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BREAKDOWN OF TIP LEAKAGE VORTICES IN COMPRESSORS AT FLOW CONDITIONS CLOSE TO STALL

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ABSTRACT

The breakdown of tip leakage vortices at operating points close to the stability limit of transonic compressor rotors has been detected. The aerodynamic phenomenon is considered to have a major impact on stall inception. Computations have been carried out and a detailed visualization of the phenomenon is given. In addition the connection of vortex breakdown to rotating instabilities and stall is discussed. Furthermore the tip flow field of the axial rotor is compared to the results for a centrifugal and a mixed flow compressor operating at similar tip speeds.

INTRODUCTION

In the present case, vortex breakdown has been located in a transonic compressor rotor downstream of the shock-vortex interaction at design speed near stall (Fig.1). The location of the shock-vortex interaction is a function of the back pressure. If the back pressure reaches a certain level the position of the interaction of the shock with the tip leakage vortex moves upstream and the burst vortex no longer heals up. The endwall blockage grows rapidly and the compressor stalls. In this case the blockage extends upstream of the leading edge.

Initially vortex breakdown was observed for leading edge vortices formed on delta wings at large angle of attack. The non-linear aerodynamic phenomenon which has been intensively investigated under laboratory conditions (Krause,1990), (Althaus et al.,1995) has a significant impact on the flow field. Referring to Sarpkaya (1971) the term 'vortex breakdown' describes 'an abrupt change in the structure of the core of a swirling flow'. Even though several vortex structures at breakdown (bubble type & spiral type) have been observed and investigated, vortex breakdown cannot be considered fully understood and stability criteria are still missing. Nevertheless, from experiments in tube flows it is known that vortex breakdown is strongly influenced by the boundary conditions (in case of transonic rotors by the annulus wall contour). The effect of the boundary conditions on the initiation and development of the breakdown phenomenon is significant in so far that the axial pressure gradient is affected. This will be demonstrated by comparing the axial compressor case with centrifugal compressors operating at similar flow conditions.

The tip flow field of high-speed transonic compressor rotors is highly sensitive to the size of the tip clearance. It is well known that increased tip clearances have a deleterious effect on stall margin and efficiency. Especially in case of high inlet Mach numbers greater than 1.4, the interaction of the shock wave with the tip leakage vortex is crucial to the flow physics of the endwall region. In the past, experimental and computational techniques have been used to investigate the flow field close to stall. Although sophisticated experimental tests with high frequency pressure transducers (Russler et al.,1995) and laser anemometry (Suder et al.,1994) have been applied to describe the tip flow structure, it has been impossible to predict which phenomenon causes the compressor rotor to stall. Of course experimental investigations in a rotor are difficult per se, especially close to the stability limit of a compressor where the experimentalist needs to be careful to avoid the loss of the test model. Concerning the resolution of the inherent flow structures it is necessary to install high frequency pressure gauges. However, since pressure measurements are restricted to the shroud, they give only a limited insight into the complex three-dimensional flow phenomena at the tip. In principal, laser anemometry can be used to measure the flow field within a rotor but, due to the restriction to one fringe in the inertial frame of reference, the data does not resolve the flow phenomena close to stall which only occur during a very small time interval.

Because of the limited availability of detailed measurements, most of the understanding of the tip flow field in high speed compressor rotors has been obtained from theoretical considerations and numerical simulations.

Studies of the shock/vortex interaction were carried out by Suder et al. (1994), Adamczyk et al. (1993) and Chima (1996) using three-dimensional Navier-Stokes solvers. These studies have linked the rapid growth of the tip leakage vortex due to shock/tip leakage vortex interaction to stall. Especially the effect of the tip clearance size on the compressor stall margin has been investigated by Dring et al. (1995). It has been demonstrated that today's Navier-Stokes codes are capable of predicting the pressure rise characteristic through the point of slope change from negative to positive. In addition it is pointed out that although compressor instability is an

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unsteady event, computations which assume the existence of a steady solution can be correlated with an experimentally observed stall margin.

The vortex breakdown described in this paper is caused by a sudden deceleration of the flow across the passage shock. The flow structure is similar to the interaction of a supersonic wing-tip vortex with a shock wave (Cattafesta and Settles, 1992; Smart and Kalkhoran, 1995) and thus differs from the widely investigated breakdown phenomenon taking place under the influence of a more gradual adverse pressure gradient (Krause, 1990). Furthermore, in the present case the streamwise location of breakdown is fixed by the passage shock whereas it is known to fluctuate significantly in the low-speed cases. Nevertheless, the appearance of the bubble indicates a strong similarity to the low-speed bubble-type vortex breakdown.

