown in Figure 4 indicates that supersonic flow was obtained throughout the double cascade, that the flow in the nozzle was well behaved and fully expanded, and that the individual shock waves from the rotor leading edges were fairly well attenuated before reaching the nozzle. Separation occurs inside the rotor blades - as might be expected at such a high-off-design Mach number[9].

The Schlieren photographs for Tests 1, 2 and 4 were very similar to those for Test 3. In the photographs for the remainder of the tests, i.e., Tests 5 to 8, the shock waves were much weaker and lay at slightly smaller inclinations to the centre lines of the passages, except at the rotor exit, where the Mach angles indicated rather lower Mach numbers than for Test 2.

In Figure 5 liquid on the end walls indicates something of the type of flow existing at the ends of the span. The flow visualisation first occurred by accident, phosphoric acid from the manometers reaching the tunnel through the inter-blade row static connections in Figure 5a.

Subsequently, liquid was allowed to leak into the cascade at the downstream pin locating the right-hand nozzle blade (Figure 1) and gave the pictures of Figures 5b to 5f. The liquid traces show that the flow near the end walls is highly three-dimensional, the boundary layer apparently having difficulty in penetrating the bow shock system. In photographs b and c, where the liquid is injected to give flow visualization at the end of the span having tip clearance, the flow at the leading edge appears to be almost entirely in a peripheral direction, suggestive of reverse flow through the tip clearance. In photograph d the liquid penetrates the cascade in some blade passages but not in others.

3.2 The analysis for incidence

The static pressure distribution $\frac{1}{8}$ in. axially downstream of the plane of the trailing edges are shown in Figures 6 and 7. These show high pressures in passage 1 when the wall AB is at 78°, and low pressures for 75°, the actual values agreeing with those of Reference 8. Unlike the results from Reference 8, however, the influence of the wall angle AB is lost as one crosses into passage 2. It will be seen on a close examination that the pressure levels to which the distributions are tending may be correlated with the thickness of the rotor leading edge - the lower pressures occurring with the thinner leading edges. Thus Tests 6 and 8 with nominally zero leading edge thickness and no tip clearance give a static pressure tending to settle at a steady 0.050 of the inlet total pressure, while Tests 2 to 4 with 6.7% leading edge thickness give pressures oscillating around about 0.064 of the inlet total pressure. (The sharp rise of pressure at statics 24 to 26 is attributed to an end effect in the cascade, perhaps associated with an accumulation of the three-dimensional flows noted in Figure 5.) Figure 8 has been prepared from the results of the nozzle cascade tests of Reference 8 in order to relate the static pressure at the nozzle outlet to the gas exit angle. From Figure 8 the estimated asymptotic pressure levels in Figures 6 and 7 have been converted to estimated flow angles, and hence rotor incidences, and these are tabulated in Table I.

TABLE I

Values of rotor incidence deduced from the estimated values of the asymptotic static pressure at the nozzle exit

Test No.	1	2	3	4	5	6	7	8
P_{stat}/P_{tot}	0.059	0.063	0.065	0.063	0.054	0.050	0.053	0.050
gas exit angle	74.3°	74.9°	75.2°	74.9°	73.2°	71.9°	72.9°	71.9°
rotor incidence	4.3°	4.9°	5.2°	4.9°	3.2°	1.9°	2.9°	1.9°

A somewhat similar process of comparison with Reference 8 allows values of the rotor incidence to be deduced from the nozzle shock positions in the Schlieren photographs. The results are shown in Table II, together with the mean values obtained from both techniques.

TABLE II

(a) Values of rotor incidence deduced from a comparison of the Schlieren photographs with those of Reference 8

(b) Mean of Table I and/Table II (a)

Test No.	1	2	3	4	5	6	7	8
(a) rotor incidence deduced from Schlieren photographs	4.4° (uncertain)	5.4°	5.3°	5.5°	2.5°	2.55°	3.2°	2.6°
rotor incidence from Table I	4.3°	4.9°	5.2°	4.9°	3.2°	1.9°	2.9°	1.9°
(b) mean values*	4.3°	5.1°	5.3°	5.2°	2.8°	2.2°	3.0°	2.2°

*When rounding for mean values reference was made to previous rounding

The mean experimental values of the incidence from Table II are plotted in Figure 9, where comparison may be made with theory. It will be seen that the incidences from the experiment are larger than those of the theoretical curve 'a' - deduced for two-dimensional inviscid flow with zero shock loss, as in Reference 8.

Curve 'b' in Figure 9 represents an extension of the theory to take into account a boundary layer on the convex surface of the blade, the derivation being indicated in the Appendix. The agreement between the experimental points and the theoretical curve 'b' is good.

The agreement just noted indicates that at least the trends presented are probably correct. The exactness of the agreement - to within a $\frac{1}{4}$° for the results without tip clearance - must be fortuitous as the experimental techniques are scarcely this accurate, while the theory neglects shock losses as well as the boundary layer and secondary flow at the ends of the span (see Figure 5); furthermore, the calculation for the boundary layer on the convex surface has been much simplified. Consequently, further work would be needed to provide an absolute reliability of say one degree or better. In the present tests, an additional complication occurred in that inspection subsequent to the completion of the experiment showed that the nozzle dimensions had been somewhat different in Tests 2 to 4 from the values in Tests 1 and 5 to 8 although the resulting effect on the apparent variation of incidence is thought to be less than $\frac{1}{2}$°. In future tests care should be taken to maintain the nozzle dimensions constant. (In the present tests the mean value of the exit to throat area ratio for the left-hand two passages - passages 2 and 3 - remained constant throughout the tests, and the static pressures in passage 1 indicate that the dimensions here must also have been near constant.)

The general level of incidence in the present tests agrees with that found by Johnston and Dransfield for the turbine. An exact comparison is not appropriate as the upstream Mach numbers differ.

3.3 Losses, and application to a turbine

From the Mach angles and static pressures at the rotor exit independent rough estimates could be made of the rotor exit Mach number and the overall losses. According to these calculations the average Mach number just downstream of the rotor exit was about 1.65, whereas for isentropic flow it would be 2.83. The corresponding overall total pressure coefficient for the double cascade is about 0.36 and the velocity coefficient, based on an isentropic Mach number of 2.50 (see Reference 8), is 0.85. Choking would have occurred inside the rotor passages at a velocity coefficient of 0.815, i.e., for about a 25% greater loss.

In a turbine installation energy is extracted from the flow, while three-dimensional losses are probably greater than in the tunnel because of the centrifugal field. Consequently one might expect to require considerable flare in order to prevent choking of the rotor when installed in a turbine. On the other hand the rotor in a turbine could be operating at its design Mach number, while the level of Mach number in the rotor would be somewhat lower than in the tunnel; both of these factors should tend to reduce the losses in the turbine installation. The additional three-dimensional effects present in the turbine as a result of the centrifugal field and of the flare could modify somewhat the interactions between the rotor and the nozzle.

