Influence of cooling intensity on shock wave boundary layer interaction region in turbine cascade

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Abstract. The shock wave boundary layer interaction on the suction side of a transonic turbine blade was one of the main objectives of the TFAST project. For this purpose a model of a turbine passage was designed, manufactured and assembled in a transonic wind tunnel. The paper presents the experimental investigations concerning the flow structure on the transonic turbine blade. A clean case (without a cooling system) with a normal shock wave interacting with a laminar boundary layer and also the influence of the blade cooling system with three different coolant blowing intensities on the laminar interaction region were investigated.

1. Introduction

The research of which some parts are presented in this paper was carried out within the European Commission 7th Frame Work project under the acronym TFAST dedicated to transition location effects on the shock wave boundary layer interaction. A more restrictive regulation concerned fuel gas emissions and increasing demands of transport system effectiveness enforcing modification in aircraft engines. It is usually realized by reduction of mass and size of newly designed engines. This can be achieved by a reduction of the number of blades in specific turbine cascades. This leads to an increase in the blade load, and as a consequence, to the appearance of a local supersonic region terminated by the shock wave in the blade passages. In turbine passages the strong acceleration favours maintaining a laminar boundary layer in a long distance and very often in these cases the laminar shock wave boundary layer interaction (SWBLI) takes place which could be problematic for the engine performance in case that separation appeared. The process of separation usually becomes unsteady and is connected with shock oscillations in the internal flows. Shock oscillation causes the pulsation of pressure, thus resulting in blade load fluctuation. From that point of view, a more desirable type of interaction is a turbulent one [1] which can be induced by introducing disturbances upstream of the interaction area.

The main objective of the TFAST project was to study the transition location effects on the interaction structure of the shock wave with the boundary layer (BL). Steps, roughness elements, and air-jet vortex generators [2] [3] were used for the transition control. Moreover, the film cooling introduced into the flow by rows of holes has influence on the boundary layer development and furthermore on the SWBLI region. Therefore an objective of TFAST was also to investigate the boundary layer development on the gas turbine blade suction side and the effects on the shock wave boundary layer interaction [4]. Despite the cooling introduced into the boundary layer small disturbances, the strong acceleration of the flow is leading to relaminarisation of the boundary layer and there is a possibility that still the laminar boundary layer will be exposed to interaction with the shock wave.

In order to investigate the flow structure on the suction side of the turbine blade in a rectilinear test section of the transonic wind tunnel, a model of a turbine passage was designed, manufactured and

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assembled in the wind tunnel. The model can reproduce the flow structure, pressure distribution and boundary layer development similar to the obtained in a reference turbine cascade profile. The blade is a Nozzle Guide Vanes (NGV) with cooling in the gas turbine. The blade model was equipped with a cooling system which is a crucial part of measurements. The thermal efficiency of the turbine stage requires operating on a high turbine entry temperature. Such heat loads of blades are intended to work in temperatures between 800÷1000 °C which are typical for aircraft and heavy duty gas turbines [5]. To avoid heat-up of blades a thermal barrier coating is used and coolant air is injected to create a film cooling layer on the cooled surface [6]. The air for film-cooling in the turbine cascade is delivered from the compressor stage, thus the increase in the coolant mass flow reduces the overall engine performance. This approach imposes that the film-cooling has to be as efficient as possible. The film cooling effectiveness and its interaction with the main flow has been studied for many decades in various configurations [7], [8], [9].

The paper presents the experimental investigations concerning the flow structure on the transonic turbine blade. A clean case (without a cooling system) with a normal shock wave interacting with a laminar boundary layer and also the influence of a blade cooling system with three different coolant blowing intensities on a laminar interaction region were investigated. A cooling design process at most takes into account the cooling effectiveness and the coolant is introduced to the flow smoothly, at a small angle not to induce too many mixing losses. The conducted investigations will help to estimate how the increase in the cooling intensity upstream of the interaction region influences the losses downstream of the passage. Such study can also cast more light on some phenomena which are not always taken into account during the very complex turbine cascade design process.

2. Experimental arrangement

The measurement campaign was carried out in the Transonic Wind Tunnel of IMP PAN. Figure 1 shows the measurement system which consists of an inlet system with a drying unit, a test section, a vacuum tank, vacuum pumps and an additional pressure tank for dedicated tasks.