METHOD

Algorithm

The computations have been carried out using the Visiun commercial code based on the Nasa code Arc3d. The algorithm is based on the implicit Beam and Warming approximate factorisation algorithm (1977) to solve either the three-dimensional Euler or Navier-Stokes equations. It is capable of treating general geometries as long as the grid is structured and varies smoothly.

The algebraic set of equations resulting from the approximate factorisation is diagonalized following Pulliam and Chaussee (1981) and then solved by the method developed by Pulliam. The finite-difference method uses artificial viscosity to prevent numerical instabilities (Pulliam, 1986).

Boundary Conditions

At solid boundaries, the pressure distribution is found by solving the normal momentum equation. Periodic boundaries are treated explicitly.

Due to the theory of characteristics four flow values have to be specified in case of subsonic inflow. The common choice is the specification of total pressure, total temperature and in-plane velocity components or flow angles at an inflow boundary. Alternatively total pressure and total temperature can be replaced by the Riemann-invariants. Their use is especially advantageous concerning the implementation of non-reflecting boundary conditions, because Riemann invariants do not change along the characteristic direction. In this case the flow is considered inviscid at the inlet boundary. At the exit boundary only the static pressure is specified using a prescribed value at the hub and for the radial variation a solution of the radial equilibrium equation. To approach the surge line the static pressure is raised in small increments:

Suder et al. (1994) and Adamcyk et al. (1993) report so-called 'numerical stall' which basically describes the following phenomenon.

Once the flow field of a rotor at design speed is obtained, one usually increases the pressure ratio by raising the static back pressure to approach the surge line. The numerical simulation responds to the artificial throttle in a similar way to a real rotor. Once a certain back pressure is reached, the mass flow keeps on decreasing, i.e. 'numerical stall'.

Nevertheless, it is possible to carry out the described procedure carefully to capture the last point of stable operation and still have a converged solution for a pre-stall condition. Therefor it is necessary to increase the back pressure in very small increments. The increments have to be small enough to avoid the generation of pressure waves at the downstream boundary which would influence the sensitive tip flow dramatically.

All the results concerning operating points close to stall have been obtained using the tedious procedure of raising the back pressure in very small intervals. It is worth mentioning that the exit boundary condition has to be located at least one chord length downstream of the rotor trailing edge so that pressure waves due to the changed exit pressure are damped out before they reach the rotor.

Turbulence Model

An algebraic mixing length model is used to approximate the effect of turbulence. Different modifications of the model are applied in regions like the wake, hub and shroud, or blade surface.

Although it is well known that lower order turbulence closures usually fail to reproduce secondary flow effects which are turbulence driven and take place at very small time scales, normally such turbulence models are successful in predicting the pressure driven secondary flows.

The basic Baldwin-Lomax model (1978) has been extended. Because of the difficulty to find appropriate vorticity values in the complex internal flow situation the following algorithms have been implemented:

1. The length scale calculation in the three-dimensional flow field is based on the reduced vorticity value which is the vorticity component parallel to the current wall and normal to the local velocity vector.

2. In the code the search for the maximum vorticity is restricted to the boundary layer. This eliminates false values of the turbulent viscosity.

3. In the wake region the Thomas turbulence model (Thomas,1979) for jets is used. An inviscid wake can be described by a so-called contact discontinuity and the shed vorticity is proportional to the jump of the tangential velocities of the upper and lower sides of the wake. Based on this consideration it is a straightforward approach to define a mixing length which is proportional to the jump of minimum and maximum velocity in the wake region.

4. Values for the turbulent viscosities in the case of multiple walls are provided by using an averaging formula which uses a weighting function depending on the distance to the walls.

Concerning the computation of the complex flow phenomena in the radial gap, several authors claim that the use of sophisticated turbulence models is necessary. It needs to be mentioned that the vortex breakdown phenomenon occurs no matter what flow condition (either inviscid, laminar or turbulent) is specified within the tip clearance and the shroud boundary layer. Computational results show that the phenomenon is basically a convective phenomenon and its occurence does not depend on the specified amount of dissipation.

Grid Generation

The grid generator is based on an algebraic solver which uses transfinite interpolations to generate H-type grids. In order to conduct a flow analysis of tip leakage flows and to resolve leading edges and blunt edges (tip) a compound grid structure has been implemented. This requires a block-structured analysis code which divides the computations into regions.