4. Conclusions

In a supersonic cascade having the axial component of velocity subsonic the small region of the blade profile at the leading edge can significantly alter the incidence at which the cascade operates. The theory so far developed includes only a simplified treatment for the boundary layer and takes no account of three-dimensional effects at the ends of the blade span; moreover, in the simple continuity form of the theory used here no account is taken of shock losses. Similarly the present experiment is subject to uncertainties of techniques and may not be reliable to better than, say, 1°. Nevertheless, very similar trends are shown by the theory and the experiment. A blade of zero thickness at the leading edge operates at a small positive incidence - which is about 2 to 3° for the present cascade. This incidence is largely due to the presence of the boundary layer on the convex surface of the blade, the theory for two-dimensional inviscid flow predicting zero incidence for a blade of zero leading edge thickness. A blade of non-zero thickness at the leading edge operates at an incidence a few degrees higher than that of the blade with the zero thickness leading edge. Only that portion of the thickness on the upper, or convex, side of the leading edge tip is effective in altering the incidence and here a thickness equal to 7% of the passage width (s cos β_1) increases the incidence in the present cascade about 2 to 3°. The effect is non-linear.

In general in the present type of flow it is the thickness of the leading edge and the value of the upstream Mach number - rather than the blade outlet angle of the upstream blade row - that governs the incidence and, hence, the gas inlet angle.

The losses of a two-dimensional double cascade with supersonic flow can be fairly high and considerable flare may be needed in the rotor of a turbine in order to prevent choking. The three-dimensional effects present in a turbine could somewhat modify the results on flow angle.

List of symbols

s	blade pitch
β_1	blade angle relative to the turbine axis
t	thickness of the leading edge as defined by Figure 3a
x	distance from the leading edge
δ^*	displacement thickness of the boundary layer
x_1	distance defined in Appendix I and Figure 10
ε_1	quantity defined in Appendix I
M	Mach number
R_x	Reynolds number based on x and the free-stream conditions
γ	ratio of the specific heats

REFERENCES

No.	Author(s)	Title, etc.
1	A. Ferri	Preliminary analysis of axial-flow compressors having supersonic velocity at the entrance of the stator. N.A.C.A. Research Memo.L9G06.TIB/3115. September, 1949.
2	A. Kantrowitz	The supersonic axial-flow compressor N.A.C.A. Report 974. 1950.
3	C. H. Hauser H. W. Plohr G. Sonder	Study of flow conditions and deflection angle at exit of two-dimensional cascade of turbine rotor blades at critical and supercritical pressure ratios. N.A.C.A. Research Memo.E9K25.TIB/2260. March, 1950.
4	M. E. Deich	Flow of gas through turbine lattices [Translation "Technical Gasdynamics" (Tekhnickeskaia gazodinamika), Chapter 7. 1953] N.A.C.A. Tech. Memo. 1393. May, 1956.
5	R. Staniforth	On the testing of supersonic compressor cascades. A.R.C. 19,877. September, 1957.
6	J. H. Keenan	Reaction tests of turbine nozzles for supersonic velocities. Trans. A.S.M.E., Vol. 71, p. 773 to 780. 1949.
7	I. H. Johnston D. C. Dransfield	The test performance of highly loaded turbine stages designed for high pressure ratio. A.R.C. R. & M. 3242. June, 1959.
8	B. S. Stratford G. E. Sansome	The performance of supersonic turbine nozzles. A.R.C. R. & M. 3273. June, 1959.
9	B. S. Stratford G. E. Sansome	Theory and tunnel tests of rotor blades for supersonic turbines. A.R.C. R. & M. 3275. December, 1960.
10	A. D. Young	Skin friction in the laminar boundary layer in compressible flow. Aero. Quarterly, Vol. I, pp. 137 to 164. 1949.

APPENDIX I

Extension of the theory of rotor blade incidence to allow for a boundary layer on the convex surface of the blade

When calculating the incidence for a blade with boundary layer the effective profile is that containing the displacement thickness δ^*. Figure 10 shows a blade with its boundary layer displacement thickness, the thickness of the leading edge of the blade being shown as zero for clarity. It will be seen that the whole of the effective profile of the upper, or convex, surface is now curved, instead of containing an extensive flat region as in the basic blade. Such a flat region is a necessary condition for the validity of the theory of Reference 8. The latter theory requires generalisation in order to apply to a fully curved surface, as the uniform one-dimensional flow assumed in the theory inside the blade passages no longer exists. The result of such generalisation may be expressed as follows. Let the point B at $x = x_1$ in Figure 10 be the position such that a Mach line drawn from it becomes incident on the leading edge E of the next blade. The inclination of the effective profile, at $x = x_1$, to the original blade surface is $(d\delta^*/dx)_1$, while the depth of the profile relative to a line through the leading edge A parallel to the tangent to the profile at B is $[\delta_1^* - x_1\,(d\delta^*/dx)_1]$. Let us suppose that that blockage on the basic blade which would have caused an incidence equal in value to $(d\delta^*/dx)_1$ is denoted ε_1. Then the curve 'b' in Figure 9 for the profile including boundary layer is derived from that without boundary layer, i.e., 'a' in Figure 9, by displacing the latter along the axis of blockage a distance $-[\delta_1^* - x_1\,(d\delta^*/dx)_1 + \varepsilon_1]$.

The point B is important in the analysis as it is only the profile upstream of B that can influence the flow upstream of the blade row. Consequently the incidence is determined by the profile AB alone, whatever the profile downstream of B (provided it does not cause shock waves which could intersect the Mach line BE). Consequently for the purpose of analysis the simplest profile may be chosen downstream of B, Figure 10 showing a profile consisting of the tangent BC. If AD is now drawn parallel to BC, and if AD is regarded as a datum direction for a calculation of the type explained in Reference 8, the result quoted in the previous paragraph may be obtained.

For the present cascade the Reynolds number based on the distance x_1 is about 25,000, so that the boundary layer is assumed to be laminar. For a laminar boundary layer