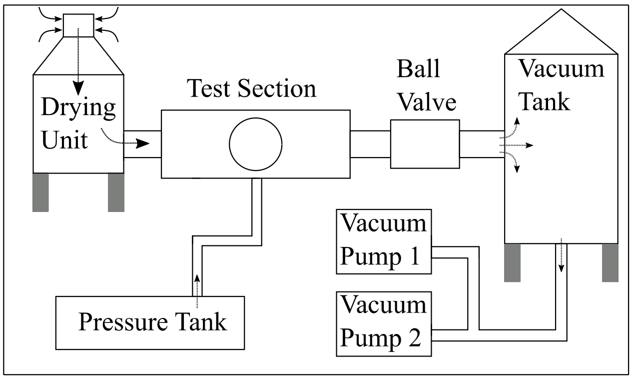

Figure 1. Measurement system scheme.

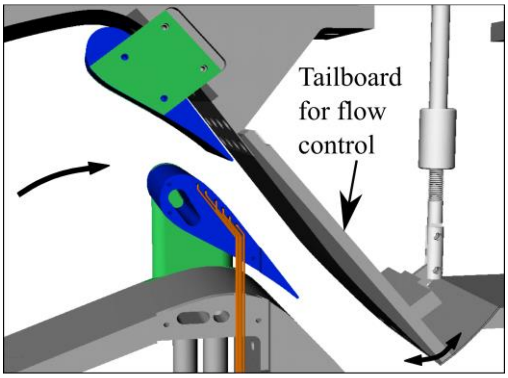

Figure 2. Sketch of turbine passage model.

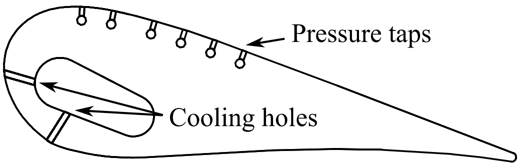

Figure 3. Cross section of lower profile.

In order to investigate the flow structure on the suction side of the blade, a turbine passage model was designed and assembled in the wind tunnel. The model can reproduce the flow structure, the pressure

distribution and the boundary layer development similar to the one obtained in a reference turbine cascade profile [10].
The single passage turbine test section (Figure 2) is a model of the reference linear turbine cascade supplied by Rolls-Royce Deutschland (RRD). The measurement configuration was also adjusted according to the reference cascade and was set up by RRD. The test section consists of two profiles and limiting upper and lower walls. The shape of the limiting walls and also the position of blades have significant influence on the Mach number distribution in the whole test section and in particular in the passage and the provided specific inflow conditions. The lower (Figure 3) and upper blade profiles are located in a similar configuration as in a linear cascade in order to keep a similar flow structure as in the reference passage. Both profiles are mounted in the channel at the required inlet Mach number M=0.13. The Reynolds number at the inlet of the passage based on the blade cord length is $Re_C=3\times10^5$, and the cord length c is 112.5 mm. The axial cord length Cax =52 mm.
In order to achieve the required flow structure the upper limiting wall was equipped with a movable part, the so-called tailboard. This part starts just above the upper blade and can influence the flow structure in the investigated passage. The investigation was focused on the flow analysis in the turbine passage, in particular on the suction side of the blade. The blade profile was equipped with two rows of cooling holes located near the leading edge. The cooling holes were 1.35 mm in diameter. The hole angle of the first row to the stream direction (tangential) was 90°, whereas it was 16.25° in second row. The measured profile was also equipped with pressure taps located along the blade at the middle span. The cooling holes were connected to the main cavity where pressurised air was delivered from the pressure tank. Three different cooling intensity blowing ratios (cavity pressure in relation to the main flow stagnation pressure) were tested in the experiments. The investigation concerned the influence of the cooling intensity on the shock wave boundary layer interaction at the suction side of the turbine blade, therefore four configurations were tested:

- Clean case, uncooled.
- Film-cooling with blowing ratio Pc/P0=1.05.
- Film-cooling with blowing ratio Pc/P0=1.10.
- Film-cooling with blowing ratio Pc/P0=1.15.

