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The Theoretical Pressure Distributions around some Conventional Turbine Blades in Cascade

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Summary.—By means of Relf's analogy between aerodynamic streamline flow and electric potential flow, the theoretical pressure distributions around a series of conventional turbine blades in cascade have been determined over a range of incidence covered in some previously reported aerodynamic tests.

The theoretical pressure distributions and their variation with incidence provide the basis of an explanation of the observed aerodynamic performance.

1. *Introduction*.—With the development of the aircraft gas turbine and the attainment of high pressure ratios and high efficiencies in the axial-flow compressor, an increase in the overall gas-turbine efficiency entails either the use of a higher turbine inlet temperature, thus increasing the simple gas-turbine efficiency, or an improvement in turbine design and efficiency comparable with that which has been achieved in the case of an axial-flow compressor. In the latter case much useful information has been obtained using a knowledge of the theoretical pressure distributions around an aerofoil in cascade (R. & M. 2095[1] and 2384[2] and Ref. 3) but little has been attempted in the case of a turbine cascade, although certain results have been used for heat transfer calculations in the development of cooled turbine blades (R. & M. 2699[4] and Refs. 5 and 6). The correlation of turbine cascade tunnel results with turbine performance is a matter of great difficulty because of the wider range of Reynolds numbers and Mach numbers which are normally encountered[7], whereas in normal high-speed cascade-tunnel tests the Reynolds number and Mach number are uniquely related for a given cascade. It thus becomes of interest to determine whether the form of the theoretical pressure distributions around a series of conventional turbine blades in cascades can interpret the observed cascade performance and thus explain the mode of operation of the aerofoil in cascade. Accordingly the theoretical pressure distributions around a series of conventional turbine sections in cascade, which have been extensively tested in the National Gas Turbine Establishment No. 3 High-Speed Cascade Tunnel (R. & M. 2697[8] and 2728[9]), have been determined and will be compared with the observed aerodynamic performance. The cascade details are presented in Fig. 1 and Appendix I.

2. *The Determination of the Theoretical Pressure Distribution*.—2.1. *Apparatus*.—The theoretical pressure distribution around each turbine blade in the cascades listed in Appendix I and Fig. 1, were determined over the range of incidence covered in the aerodynamic tests (R. & M. 2697[8] and 2728[9]) by means of Relf's analogy between streamline aerodynamic flow and electrical potential flow. The apparatus has been described in R. & M. 2699[4]. An

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electrical refinement has however been incorporated which enables the balance point of the electrical potential measuring bridge to be reached rapidly and accurately. The original battery powered pre-amplifier has been replaced by a mains powered two-stage push-pull amplifier which exhibits negligible harmonic distortion over a wide range of input voltages. This enables the out-of-balance component of the measuring bridge to be indicated by the phase relationship between the signal wave form and that produced on the second beam of the oscilloscope by a synchronised signal from the master oscillator (R. & M. 2699[4]), which provides the alternating current required for the electric tank.

2.2. *Air Outlet Angle.*—The observed aerodynamic air outlet angle for each of the blades varied with Reynolds number and thus it would have been desirable to determine the theoretical air outlet angle by satisfying the Kutta-Joukowsky condition at the trailing edge of the blade. Since the turbine blades possessed a chamfered trailing edge, however two sets of Kutta-Joukowsky conditions could be satisfied, and thus it was decided to use a nominally geometric condition to fix the outlet angle for each blade. This geometric condition was that the outlet side walls of the electric tank should be placed parallel to the straight line portion of the upper surface profile. This corresponded with the reasonable aerodynamic condition that little or no diffusion occurred on this part of the profile.

The derived air outlet angle agreed closely with the outlet angle observed in the aerodynamic tests at a Reynolds number of approximately 2×10^5. This condition meant that the rear stagnation point was on the chamfered flat and moved but little as the inlet angle was changed. The derived pressure distributions for each blade at several incidences are shown in the Figs. 2 to 8.

3. *The Theoretical Pressure Distribution and Performance.*—In the interpretation of the performance of aerofoils in a compressor cascade the form of the theoretical pressure distribution has been shown to indicate qualitatively the mode of operation of the aerofoil in cascade (R. & M. 2384[2] and Ref. 3). In the study of turbine cascades, however, this method has not been used extensively because of the computational difficulties produced in the calculation of the theoretical pressure distribution around a thick aerofoil in the close pitching of a turbine cascade. Since in this type of cascade a wide range of Reynolds number and Mach number is normally used in aerodynamic testing the influence of the two assumptions, firstly of inviscid flow and secondly of incompressible flow, which are made in obtaining the theoretical pressure distribution must thus be considered.

