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Some Tests on Cascades of Compressor Blades fitted with Vortex Generators.

by

R. Staniforth

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Some tests on cascades of compressor blades fitted with vortex generators

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SUMMARY

Cascade tests showed that the fitting of vortex generators to the convex surface of compressor blades increased the two-dimensional losses under all conditions. A more detailed investigation indicated that the vortex generators were, in fact, suppressing shock-induced flow separation, but that boundary layer separation still occurred at about 50 per cent chord - probably due to the increased boundary layer thickness produced by the vortex generator drag.

Whilst there is always a possibility that some different configuration will be successful, the scheme of increasing the drag critical Mach number of a compressor cascade by means of vortex generators is, on the basis of available test evidence, a failure.

CONTENTS

ILLUSTRATIONS

1.0 Introduction

It can be shown that the stage pressure ratio of a compressor depends to a large extent on the Mach number of the air relative to the blading, so that in an engine compressor, where weight and bulk are of importance, high Mach numbers must be employed. There is, however, a Mach number limit above which loss associated with compression shocks causes a large drop in efficiency. By using special blade forms, an improved high speed performance can be obtained, but as this is generally at the expense of efficient flow range, it is not an ideal solution.

When a high speed axial-flow compressor is operating near its normal working point, a near normal shock occurs above the convex blade surfaces. As the Mach number through the blading is increased, this shock strengthens, moves downstream a little, and finally forms the bow wave before the leading edge of the next blade. In inviscid flow, the shock adjacent to the blade surface is normal to the local flow direction, and is transonic, and forms part of the compressing process. If however flow separation at the shock were to occur, not only would a substantial part of the shock compression be lost, but diffusion in the remainder of the blade passage would be seriously impaired.

As the contribution of direct shock loss to the total loss in a cascade operating at high subsonic relative speeds is quite small, the increase in losses due to mixing of the separated flow completely swamps any decrease in shock loss which may occur. Thus it is wholly advantageous to suppress shock-induced flow separation in compressor blading.

Previous work has shown that vortex generators can be used to inhibit turbulent flow separation following a compression shock and it was therefore considered possible that they could be used to delay shock-induced separation on conventional aerofoil shaped blades in cascade, thereby increasing the drag critical Mach number.

This paper concerns tests made on cascades of blades fitted with vortex generators to test the practicability of the above scheme.

2.0 Voxtex generator design

Initially it was decided to use an existing nozzle, cascade etc., as these tests constituted only a minor investigation. Suitable tunnel nozzles were available[1] together with a cascade of blades of section 10C4/35C50, stagger - $27\frac{1}{2}^{\circ}$ pitch/chord 0.75 and chord 1.3 in. Preliminary work to determine how small vortex generators could be made showed that they could be made 0.01 in. high, 0.03 in. chord and 0.005 in. thick without much difficulty (Figure 1A). Smaller ones could, no doubt, have been made but the additional work involved was not justified for the present tests. Thus, to start with, the vortex generator height was fixed at 0.77 per cent chord.

For the vortex generators to be effective, both their height and location with respect to the shock must be estimated fairly accurately. A suitable generator height has been found to be 1.4δ where δ is the boundary layer thickness at which the velocity is 0.95 of the free stream value[2]. Calculations show that the laminar boundary layer thickness at $7\frac{1}{2}$ per cent and 15 per cent chord is about 0.026 and 0.037 per cent chord respectively for the Reynolds number of the present tests. It is the adverse pressure gradient which follows the peak velocity point which usually promotes transition on a compressor blade convex surface, but the fitting of vortex generators introduces a flow blockage and so upstream an adverse pressure gradient which will probably cause earlier transition. Thus assuming that transition to turbulent flow occurs just before the vortex generators and that the velocity profile then follows a 1/7 power law, the height of the vortex generator from boundary layer considerations would be 0.24 and 0.34 per cent chord at $7\frac{1}{2}$ per cent chord and 15 per cent chord respectively - roughly half that to be used in the tests. This discrepancy was not considered serious at the start of this work.

A study of Schlieren photographs of the flow through similar cascades indicated that the shock causing flow breakdown at high Mach numbers was located at about 20 per cent chord. From Reference 2, the optimum location of the vortex generators has been found to be roughly twenty boundary layer thicknesses upstream of the shock which, in this case, comes to 3.5 to 4.5 per cent chord - depending on the shock position. Thus it would seem that if they are fitted at 15 per cent chord, and the shock should meet the blade surface at 20 per cent chord, the arrangement should be close to the optimum and shock-induced separation should be inhibited. On the other hand, if the shock were located further upstream, the $7\frac{1}{2}$ per cent chord position would be the better.

The vortex generators for these experiments were manufactured with a base which could be cemented to the blade surface. Most of the tests described in this paper were carried out with this base projecting above the aerofoil surface, so adding to the drag. Rough estimates of the vortex generator drag can be made as follows. Of the assumptions necessary for this calculation, the most important is that the local Mach number at the vortex generator is 1.0 when the flow inlet Mach number is 0.6 which gives the ratio of velocity at vortex generator to inlet velocity as 1.58. The approximate drag coefficient of the vortex generator itself can be estimated from the low speed tests on one of similar dimension[2], due allowance being made for the greater relative thickness of the blades used in the present tests, but not for Mach number effects or scale because of a lack of data. This gives a drag coefficient (based on the generator length and height) of roughly 0.37, which implies for the cascade as a whole a drag coefficient of 0.71×10^{-2}, or as a percentage loss in inlet dynamic head 0.62. In a similar manner, by assuming that its drag coefficient is unity, the loss caused by the 0.005 in. thick base becomes roughly 1.1 per cent inlet dynamic head. The total of the above losses, 1.7 per cent inlet dynamic head, is by no means negligible compared with the normal two-dimensional losses of 2 per cent inlet dynamic head but small enough to allow any changes in blade drag in the shock stall region to be studied.