Concerning the computation of the flow about standard unshrouded compressor blades with tip leakage flow the block-structured method is very useful. Figure 2 shows the computational grid. It consists of 258000 grid points (120 in the flow direction, 50 in the blade to blade direction and 43 in the spanwise direction) for all computations presented in this paper. The tip clearance itself has been discretized with 10 grid cells in the radial direction and 16 grid points in the circumferential direction above the last discretized rotor blade section.

RESULTS

The fan design contains 24 main blades. The tip clearance gap is set to a constant value of 0.4 per cent of span from leading edge to trailing edge.

At the design point operating condition the total pressure ratio is about 1.8 and the tip speed is 460 m/s. The rotor has transonic relative flow with a maximum inlet tip Mach number of 1.6. In Fig. 3a,b the comparison of rig data and the computational results are shown for the design point. The computational results of the circumferentially averaged total pressure ratio and the absolute whirl angle distribution at the trailing edge are in good agreement with the experimental data.

Figures 4a,b,c show the computed relative Mach numbers at a cutting plane which passes approximately through the center of the tip leakage vortex for three different back pressures. The shock/tip vortex interaction at the tip is visualized using streamlines. By comparing the three figures the evolution of the shock/vortex interaction is clearly visible. While the vortex shown in Fig. 4a only thickens and heals up further downstream, a recirculation area is visible in Fig. 4b. A further increase of the static back pressure leads to the situation shown in Fig. 4c. A large recirculation area covers most of the passage. All further attempts in increasing the back pressure lead to a severe stall, i.e. the blockage extends in front of the leading edge and the solution cannot be stabilized any more.

The radial extension of the flow phenomenon is shown in a radial cutting plane (Fig.5) which corresponds to the solution depicted in Fig. 4b. The cutting plane intersects the interaction area in the streamwise direction. The shock front interacts with the vortex and a recirculation area occurs.

For the computations described in the paragraph above the viscous effects were simulated in order to obtain a shroud boundary layer and a realistic tip clearance flow. For comparison and to investigate how dissipation influences the vortex breakdown phenomenon, computations have been performed where these areas have been treated inviscidly only. One result is presented in Fig. 6. It shows the vortex breakdown at about the same flow condition as described in Fig. 4b. No blockage occurs close to the blade surfaces because boundary layers are absent (inviscid treatment). The location of the shock/vortex interaction differs from the viscous solution. Since the boundary layers are missing the burst vortex remains closer to the suction surface. To highlight the shock-vortex flow structure, density contours are drawn on top of the Mach number distribution. Mach numbers greater than 0.6 are indicated by the red colour. The density contours do not only highlight shock fronts but also vortex sheets or contact discontinuities. Here, the passage shock is clearly visible and the kink indicates the location of the interaction with the tip leakage vortex. The interaction area has a specified diameter which is represented at the upper and lower boundary by triple points. Density contours show that vortex sheets emanate from these locations. They enclose the region of the burst vortex. In Fig. 1 a sketch of the actual flow situation is depicted which underlines the importance of the shock/vortex interaction by ignoring all the boundary layer effects which tend to smear out the physical phenomenon of vortex breakdown.

DISCUSSION

Vortex breakdown is a highly non-linear flow phenomenon. It covers a large part of the flow field and dominates other inherent flow structures like boundary layers and shock/boundary layer interactions. In compressors, vortex breakdown is the reason for a sudden growth of the blockage in the tip flow region with a dramatic influence on the mass balance of the whole passage.

In this case the gross exterior appearance of the vortex breakdown is bubble-type and the internal structure of the flow phenomenon is asymmetric. The initiation process of bubble-type breakdown has been explained by a feedback mechanism (Krause et al., 1994) which is based on the redistribution of axial vorticity into the radial and circumferential components and vice versa. Brown and Lopez (1990) explained bubble-type vortex breakdown in rotating containers by a similar feedback mechanism. The positive axial pressure gradient (compressor) of the outer flow reduces the axial velocity component while the spanwise redistribution of mass increases the radial component. Therefore, a negative gradient of the circumferential velocity of the vortex in the axial direction is obtained. This gradient drives the redistribution of vorticity from the axial into the circumferential vorticity. The circumferential vorticity component induces an axial flow against the main flow direction and leads to the deceleration of the axial velocity component. In case of a transonic rotor, the mass flow through the passage does not collapse immediately. This means that, up to a certain pressure ratio the bubble does not show unlimited growth. The reason why the burst vortex can be kept stable is a self balancing mechanism: The blockage of the passage caused by the vortex bubble leads to an acceleration of the flow close to the walls which decreases the effect of the axial pressure gradient on the deceleration of the axial velocity component.

