

Recent developments of axial flow compressors under transonic flow conditions

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Abstract. The objective of this paper is to give a holistic view of the most advanced technology and procedures that are practiced in the field of turbomachinery design. Compressor flow solver is the turbulence model used in the CFD to solve viscous problems. The popular techniques like Jameson's rotated difference scheme was used to solve potential flow equation in transonic condition for two dimensional aero foils and later three dimensional wings. The gradient base method is also a popular method especially for compressor blade shape optimization. Various other types of optimization techniques available are Evolutionary algorithms (EAs) and Response surface methodology (RSM). It is observed that in order to improve compressor flow solver and to get agreeable results careful attention need to be paid towards viscous relations, grid resolution, turbulent modeling and artificial viscosity, in CFD. The advanced techniques like Jameson's rotated difference had most substantial impact on wing design and aero foil. For compressor blade shape optimization, Evolutionary algorithm is quite simple than gradient based technique because it can solve the parameters simultaneously by searching from multiple points in the given design space. Response surface methodology (RSM) is a method basically used to design empirical models of the response that were observed and to study systematically the experimental data. This methodology analyses the correct relationship between expected responses (output) and design variables (input). RSM solves the function systematically in a series of mathematical and statistical processes. For turbomachinery blade optimization recently RSM has been implemented successfully. The well-designed high performance axial flow compressors finds its application in any air-breathing jet engines.

1. Introduction

Axial flow compressors are mostly used in air breathing propulsion especially in aerodynamic applications. The design of axial flow compressor is not an easy task to the researcher because of its complexity in profile and its flow characteristics. Compressor size can vary from few meters to tens of



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meters in diameter, based on their applications. Till today recently the design of compressor two-dimensional compressor under real viscous conditions or three-dimensional meridional flow or inviscid three-dimensional flow analysis was done. With the use of modern sophisticated computational techniques, it is possible to solve three-dimensional viscous flow analysis and also potential to forecast better results of three dimensional multistage compressors. Therefore, it is required to retool the design procedures to have efficient results and physical dependability of these modern techniques.

In [1-2] discussed that the basic operations of an axial flow compressor were known and shown in the year 1853 to the FAS (French Academia des Sciences). Since then the development of axial flow compressor has been studied comprehensively and also slowly staging of compressor progressed meaningfully. To achieve high performance of the compressor the aerodynamics of each blade and stage(s) need to be understood fully. In this paper¹ main focus is given to understanding the aerodynamic flow characteristics and recent design techniques followed to improve the performance of the multistage compressor by referring to various research articles.

2. Understanding the Axial Flow Compressor Operation

The compressor is the first component in the core jet engine among various parts shown in the figure 1. The main purpose of the compressor is to increase the pressure ratio per unit stage working fluid using rotating shaft work. The axial flow compressor consists two set of blades one rotor and stator. The combination of rotor and stator together is called stage [3]. The working fluid which is entering from the earth atmosphere region accelerated through the rotor blade where the kinetic energy increases, then high kinetic energy of the fluid is further decelerated through the stator blades to form the high pressure. Stator blade is important to have proper pressure flow, by changing the energy associated with the swirl into pressure.

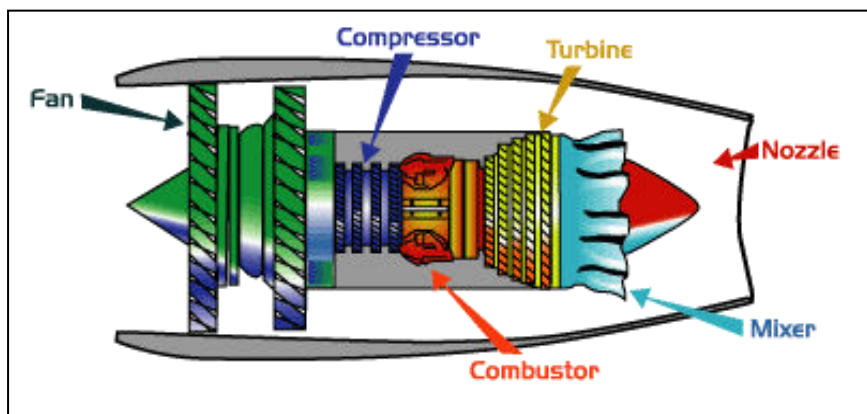


Figure 1: Various components of a commonly used turbo-engine [63].

