Compressors Benefit from the NASA Rotor 37

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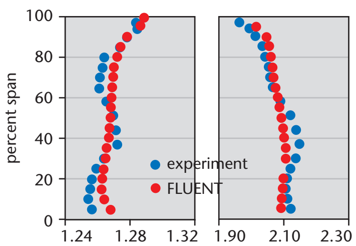
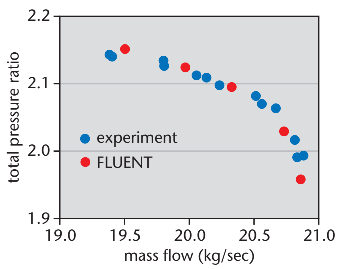


The total temperature ratio (left) and total pressure ratio (right) along the wing span, ranging from the blade root (0%) to the blade tip (100%), approximately 10cm from the leading edge at 98.7% of the maximum (experimentally measured) flow rate (20.93kg/s)

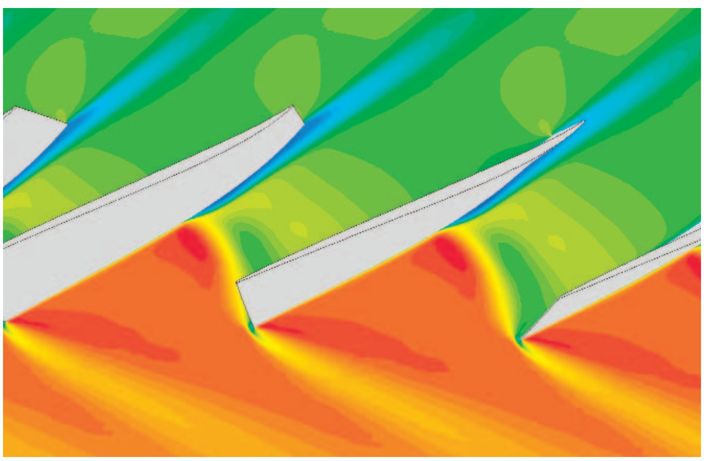


The predicted compression ratio is within 1% of the experimental values throughout the range of mass flow rates studied

THE NASA ROTOR 37 is among the most popular rotors that have been manufactured to date. It was designed and tested in the early 1980s by Reid and Moore at NASA Glenn Research Center [1]. It consists of 36 blades with multiple arc profiles, and was designed for compressors and turbines with a compression ratio of 2.05 at a mass flow rate of 20.19 kg/s.

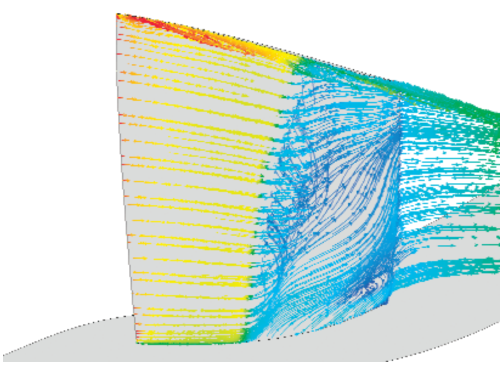
Steady-state numerical simulations of the NASA Rotor 37 have been performed using FLUENT for one operating curve. Using a constant rotational speed (17,188 rpm), the simulations covered a range that extends from the maximum flow rate to the pumping regime. A fully structured mesh of 420,000 cells was built using GTurbo, Fluent's turbomachinery preprocessor. During the preliminary runs, the shock wave accuracy was found to strongly depend on the mesh resolution around the blade. Consequently, most of the cells were concentrated near and between the blades. A comparison of turbulence models was also conducted early on, and the most suitable one for this case was found to be the realizable k-ε model. Since the mean y^+ value was around 33, non-equilibrium wall functions were used. An axial flow direction was set at the inlet, and rotationally periodic lateral boundaries were used. An inlet temperature of 288.16K was assumed, and the exit pressure was altered for each of seven points on the operating curve that were simulated.

Using data from published experimental results [2] a comparison of the radial variation of the pitch-averaged total pressure ratio was made, 10.19 cm from the blade leading edge. The results indicate that the total pressure is lower at the blade tip (100% span) resulting in losses due to the tip gap. A comparison of the total temperature ratio along the blade span, measured at the same location, shows an increase in the total temperature at the blade tip due to the friction between the fluid and the shroud, and to the detaching of the boundary layer. In both cases, the FLUENT results were in very good agreement with the data.



Contours of Mach number at 70% of the blade height at 98.7% of the maximum experimental flow rate

Predictions of the efficiency of the compression process along the operating curve were found to be good at low flow rates, but to deviate from the data at large flow rates. The total temperature ratio along the operating curve was found to be in good agreement throughout the range, even though the computed values were higher. A comparison of the compression ratio as a function of the flow rate was found to be within 1% throughout the range studied.



Pathlines colored by relative Mach number at 98.7% of the maximum experimental flow rate

Contours of Mach number on a plane through the blades illustrate some of the characteristics of flow in transonic compressors. The detached bow shock wave at the leading edge of each blade generates a normal shock wave in the flow channel that is the main compression mechanism for this type of compressor. There is a low momentum fluid region behind the shock wave, which is the second main source of total pressure loss and compression efficiency. ■

References

1 Reid, L. and Moore, D.: Performance of Single-Stage, Axial-Flow Transonic Compressor With Rotor and Stator Aspect Ratios of 1.19 and 1.26 Respectively, and with Design Pressure Ratio of 2.05. Tech Rep. TP-1338, NASA.

2 NASA/TM-2003-212457 Report, http://gltrs.grc.nasa.gov