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Derivation of Axial Compressor Stage Characteristics from the Overall Performance Map.

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ABSTRACT

The purpose of this paper is to present an approximate procedure for the derivation of individual stage characteristics for a multistage axial flow compressor from the given steady-state overall performance map and the geometry of the annulus and stage/blade and blade section profile data. This was achieved through the use of a one-dimensional inviscid compressible flow computer model STGSTK, which was developed at NASA Lewis Research Centre for predicting the overall performance of the multistage axial flow compressors utilising the stage-stacking technique at mean radius. This program was modified to accommodate adjustments for losses and real flow effects, which were estimated through the use of an AMRL modified code CASCAD, which utilises NACA and most of the known cascade correlations. A set of stage characteristics for a seven stage high-pressure axial flow compressor was derived. When these calculated characteristics were subsequently input to the program, the derived overall characteristics of the compressor were found to agree reasonably well with the original characteristics at the same corrected speeds.

INTRODUCTION

Distortions in the air intake flow at the engine face of a jet aircraft, in the form of ramps and transients in inlet total pressure or temperature, can produce adverse effects on the performance and stability of an aircraft's propulsion system. A current research program at AMRL is concerned with distortion-induced aerodynamic instabilities in engines, which are normally initiated in the compression system. A US Air Force generic code DYNTECC has been adapted to determine the effect of these on the stability and performance of the Pratt and Whitney TF30 engine in the F-111 aircraft. This code was developed by Sverdrup at the Arnold Engineering Development Centre, USA for a single spool engine, and was upgraded to accommodate a generic dual spool engine. It was further modified at AMRL for triple spool applications. As this is an axial compression system stage-by-stage mathematical model, it is essential that the stage characteristics for all stages are determined as they form a major part of the input to run the model.

The individual stage characteristics needed for the DYNTECC operation were available for the fan and low pressure compressor but not for the high pressure compressor of the dual spool TF30 compression system.

The flow in multistage axial flow compressors is highly three dimensional, and viscous effects play a major role in determining the compressor performance. An exact solution for such a complex flow field is not available, and calculation procedures must therefore be approximate. Several methods have been used to predict the stage and overall compressor performance, with some claimed to be more accurate than others. The streamline curvature method, which can be applied to estimate the flow radial stations between the axial blade rows is the most favoured by researchers and designers , such as: Jansen and Moffat, 1970, Oldham, 1968, and Glenny, 1974.

Within the time constraints, the only available option to obtain the characteristics for the TF30 high pressure compressor was through the use of the approximate method of the simple one-dimensional stage stacking technique. This was achieved with an adapted one-dimensional inviscid compressible flow computer model STGSTK, which was developed at NASA Lewis Research Centre by Steinke, 1982, for predicting the overall performance of the multistage axial flow compressors utilising the stage-stacking technique at mean radius. In the present study this program was modified to accommodate stage characteristic adjustments for losses and real-flow effects. These were calculated through the use of another modified program CASCAD.

With real gas effects taken into account, this method was also used by Raw and Weir, 1980, for the prediction of the off-design stage and overall characteristics of a multistage axial flow compressor. In this method,

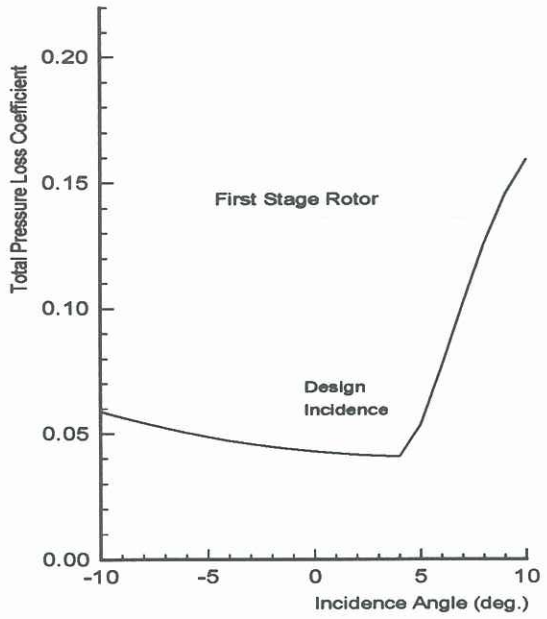

FIGURE 1: CALCULATED TOTAL PRESSURE LOSS COEFFICIENT VERSUS BLADE INCIDENCE ANGLE OF THE FIRST STAGE ROTOR.

conditions at mean diameter are taken as representative for the stage and full blade span. For inviscid flow, this method was presented in Johnsen and Bullock, 1965, and was fully described and analysed by Steink, 1982.