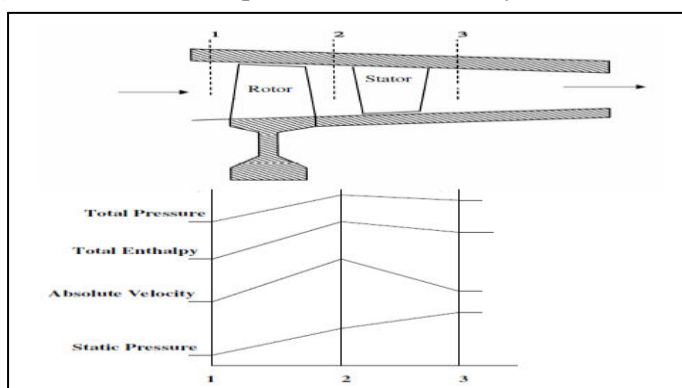


Figure 2: Various components of a commonly used turbo-engine [63]

Figure 2 represents a single stage axial flow compressor having a set of one rotor blade and one stator blade. This also represents the direction of working fluid which passes from left to right and for the rotor blades direction of rotation from bottom to top. Generally, every aircraft engine axial compressor has inlet guide vanes. These vanes will ensure that working fluid flow streamlines should be straight to the rotor blades to achieve high acceleration of mass flow rate in the blade rows when high accelerated working fluid leaves from the rotor blade exit, the flow is decelerated in the stator blade row and further enters into the next stage of the axial compressor.

Depending on the application of the aircraft engine specifications the number of stages in a compression system varies. In any axial flow compressor increasing the number of stages increases the weight, cost of the system, decreases the overall efficiency but increases the total pressure ratio for the system. In the working axial compressor, the fluid enters into gradually smaller volumes resulting in an increase in stagnation enthalpy of air and an increase in the stagnation pressure.

3. Axial flow compressor design

The compressor undergoes wide variety of operating conditions during the flight. During the take-off operation of the aircraft jet engine it requires more pressure ratio through the compressor to generate enough thrust. To control the thrust the compressor operates at different mass flow rates to vary the compressor and turbine speeds [4]. Here the staging plays a vital role in maintaining the desired efficiency. Through literature review it is evident that early designs of compressor at transonic and supersonic conditions were failure⁴. In [5] discussed more about subsonic compressor designs. It is found that low reliability and poor efficiency of the compressor blade steered to bad designs. In the early designs the researchers believed that poor efficiency was because of shock patterns alone, but later after successful failure of designs it was renowned that losses traits to flow blockages those are because of shocks. Since then drastic improvement have been made in compressor blading profile design, hub to tip design and also tip to hub design. In [6] discussed briefly about the early designs of compressor blade.

Through the existing designs it is evident that by increasing the number of stages in the compressor results in higher overall pressure rise, but at the same time the overall length of the compression system and weight also increases. Comparing with last stage the first stage of the compressor volume is larger, therefore the typical modern compressors are wide at the inlet. They generally have a conical shape and have somewhere amid 9 to 15 stages. Figure 3 illustrates the straightforward compressor design difficulty. The main components that need to be designed are the geometry of the blade rows and geometry of the blade end-wall contour. To accomplish design study on a mean stream surface (shown in figure 3) in an axisymmetric three dimensional multi stage compressor designs approach generally used. To achieve both the end wall geometry and blade geometry the mean stream surface was used as a baseline. This type of methods and findings may be found in reference [5].