The effect of the first assumption, that of inviscid flow on the interpretation of performance by means of the theoretical pressure distribution may be minimised by the consideration of the performance of the boundary layer as the fluid flows over the aerofoil surface into the regions of acceleration and diffusion indicated by the theoretical pressure distribution. Indeed if the position of boundary-layer transition is assumed a quantitative estimate of the total-head loss at low speed may be obtained from the theoretical pressure distribution by the methods available for the step-by-step integrations of the boundary-layer equations.

As the Mach number of the flow entering the cascade increases however the assumption of incompressible flow becomes less and less valid until finally with the onset of sonic conditions and the production of shocks on the surface of the boundary layer, compressibility effects begin to predominate. The limit on the range of applicability of an interpretation of cascade performance which is based on the theoretical pressure distribution may be ascertained using either the Glauert-Ackaret or the von Kármán relationship between compressible and incompressible flow. In this report the von Kármán relationship which gives a lower value for the critical Mach number, has been used and the values are shown on the loss curves (Figs. 2 to 8). Where the peak velocity occurs near the leading edge where the boundary layer is thin, the calculated critical Mach number agreed broadly with the Mach number at which the shocks were first observed in the cascade-tunnel schlieren tests (R. & M. 2728[9]). If the maximum velocity occurs near the mid-chord position along the blade where the boundary layer is thick, the velocities outside the boundary layer will be increased and thus the true value of the critical Mach number will lie below the calculated critical Mach number. In all cases therefore the calculated critical Mach number indicated the maximum Mach number at which shockless flow around the turbine cascades which are considered here, is possible.

The forms of the theoretical pressure distributions which were obtained for this series of cascades, are similar to those which were previously obtained for some turbine cascades used for heat transfer investigations (R. & M. 2699[4] and Refs. 5 and 6). At all incidences an indication of breakaway was given by the degree of diffusion which is shown at positions of about 70 per cent chord. The classification of impulse type cascades and reaction type cascades will be retained in this report as this nomenclature represents the basis for which the blade passages were designed. It will be seen that some of the details of the discussion of the blades' performance previously reported (R. & M. 2728[9]) will have to be modified in the light of the theoretical pressure distributions which clarify many of the seemingly anomalous effects observed.

3.1. *Impulse Type Blades with Constant Passage Area.*—In the impulse type cascades numbered 1, 5 and 6 a constant passage area was chosen as a basis of design but the design inlet angle and design outlet angle were varied (Appendix I). There is a marked similarity between the forms of the pressure distributions around the three cascades as can be seen from Figs. 2, 6 and 7.

At a positive incidence of ten degrees all three cascades possess a highly peaked pressure distribution, reminiscent of a compressor cascade near stall, with a high degree of diffusion of the boundary layer. In such a pressure distribution the boundary layer would be turbulent (R. & M. 2384[2] and Ref. 3), at a very low tunnel Reynolds number, tending to stall rapidly with detachment of boundary layer (R. & M. 2728[9]).

In cascade No. 6 the pressure distribution at a positive incidence of five degrees shows that a similar degree of turbulence would be possessed by the boundary layer as in the previous instance. This turbulence would increase with tunnel Reynolds number and by causing a later breakaway would tend to reduce the total-head loss as the tunnel Reynolds number increased. The theoretical critical Mach number shows that sonic conditions are then rapidly attained. A shock would then tend to occur which would thicken the boundary layer, causing an increase in loss. It is clear therefore that the rise of loss previously ascribed to a turbulence transition effect in an earlier report, is due to shock which would not be observed in the schlieren system used in the cascade tunnel, for conditions at the leading edge were obscured (R. & M. 2728[9]).

At zero incidence the effect of the varied blade parameters, as distinct from the constant passage area concept, begins to appear. In the cascades No. 1 and 6 a relatively smooth acceleration is followed by a gentle diffusion, before the abrupt diffusion region characteristic of the conventional type of turbine blade, in cascade No. 5, however, at this incidence there is a small peak in the pressure distribution followed by a slowly diffusing region. It would thus be expected that the rate of reduction of loss with Reynolds number would be less for cascade No. 5 than for cascades Nos. 1 and 6 at this incidence, due to localised transition[3]. That this is indeed the case can be seen from the loss characteristic at this incidence.

At a negative incidence of fifteen degrees the pressure distribution with cascade No. 1 shows a relatively sharper junction between a region of acceleration and of diffusion than the others. Thus again due to the increased turbulence of the boundary layer in this case, the rate of reduction of loss with increase of Reynolds number is less than in the other two cascades at the same incidence.

This effect of a slower reduction of loss with increase of Reynolds number where a peaked pressure distribution is present is also shown on cascade No. 1, at a negative incidence of thirty degrees and in cascade No. 5 at a negative incidence of twenty-five degrees.