$$\delta^* \propto x^{\frac{1}{2}}$$

so that

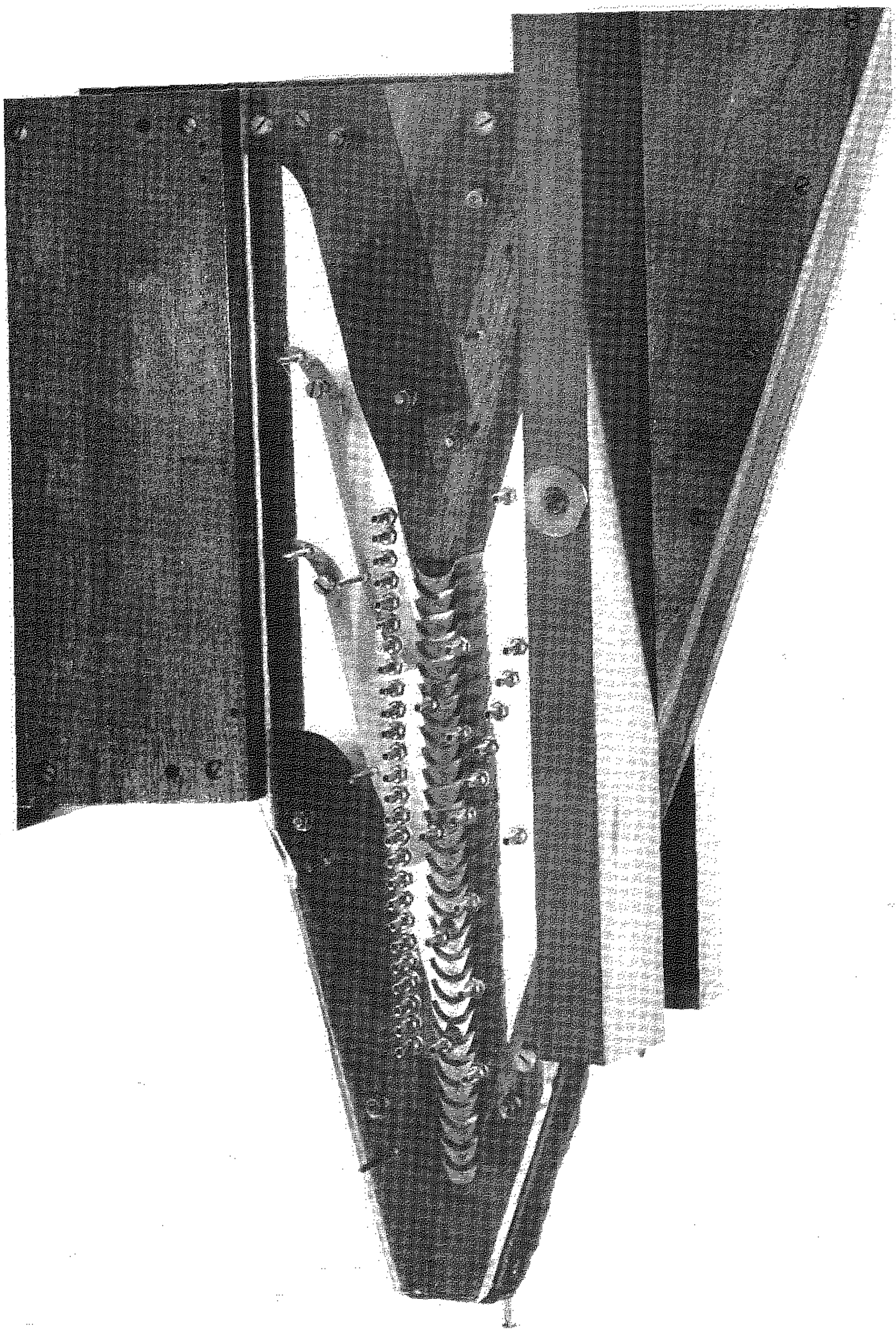
$$\delta^* - x(d\delta^*/dx) = \delta^*/2$$

From Reference 10 an adequate formula for δ^* may be taken as

$$\delta^* = 1.721\ [1 + 0.693\ (\gamma - 1)M^2]\ xR_x^{-\frac{1}{2}}$$

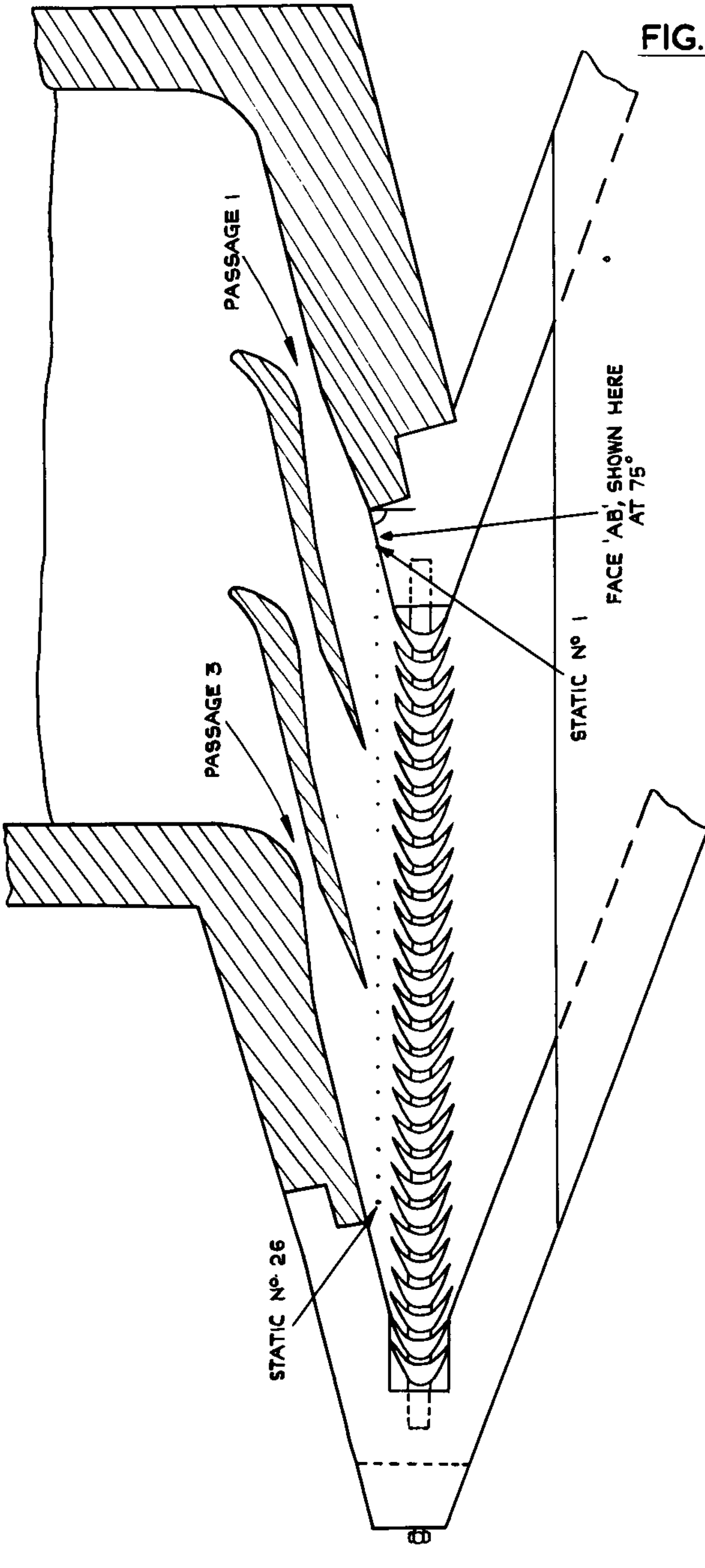
The value of $\delta_1^*/2$ is calculated to be about 1.0% of the passage width $s \cos \beta_1$ (see Figure 3) and $(d\delta^*/dx)_1$ is about 1.0°. Figure 9, curve 'a', shows that ½% blockage on the basic blade would be required to cause 1.0° incidence. Hence curve 'b' is obtained from curve 'a' by displacing it (-1½%) along the scale of blockage. The numerical values just given have to be obtained by successive approximation as x_1 is initially unknown, the angle of the Mach line depending upon δ_1^*, $(d\delta^*/dx)_1$ and, strictly, the incidence. In the calculations a constant static pressure has been assumed between the points $x = 0$ and $x = x_1$.