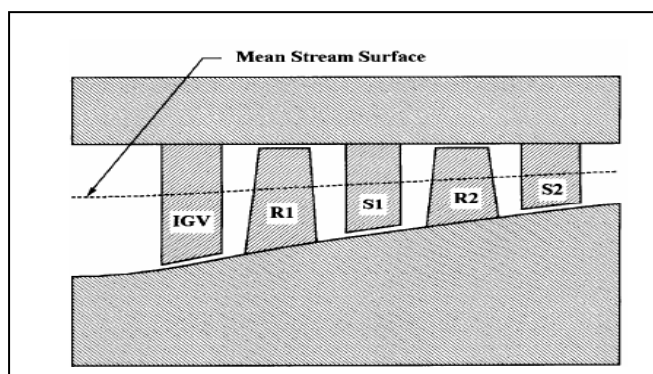


Figure 3: Compressor geometry design with endwalls and blade rows [1].

Considering from an aerodynamic perception, a more accurate three-dimensional design of compressor blade is very much essential as this safeguards the maximum possible blade loading. Structural and aeroacoustics responses are closely connected to aerodynamic design of a compressor blade. Through the literature it is evident that, the compressor blade load increases eventually structural deformation on the blade also increases ultimately leading to structural failure. An ineffective design of blades can naturally lead to greater than before acoustic response of the blade, from shock formations and rotor-stator interactions. Finally, it clears that the design of compressor system is a multidimensional problem.

4. CFD studies on axial flow compressor performance

Compressor flow solver is the turbulence model used in the CFD to solve viscous problems. In order to improve compressor flow solver and to get agreeable results careful attention need to be paid viscous relations, grid resolution, turbulent modeling and artificial viscosity. Figure 4 shows the harmonic balance simulation of complete axial flow compressor. Initially turbomachinery compressor performance calculations using CFD were frequently done in two dimensions. Such two-dimensional cascade analysis for both turbines and compressors were calculated by many authors [7-10]. By nature, the flow in an axial flow compressor is unsteady and vertical [11-14] and therefore huge turbulence and perfect modeling of the viscous effects is imperative.

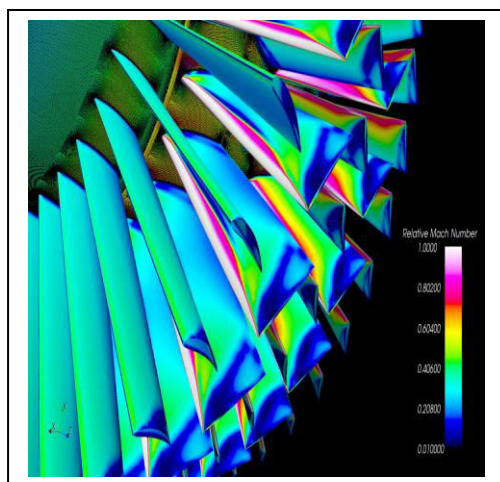


Figure 4: Harmonic balance simulation of multistage axial flow compressor relative mach number [64].

Ample of work done in progressing phenomenological and sophisticated simulation techniques to understand in addition simulate the stream flow behavior in compressor over the past two decades. The area of computational fluid dynamics during this time has undergone significant changes, giving engineers and designers having useful competence to model and study the intrinsically three-dimensional flow in axial flow compressor. Most recently having increased computational power and developments in visualization tools and post processing three dimensional simulations of axial flow compressor rotor machines are feasible. Also researchers now have access to a plethora of data for justification of compressor codes with the accessibility of dependable experimental test data by [15]. Since then the analysis of single and multistage axial flow compressor blade passage flow has been developed using a number of three dimensional CFD codes [16-23]. Computational codes for turbomachinery multistage three dimensional compressor configuration have been developed by different authors [24-26]. Very recently initially developed compressible unsteady flow solver for an

isolated three-dimensional blade row, which was altered effectively to add in multistage competencies [27-28].

5. Background of axial flow compressor design methods

The axial flow compressor blade design is broadly classified into two approaches, direct and inverse methods. In the direct design method, the edge geometry is analyzed directly by experiments and / or by a computational analysis. In this method either the existing coordinates (x, y, z) of the edge are changed or performance constraints that directly affect the blade profile are changed. The profile of the blade is consequently analyzed and the effect of performance variables on its overall efficiency is evaluated.