STAGE STACKING MODEL

In this model, the annulus and stage/blade geometry for all stages, the rotor rotational speed and the overall conditions at the inlet of the first stage are specified as input. In relation to stage characteristic data, the program has two options: to input the full characteristic for individual stages; or to input the stage performance at design or a reference point. In the case of the first option, the code will then " stack-up" the compressor stages to give the overall performance of the compressor. In the absence of full characteristics for individual stages, the program has the capability to calculate them based on the overall stage performance data at design or a reference point for each stage. This is achieved through a procedure of a parabolic fit between the reference flow coefficient and assumed minimum and maximum flow coefficients corresponding to stall and choke conditions respectively.

As the stage characteristics data for both options are not available for the TF30 compressor, the latter option was chosen as it would require fewer program iterations to match the overall compressor performance. In either case, the stage and overall performance of the compressor can be calculated in the program by utilising the velocity triangles located at rotor inlet and outlet along a single stream line which divides the flow into two equal halves.

These velocity triangles are used in the program to represent the overall performance of the stage and not the blade element performance at midspan. The code incorporates the options to modify the stage characteristics and rotor design deviation angle for off-design flow and speed, and inlet guide vane setting angle.

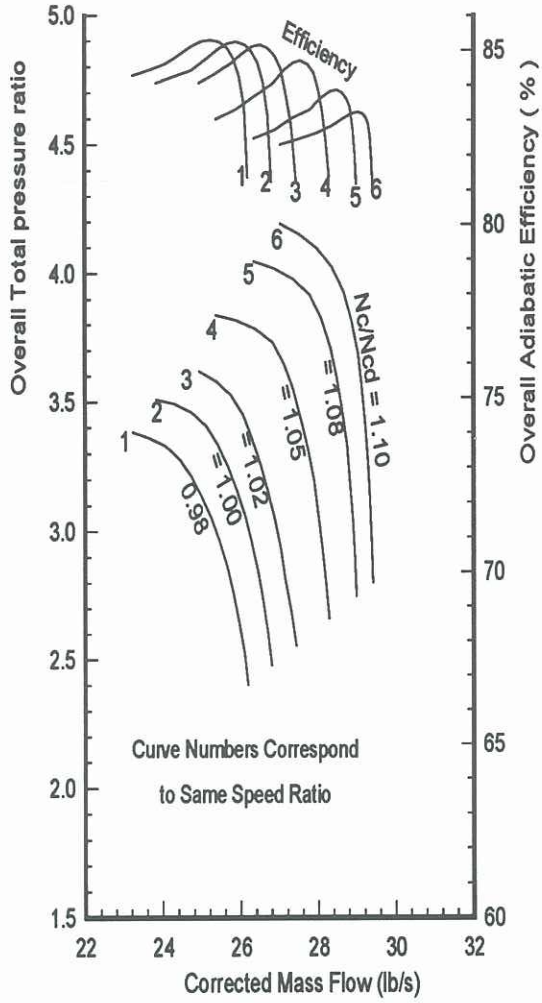

FIGURE 2: THE COMPRESSOR OVERALL PERFORMANCE MAP OBTAINED FROM THE MANUFACTURER'S ENGINE MODEL.

LOSS MODEL

Detailed gas-path and cascade geometry and blade/vane section profile data, together with NACA and other published cascade correlations were coded into an adapted computer program CASCAD. This was developed by Davis, 1970, and was presented as a subroutine in a major computer program used for the analysis and design of turbomachinery. CASCAD was used to determine realistic flow parameters such as deviation angles and total pressure loss coefficients throughout the compressor. Most of these correlations were compiled by Davis, 1970, and some can be found in Jensen and Moffat, 1976, Horlock 1958, Johnsen and Bullock, 1965, Miller, 1969, and Gostelow 1984. The total pressure loss coefficient and the blade exit angle or deviation angle depend upon the cascade geometry and, for a fixed geometry, the inlet flow conditions. Design or minimum loss and off-design conditions can be predicted by this program which incorporates most of the well-known cascade loss and deviation correlations for conventional blade profiles deduced from cascade test results and observations of turbomachinery performance.

