EXPERIMENTAL AND NUMERICAL STUDY OF THE TRANSONIC COOLED TURBINE BLADES

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ABSTRACT
Modern gas turbines operate at high temperatures, which exceed the endurance limit of material, and therefore the turbines components are cooled by the air taken from the compressor. The cooling provides positive impact on lifetime of GT has negative impact on its performance. In convection-cooled turbine blade the coolant is usually discharged through the trailing edge and it leads to limitation on the minimal size of trailing edge and thereby negatively affects the losses. Moreover, the injection of cooling air in the turbine disturbs the main flow, and may lead to additional increase of losses, and the trailing edge loss is a significant part of the overall loss in modern gas turbines.

This study comprises investigations of losses in cooled blades. Four cascades with different unguided part of aerofoil with and without coolant injection were studied both experimentally and numerically. This analysis provides split of losses caused by different factors, and offers the opportunities for efficiency and lifetime improvements of real engine designs/upgrades. In particular it is shown that an increase in the unguided turning angle results in a reduction of loss in case of relatively thick trailing edge. It is also shown that injection through the trailing edge slot parallel to the main flow leads to neutral loss impact and even loss reduction in subsonic range and loss increase in the supersonic range of exit Mach numbers.

KEYWORDS: cooled transonic blade, profile losses, unguided flow angle impact, cooling air injection

NOMENCLATURE
\( a_2 \) – throat width
\( C_x \) – axial chord length
\( C_{pb} \) – base pressure coefficient
\( c \) – velocity
\( d_2 \) – trailing edge thickness
\( d_2/a_2 \) – ratio of trailing edge diameter to throat width
\( L \) – true chord length
Loss – profile energy loss
\( M \) – Mach number
\( P_0 \) – total pressure
\( P \) – static pressure
\( P_b \) – base pressure
\( P_{2av} \) – averaged exit static pressure
\( R \) – gas constant
\( Re \) – Reynolds number
\( s \) – coordinate along wetted surface of profile
\( T_0 \) – total temperature
\( t \) – pitch
Tu – turbulence intensity  
w_2 – trailing edge wedge angle  
z – distance between planes

**Subscripts**  
1 – row inlet  
2 – row exit  
is – isentropic

**Abbreviations**  
LE – leading edge  
PS – pressure surface  
SS – suction surface  
TE – trailing edge

**Greeks**  
\( \beta_{1m} \) – inlet metal angle  
\( \beta_{2e} \) – outlet effective angle;  
\( \beta_{2e} = \arcsin(a_2/t) \)  
\( \gamma \) – stagger angle  
\( \delta \) – unguided turning angle or uncovered turning  
\( \theta \) – perimeter of aerofoil  
\( k \) – specific heat ratio  
\( \nu \) – kinematic viscosity

**INTRODUCTION**

Modern gas turbines operate at high temperatures which exceed the endurance limit of material and therefore the turbine components are cooled by the air taken from the compressor. The cooling providing positive impact on lifetime of gas turbine (GT) but has negative impact on its performance.

The cooling air bypasses combustor and therefore provides less work in the turbine than hot gas due to lower enthalpy. Moreover in multistage turbine this coolant provides as less work as is ejected further downstream of flowpath. However the injection of cooling air also has a second effect – increase of aerodynamic losses.

The aerodynamic losses in turbines were intensively investigated since the mid of 20th century. The different types of losses were identified (e.g. friction losses, trailing edge losses, secondary losses, etc.), and several correlations were proposed for each loss type. The results of these investigations have been published in numerous papers and books, where the references [1-8] are only a small portion of them, representing the most often cited works.

One source of losses is cooling, and its impact on turbine aerodynamic is twofold. Firstly cooling requirements lead to limitation on the blade shape, where the most important factor is the limitation on the trailing edge size. To ensure cooling air injection the blade should have a slot or holes at the trailing edge, and therefore the trailing edge thickness cannot be reduced according to aerodynamic requirements. This feature causes additional losses. Secondly coolant injection disturbs the main flow, and may lead to increased losses. As far as the coolant is mixed with main flow these losses are called mixing losses. The models for these losses are presented in [2, 5, 7, 9-11].