FIG. I



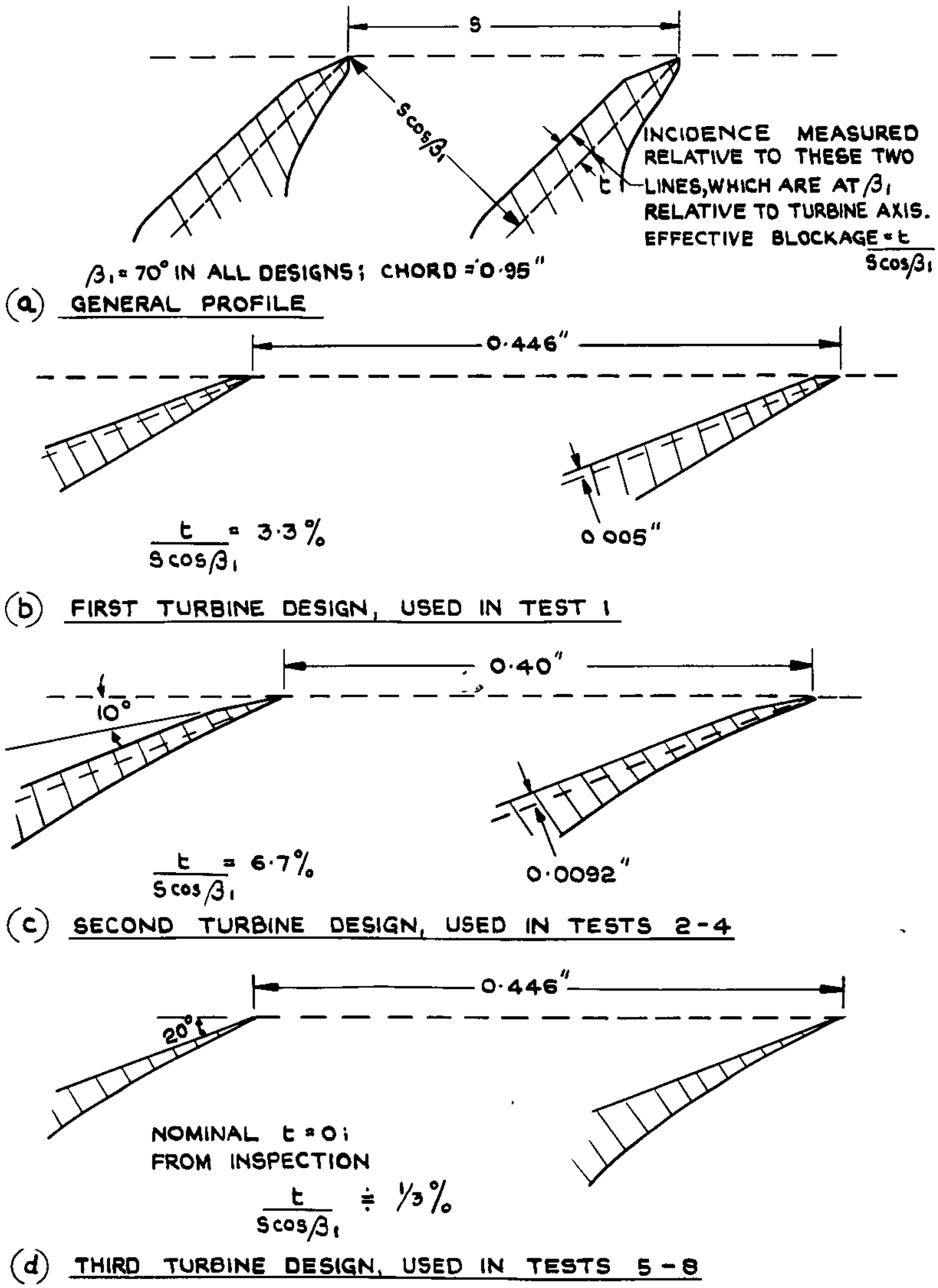
PHOTOGRAPH OF THE WORKING SECTION

FIG. 2



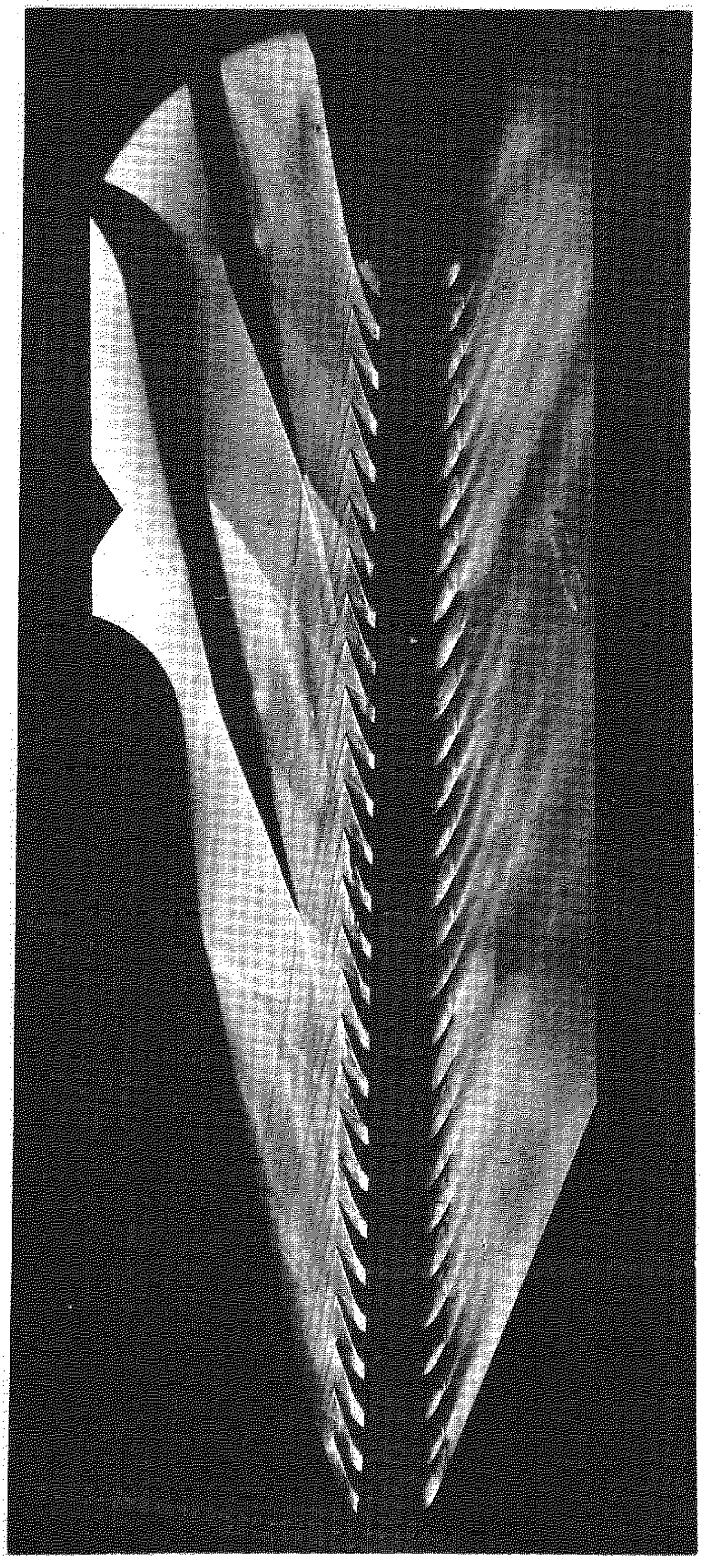
SKETCH OF THE WORKING SECTION.

