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DESIGN PROCEDURES FOR COMPRESSOR BLADES

H. STARKEN

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16. Abstract

The conventional methods for the design of the blades in the case of axial turbomachines are considered, taking into account difficulties concerning the determination of optimal blade profiles. These difficulties have been partly overcome as a consequence of the introduction of new numerical methods during the last few years. It is pointed out that, in the case of the subsonic range, a new procedure is now available for the determination of the form of blade profile on the basis of a given velocity distribution on the profile surface. The search for a profile form with favorable characteristics is consequently transformed into a search for a favorable velocity or pressure distribution on the blade. The distribution of velocities depends to a large degree on the characteristics of the profile boundary layers. The considered concept is not new. However, its practical implementation has only recently become possible. The employment of the new design procedure is illustrated with the aid of an example involving a concrete design problem.

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Design Procedures for Compressor Blades

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Abstract

The capacity of axial compressors depends decisively upon the profile /49* of the blades. Until recently, this profile has been based upon empirical experiences summarized in correlations. Today, new calculation methods allow profile designs that take into consideration the actual flow-mechanical activities in the turbo device. This permits targeted optimization.

1. Conventional Design Methods

The blades of axial turbo devices, especially of compressors, are composed of individual silhouetting sections during construction. Even though this procedure has some similarity with the profiling of air plane wings--much experience could be taken from this area--fundamental differences do exist. The differences arise primarily from the quite different boundary conditions to which a compressor blade, as compared to a wing, is subjected. For example, the relative inflow velocity toward a compressor turbine wheel varies strongly from the hub to the housing based upon different volume velocities. In the front transonic compressor stages of the reaction propulsion engines relative local flow mach numbers well above one appear in the external sections. Mach numbers under one appear on the hub. (See figure one). During operation a compressor blade of this type must function partially in the subsonic and partially in the supersonic and wide ranges in the transonic range. This is a velocity range that airplanes traverse as fast as possible unsteadily. Concurrently, the blade is supposed to deliver a constant supply of energy to the flow. On the sections closest to the hub this can only be achieved by means of large-scale flow diversion due to low relative velocities. In addition, a compressor should not only function at its rating point; it should also function under adverse conditions. For example, the compressor must stand up to deviations of counter pressure without the flow collapsing altogether. For the blade foils this means that

* Numbers in the margin indicate pagination in the foreign text.

they must function without fail in a sufficient angle of attack range.

All of these flow-mechanical boundary conditions of the blades are increased due to rigidity problems based upon centrifugal force stress and oscillation excitations. It is therefore understandable that finding favorable blade profiles under these conditions is unspeakably difficult. In the past one relied primarily upon a few basic foil designs that were adapted to the respective boundary conditions by means of differentiated camber and foil thicknesses. With the help of empirical experience and correlation deduced thereof, resistance and/or total pressure loss coefficients as well as the diversion characteristics of the foils are estimated. It is apparent that this procedure does not permit quick and goal-oriented optimization due to the large number of parameters. That also means that systematic improvement of axial compressors by means of the profile is very difficult or perhaps even impossible until now. This situation has improved considerably during the past few years due to the development of new numeric procedures for calculating flow. /50

2. New Design Method

Today there is a new method at our disposal for the subsonic range. This method was developed at the University of Stuttgart and allows determination of the foil form according to a predetermined velocity distribution on the foil surface. Thus, the search for a favorable profile form becomes a search for a favorable velocity of pressure distribution on the blades. The latter depends decisively upon foil boundary layers, so boundary layer behavior is given first consideration at the outset of a foil design of this type. This procedure is not actually new; it has been known for some time. As far as possible, it has been verified in individual cases. A continuing practical realization has only become possible today. The realization was undertaken and worked at several years ago by the Institute for Propulsion Technology in cooperation with the MUT and the Institute for Aerodynamics and Gas Dynamics at the University of Stuttgart. Here is the result of this new foil development.

The basic problem of an axial compressor grid lies in delaying one flow from a relatively high velocity to a lower one. This is accomplished in the blade grid by means of a suitable cross-section guide, i.e. by flow diversion from a lesser subsonic inflow to a greater overflow cross-section. (figure 2) The diversion means an impulse change for the flow, which is carried by the blades in the form of lift. Depending upon the distance of the blades from one another--the so-called separation t--the lift varies in strength. This permits changing the stress of the individual blades. Generally, attempts are made to keep the number of blades as low as possible due to weight and cost considerations. This leads to high separation and high stress. On the foil surface high stress is expressed in high flow overvelocity on the suction side as opposed to the pressure side. If the speed of sound is surpassed locally, one speaks of overcritical flow. A foil that delays this flow to subsonic velocity without rebounding is called a supercritical foil. Foils of this type are interesting for several reasons. They promise high stress and low losses due to lack of thrust. The respective geometric form of these foils can only be found with the assistance of the new design procedures, i.e. by calculations.