3.0 Description of first cascade tunnel

As mentioned previously, the first series of tests was carried out in the tunnel described in Reference 1, using a cascade of nine blades, profile 10C4/35C50, chord 1.3 in., aspect ratio 2.28, set at a pitch/chord 0.75, and a stagger $-27\frac{1}{2}^{\circ}$ (Figure 1(B)). The blade section C4, whilst not exceptional in its high speed performance, was considered satisfactory because of the purely comparative nature of these tests.

For its original purpose, the tunnel was provided with slots for removal of the boundary layers on the two end walls (i.e. the walls roughly in the same plane as the cascade blades) so that substantially uniform conditions existed along the cascade. However, since suction facilities were not available in this instance, these slots were sealed. This, of course, resulted in the flow conditions along the cascade being non-uniform and it was necessary to assume that the flow over the two centre blades was sufficiently close to that in an infinite cascade.

The blade profile loss was measured at the mid blade height, initially using a remotely operated pitot traverse gear, but later with a fixed pitot comb. Although the angle of the pitot and rake was adjusted to be equal to the air exit angle calculated from the standard deviation rule, an apparent loss in total head was sometimes recorded across the cascade in midstream. Thus, in calculating the true loss through the cascade, the midstream total pressure as measured was used as a datum.

The vortex generators were normally attached to the blades with "Durofix". However, when they were fitted at the forward station, or when they were recessed into the blade surface, it was found necessary to soft solder them in place.

4.0 Test procedure in first tunnel and results

The test procedure consisted of taking total head pressure readings after the centre two blades of the cascade at mid blade height, over a Mach number range of from 0.2 to the choking Mach number at nominal air inlet angles of 30°, 40°, 45°, 50° and 55°(the blade inlet angle being 45°). The inlet Mach number was taken as given by the total pressure before the tunnel nozzle and the pressure at a static hole about one chord upstream of the centre blade of the cascade, and the profile loss as $\frac{1}{s}\int_0^s P_{TOT}\,dy$ divided by the inlet dynamic head (see Appendix for notation).

Tests were made both without the vortex generators fitted, and with them at 15 per cent chord from the leading edge, the results being shown in graphical form in Figures 2 and 3. It is evident from these test results that the vortex generators are totally detrimental to the performance of the cascade; the minimum profile loss is doubled, the drag rise associated with compressibility appears to occur at a lower Mach number, and the unstalled incidence range is reduced. Nevertheless, it was noted on examination of the blades after these tests that there were dust trails downstream of the vortex generator extending up to half chord which indicates that vortices were being generated after the desired pattern.

As it was thought possible that the 0.005 in. base of the vortex generator was promoting boundary layer separation and so causing this failure, a test was carried out with the base fitted into a recess milled in the blade surface. This and all later tests were made at only one air inlet angle so as to reduce the amount of labour involved in obtaining and reducing the results. From this test (Figure 4(A)) it can be seen that mounting the base flush with the blade surface reduces drag of the vortex generators by about one third, but they are still completely detrimental to the cascade performance.

Because the drag critical Mach number of the cascade was adversely affected by the addition of vortex generators, it was thought possible at that time that they were located too far downstream to be effective, or even that the shock was occurring before the generator. To investigate this possibility, the cascade was next tested with the vortex generators fitted to the blades at 6.2 per cent chord instead of 15 per cent chord as before, both with the base proud and with it recessed. These tests (Figure 4(B)) showed, as did the previous ones, that the vortex generator increased the loss at all Mach numbers; the loss increase being greater with the generator at the 6.2 per cent chord position than at the 15 per cent chord position. Fitting the base flush with the blade surface did not, in this test, cause any significant change in the loss.

In an endeavour to determine the position of the shock on the blade surface, a traverse was made chordwise across a blade with a small pitot tube in contact with the blade surface. The readings taken indicated that an abrupt pressure increase occured in the region of 20 per cent chord which, it was believed, was caused by the shock in question. Thus it was quite certain that the vortex generator positions tried covered the range over which it would be reasonable to expect them to function satisfactorily.

The fact that the above tests did not show any beneficial effect whatsoever on the cascade performance was puzzling as previous tests in a wind tunnel were promising. Additionally, at about 2.5 per cent the drag increase due to fitting the vortex generator to the blades was considerably greater than the 1.7 per cent estimated (see Section 2.0), which pointed to the presence of undesirable secondary effects. As the tunnel used for the above tests did not lend itself to the more detailed studies which would be required to resolve these difficulties, a new tunnel was constructed, as described below.

5.0 Description of second cascade tunnel

In Section 2.0, it has been shown that the height of the vortex generators used in the first tests were too large compared with the blade chord. To remedy this, the new tunnel was designed to accommodate 4 in. chord blades - that is roughly three times the original size. The tunnel was constructed of wood, except for the sidewalls of the working section which were made of "Perspex" (Figure 5).

To reduce air loads on the blades and tunnel structure, air was sucked through the tunnel using an ejector, instead of blowing it through as in previous tests. The flow capacity of the ejector used limited the working section area to roughly 34 sq.in. so that with an aspect ratio of 2 - the minimum which would give acceptable results - only three blades could be used. Because of the small numbers of blades, the two outer blades must operate unstalled, so it was essential to provide for endwall suction. An attempt was made to improve the flow in the tunnel at high speeds with a perforated wall (Figure 5), but initial tests showed that this had little effect on the flow in the tunnel, and it therefore was not used.

A coarse gauze was fitted across the tunnel inlet, not only for safety, but also to increase the tunnel turbulence to guard against Reynolds number effects.