As described above, the axial velocity of a vortex core is an important parameter concerning the stability of vortices. For several reasons common fan designs have an inclined annulus contour or have at least a constant annulus which leads to a local deceleration of the flow with a negative effect on the stability of the tip vortex.

In comparison to the axial fan case, mixed flow and centrifugal compressors which operate at transonic conditions have massive acceleration at the shroud in the inducer. Therefore, one can expect that breakdown of the tip leakage vortex due to the interaction with the inducer bow shock will not occur.

Computations of the flow field of a transonic, mixed flow compressor and a transonic, centrifugal compressor have been performed to investigate the structure of the tip clearance vortex close to the stability limit.

The mixed flow and centrifugal compressors consist of main and splitter blades and have a tip speed of about 400m/sec. The tip clearance has been set to a constant value of 0.3 per cent span similar to the fan case discussed above. Figure 7 shows a sketch of the two compressors to indicate the geometry of the annulus contour. At the design point the total pressure ratio is 7.5 for the mixed flow and 6 for the centrifugal compressor.

The rotors have transonic relative flow with a maximum inlet tip Mach number of about 1.45. Extensive experimental and numerical investigations have been carried out to investigate the flow field of these compressor rotors (Eisenlohr and Benfer, 1993, Krain et al.,1995). In Fig. 8 the computed relative Mach number distribution is shown at a cutting plane at 98% blade height and streamlines have been used to visualize the tip leakage vortex. No vortex breakdown has been observed at design speed conditions even for operating conditions close to the stability limit.

Because the pressure rise in radial impellers is produced by centrifugal acceleration of the flow, performance charts of centrifugal compressors do not show a discontinuous jump at the stability limit to an operating point on a second characteristic with lower pressure ratio and mass flow. Instead, they tend to maintain a certain pressure level at a constant speed line. At these conditions strong pressure fluctuations occur due to different stall phenomena and of course this condition is out of the operating range of a centrifugal compressor. Nevertheless, no discontinuous jump to a low pressure ratio and no breakdown of the tip leakage vortex can be observed as in axial compressors.

In the following summary, the reasons why the phenomenon of vortex breakdown plays an important role in the stall and surge behaviour of axial compressors are given:

1. The computations show that vortex breakdown leads to a large blockage area in the tip region. Although the application of different viscosity models changes the position of the shock/vortex interaction, the phenomenon of vortex breakdown is observed always, independant of the specified dissipation level.

2. Computations of the flow field close to the stability limit of a mixed flow and a centrifugal compressor do not show any indication of vortex breakdown. The reason is that the local acceleration at the shroud stabilizes the vortex.

3. Unsteady pressure fluctuations were detected with high-response transducers upstream of compressor rotors operating close to stall (Baumgartner et al.,1995). It has been discussed to use the fluctuations for indicating stall inception. The pressure fluctuations are caused by upstream propagating pressure waves which are probably related to the inherent unsteady modes of the burst vortex.

SUMMARY AND CONCLUSIONS

The paper describes the detection of tip leakage vortex breakdown in a modern, wide-chord, transonic rotor caused by the interaction of the vortex with the passage shock. Navier-Stokes computations were carried out, using different viscosity models in the near shroud and tip clearance region, to verify the occurence of the phenomenon.

At a certain back pressure the burst vortex covers a large part of the flow field and results in significant blockage which finally leads to stall. In order to improve efficiency and surge margin, studies of different tip and annulus geometries which positively affect the shock-vortex interaction are needed.

A first step in that direction has been done by comparing the axial fan situation with a mixed flow and a centrifugal compressor operating at tip Mach numbers similar to the transonic rotor. Here, no indication of tip vortex breakdown has been observed. The reason is the local acceleration of the tip flow which leads to a stabilization of the vortex after it has been distorted by the passage shock.

In the future further studies of appropriate modifications of the annulus contour and the blade tip design will be carried out. This includes an investigation of the bubble growth as a function of the hade angle.