FIG. 3



PROFILES AT THE LEADING EDGES OF THE ROTOR BLADES.

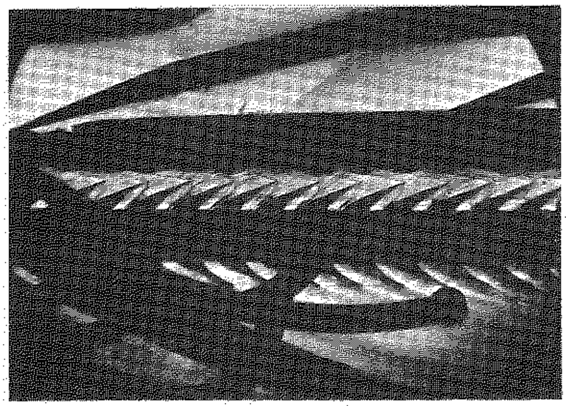
FIG. 4



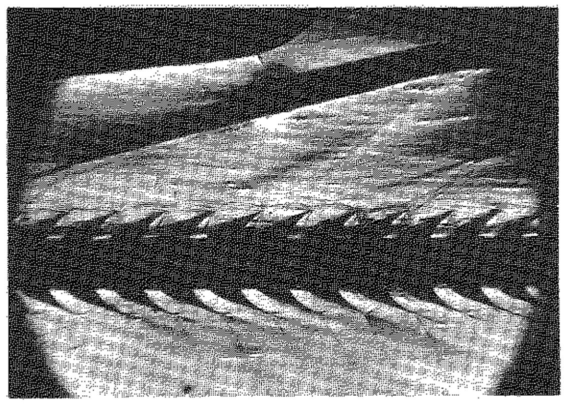
COMPOSITE SCHLIEREN PHOTOGRAPH

FOR TEST 3

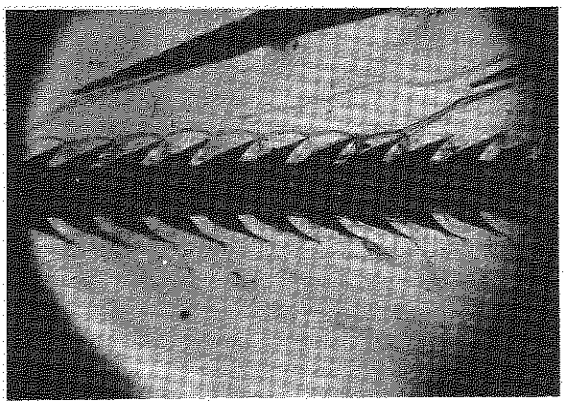
FIG. 5



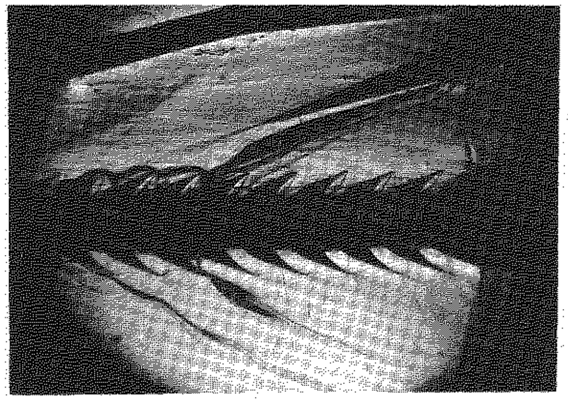
a. TEST 2



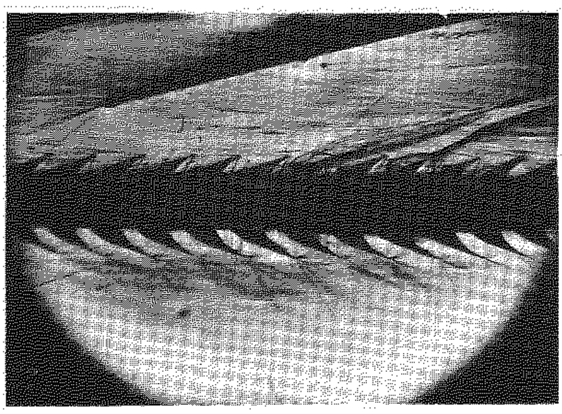
b. TEST 5



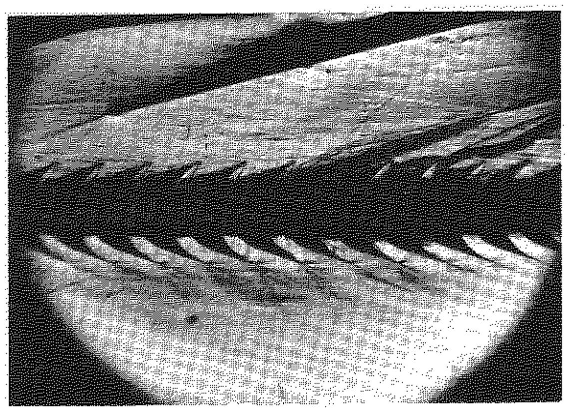
c. TEST 5



d. TEST 6



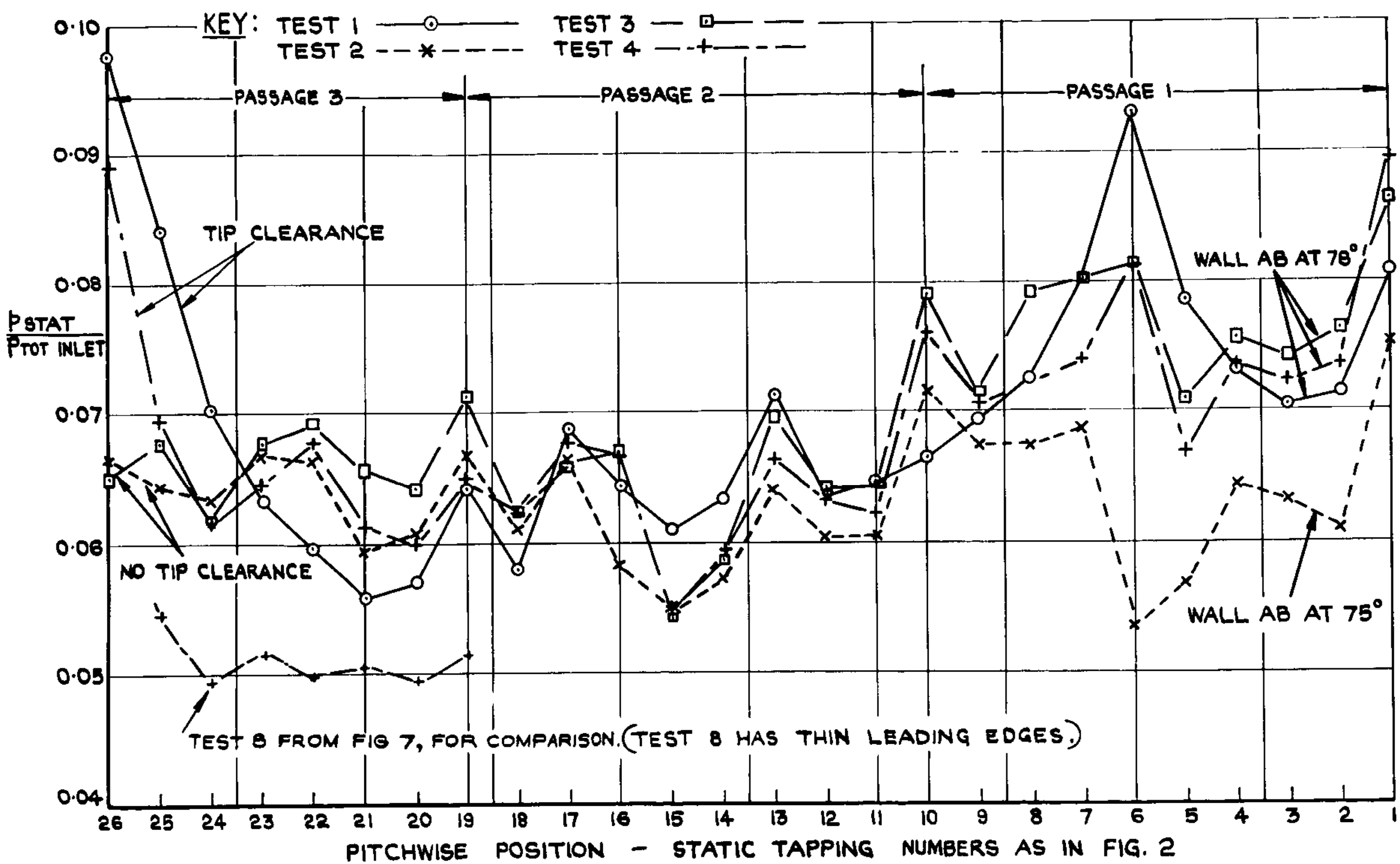
e. TEST 8



f. TEST 8

IN PHOTOGRAPHS b AND c, TEST 5, THE LIQUID IS INJECTED TO GIVE FLOW VISUALIZATION AT THE END OF THE SPAN HAVING TIP CLEARANCE