The blades used in this tunnel were made of wood to the same section as used before (10C4/30C50) but the accuracy to which the aerofoil profile was manufactured was inferior to that of the first set of blades. Additionally, the air loading on the blade at high tunnel speeds caused an appreciable deflection of the blade at midspan so that small discrepancies in behaviour between the two test series might be expected. In all cases, the vortex generator strip was attached to the blade surface with "Durofix".

To enable clear Schlieren photographs to be taken of the flow a section of the "Perspex" tunnel wall by the cascade was replaced by glass, the latter being drilled to accommodate pins for mounting the blades. The Schlieren system used was of a simple type using two lenses and a spark light source.

Fixed pitot rakes were used to measure the profile loss on the centre blade. Rakes were also fitted behind the two outer blades in some tests to ascertain that they were unstalled.

The suction at the blade ends was adjusted in every case until water jets from $\frac{1}{2}$ mm hypodermic tubes projecting through the sidewall about 1 in. directly upstream of the end two blades impacted on the respective leading edges and split roughly into two equal parts. In theory, this setting is not quite correct but it was found to be quickly and easily repeatable - two qualities essential in comparative testing.

6.0 Test procedure in second tunnel and results

The test procedure was identical to that used earlier; the profile loss was determined over a range of Mach numbers up to the choking Mach number, without vortex generators fitted and with them fitted in turn at 15 per cent chord and $7\frac{1}{2}$ per cent chord (Figure 6). In these tests also, the blade profile loss was increased considerably by adding the vortex generators, although not as much as in previous tests due, no doubt, to the relatively smaller generator height. If the vortex generator base had been fitted into the blade surface, this loss increase would presumably have been reduced to about 1 per cent of the inlet dynamic head.

An interesting result was observed when the vortex generator was fitted at 15 per cent chord: as the inlet Mach number approached 0.64, the profile loss rose rapidly, but in the region of Mach 0.7, the loss dropped. This phenomenon was not noticed on any other test results of this series. On investigating this matter further, it was noticed that the loss peak occurred when the shock position coincided with the vortex generator, and at higher Mach numbers, when the shock met the blade surface further downstream no separation occurred. Comparing the photographs of the flow at roughly the same Mach number, (Figure 7), it is evident that the vortex generators are, in fact, suppressing shock-induced separation, but that flow separation still occurs further downstream, so giving a gross loss much greater than in the case where separation occurs at the shock. Judging from the angles of the waves from the vortex generator, it appears that separation is inhibited at the shock at least up to a Mach number of 1.4 so there can be no doubt as to the correct disposition of the generator.

It was noticed during the above tests that the boundary layer on the convex blade surface near the trailing edge was unstable. Thus it was considered possible that the blade loading was too high and the addition of the vortex generator drag to the blade upper surface boundary layer caused a complete flow breakdown on the rear portion of the blade. To investigate this, tests were made on a blade of identical profile, but lower camber - $27\frac{1}{2}^{\circ}$ as against the original 35°. Tests on the plain blade showed that incipient flow separation was now absent but, as in all previous tests, fitting the vortex generators increased the drag at all Mach numbers: indeed, at no time was shock-induced or other separation noted before they were fitted, but afterwards at all Mach numbers a separated region was present near the blade trailing edge. Obviously, as this cascade design did not exhibit shock-induced boundary layer separation, adding the vortex generators could not improve matters. However, this test did show that they were causing a harmful boundary layer disturbance - probably thickening the boundary layer - and so, flow separation.

7.0 Conclusions

In all the tests described, vortex generators had a harmful effect on the performance of a cascade of aerofoils, even though the primary aim of suppressing shock-induced boundary layer separation was achieved. This, it appears, is principly due to a flow separation at about 50 per cent chord caused by the thick boundary layer following the vortex generator.

Possibly, but not very likely, a different vortex generator design having, say, thinner blades and so lower drag may overcome this difficulty or perhaps even a second row at about 40 per cent chord would be effective in suppressing the separation observed at 50 per cent chord. However, on the basis of the available evidence, the scheme of increasing the critical Mach number of a cascade blade with vortex generators is a failure.

ACKNOWLEDGEMENT

To Mr. D. W. Kew of Loughborough College for his assistance in the early stages of this work.

REFERENCES

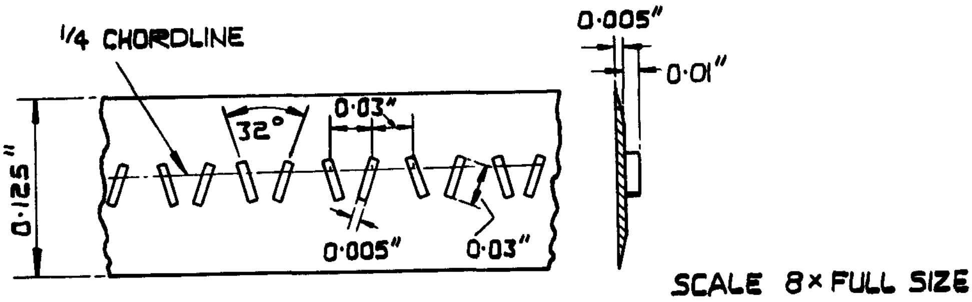
No.	Author(s)	Title, etc.
1	S. J. Andrews	Tests related to the effect of profile shape and camber line on compressor cascade performance. A.R.C. R and M 2743, October, 1949.
2	L. H. Tanner, H. H. Pearcey and Miss C. M. Tracey	Unpublished work at the National Physical Laboratory.