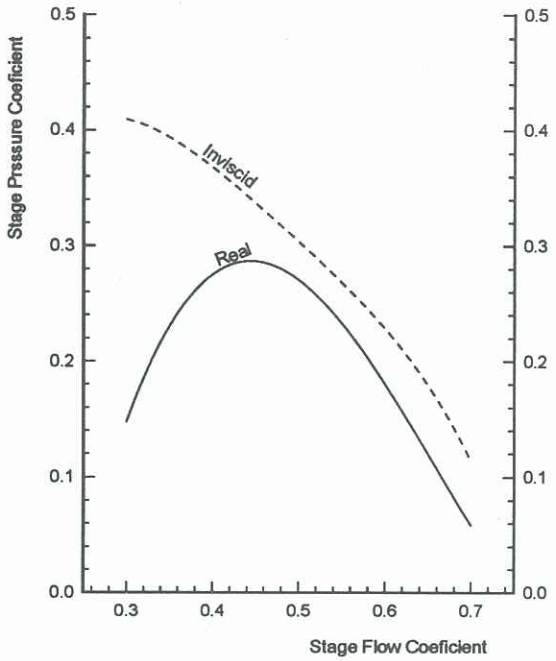

FIGURE 3: THE DERIVED STAGE CHARACTERISTIC FOR THE FIRST STAGE FOR BOTH INVISCID AND REAL FLOW SOLUTIONS.

A number of options are provided for the user regarding the method used to make the calculations, and the particulars of various options are included and evaluated for conventional blade profiles. The program provides analytical correlations and corrections for effects of parameters such as; compressibility, annulus wall boundary layer effects, streamline slope, axial velocity variation, blade thickness and solidity. It also includes provisions for approximating the shock contribution to total element loss and critical Mach number. As an example, some of the results in terms of predicted total pressure loss coefficient as a function of blade incidence angle for the first stage rotor of the compressor are shown in figure 1. This non-dimensional parameter is defined as the ratio of the difference in total pressure between the inlet and exit of a blade row to the velocity head at the inlet of the blade row at mid radius.

RESULTS AND DISCUSSION

Details of stage geometry, blade section profile data and the realistic flow parameters and losses derived above using CASCAD were incorporated in the STGSTK code to predict the individual stage characteristics for a given input of overall design parameters. The compressor design overall performance in terms of total pressure and efficiency as functions of corrected rotor speed, were estimated from the steady-state performance map of the of the high-pressure compressor, which are shown in figure 2. This map was in turn extracted from the manufacturer's thermodynamic transient model of the TF30 engine. A typical characteristic in terms of pressure - mass flow coefficients calculated for the first stage for both inviscid and real flow solutions is presented in figure 3. The pressure coefficient is defined as the ratio of

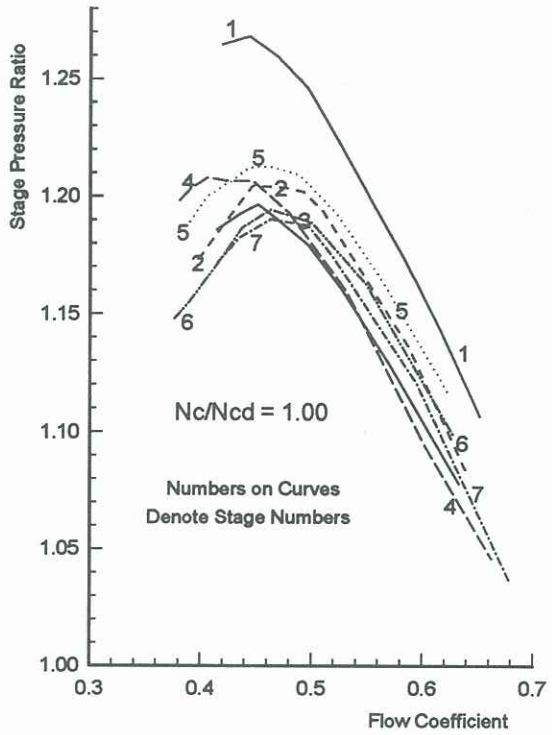

FIGURE 4: THE DERIVED CHARACTERISTICS FOR ALL STAGES AT THE FIXED DESIGN CORRECTED SPEED.

the product of the adiabatic efficiency, ratio of specific heats at constant pressure and the temperature rise across the blade row to the square of the tangential velocity of the rotor mean radius at the exit of the blade row. The stage flow coefficient on the other hand is presented in the conventional form of axial flow velocity normalised with rotor tangential wheel speed at blade row inlet at mean radius.

A set of stage characteristics for the seven stage high - pressure axial flow compressor was derived for design and various selected off-design corrected rotational speeds of the rotor. These are presented in figure 4, in terms of stage pressure ratio versus stage flow coefficient, for all stages at the fixed design corrected rotor speed, and in figure 5 for the first stage at various off-design corrected rotor speeds. The main criterion to be satisfied is that when these calculated characteristics were subsequently input to the program, the derived overall characteristics of the compressor should agree with the original manufacturer's characteristics shown in figure 2 at the same corrected rotor speeds. With adjustment for real-flow effects the calculated results are compared with the manufacturer's results in figure 6. This figure shows that the agreement between the stacked and the manufacturer's overall characteristics in the vicinity of design corrected speed is better than that in the higher range of corrected rotor speeds. However, it is evident that reasonable accuracy can be achieved with this simple stage-stacking technique.