3. Results of Experimental Investigation
The results are presented in form of charts obtained from the measured data and in form of photographs taken from the schlieren and oil visualizations. To investigate some aspects of the cooling intensity effects on the SWBLI region the following measurements were carried out:

- Static pressure along the lower wall and blade.
- Schlieren visualization.
- Oil visualization.
- PSP (Pressure Sensitive Paint) measurements.
- Shock oscillations measurements.
- Velocity traverse measurements by means of LDA.

3.1. Pressure distribution on blade
Film cooling introduces some disturbances into the flow which have influence on the SWBLI region, and consequently on the shock wave system on the suction side of the profile. Figure 4 shows normalized static pressure distribution on the suction side along the profile at middle span versus normalized cord length. Static pressure was measured by means of PSI scanner with accuracy of 0.05% of full scale (1 bar), i.e. ±0.5hPa. For all measured configuration the inlet stagnation temperature was the ambient one. The plot shows the results for flow without cooling (clean case) and with cooling of various blowing intensity. One can observe that in the uncooled configuration (the green line with triangles) the pressure jump is spread over a longer distance opposite to the film-cooling case. For cooling configurations the

pressure lines mostly coincide, except for the main shock wave region. The differences in pressure distributions result from the shock system distinction for specific cases.

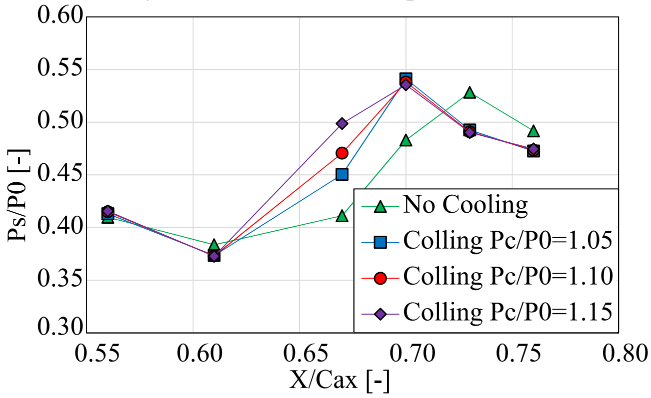

Figure 4. Pressure distribution on turbine blade suction side.

3.2. Schlieren visualization

Figures 5-7 show the schlieren visualization for a clean case and for cooling cases with pressure ratios of Pc/P0=1.05 and 1.15. In every single measurement in specific set-up, pressure and velocity distribution were repeatable. For all configurations one can observe a shock wave interaction with the laminar boundary layer. The laminar type of interaction indicates that a very large λ-foot (as it's visible on schlieren pictures) appears in the flow at such a relative small Mach number upstream of the shock (M=1.26). Moreover, the shock pressure jump is realized by means of two shock waves in the clean case. Film cooling just downstream of the jets locally turbulises the boundary layer, nevertheless, when the flow accelerates one can observe relaminarization of the boundary layer upstream of the interaction area. Downstream of the interaction region the flow accelerates again to a local supersonic region which is closed by the shock at the blade trailing edge.

Figure 5. Schlieren visualization, clean case.

Figure 6. Schlieren visualization, Pc/P0=1.05.

Increasing the cooling intensity modifies the shock wave system topology where the second shock gradually vanishes with the increasing blowing ratio. The case with a maximal cooling intensity produces only one main shock wave. In general one can notice that the cases with cooling are characterized by smaller λ-foot in comparison to the clean case. The film cooling introduces some energy into the boundary layer, and as a result, the shock pressure jump can be realised at a shorter distance, thus the λ-foot height decreases.

Figure 7. Schlieren visualization, Pc/P0=1.15.

3.3. Oil visualization

The flow structure on the suction side of the turbine profile from oil flow visualization in Figures 8-9 is shown. The view onto the blade is through the sidewall window of the test section. The flow is from left to right. One can observe a two dimensional flow with a separation region induced by the SWBLI. The separation onset starts at the front λ-foot of the first normal shock wave (see Figure 5). It is worth noticing that the interaction at the sidewall between the main flow and the corner vortex is very weak, thus the separation region spreads almost from one lateral wall to the opposite one. For the clean case, the separation size length is about 9 mm (Figure 8).