3. Design Example

As an application example the hub section of a transonic compressor guide blade was chosen. The foil of this blade must attain a flow diversion of $\beta_1 - \beta_2 = 36{,}7^{\circ}$ based on a local current mach number of $M_1 = 0{,}77$ and a separation ratio of $t/l = 0{,}82$.

Based on experience with other foil designs and with the assistance /51 of boundary layer calculations, velocity distribution, respectively the mach number distribution on the blade surface was varied until a theoretically separation-free, super critical circumventing flow was achieved. This distribution is plotted as the extended curve in figure 3. The upper curve depicts the critical suction side development with the local supersonic range ($M > 1{,}0$) and a maximum mach number of $M = 1{,}25$. The flow acceleration up to 30% of the foil depth and the subsequent delay was supposed to fixate the point of change of the boundary layer from laminar to turbulent at this point. The measurements in the grid wind tunnel confirm this design procedure. This is demonstrated in the high agreement of design mach number distribution and measurement points in figure 3. Slight deviations appeared on the foil leading edge as well as in the middle of the suction side. The slight deviations are attributed to the fact that the design procedure results in a sharp foil which must be smoothed off for applications. The latter is due to a laminar separation bubble, which in turn is attributed to the low degree of turbulence in the grid wind tunnel. Because of this the point of change shifted upstream with regard to the rating calculation, and the laminar boundary layer was incapable of overcoming the rise in pressure. Figures 4 and 5 show that the measured total pressure loss coefficient (1) of this foil amounts nonetheless to a very low value of 2% due to turbulent reformation of the boundary layer. These figures plot the measured loss coefficient (1) dependent upon the inflow mach number M_1, respectively from the local flow angle β_1

The value Ω appears as an additional parameter. It is a measurement for the radial flow tube contraction in the turbo. It is also designated an axial flow density ratio. The value $\Omega = 1,08$ equals the foil rating. The diagrams show that the foil constructed according to the new method has a sufficient working range with low losses with regard to the inflow mach number as well as the inflow angle.

4. Outlook

The above example shows that it is possible today to design blade profiles within limited velocity ranges in close cooperation with relative experiments specifically with regard to the existing boundary conditions by applying numerical calculation methods.

Similar results may be expected in the future for transonic and supersonic velocities. Today it is already possible to systematically improve a large part of axial turbos from the basis with the method described above. The possibility of increasing blade stress without decreasing the degree of effectivity must be especially emphasized. This helps to reduce manufacturing costs and machine weight.

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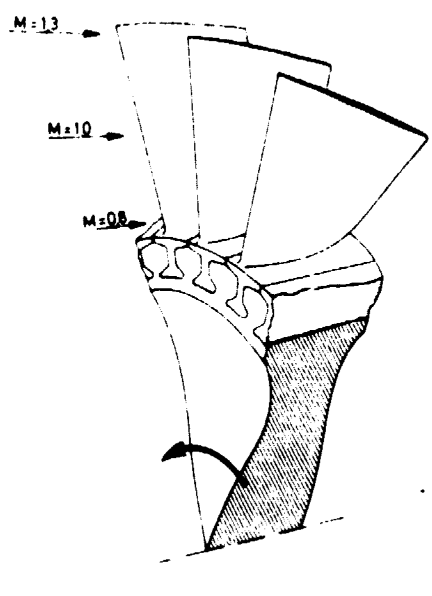

Figure 1: Segment of a transonic axial compressor turbine wheel indicating the relative inflow mach numbers.

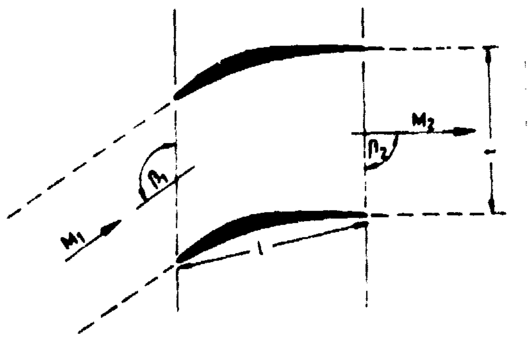

Figure 2: Development of the flow cross-section in the axial compressor grid.