3.2. *Turbine Blades with Reaction.*—The turbine cascades numbered 1, 2, 3 and 4 were designed to have the same air outlet angle but to have higher degrees of reaction at design incidence (*see* Appendix I) while cascade No. 7 was designed to have the same incidence angle as cascade No. 4 but with a lower air outlet angle. The lack of uniformity in the variation of blade parameters and passages however tends to make any comparison between them extremely difficult but there are certain broad resemblances in the type of pressure distributions which unifies the apparently disconnected phenomena observed in the aerodynamic tests, and reveals the underlying unity.

It will be observed from Figs. 2, 3, 4 and 5, that, at design incidence, as the design reaction increases the initial acceleration on the upper surface becomes smoother, the region before diffusion more rounded, and the amount of diffusion required in the abrupt diffusion region near the tail diminished. One would thus expect the skin friction loss and the eddying losses produced by boundary-layer breakaway to be reduced as the degree of reaction increases. This is confirmed by the loss characteristics.

At positive incidences (*see* Figs. 3, 4, 5 and 8) as in the case of the impulse type cascades a peak begins to appear near the leading edge but unlike the impulse type with constant passage area, the resulting diffusion is followed by an acceleration. This acceleration would keep down the value of the turbulence in the boundary layer. Thus despite the early turbulence induced by the peak, the boundary-layer thickness, and then the loss, is kept down by the acceleration until the magnitude of the later acceleration is less than the initial diffusion, when the loss in the reaction blades 3, 4 and 7 begins to increase. In cascade No. 2 where the degree of diffusion near the leading edge is pronounced this stabilised turbulence produces a lower loss than observed with cascade No. 1 at a similar inlet angle. This effect can be observed in Figs. 4, 5 and 8 for all positive incidences below stall. It will be noted in cascade No. 3 at an incidence of fifteen degrees the presence of the peak is revealed by the slow initial decrease of drag with Reynolds number increase. In cascades No. 4 and 7 the effect of this ‘ stabilised turbulence ’ produces little change of drag with Reynolds number. As the incidence further increases, the magnitude of the diffusion, which the boundary layer has to surmount, increases until the boundary layer would break away. This can be seen from the pressure distributions for the highest positive incidences shown in Figs. 3, 4, 5 and 8.

An indirect confirmation of the actual occurrence of the peaks near the nose for a reaction type cascade at positive incidences is obtained from the schlieren photographs which, while showing a shock about 60 per cent chord, reveal a series of λ-shocks near the nose.

4. *Conclusion.*—The consideration of the theoretical pressure distributions provides an explanation of the variation of the performance of a given cascade with incidence, and enables a general comparison of the turbine cascades to be made. The onset of sonic conditions can be indicated although the effect of boundary-layer thickness may reduce the actual value of the inlet Mach number at which shocks occur on the boundary layer. The occurrence or non-occurrence of Reynolds number effects can be explained although a quantitative estimate of the loss would have to be made by an integration of the boundary-layer equations.

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9	J. A. Dunsby	Schlieren Tests on some Conventional Turbine Cascades. R. & M. 2728. September, 1949.

APPENDIX I

Cascade	*Aerofoil*	*Stagger*	*Pitch/Chord*	*Design inlet angle (deg)*	*Design outlet angle (deg)*
1	31.3T6/110.5P41.7	8·6	0·627	55	— 60
2	28.35T6/102P41.8	14·0	0·613	45	— 60
3	23.55T6/92.5P41.3	21·5	0·584	30	— 60
4	18.08T6/76.8P40.8	28·5	0·551	15	— 60
5	21.7T6/91.3P43.5	9·0	0·625	45	— 50
6	14.75T6/74P44.5	6·3	0·627	35	— 40
7	15.15T6/66P42.2	21·2	0·584	15	— 50

Note. : Aerofoil profile quoted approximates to circular-arc and straight-line construction (*see* R. & M. 2697[8]).