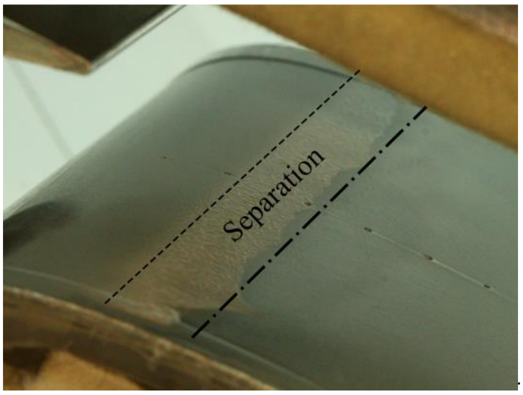

Figure 8. Oil visualization, clean case.

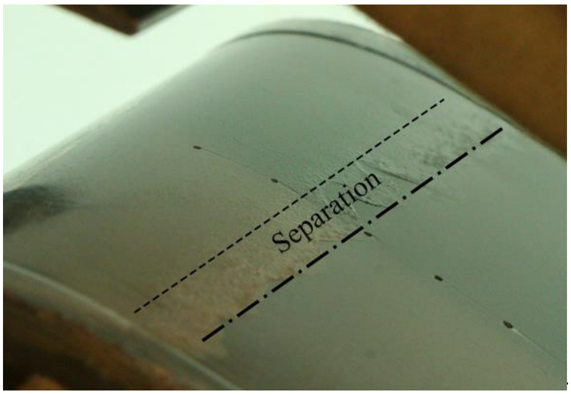

Figure 9. Oil visualization, cooling Pc/P0=1.15.

The film cooling introduced to the flow (Figure 9) reduces the separation region in relation to the clean case (Figure 8). For this case the separation length is about 6mm. The increasing of cooling intensity has no effect on the separation length. The cooling increases the fluid momentum in the boundary layer and therefore the separation region decreases. Nevertheless, the differences in the fluid momentum for various cooling intensities are probably too small to influence the separation size.

3.4. Pressure distribution on blade by means of PSP

In the PSP (Pressure Sensitive Paint) measurements the ‘in situ’ calibration was used, what means that the paint light intensity response was determined based on the pressure measured by the taps (see Figure 3) located along the blade. Figures 10 and 11 present normalized static pressure distribution on the blade suction side measured by means of PSP. The results are presented only for the flow with cooling (Pc/P0=1.05 and 1.15). The results for the clean case are not presented because the PSP in this flow modifies the shock system on the blade, hence the pressure distribution of the same flow with and without paint is not comparable. As in the foregoing oil visualization analysis one can clearly notice in

PSP measurements a two dimensional flow structure on the suction side of the turbine profile. Furthermore, very weak corner effects are also observed. Figures 10 and 11 show a very well displayed pressure jump (dashed lines) on the shock wave following a strong acceleration region (blue area). Downstream of the shock wave the flow accelerates again to the supersonic region which is terminated by another shock at the trailing edge.

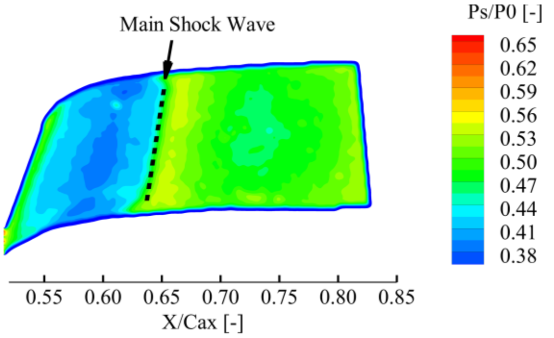

Figure 10. Blade pressure distribution for cooling of blowing ratio Pc/P0=1.05.