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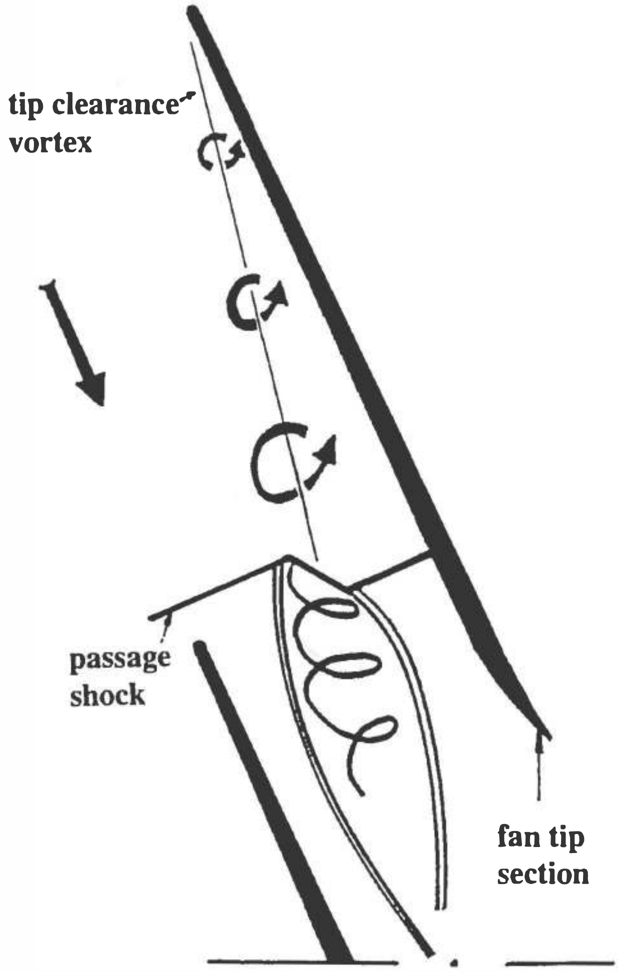


Fig.1: Sketch of the tip flow field of a transonic rotor

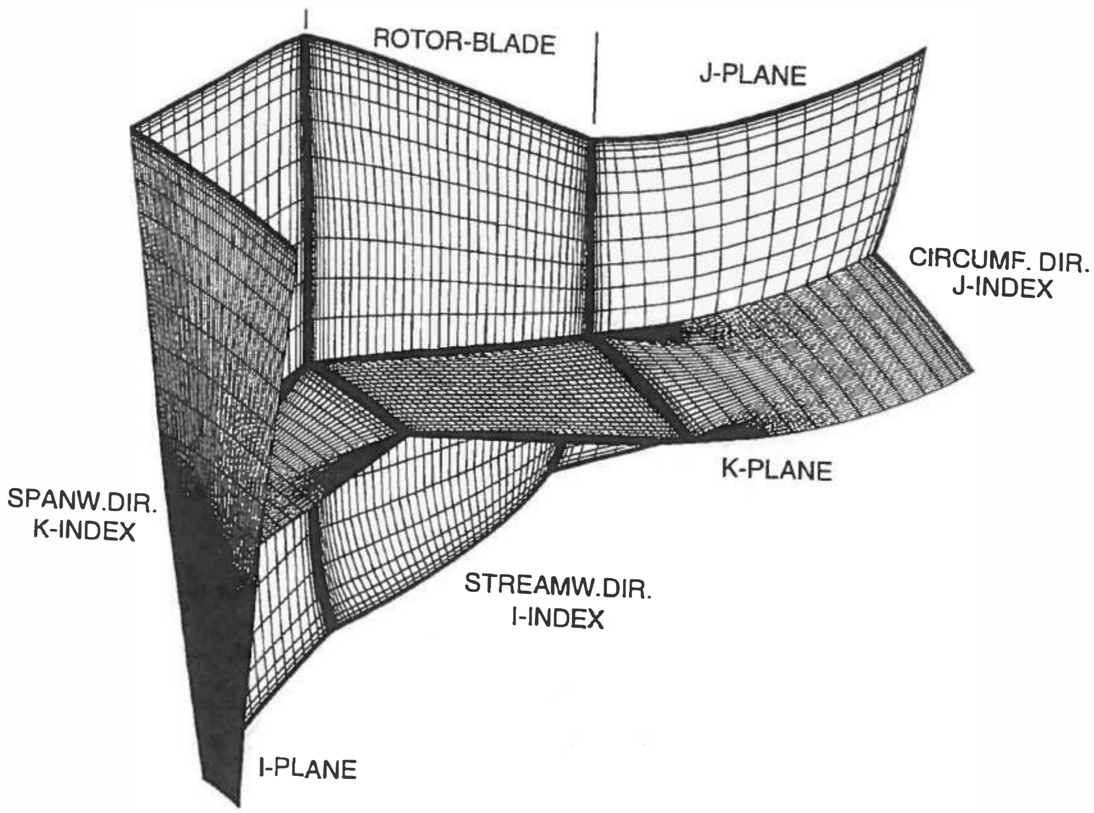
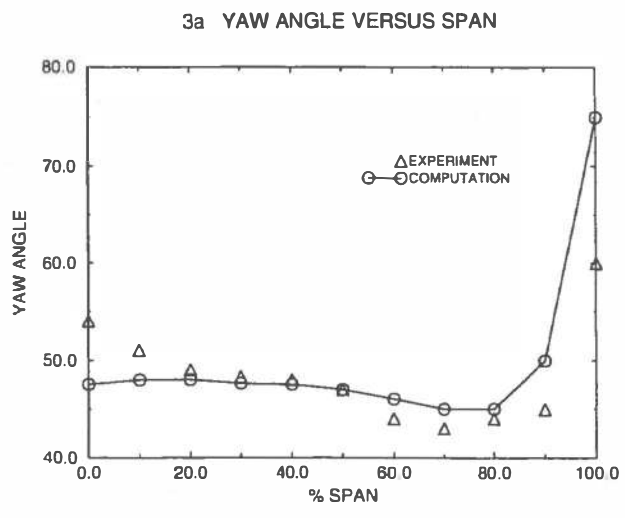