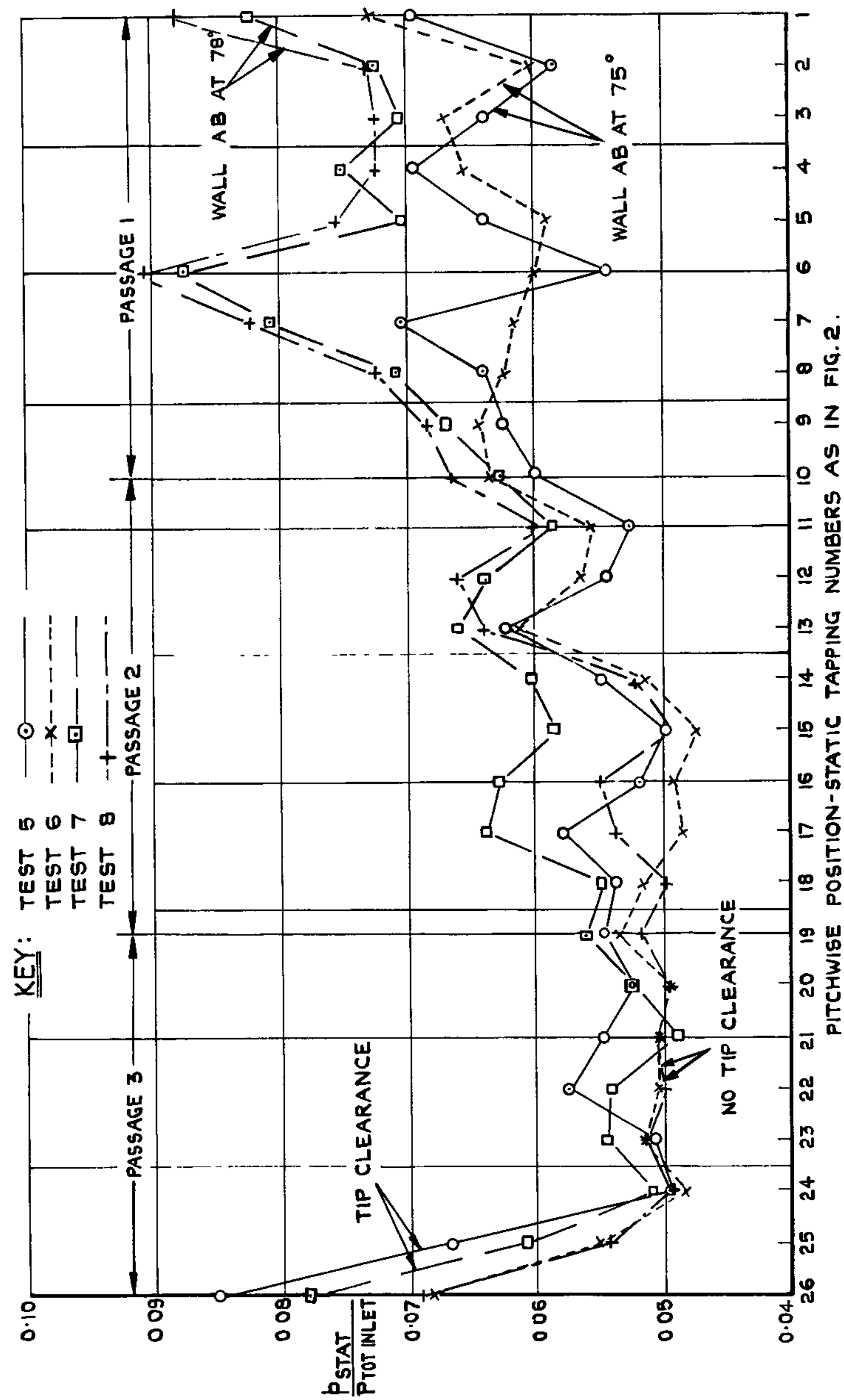
INJECTION OF LIQUID TO VISUALIZE THE FLOW ON THE END WALLS



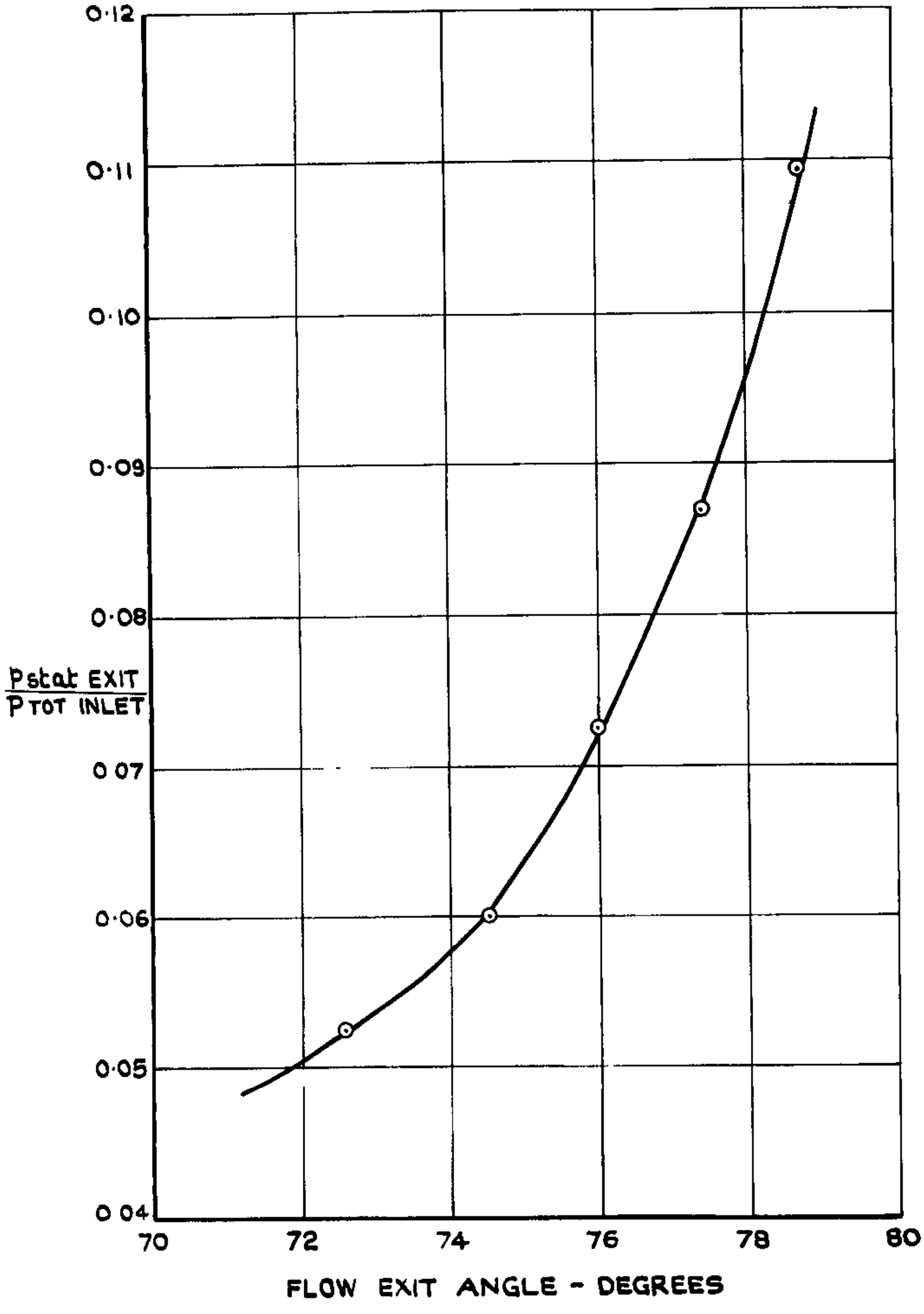
STATIC PRESSURE DISTRIBUTIONS