APPENDIX

List of symbols

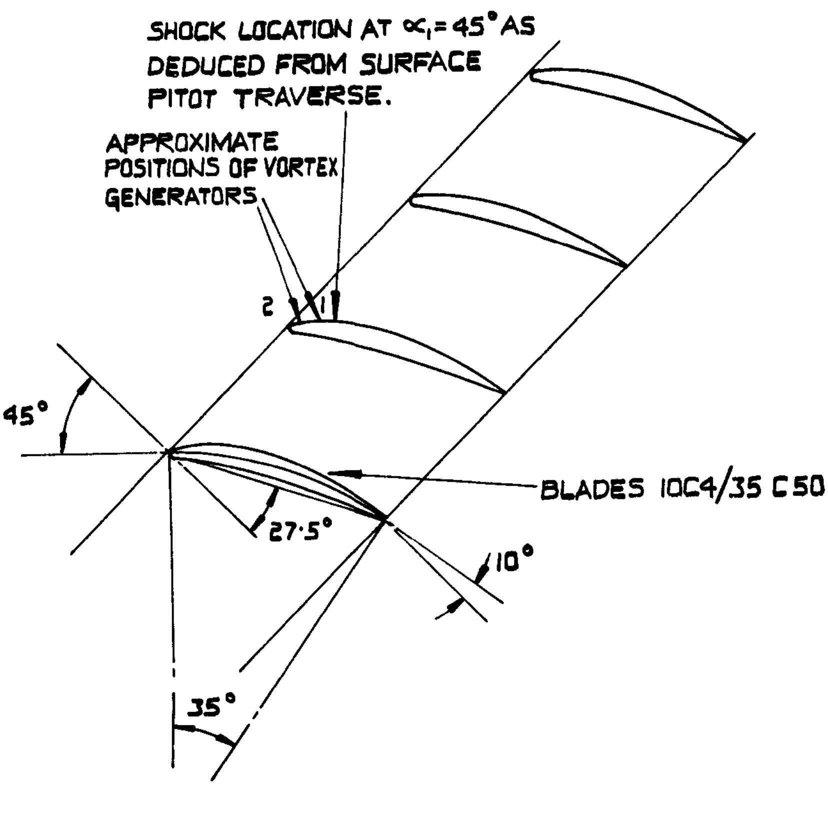
P_{TOT}	Total pressure
P	Static pressure
s	Blade pitch
y	Distance measured parallel to a cascade
α_1	Air inlet angle to cascade
δ	Boundary layer thickness at which velocity is 0.95 free stream value

FIG. I



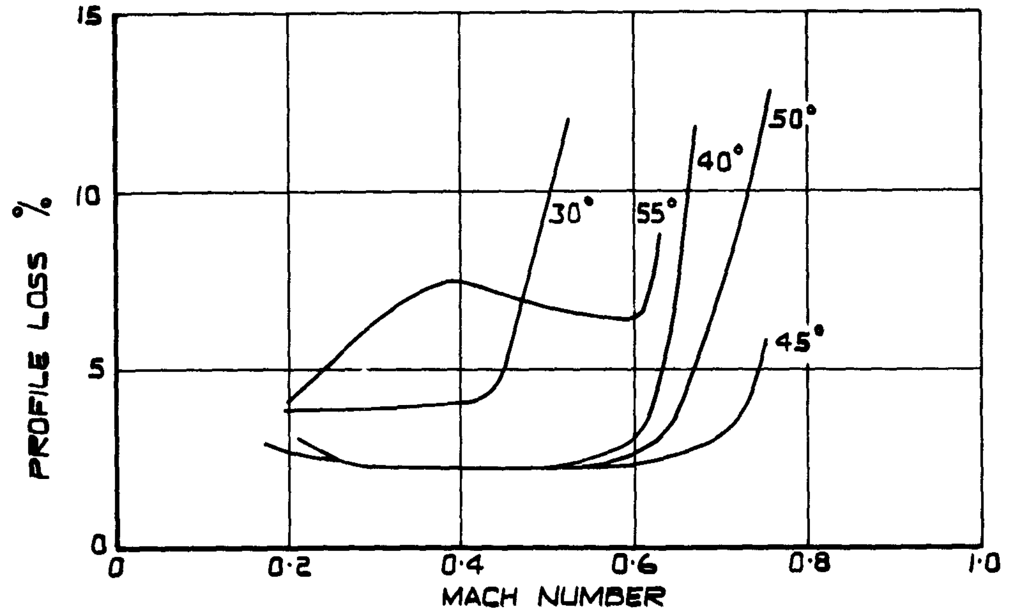
(A) DETAILS OF VORTEX GENERATOR STRIP

(COUNTER-ROTATING TYPE)