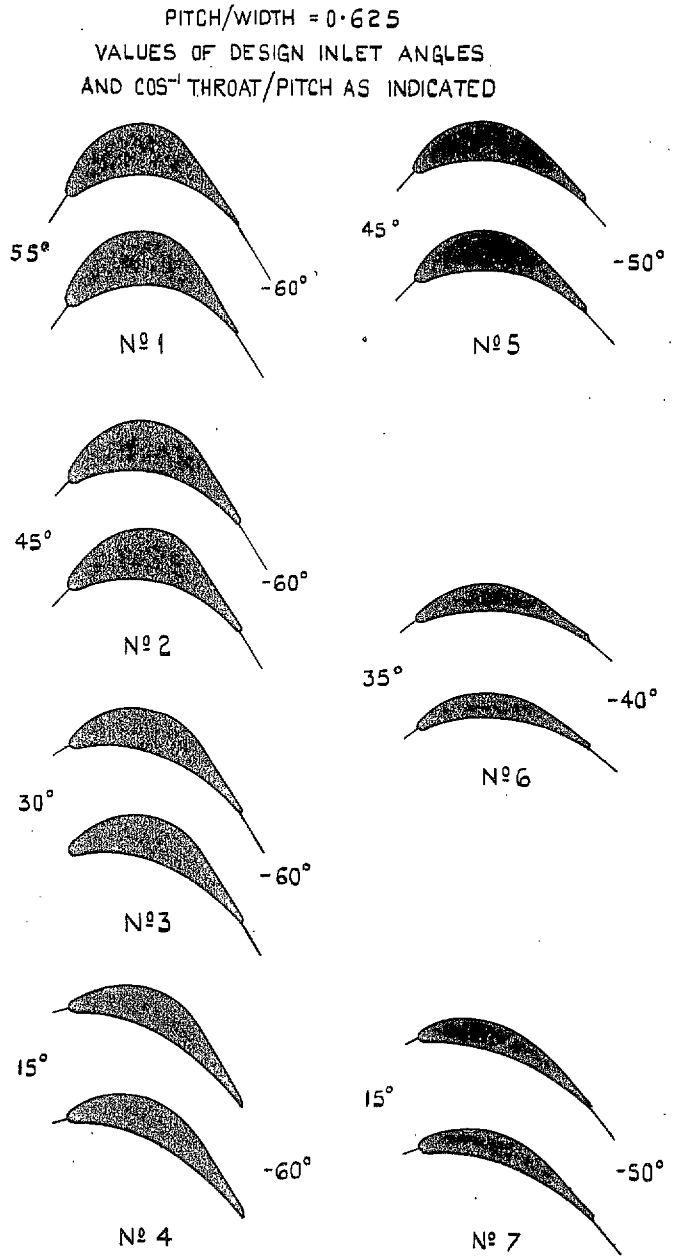

FIG. 1. Cascade details.

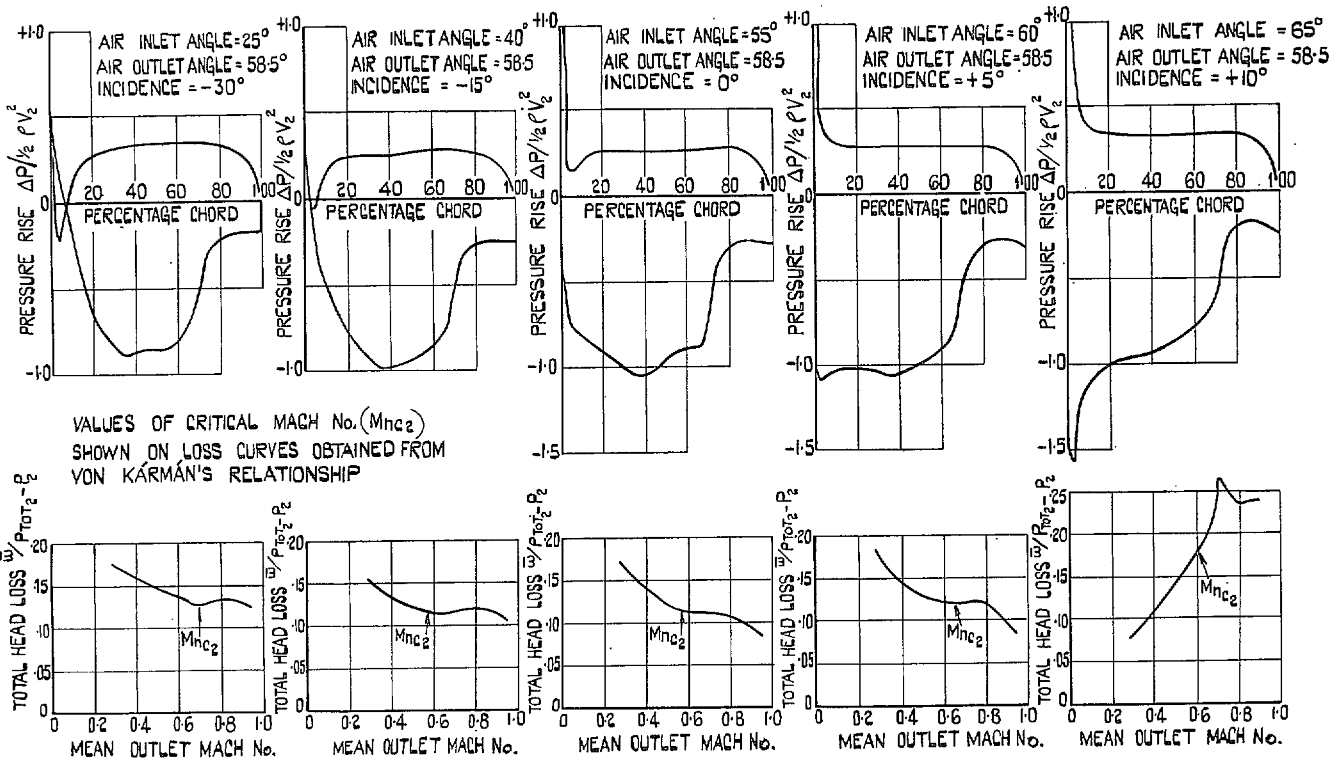

Fig. 2. Theoretical pressure distributions and loss for turbine cascade No. 1.