AT THE NOZZLE EXIT FOR TESTS 1 TO 4.

FIG. 6

FIG. 7

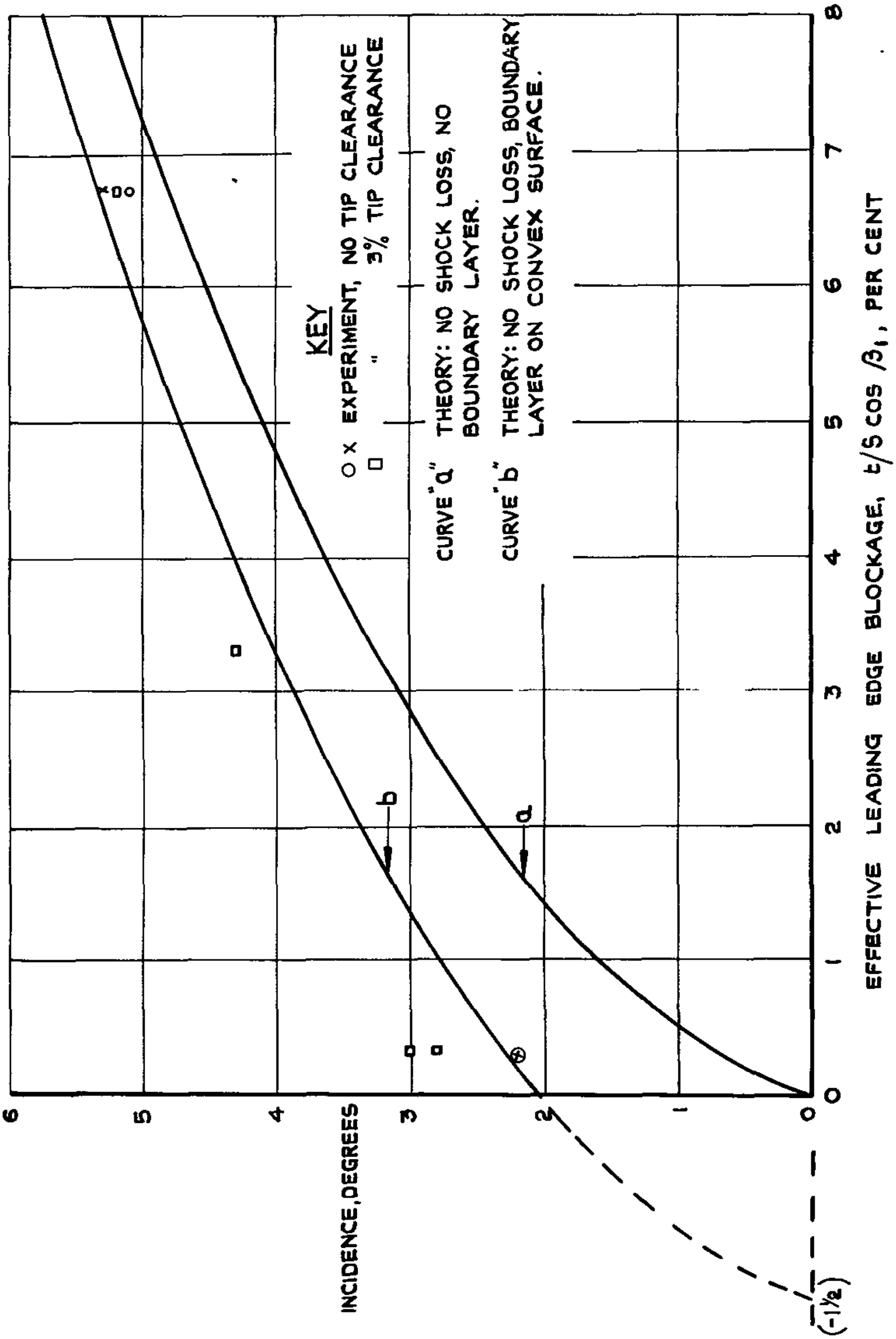


STATIC PRESSURE DISTRIBUTIONS AT THE NOZZLE EXIT FOR TESTS 5 TO 8.



