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Numerical Analysis of the Unsteady Strong Shock Interactions in a Transonic Turbine

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ABSTRACT

The requirements for modern gas engines concerning greater efficiency and increased power to weight ratio not only lead to an increase of the turbine entry temperature, but also to higher loads per stage which raise the exit Mach number of turbine's nozzle guide vanes (NGV) up to transonic flow regimes. Such flow-fields are characterised by NGV trailing edge shocks propagating periodically into the moving rotor passages. In the frame of the European research program TATEF2 (Turbine Aero Thermal External Flows) coordinated by SNECMA, experimental investigations have been carried out on the VKI's CT3 turbine rig at high vane exit Mach number in order to understand the progression of theses shock waves and the unsteady mechanisms by which it affects pressure and heat transfer fluctuations on the blades [1]. Numerical investigations associated to this case have been performed at ONERA .

Three dimensional unsteady RANS computations have been performed with *elsA* CFD software on a structured multi-block grid including 3.10^6 points (figure 1). The phase-lagged technique was used for the unsteady simulation in order to reduce the computational domain to one single blade passage for each row. Convergence of the computation takes 25 CPU hours on the vectorised computer NEC-SX8+. Three calculations have been accomplished corresponding to the different stage pressure ratios (P_{01}/P_{s3} = 2.42, 3.86, 5.12). Flow visualisation represented in figure 2 reveals a complex shock system for the high pressure ratio configuration at 25% of span. The trailing edge shocks emanating from the NGV interact periodically both with the wakes and with the adjacent blades, generating reflected waves.

Comparisons between CFD and experimental results have been achieved. Experimental and computed pressure distributions at mid-span on the stator and rotor airfoil are compared in figure 3 (left) for the three pressure ratios. On the vane the increase of pressure ratio shifts downstream the impact of the trailing edge shock, while the loading is increased in the rear suction side. On the rotor the increase of the pressure ratio leads first to positive incidence, once the airfoil becomes supersonic the rear suction side is submitted to higher acceleration rates. Satisfactory agreement between CFD and experiments is also obtained on the blades heat transfer distributions (figure 3-right), especially on the vane where the impingement of the shock on the rear part of the suction side increases heat transfer as the boundary layer becomes turbulent. Figure 4 represents a pitch-wise pressure distribution at shroud and hub at the stator /

rotor interface. Both experiments and CFD reveal the pressure rise due to the trailing edge shocks. Unsteady results have also been compared as the unsteady static pressure fluctuations which is plotted on figure 5 for 3 gauges located at the front suction side of the rotor. These gauges clearly identify the sweeping effect of the left running shock at the origin of pressure peaks. Finally, an analysis of the total pressure losses downstream has shown that the impingement of the vane trailing edge shocks on the rotor crown is at the origin of significant total pressure deficit. The detailed CFD analysis allowed identifying the creation of a small separation bubble on the rotor crown, downstream of the rotor impact. This vortical structure containing hot gases is then convected downstream. This phenomenon was named as 'vortical bubble' in 1990 by Johnson et al. (ASME Paper 90-GT-310) while performing shock interaction studies in linear cascades.

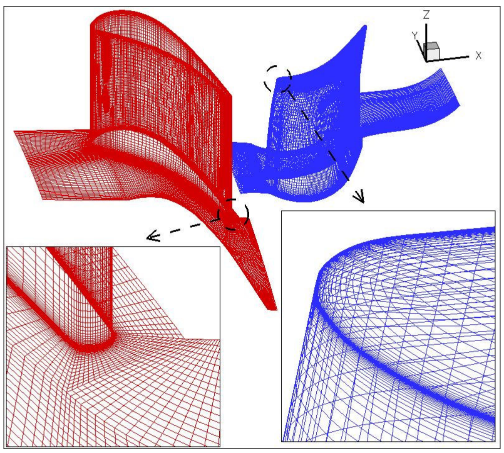


Fig 1: View of the computational grid.

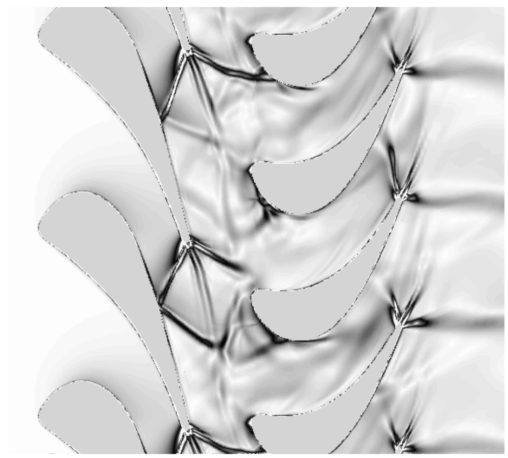


Fig 2: Density gradient at H/Ho~25%.