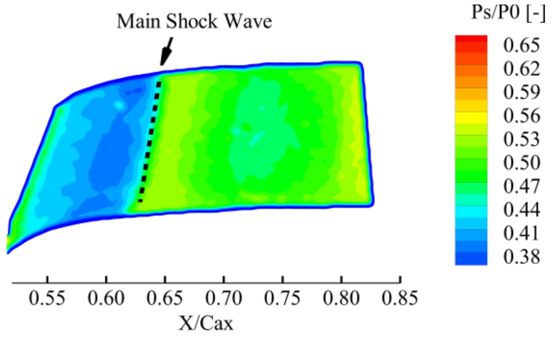

Figure 11. Blade pressure distribution for cooling of blowing ratio Pc/P0=1.15.

The plot in Figure 12 shows normalized static pressure distribution at the blade middle span obtained by PSP measurements. In the region just upstream of the shock wave the flow slightly decelerates (the pressure increases) showing that the onset of the interaction starts upstream of the shock wave what is a typical phenomenon for a laminar type of interaction [11] [12] and [13]. This observation is in agreement with schlieren visualization where one can also observe that a boundary layer upstream of the shock system is laminar. It is also visible that increasing the cooling intensity shifts the shock pressure jump upstream of the flow (see Figures 10 and 11).

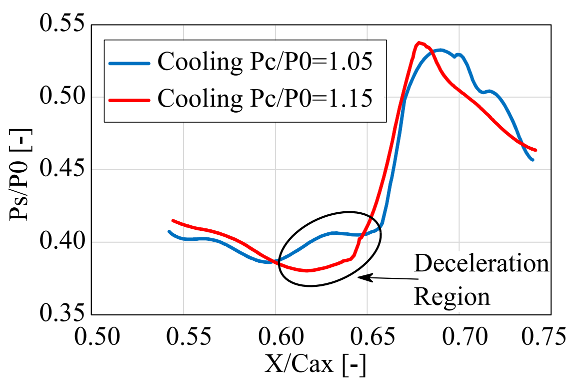

Figure 12. Pressure distribution on suction side of lower profile from PSP measurements.

3.5. Velocity profiles

The location of traverses for velocity measurements by means of LDA (Laser Doppler Anemometry) is presented in Figure 13. The first traverse is placed 10 mm upstream of the main shock wave position at X/Cax=0.55 and the second is located 20 mm upstream of the trailing edge at X/Cax=0.94. All traverses were measured in the channel centre. In every traverse the Y coordinate (distance from the wall) is perpendicular to the wall. Spatial resolution of LDA measurements in perpendicular direction towards the wall was approximately 80 µm.

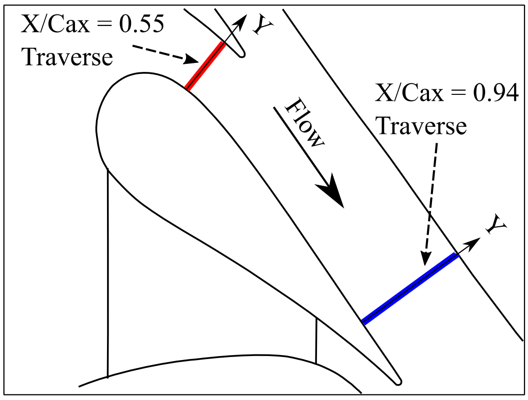

Figure 13. Locations of traverses for velocity measurements by means of LDA.

Figure 14 shows the velocity profiles in the passage upstream of the shock wave. The curves for all configurations nearly coincide. One can notice a supersonic region (velocity ratio above U/Umax=0.87) in the vicinity of the lower profile suction side. At some distance from the lower wall towards the upper blade the flow decelerates to subsonic values. The distribution of seeding in the laminar boundary layer was poor (due to the weak mixing), thus the reliability of the results permits only to establish the BL thickness which reaches approximately 0.5 mm. The film cooling introduced into the flow has no impact on boundary layer thickness in that region.

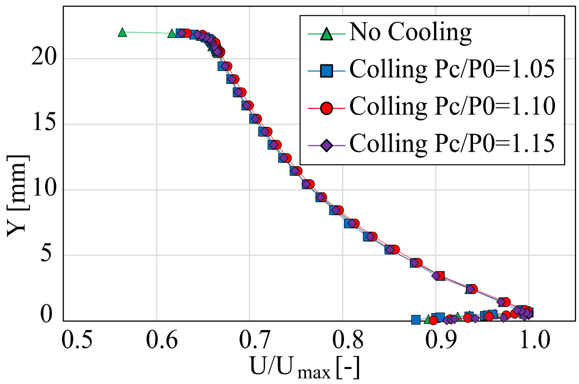

Figure 14. Velocity profile upstream of interaction, X/Cax=0.55.