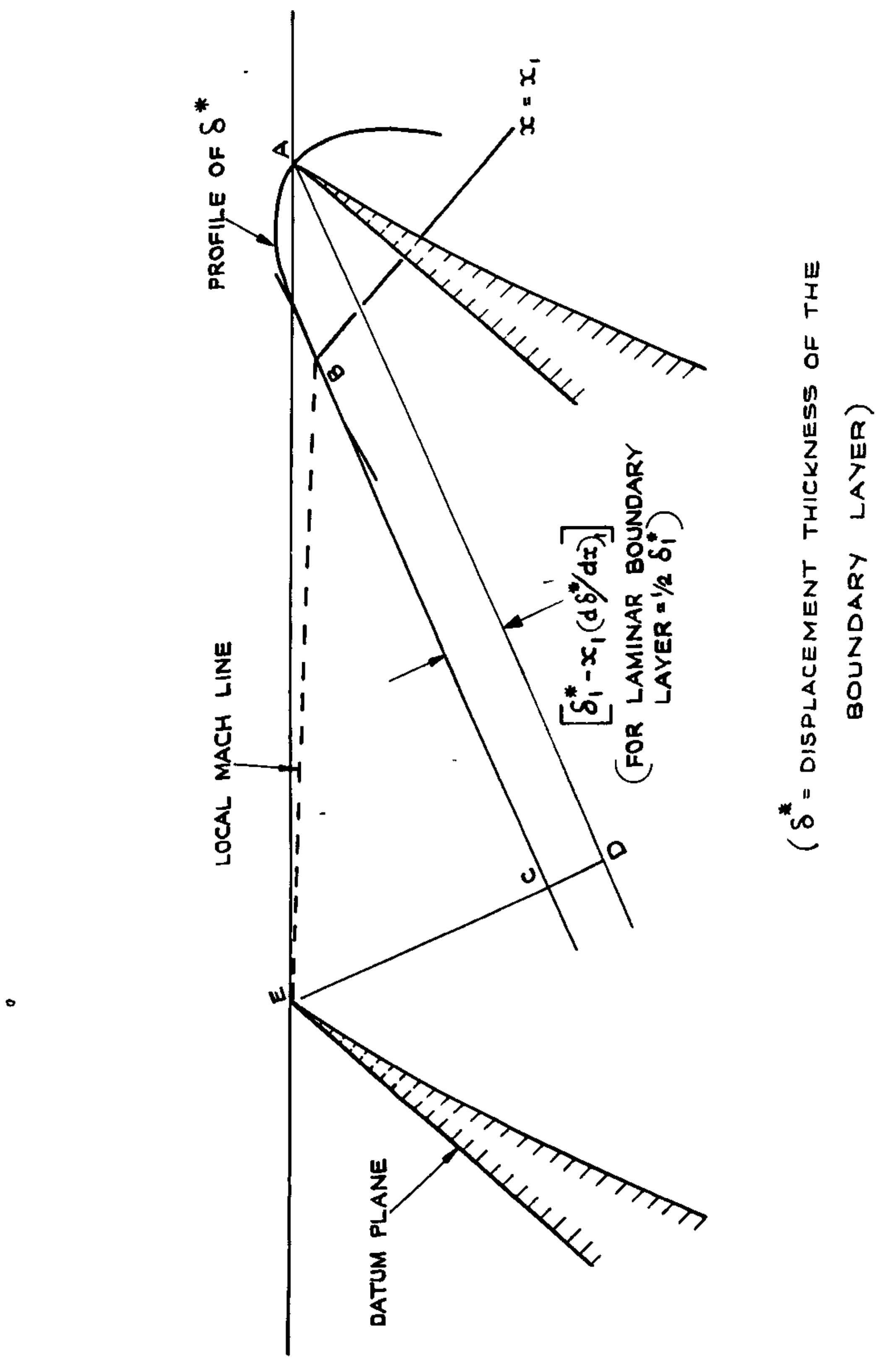
CURVE FOR THE CONVERSION OF STATIC PRESSURE TO FLOW ANGLE AT THE NOZZLE EXIT, ESTIMATED FROM REFERENCE 8.

FIG. 9



VARIATION OF INCIDENCE WITH
EFFECTIVE LEADING EDGE BLOCKAGE.

FIG. 10



EXTENSION OF THE THEORY TO ALLOW FOR A BOUNDARY LAYER ON THE CONVEX SURFACE OF THE BLADE.

A.R.C. C.P. No. 693. August, 1962. 621.438-253.5: 533.6.011.5

Stratford, B. S. and Sansome, G. E.

TUNNEL TESTS ON A DOUBLE CASCADE TO DETERMINE THE INTERACTION BETWEEN THE ROTOR AND THE NOZZLES OF A SUPERSONIC TURBINE

Experimental confirmation has been required concerning the interaction between the rotor and the nozzles in a supersonic turbine.

The present test cascade represented the stationary configuration of a turbine of 2.5 nozzle Mach number and 74° swirl angle, the rotors being designed to operate at 1.9 relative Mach number and to provide a turning angle of 140°. In the tests, fully supersonic flow could be established through the system, but the losses were fairly high and an increase in loss of about 25% would have caused choking in the rotor.

A.R.C. C.P. No. 693. August, 1962. 621.438-253.5: 533.6.011.5

Stratford, B. S. and Sansome, G. E.

TUNNEL TESTS ON A DOUBLE CASCADE TO DETERMINE THE INTERACTION BETWEEN THE ROTOR AND THE NOZZLES OF A SUPERSONIC TURBINE

Experimental confirmation has been required concerning the interaction between the rotor and the nozzles in a supersonic turbine.

The present test cascade represented the stationary configuration of a turbine of 2.5 nozzle Mach number and 74° swirl angle, the rotors being designed to operate at 1.9 relative Mach number and to provide a turning angle of 140°. In the tests, fully supersonic flow could be established through the system, but the losses were fairly high and an increase in loss of about 25% would have caused choking in the rotor.

A.R.C. C.P. No. 693. August, 1962. 621.438-253.5: 533.6.011.5

Stratford, B. S. and Sansome, G. E.

TUNNEL TESTS ON A DOUBLE CASCADE TO DETERMINE THE INTERACTION BETWEEN THE ROTOR AND THE NOZZLES OF A SUPERSONIC TURBINE

Experimental confirmation has been required concerning the interaction between the rotor and the nozzles in a supersonic turbine.

The present test cascade represented the stationary configuration of a turbine of 2.5 nozzle Mach number and 74° swirl angle, the rotors being designed to operate at 1.9 relative Mach number and to provide a turning angle of 140°. In the tests, fully supersonic flow could be established through the system, but the losses were fairly high and an increase in loss of about 25% would have caused choking in the rotor.

C.P. No. 693

© *Crown copyright* 1963

Printed and published by
HER MAJESTY'S STATIONERY OFFICE
To be purchased from
York House, Kingsway, London W.C. 2
423 Oxford Street, London W.1
13A Castle Street, Edinburgh 2
109 St Mary Street, Cardiff
39 King Street, Manchester 2
50 Fairfax Street, Bristol 1
35 Smallbrook, Ringway, Birmingham 5
80 Chichester Street, Belfast 1
or through any bookseller

Printed in England

S.O. Code No. 23-9013-93

C.P. No. 693