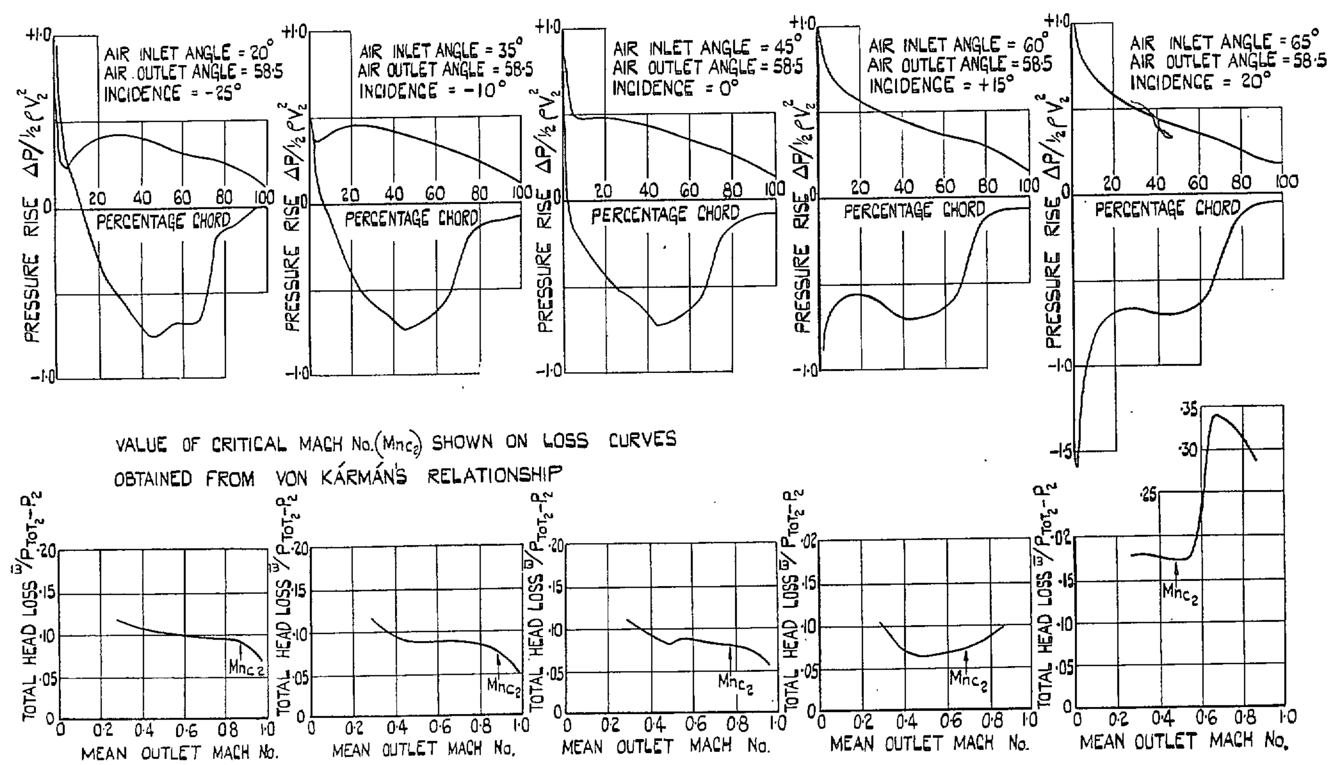

Fig. 3. Theoretical pressure distributions and loss for turbine cascade No. 2.

