IMPROVING THE FILM COOLING OF A ROTOR BLADE PLATFORM

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ABSTRACT

In modern gas turbines the achievement of high performance requires a continuous increase of turbine inlet temperature. Hence an enhancement of thermal protection of all surfaces directly exposed to the hot gases, including platform regions, is needed. This paper shows the results of an experimental activity on a high-pressure-rotor blade of a real gas turbine with the aim of assessing the impact of newly designed film cooling holes on the thermal protection of a rotor blade platform. The investigation was performed in cooperation between Ansaldo Energia and the Department of Engineering and Applied Science of Bergamo University. The original rotor blade platform featured 10 cylindrical holes located along the blade pressure side. These holes were able to cool effectively the rear platform region, exploiting the passage vortex pressure to suction side end wall cross flow (Fig. 1) [1]. The channel front side was cooled exploiting the seal purge flow exiting the stator to rotor interface gap. Purge flow was able to cool the channel entrance region and the blade front suction side upstream the three dimensional separation line (Fig. 2) [2]. The front mid channel, and particularly the region around the inter-platform gap, remained uncooled. To protect this region a row of cylindrical holes was added to the previous cooling scheme. Besides the original 10 holes cooling the rear of the passage, the new platform featured five or seven cylindrical holes distributed inside the channel front part along the blade to blade inter-platform gap (Fig. 3). Two new blade platforms were manufactured for experimental verification: the first one with 5 holes and the second one with 7 holes. Tests have been performed at the Turbomachinery Laboratory of the University of Bergamo in the wind tunnel for linear rotor cascades. It is a continuous running suction type wind tunnel that assures a complete optical access because entirely made up of Plexiglas. The cascade model consists of a 7 blade cascade. Aerodynamic and thermal tests were carried out at Mach number Ma_{2is}=0.3, with a low inlet turbulence intensity level (Tu1 = 0.7 %). Coolant-to-mainstream mass flow ratio has been varied in a range of inlet blowing ratios M_1 from 0 to 2.0. To evaluate the interaction of injected flow with secondary flows a 5hole probe was traversed downstream of the trailing edge plane. The thermal behaviour was analysed by using Thermochromic Liquid Crystals technique, so to obtain film cooling effectiveness distributions. The comparison of the results at different blowing conditions allow to identify the optimal hole configuration and operating condition for the new cooling scheme.

References

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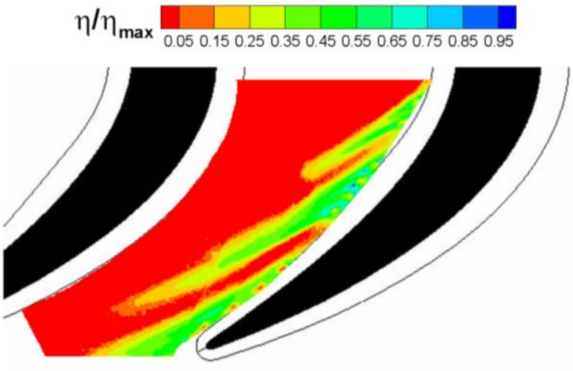

Figure 1.

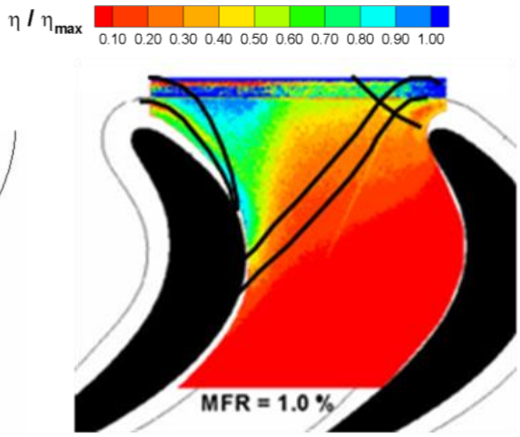

Figure 2:

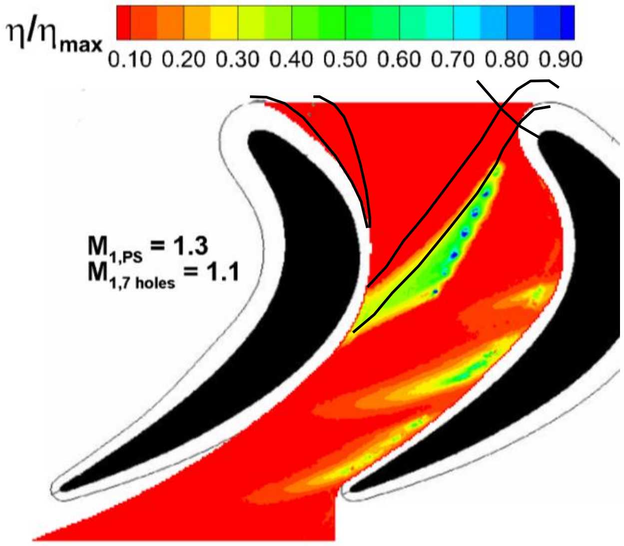

Figure 3.