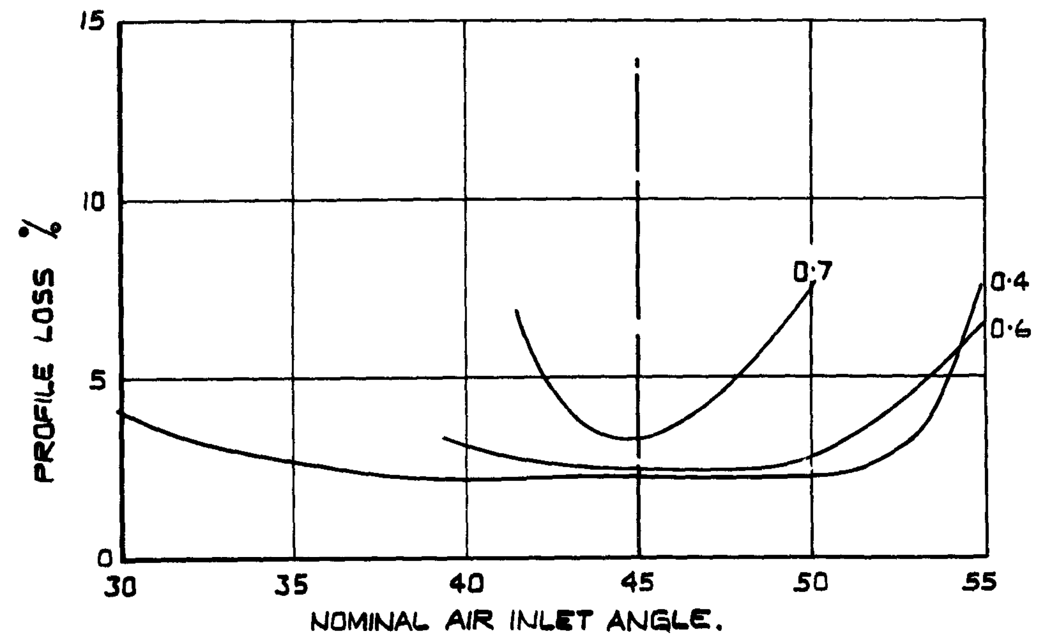


(B) CASCADE AS TESTED

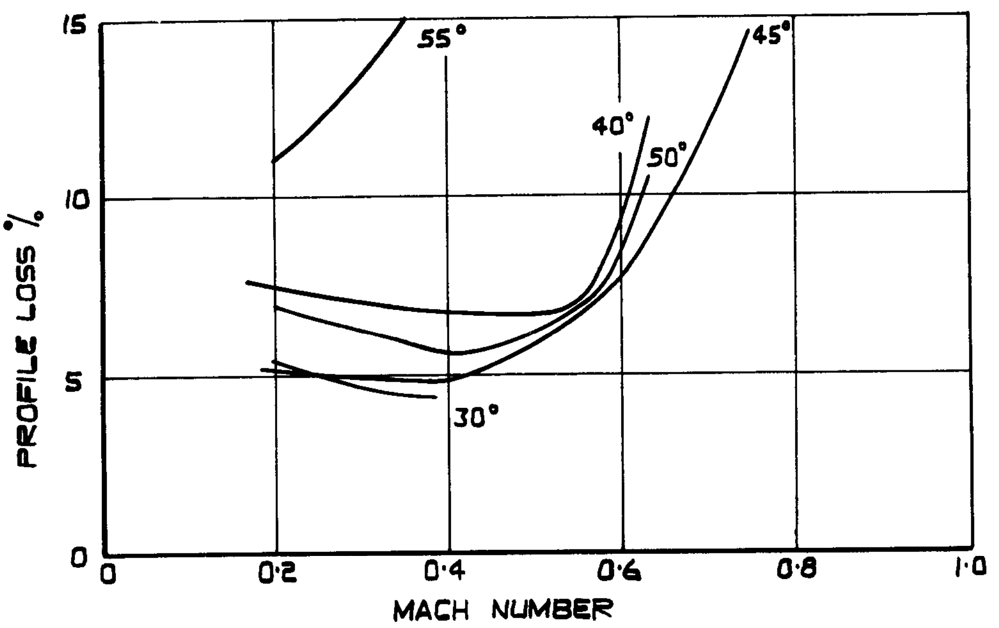
FIG. 2



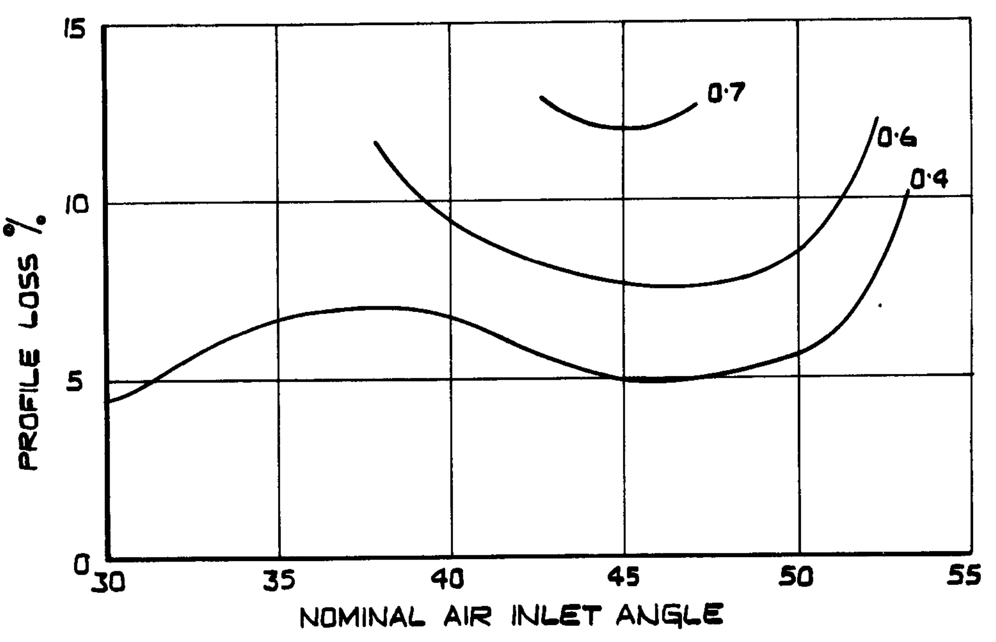
(A) PROFILE LOSS AGAINST MACH NUMBER FOR PLAIN BLADE.



(B) PROFILE LOSS AGAINST NOMINAL AIR INLET ANGLE FOR PLAIN BLADE.