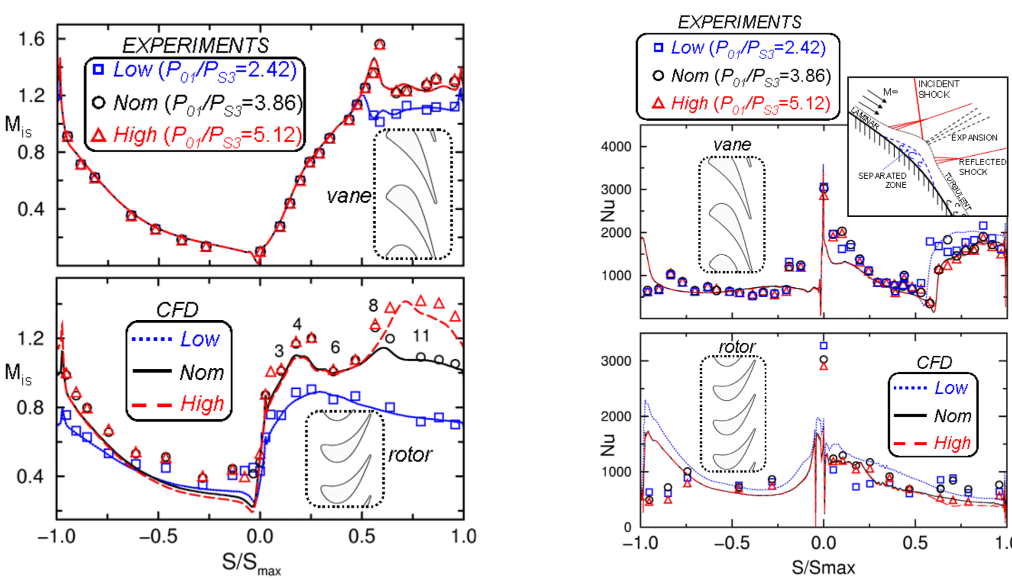
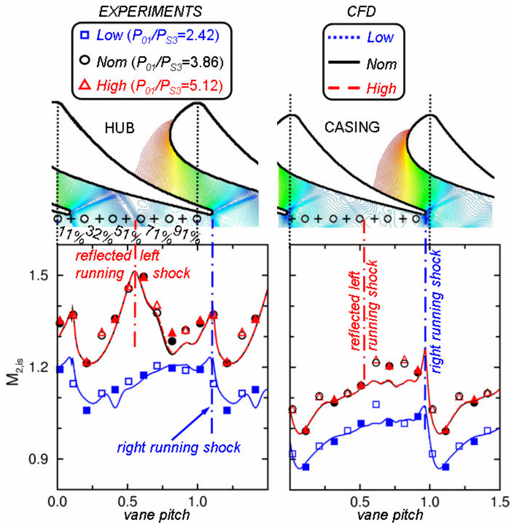


Fig 3:Time-averaged isentropic Mach number (left) and Nusselt number (right) at mid-span.



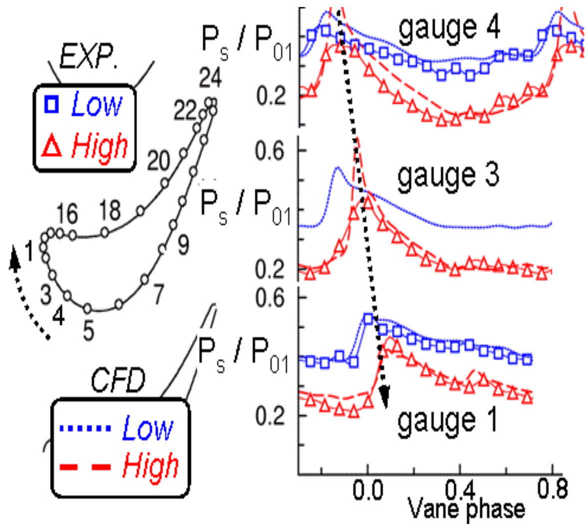


Fig 4: Pressure distribution at hub (left) & casing (right) at interface. Fig 5: Pressure fluctuations on the rotor.

REFERENCES

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