Introducing in the year 1970s the popular techniques like Jameson's rotated difference scheme [29-30] and used method were possible to solve potential flow equation in transonic condition for two dimensional aerofoils and later three dimensional wings. These advance techniques had most substantial impact on wing design and aerofoil. The same techniques also had significant impact on axial flow compressor design as the new schemes may possibly be coupled with early established potential flow solvers.

In papers [31] and [32] investigated transonic direct method into inverse design methods. CAS22 a programme developed by Dulikravich which has capable of shock free aerofoil cascades designing and valid to transonic shock free redesign of existing cascade two dimensional aerofoil and also to aerodynamic analysis. Meauze was developed some of other applications of inverse design approach, thereafter Dulikravich passing few years implemented same technique for three dimensional blades.

In the year 1990's drastically improvements in computational fluid dynamics, the stage three-dimensional design and the flow analysis of compressor geometry configuration has been made possible. Significant work has been done on a four stage compressor through direct design analysis and showed that reiterating stage phenomenon where the blockage raises across the flow [33]. Complete three-dimensional inverse design approach for compressor blades in transonic flow was developed in 1993 [34]. After passing few years Dang et al redesigned NASA 67 configuration using three-dimensional inverse design method. In another investigation done on multi objective optimisation of compressor blades to find best blade profile with respect to working range and loss by [35].

In the recent years' researchers introducing the concept of 'sweep lean'. Where the term sweep describes about the moving blade in axial flow direction, whereas the lean explains the moving blade in circumferential direction. This concept is applying by many researchers [36-38] have applied the sweep and / or lean idea to multistage and single stage compressor configuration and found positive results.

When the design constraints are selected in an ad-hoc manner direct design method can sometimes be trial and error. Provided it has own benefits compared with the inverse design method, which normally required more number of inputs out of which few were always unknown (like pressure distribution over 3D blade) so as to produce the preferred flow features. Hence, the inverse direct design method always advisable and usually preferred for finding the unknown performance parameters to acquire detailed knowledge of the flow.

In the recent years' researchers have described the importance of compressor blade aerofoil curvature scattering on its performance and showed the potential methods that can be used to design extremely differentiable blade planes [39-41]. A set of Chained G Bezier polynomials capable of generating compressor aerofoils developed by corral [42] and pastor [48].

6. Aircraft system level engine design overview

This study describes about an axial flow compressor blade design analysis component level. However, the design studies on turbomachinery configuration can also be done at an overall engine system level, where the focus is not on individual component⁴². At the Aerospace System Design Laboratory (ASDL) at Georgia tech has been extensively performed such system level design studies. In paper [43] proposed a probabilistic design methodology for commercial aircraft engine cycle selection. In paper [44] investigated an assessment of lost thrust method for analysis of gas turbine engine thermodynamic performance. In paper [45] also done another study on probabilistic methods to systematically make the decisions under uncertainty and rationally to solve the preliminary design calculations [62]. Another probabilistic sensitivity in the gas turbine engine system level design can be found in references [46-48].

7. Axial flow compressor optimization techniques overview

For a variety of transonic flow design problems numerical optimization techniques have been used successfully. The design of compressor blade under aerodynamic perspective itself is a challenging task to a researcher. Initially, efficiency of the transonic compressor blade is most sensitive to its shape and hence the shape of the blade must be parameterized with required number of performance parameters are to be optimized. Provided in addition aerodynamic optimization design objective function problem is repeatedly multidimensional and non-linear reason is flow field directed by a system of non-linear partial differential equations. The optimization also has constraints such as an acoustic characteristic, operating mass flow range by varying the rotor speed and others make the aerodynamics blade profile optimization an effective problem. Many more optimization techniques have been reported with degree of success such as artificial intelligence methods (neural network and evolutionary algorithms EA's), orthogonal array method, response surface method (RSM), gradient based method.

The gradient base method is most popular method especially for compressor blade shape optimization. This gradient based algorithm in which the optimum is investigated by estimating the local gradient information. Constrained minimization (CONMIN) uses gradient algorithm and has been efficiently used for wing aerofoil design by vanderplaats [48]. Gradient based method design widely used in subsonic and supersonic wing design [49-50]. Later further studied [51-52] detailed manner and implemented same gradient based method on centrifugal compressor and design of compressor aerofoil.