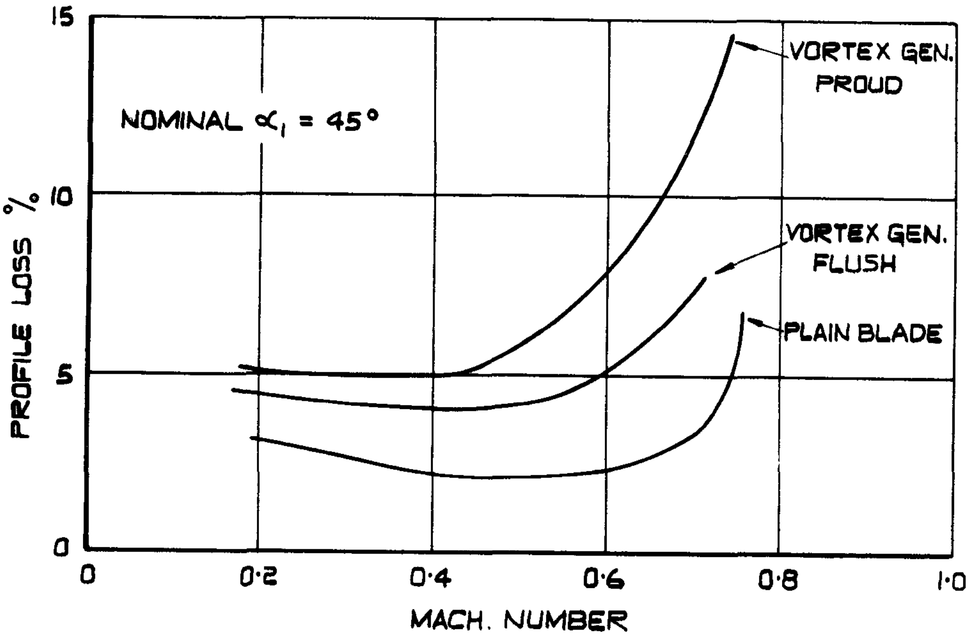


(A) PROFILE LOSS AGAINST MACH NUMBER FOR BLADE FITTED WITH VORTEX GENERATORS AT 15% CHORD.

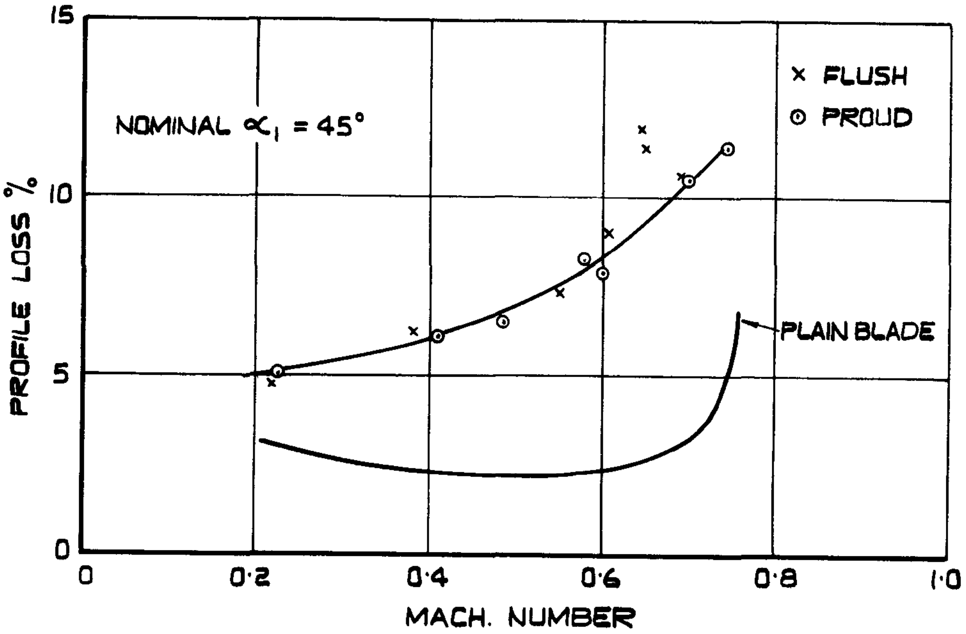


(B) PROFILE LOSS AGAINST NOMINAL AIR INLET ANGLE FOR A BLADE FITTED WITH VORTEX GENERATORS AT 15% CHORD.