Many researchers investigated the trailing edge losses without cooling air injection (e.g. [12-20]). Basically these studies show that direct way of trailing edge loss reduction is application of thin trailing edges which may be not applicable to the cooled blades.

However few reports in open literature have already pointed out that trailing edge thickness reduction is not the only way to achieve trailing edge loss reduction. Granovskiy et al. [20] reduced profile loss in a transonic vane by the suction surface curvature redistribution. Chao Zhou et al. [21] examined the effect of the trailing edge thickness on losses in ultra-high lift turbine blades. These blades had a significant unguided turning angle which allowed the profile loss reduction even with a thicker trailing edge.

Vagnoli et al. [22] demonstrated the evolution of the trailing edge base pressure in function of the downstream Mach number on the basis of the highly unsteady character of the near wake flow which affects directly the trailing edge base pressure.
In present work the results of a combined experimental and numerical investigation of the unguided turning angle and flow injection through the trailing edge influence on transonic cooled blades loss are presented.

To understand the effect of the unguided turning angle four linear cascades were tested in a transonic wind tunnel at a wide range of exit Mach number (M = 0.7-1.2). A numerical study of these cascades has been carried out using commercial code Fluent. These cascades consist of blades having nearly the same profile and other parameters but differ from each other in the value of the unguided turning angle.

To understand the impact of the trailing edge injection one of these cascades was tested and numerically simulated at different flow rates of injected air.

**EXPERIMENTAL AND NUMERICAL PROCEDURES**

**Experimental setup**

Large-scale plane cascades were tested in the transonic wind tunnel. The air from the atmosphere is sucked to the test section, and therefore the total pressure and total temperature there correspond to ambient conditions (typically P₀≈1.013bar, T₀ ≈ 295K). Inlet turbulence intensity was maintained at the level Tu= 0.02. The exit Mach number (Mₑ) varies in the range Mₑ = 0.67 - 1.12, and the Reynolds number (Re = cₑL/ν) based on profile true chord length varies in the range Re = (0.5-0.8) x10⁶.

The air injected through the trailing edge is supplied by separate high pressure line. For simplicity this air is called cooling air, although it does not have cooling function in these tests. In order to affect the properties of “coolant” CO₂ or N₂ can be added to the supply line. More details of this rig can be found in [10]. The mass flow of injected air varied in range 0< m̅ <7%, where m̅ is the ratio of injected air mass flow to the main mass flow.

The experimental cascades consist of 7 to 9 aerofoils with height h=125 mm. Basic geometric parameters of cascades are illustrated in Fig. 1.

![Fig.1. Basic geometric parameters of cascade](image)

The true chord length varies in the range L= 50-90 mm (L depends on type of aerofoil). The boundary layer accumulated on the walls of the test rig upstream of a test section is cut off by special expansion plates located near the endwalls of the cascade. The cooling air is injected through the simple slot in the trailing edge characterised by one parameter - width Δ.

The total and static pressure distributions downstream of cascade are measured using combined probe and pressure taps located at distance z behind the trailing edge. The relative value z/a₂ belongs to the range 0.3 ⋯ 1.5 (depending on throat size). The positioning of measurement planes is shown in Fig.2.
Fig. 2. Measurement planes downstream of cascade
1 - cascade; 2– combined probe; 3 - static pressure taps; 4- measurement plane (z – distance between cascade and measurement plane).

One part of the combine probe with a fang measures static pressure and is located apart from the other part with 3-holes to avoid interference between them. The 3-hole probe measures total pressure and flow angle (the central hole is used for total pressure measurements, and other two with 45 degrees oblique ports measure flow angle). The probe can be positioned in different axial planes, and the pressure distributions are obtained by traversing the probe along the cascade. Static pressure distribution in front of the cascade is measured by 20-30 static taps located at a distance (1.5-2.0) $a_2$ upstream of the leading edge. The static pressure distribution along the aerofoil and the base pressure on the trailing edge are measured by the static pressure taps at the midspan section. Traversing of flow in spanwise direction showed that total and static pressure distributions are uniform within half of span around the midspan position, and therefore the flow at midspan can be considered as 2D. Traversing in tangential direction showed that flow is periodic at least for 3 aerofoils at the middle of cascade.