Another type of optimization technique is Evolutionary algorithms (EAs). Evolutionary algorithms are quite simple than gradient based technique because it can able to solve the parameters simultaneously search from multiple points in the given design space. Recently EAs have been successfully used for solving the turbomachinery blade optimization problems found in references [53-55].

Response surface methodology (RSM) is another type of optimization algorithm. This method basically to design empirical models of the response that were observed and to study systematically the experimental data. This methodology analyses the function denotes the correct relationship between expected responses (output) and design variables (input). RSM solve the function systematically in a series of mathematical and statistical processes. For turbomachinery blade optimization recently RSM has been implemented successfully. Stacking line of the axial flow compressor rotor blades optimization was successfully carried out by Ahn [56]. To perform the shape

optimization in a single stage transonic axial flow compressor stator blade successfully used RSM by Jang in 2006 [57].

8. Literature of summary of recent developments on axial flow compressor under transonic flow conditions

There is very little literature available in the unsteady and instability area. Very little is known about its behavior and its possible configurations as function of field variables. Experimental and Computational Analysis conducted in this area are less and a lot more is to be done to conceptually understand the flow behavior and its implications on rotor. This regime itself is non-uniformity associated, and is different at different regions and time. The solution is basically guided by the type of non-uniformity. Much is required to be understood about the impacts of different possible discontinuities, on the flow structure through the rotor. Some region of cross section subjecting to discontinuities will have its implications on remaining blades flow passages and therefore full rotor analysis is very much inevitable. From the literature it is understood that there were no attempts made to generate these discontinuities at the entry face of the compressor rotor.

There are varieties of configurations on non-uniformities possible at the entry face to rotor, like the one associated with pressure field or velocity and temperature and so on. Point to note here is they are not independent but are coupled effects. One discontinuity leads to the deviation of others. Though there is some theoretical data at hand very little is on the experimentation side. Since it is stall associated which is actually rotating by the rotor of the compressor in the aircraft engine. There were few literatures on Numerical analyses done towards understanding the effects of the tip sensitivity driven Distortion flow development in the rotor, by using some circumferential grooving and also there are few papers discussing about the stall inceptions and associated distortion.

Attempts have been made to find out the characteristics of distortion associated in the compressor rotor with rotating stall in multistage compressor but not exactly to understand the distorted flow field through compressor. It was all about tip vortex and separation implications⁵⁸ and it's as such flow disturbance effects were experimented⁶⁰. Therefore, it is required that first to understand the flow structure under distorted entry flow field and its development through the rotor. In this process, simulations come very handy providing more comfort and relief, with ease of analyzing. Moreover, this area is very vast; there are lots of things to experiment and analysis, therefore in this research literature steady state simulation on axial flow fan, for which there is experimental work available have been carried out⁵⁹. So to understand its effects turbulent kinetic energy, its development and growth is also reported⁶¹ to some extent. There is still a lot more to understand with regard to vortex effects, unsteady disturbances and stall propagation, which guides passage flows. This served as the very Motivation to carryout distortion analysis of axial flow fan, and to find out the consistency of simulations and their ability to predict the unsteady flow field [58-59].

9. Conclusions

Review of literature revealed that implications of nonuniformities were studied to some extent, by specifying conventional one dimensional(axial), inaccurate profile at the entry, and no attempt was made towards possibility of velocity fields in the radial and circumferential directions, in distorted region at the entry face.

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Some region of cross section subjecting to discontinuities will have its implications on remaining blades flow passages and therefore full rotor analysis is very much inevitable. From the literature it is understood that they were no attempts made to generate these discontinuities at the entry face. This regime itself is non-uniformity associated, and is different at different regions and time. The solution is basically guided by the type of non-uniformity.

The computational work is to initiate with the intent to simulate and find out the effects and implications of the unsteady non uniform flow at the compressor inlet. Unsteady flow is not a fixed configuration problem. It is possible to understand it well upon investigation of flow through the rotor with different inlet conditions and varied problem specifications.

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