FIG. 4

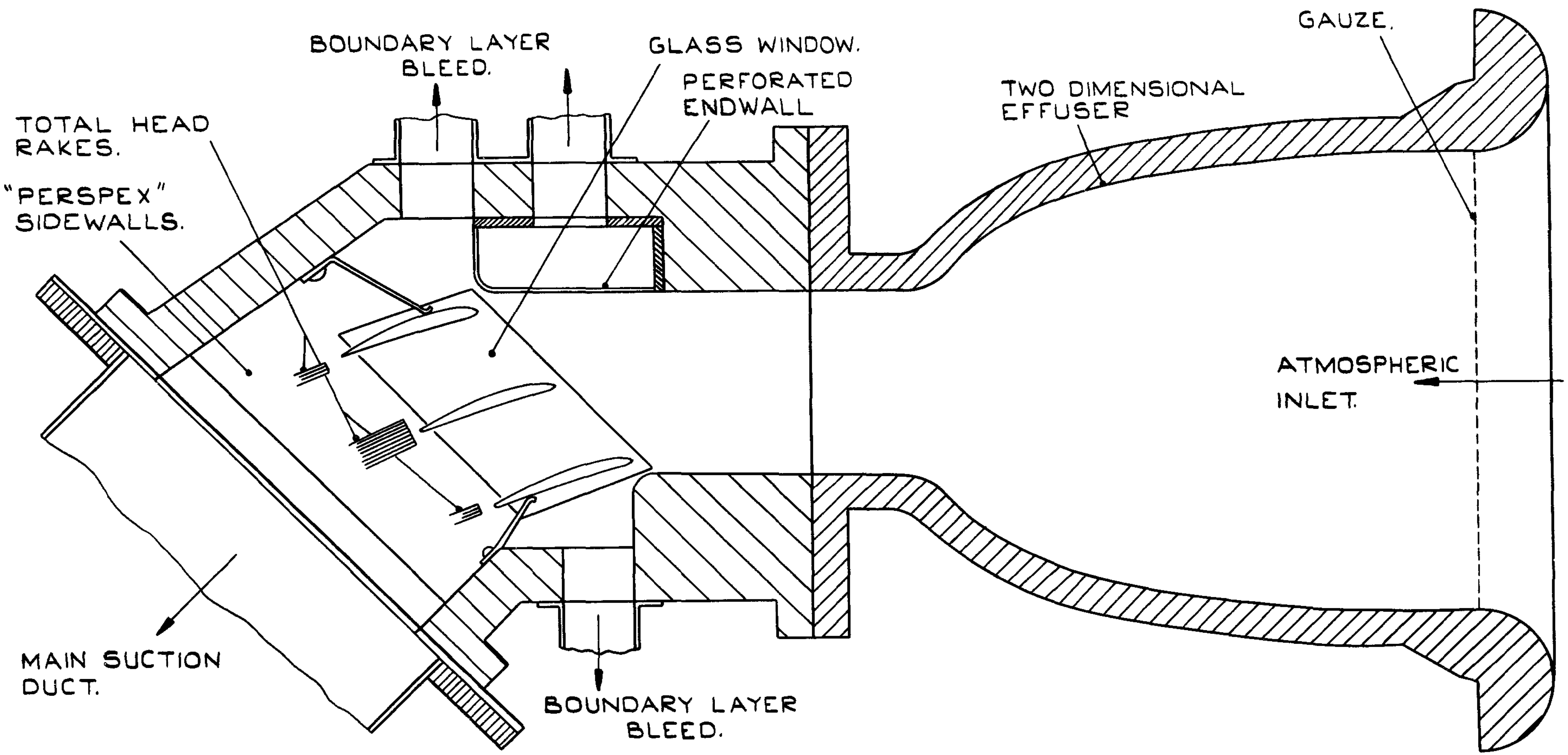


(A) PROFILE LOSS AGAINST MACH NUMBER
(VORTEX GENERATOR AT 15% CHORD.)



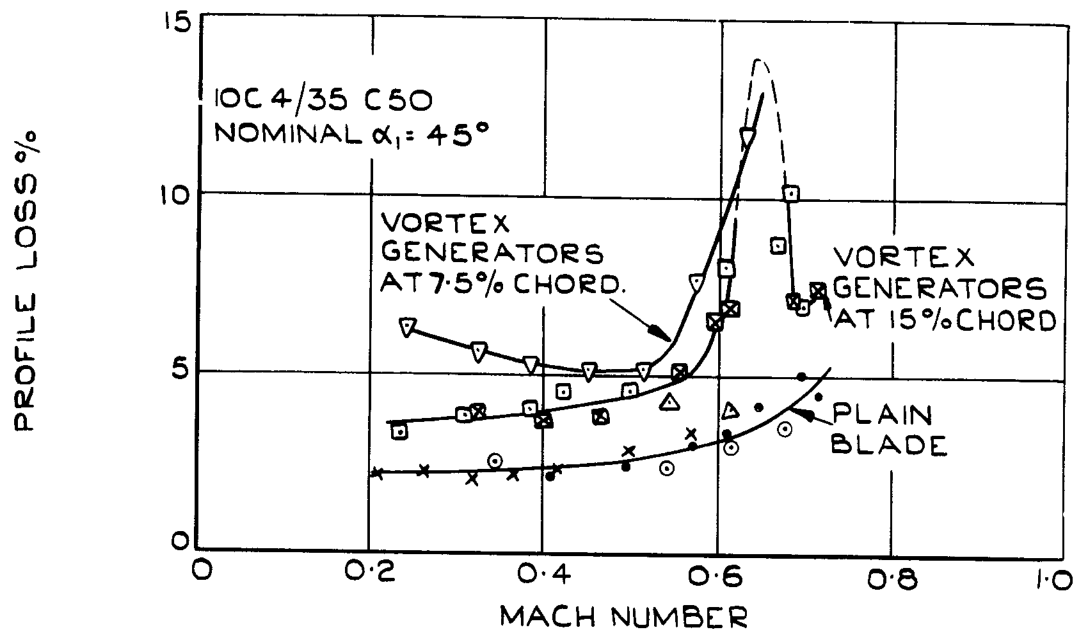
(B) PROFILE LOSS AGAINST MACH NUMBER
(VORTEX GENERATOR AT 6·2% CHORD)