Fig.2: Computational grid (i=120, j=50, k=43)



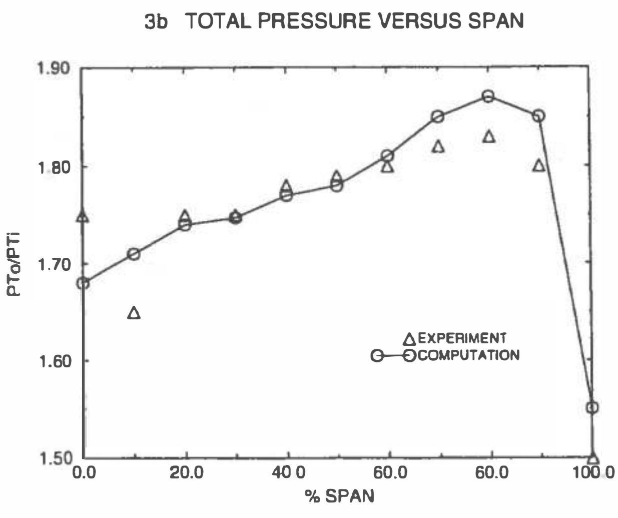
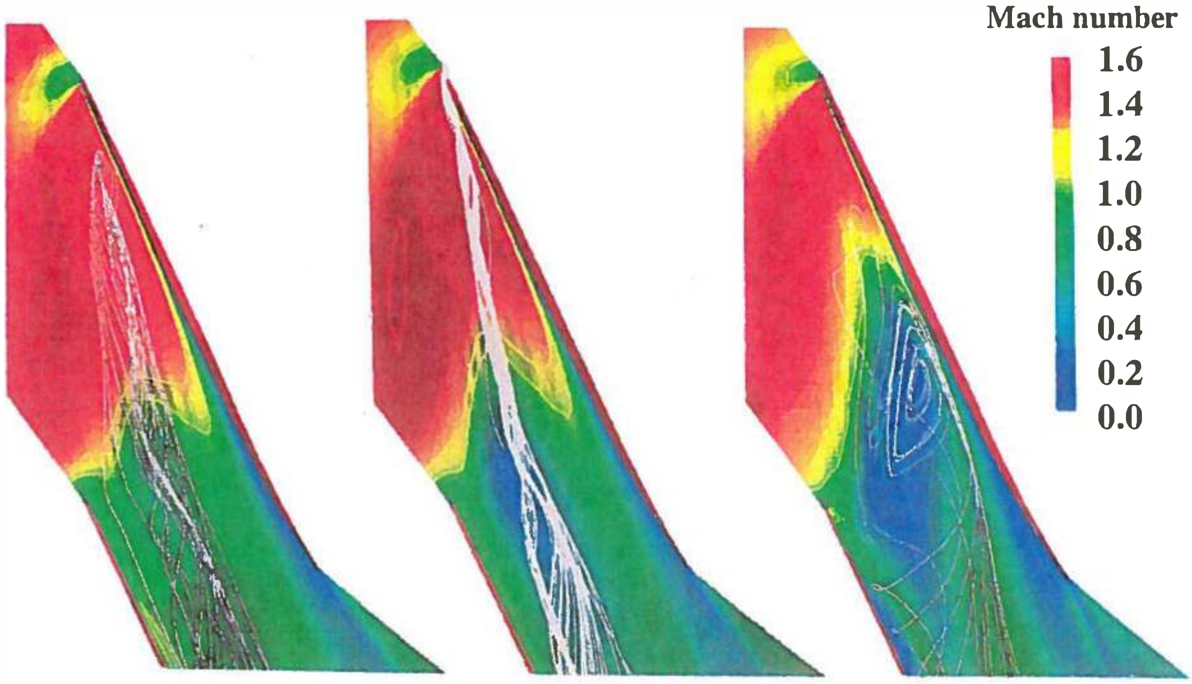