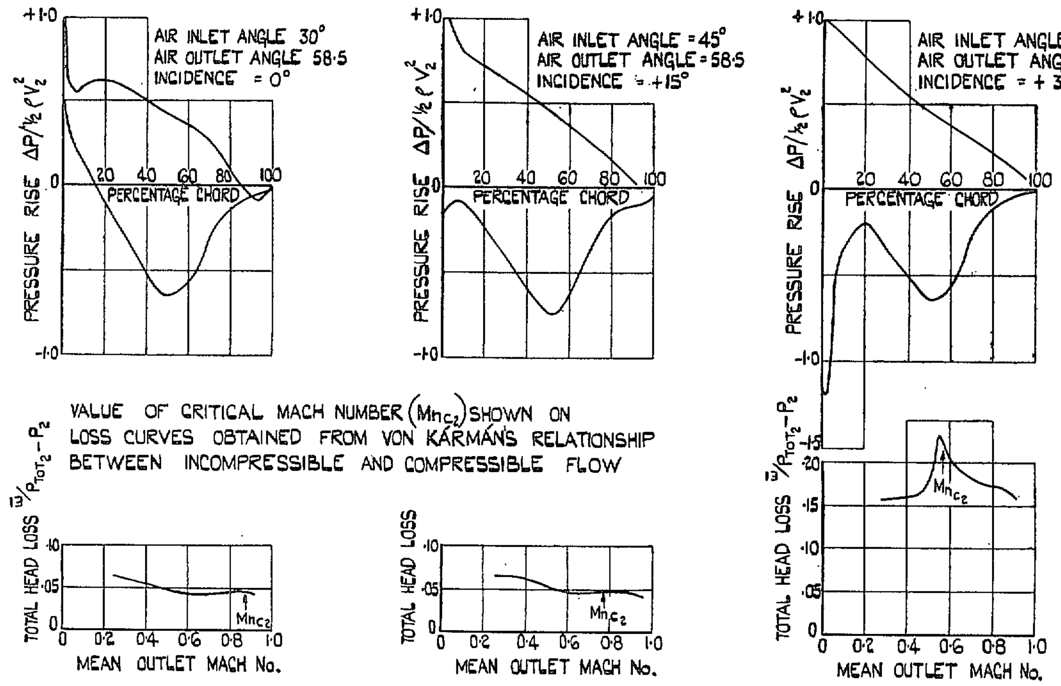

FIG. 4. Theoretical pressure distributions and loss for turbine cascade No. 3.

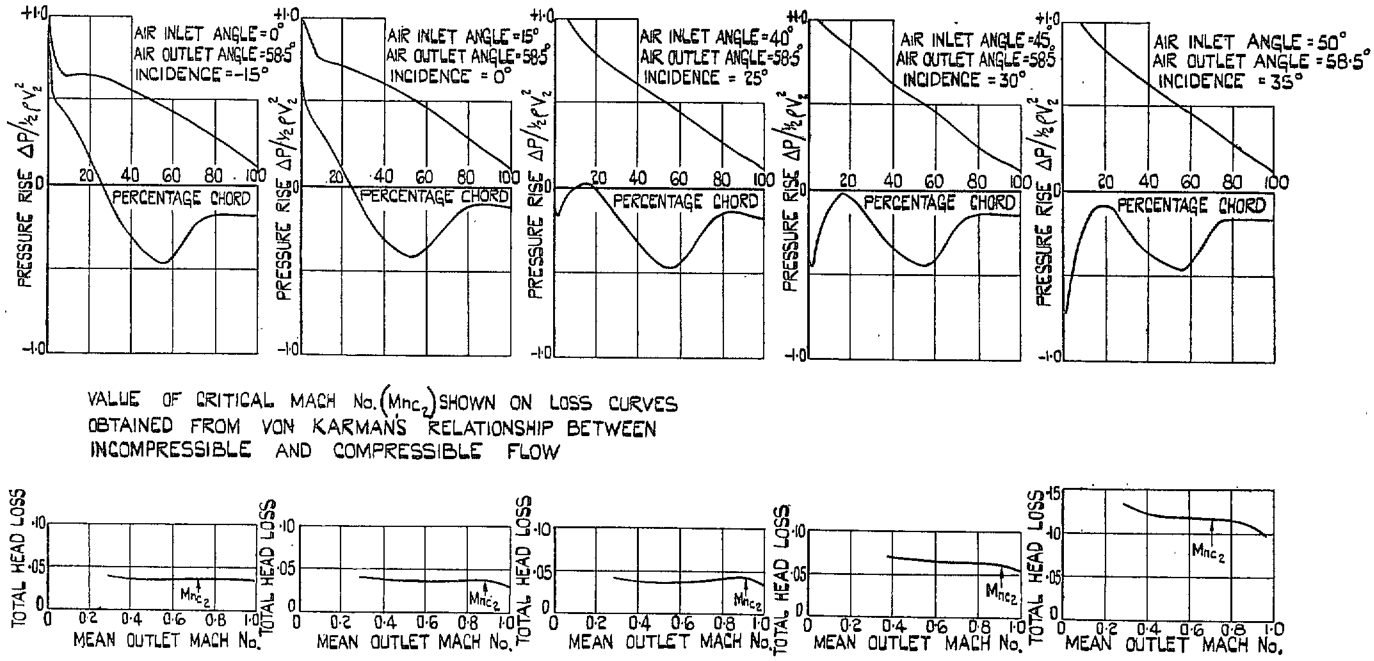

FIG. 5. Theoretical pressure distributions and loss for turbine cascade No. 4.