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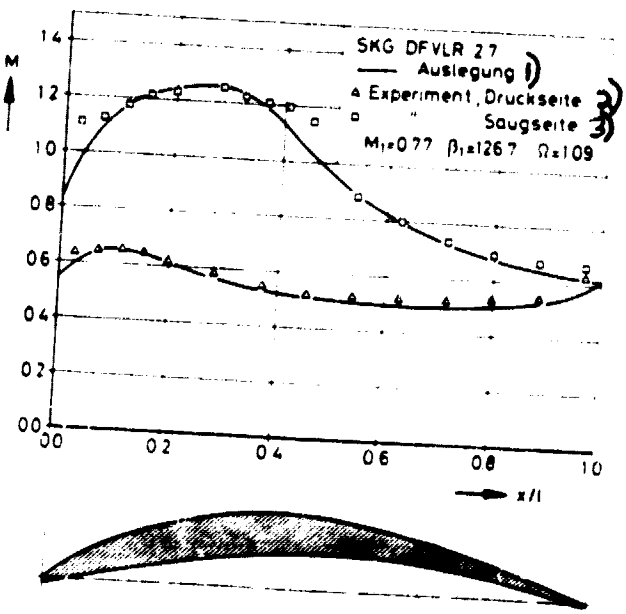

Figure 3: Existing and measured mach number distribution on the surface of the super critical compressor guide blade SKG DFVLR 2.7.
Key: 1) rating 2) experiment, pressure side 3) experiment suction side

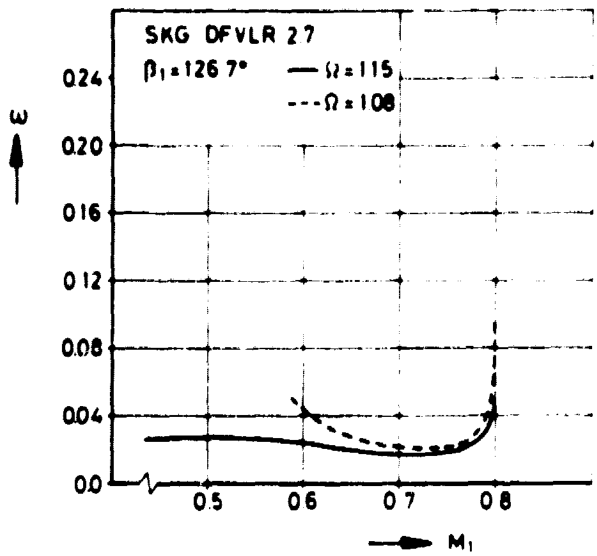

Figure 4: Measured total pressure loss coefficient w dependent upon the inflow mach number M_1 and the axial flow density ratio Ω for the foil SKG DFVLR 2.7.

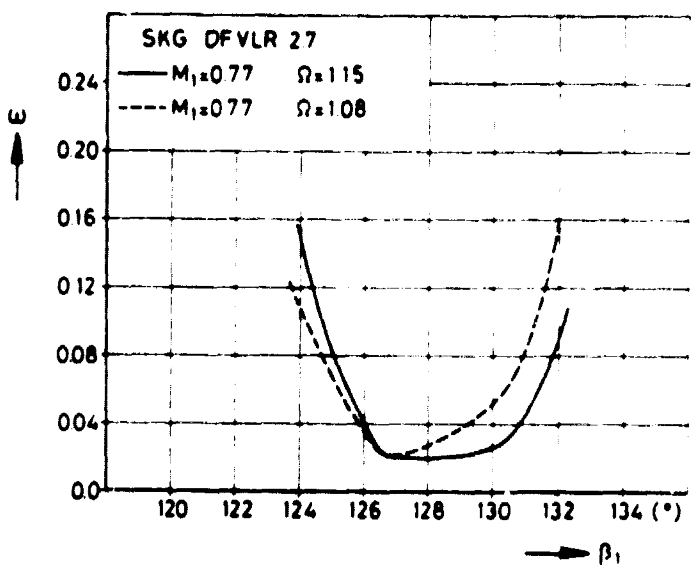

Figure 5: Measured total pressure loss coefficient 2 dependent upon the local flow angle β_1 and the axial flow density ratio for the foil SKG DFVLR 2.7.