FIG. 5

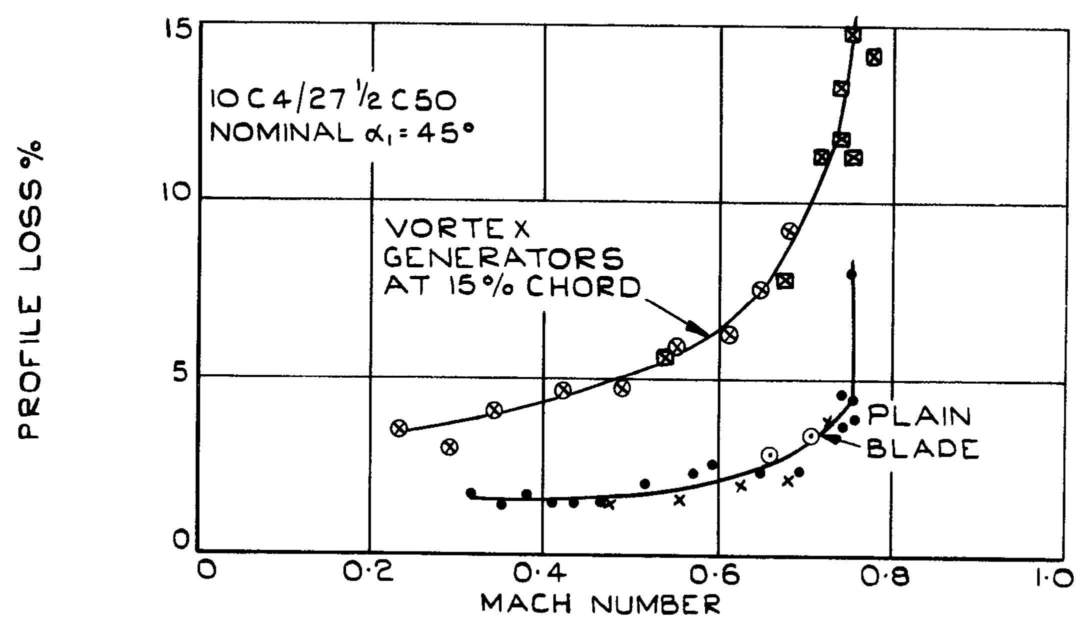


SUCTION CASCADE TUNNEL

FIG. 6



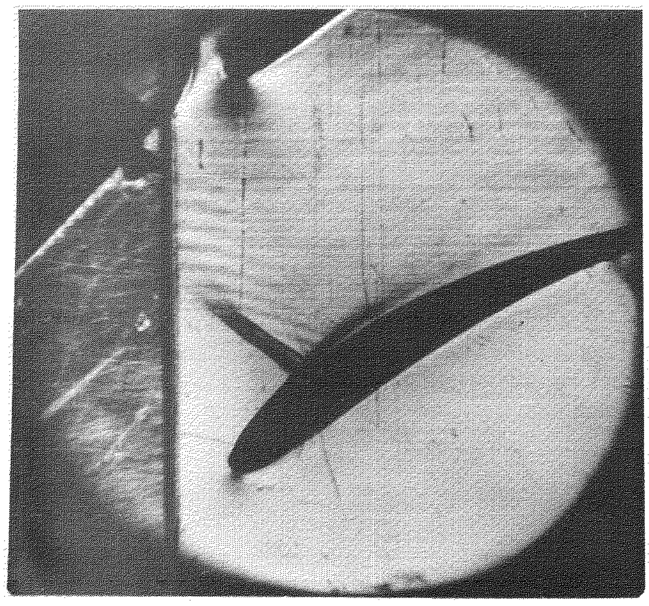
(A) PROFILE LOSS AGAINST MACH NUMBER AS DETERMINED IN SUCTION CASCADE TUNNEL-35° CAMBER BLADE



(B) PROFILE LOSS AGAINST MACH NUMBER AS DETERMINED IN SUCTION CASCADE TUNNEL-27 ½° CAMBER BLADE

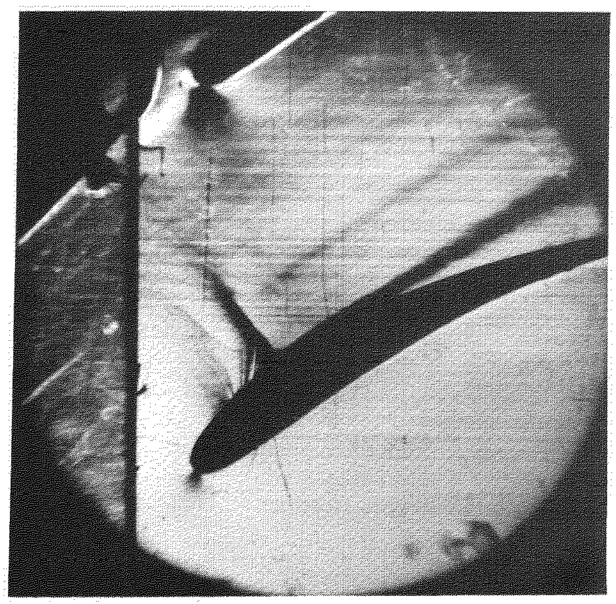
FURTHER TEST RESULTS.

FIG. 7



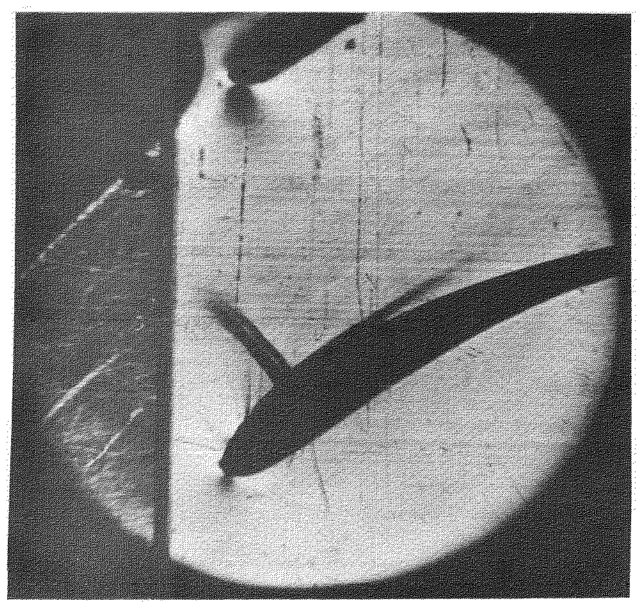
(A) NO VORTEX GENERATORS

MACH No ≏ 0·71



(B) VORTEX GENERATORS 15 % CHORD

MACH No ≏ 0·72



(C) VORTEX GENERATORS 7½ % CHORD

MACH No ≏ 0·70

PHOTOGRAPHS OF FLOW OVER 35° CAMBER BLADES

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