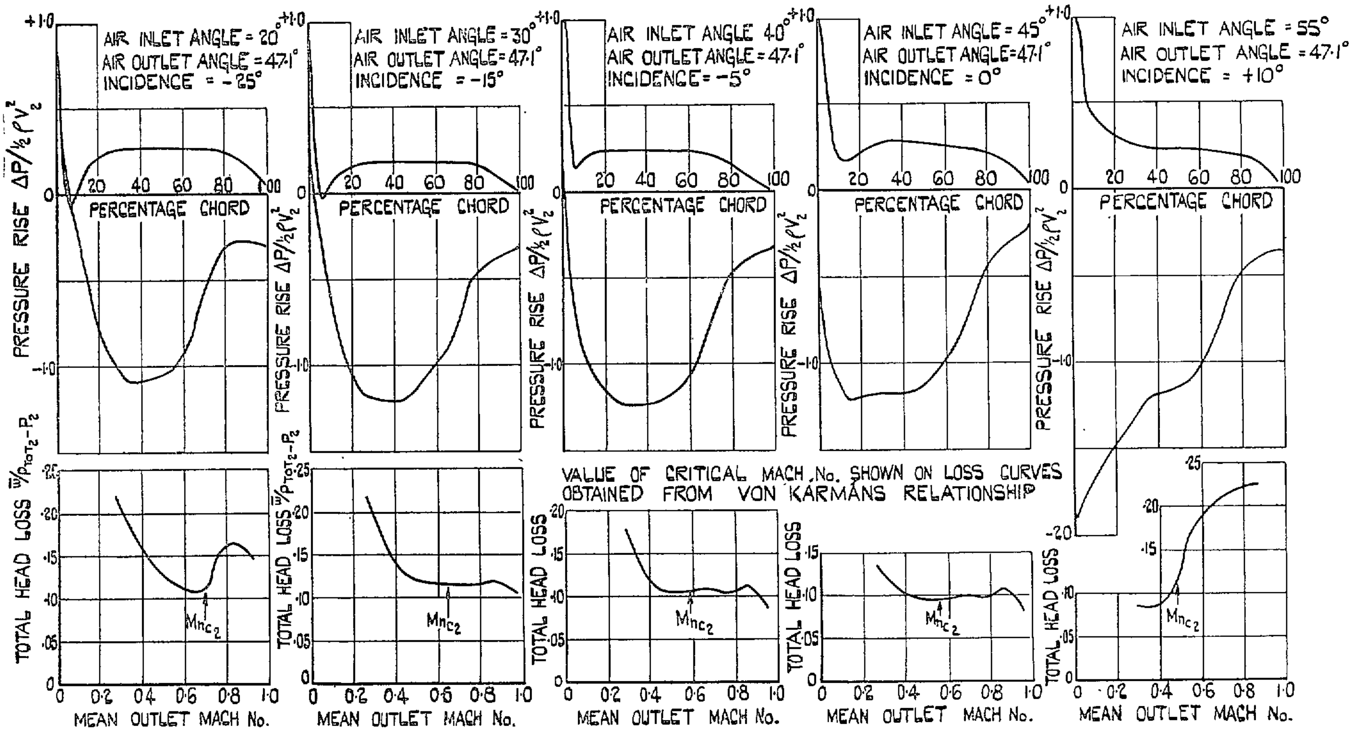

FIG. 6. Theoretical pressure distributions and loss for turbine cascade No. 5.