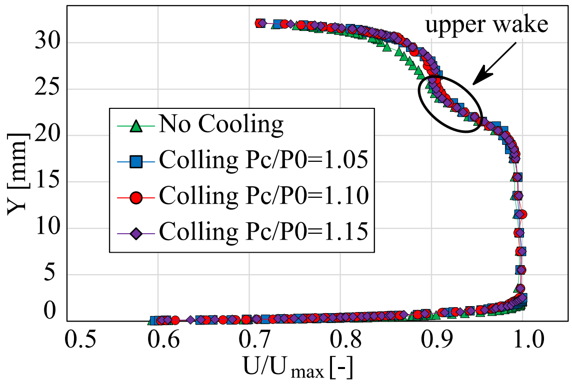

Figure 15. Velocity profiles downstream of interaction, X/Cax=0.94.

The results of measurements in traverse upstream of the trailing edge are shown in Figure 15. Velocity distribution for all cases are very similar except the boundary layers. In the plot one can observe a barely visible wake from the upper blade where the deceleration connected with the wake has almost dissipated due to a strong mixing process. The detailed velocity distributions in the boundary layer at X/Cax=0.94 are shown in Figure 16. The curves are mostly overlapped with one exception for the clean case. The BL in the clean configuration is not disturbed by any film cooling, therefore the boundary layer is thinner in relation to the cooling case.

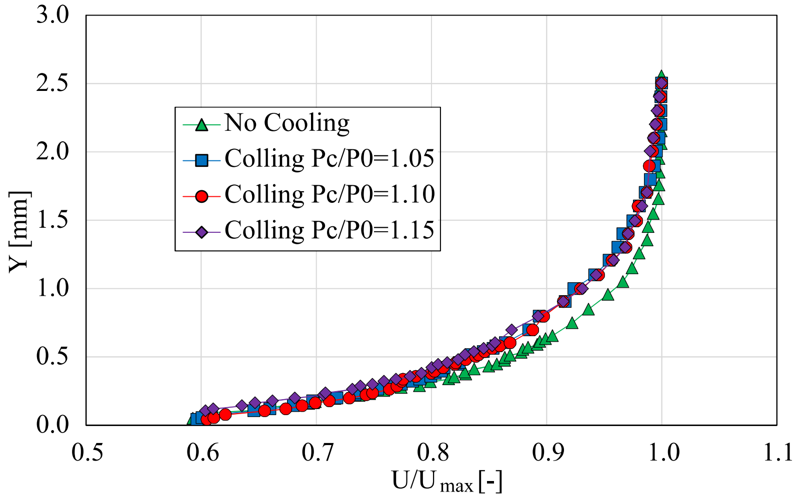

Figure 16. Velocity profiles in boundary layer downstream of interaction, X/Cax=0.94.

In order to calculate the boundary layer parameters the following formulas were used:

$$\delta^* = \int_0^\infty \left(1 - \frac{u(y)}{u_0}\right) dy \text{ - displacement thickness} \tag{1}$$

$$\delta^{**} = \int_0^\infty \frac{u(y)}{u_0}\left(1 - \frac{u(y)}{u_0}\right) dy \text{ - momentum thickness} \tag{2}$$

$$H_{12} = \frac{\delta^*}{\delta^{**}} \text{ - shape factor,} \tag{3}$$

the δ is determined as 0.99 of local free stream velocity.

Cooling increases the boundary layer thickness by about 30% in relation to the uncooled flow in the passage. The cooling intensity has very limited effect on the boundary layer size. The integral parameters like displacement and momentum thickness increase about 20% with introducing film cooling into the flow. Moreover, the shape parameter indicates that the boundary layer is turbulent downstream of the interaction region.

Table 1. Boundary layers integral parameters downstream of interaction.