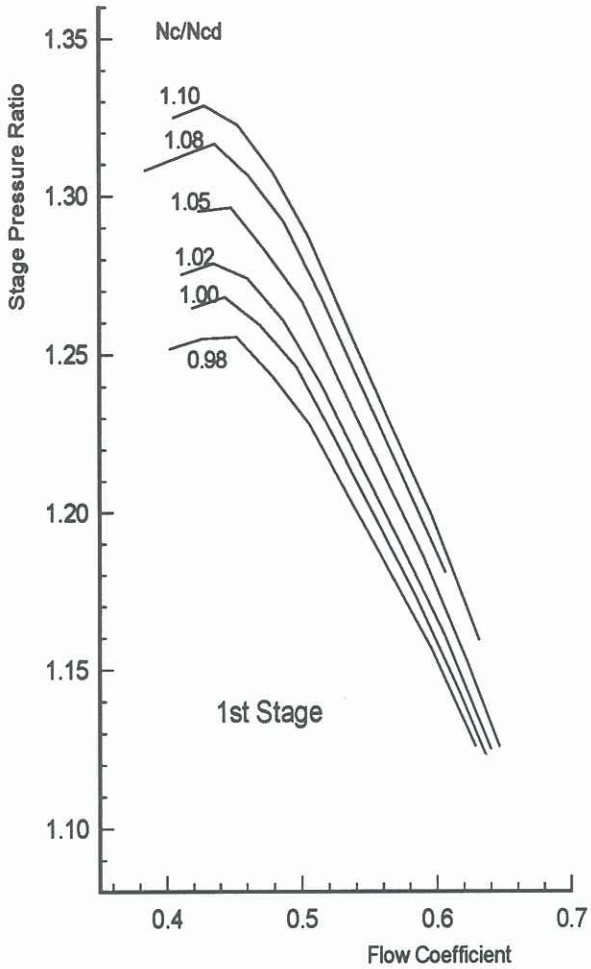

FIGURE 5: THE DERIVED CHARACTERISTICS FOR THE FIRST STAGE AT DESIGN AND VARIOUS OFF-DESIGN CORRECTED ROTOR SPEEDS.

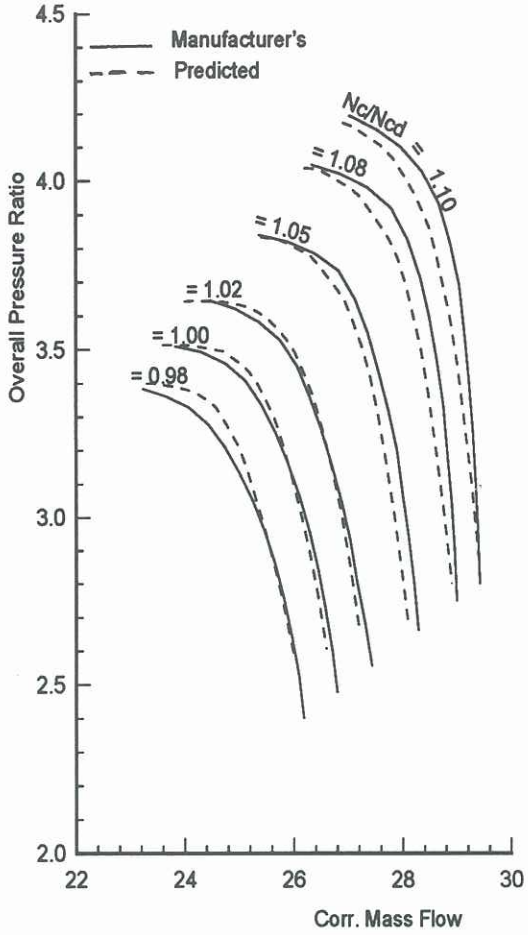

FIGURE 6: COMPARISONS OF THE ORIGINAL OVERALL PERFORMANCE MAP OF THE COMPRESSOR WITH THAT OBTAINED USING THE STAGE - STACKING CODE.

CONCLUSIONS

The main conclusions are:

1. A simple stage-stacking method at mean radius, modified for losses and real flow effects, can be used in a reverse engineering procedure to produce the individual stage characteristics for a multistage axial flow compressor.

2. The effectiveness of the method was demonstrated by the fact that when these derived characteristics were subsequently input to the program, the calculated overall characteristics of the compressor agreed closely with those of the manufacturer at the same corrected speeds.

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