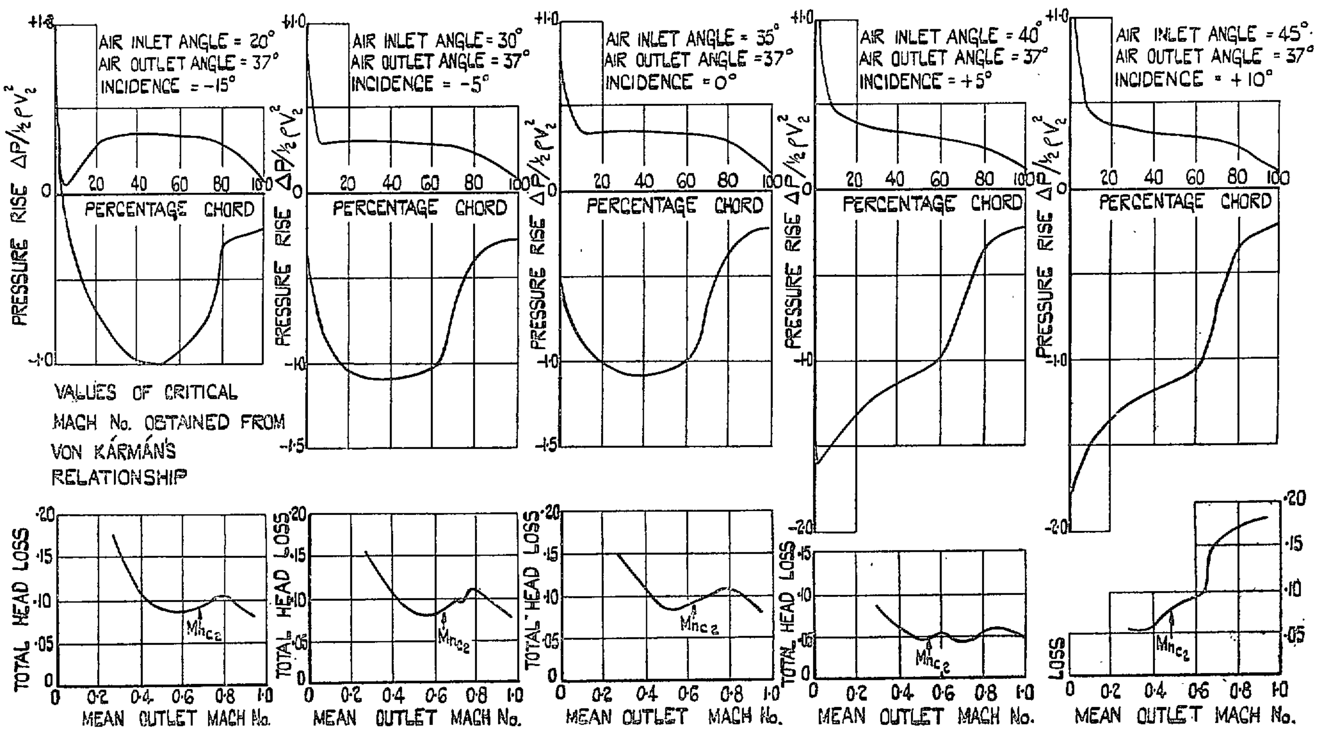

FIG. 7. Theoretical pressure distributions and loss for turbine cascade No. 6.

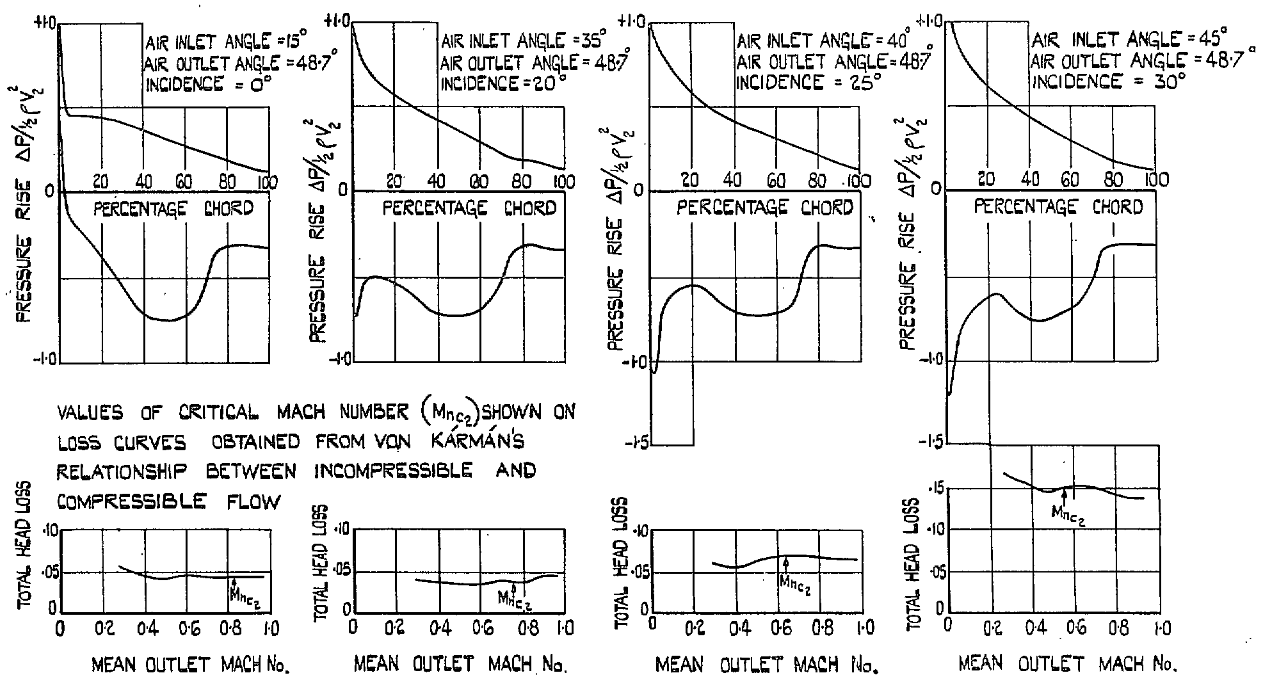

FIG. 8. Theoretical pressure distributions and loss for turbine cascade No. 7.

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