	No Cooling	Cooling Pc/P0=1.05	Cooling Pc/P0=1.10	Cooling Pc/P0=1.15
δ [mm]	1.48	1.80	2.05	1.98
δ^* [mm]	0.16	0.20	0.21	0.19
δ^{**} [mm]	0.13	0.16	0.17	0.15
H_{12}	1.27	1.25	1.24	1.24

4. Conclusions

The influence of the cooling intensity on the interaction region between the shock wave and the boundary layer was analysed in these investigations. The following conclusions can be drawn from the above presented results:

- A turbine passage model was manufactured and assembled in the rectilinear test section of a transonic wind tunnel. The flow field contains characteristic features of the flow in the reference NGV turbine cascade.
- Both in the reference flow (without cooling) and in the flow with cooling the laminar shock wave boundary layer interaction is observed; the transition is induced in the interaction region, downstream of the separation the boundary layer is turbulent.

- The blade surface cooling intensity changes influence to a certain extent the pressure distribution on the blade suction side only in the interaction region. Increasing the blowing intensity also causes gradual shifts of the shock position towards the leading edge.
- Film cooling alters the shock system in the interaction area, from a double normal shock wave in the clean case to the gradual modification to a single normal shock wave for the highest cooling intensity.
- The cooling flow causes a reduction of the separation size induced by the shock wave, nevertheless, the cooling intensity increase does not affect the separation region size.
- Film cooling energizes the boundary layer a little (the separation size decreases), however the boundary layer velocity deficit downstream of the interaction increases.
- The boundary layer upstream of the interaction area has the same thickness for all cases, approximately 0.5 mm. Downstream of the interaction the cooling system causes the boundary layer to grow by about 30%, also the displacement and momentum thickness increase.

Acknowledgments
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References
[1] Babinsky H, Harvey J K 2011 Shock Wave-Boundary-Layer Interactions *Cambridge University Press vol 1*
[2] Szwaba R 2013 Influence of Air-Jet Vortex Generator Diameter on Separation Region *Journal of Thermal Science* vol 22 no 4 p 294-303
[3] Szwaba R 2011 *Comparison of the Influence of Different Air-Jet Vortex Generators on the Separation Region* Aerospace Science and Technology vol 15 p 45-52
[4] Petersen A 2014 Numerical Transition Prediction in a Straight Turbine Cascade *11th World Congress on Computational Mechanics* (WCCM XI)
[5] Naik S, Krueckels J, Grisch M, Schnieder M 2014 Multirow Film Cooling Performances of High Lift Blade and Vane *Journal of Turbomachinery* vol 136 / 051003-1
[6] Narzary D P, Liu K-C, Rallabandi A P, Han J-C 2012 Influence of Coolant Density on Turbine Blade Film Cooling Using Pressure Sensitive Paint Technique *Journal of Turbomachinery* vol 134 / 031006-1
[7] Goldstein R J 1971 Film Cooling Advances in Heat Transfer *New York: Academic Press* vol 7 p 321-379
[8] Bogard D G, Thole K A 2006 Gas Turbine Film Cooling *Journal of Propulsion and Power* vol 22, no 2
[9] Mhetras S, Han J-C, Rudolph R 2012 Effects of Flow Parameter Variations on Full Coverage Film-Cooling Effectiveness for a Gas Turbine *Blade Journal of Turbomachinery* vol 134/ 011004-1
[10] Doerffer P, Flaszynski P, Szwaba R 2014 Design of the test section *TFAST Deliverable D4.1.1* TFAST Webside - www.tfast.eu
[11] Flaszyński P, Doerffer P, Szwaba R, Kaczyński P, Piotrowicz M 2015 Shock Wave Boundary Layer Interaction on Suction Side of Compressor Profile in Single Passage Test Section *Journal of Thermal Science* vol 24 vol 6 p 510-515
[12] Szwaba R, Flaszyński P, Szumski J A, Doerffer P 2015 Influence of Air Cooling and Air- Jet Vortex Generator on Flow Structure in Turbine Passage *TASK Quarterly* vol 19 no 2 p 153-166
[13] Gomes R A, Niehuis R 2012 Aerothermodynamics of a High-Pressure Turbine Blade With Very High Loading and Vortex Generators *Journal of Turbomachinery* Vol. 134 pp 011020 1-9