Fig.3: Comparison of yaw angle and total pressure ratio at design point conditions



4a 4b 4c

Fig.4: Evolution of the tip leakage vortex/ shock interaction due to increased back pressure from left to right, (Mach number contours)

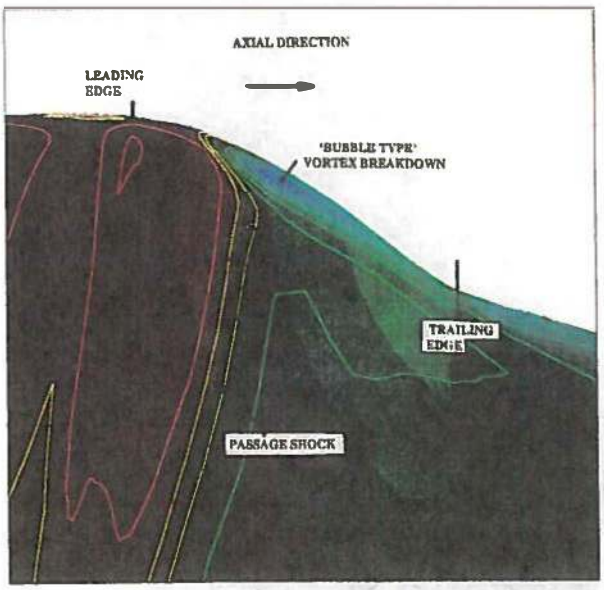


Fig.5: Flow structure in radial direction (j-plane see Fig. 2; Mach number contours)

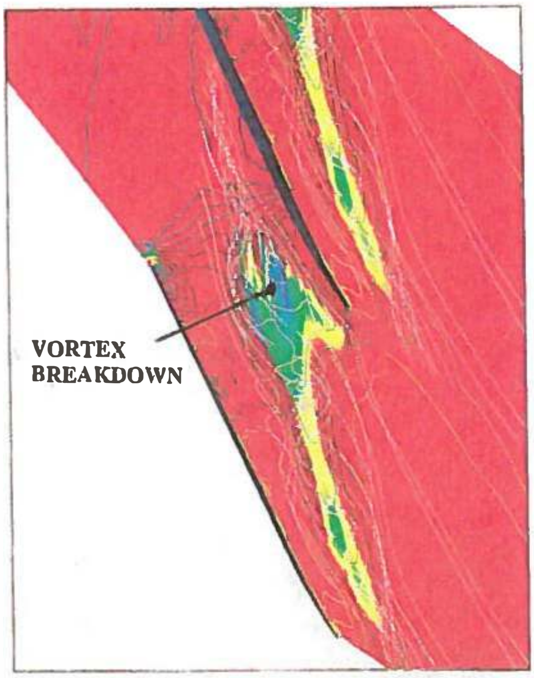


Fig.6: Tip leakage vortex/ shock interaction; inviscid treatment of the flow in the tip gap and along the shroud, (Density contours on top of Mach number spectrum plot, cut-off level M=0.6)

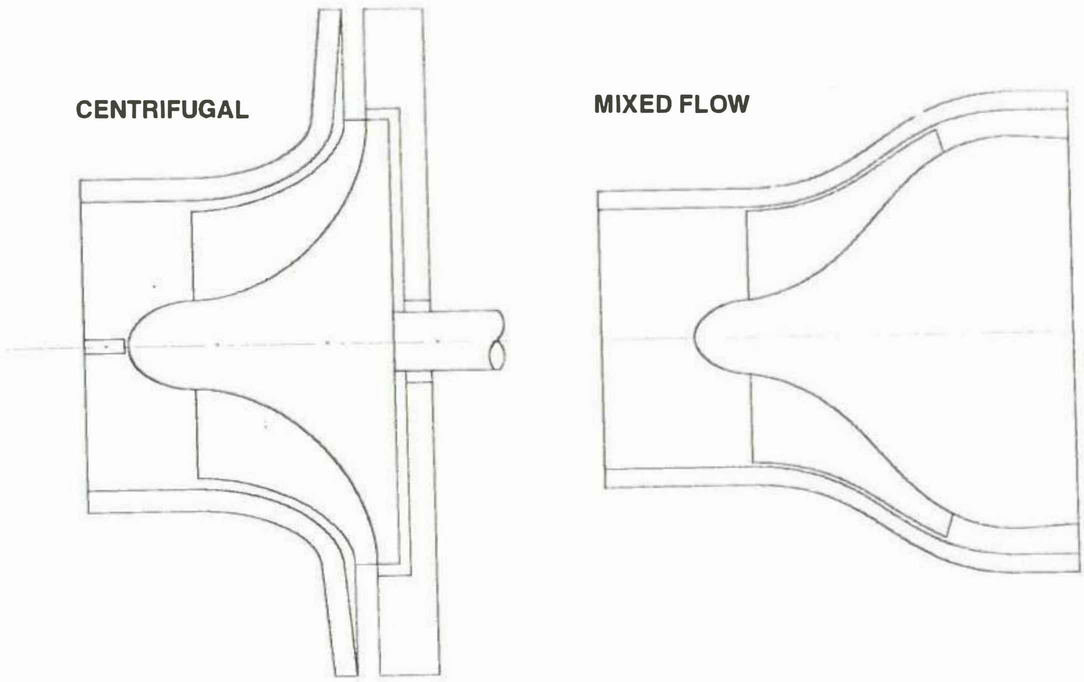


Fig.7: Sketch of the mixed flow and the centrifugal compressor

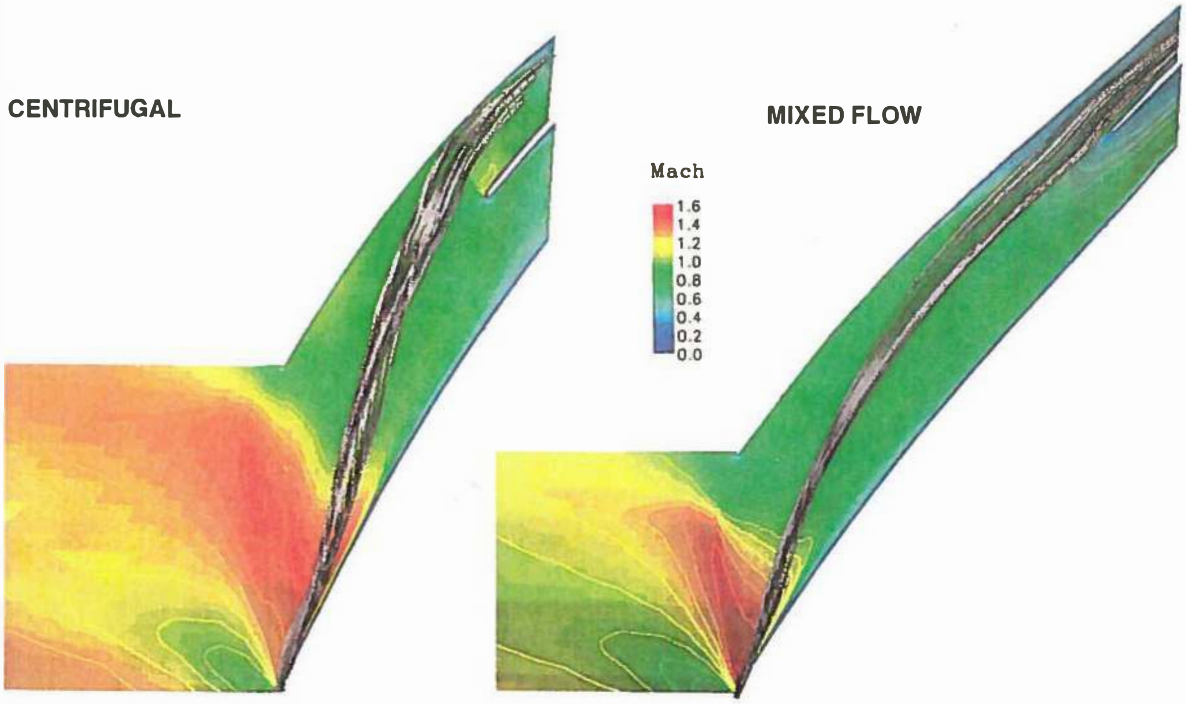


Fig.8: Mach number contours and streamlines at a cutting plane at 98 % blade height