The mass flow rate of injected cooling air is measured by orifice. The total pressure and temperature of injected air are measured in the supply line, and these parameters are practically the same at the injection point, since there is no cooling channels inside the aerofoil, but just a hollow cavity with negligible losses.

Based on these measurements the profile loss are calculated using the following expressions (see e.g. [10])

$$V_{2is}^2/2 = c_p T_{01} \left[ 1 - \left( \frac{P_2}{P_{01}} \right)^{\frac{k-1}{k}} \right]$$

$$V_{cis}^2/2 = c_p T_{0c} \left[ 1 - \left( \frac{P_2}{P_{0c}} \right)^{\frac{k-1}{k}} \right]$$

$$\text{Loss} = 1 - \frac{V_{2is}^2}{V_{2is}^2 + \bar{m} V_{2is}^2 / V_{2is}^2}$$

(1)

Here $V_{2is}$ is the isentropic velocity of the main air (i.e. velocity the air would have after expansion to pressure $p_2$ without losses), $V_{cis}$ - is the isentropic velocity of the cooling air, $V_2$ is actual flow velocity at the cascade outlet, and $\bar{m}$ - relative mass flow of cooling air. This expression is valid if the total parameters of cooling air are the same at all injection points, and therefore it is valid in cases with single injection slot considered in this work. If the total parameters at different injection locations are different, this expression should be modified appropriately.

The velocity at the outlet is related to total pressure and temperature by relationship

$$V_2^2/2 = c_p T_{02} \left[ 1 - \left( \frac{P_2}{P_{02}} \right)^{\frac{k-1}{k}} \right]$$

(2)

In cases without injection ($\bar{m} = 0$) the equation (1) can be reduced to
Loss = \left( \frac{P_{01}}{P_{02}} \right)^{\frac{k-1}{k}} - 1 \div \left( \frac{P_{01}}{P_{2}} \right)^{\frac{k-1}{k}} - 1 \right)

(3)

In cases without injections the base pressure coefficient is calculated using the following expression

\[ C_{pb} = \frac{P_b - P_{2av}}{P_{02} - P_{2av}} \] (4)

This parameter characterise the separation and wake width, and therefore is correlated with losses as it is shown in [15]. The isentropic Mach number is derived from the equation

\[ P_{01}/P = \left( 1 + \frac{k-1}{2} M_{is}^2 \right)^{\frac{k}{k-1}} \] (5)

**Numerical model**

The numerical simulations were performed using commercial solver Fluent. The calculations are done at the mid span, and the flow is assumed 2D and periodic in spanwise direction. The unstructured numerical mesh was generated using commercial code Gambit. Fig.3a shows this mesh, which contains 50000 cells. The mesh is stretched within the boundary layer according to the exponential law, and the value of \( y^+ \) in the first cell on the wall is below 1. The mesh behind the trailing edge is also refined to resolve the shocks at supersonic exit conditions.

In calculations with injection the exit part of slot is included into the domain, as shown in Fig.3b. Calculations without injections were done using two meshes: 1) unstructured mesh described above, where the slot is present but not active, and 2) structured mesh without slot. The latter has value of \( y^+ \) in the first cell on the wall below 1, but has two times less cells outside boundary layer. Nevertheless the results obtained on both meshes are close to each other and in the following discussion only the data for unstructured mesh are shown.

The turbulence was simulated using k-\( \varepsilon \) realisable turbulence model. The turbulence intensity at the inlet equals to experimental value \( Tu = 0.02 \).

![Fig.3. View of numerical grid around Cascade 1: a) view of two periodic pieces, b) fragment at the trailing edge](image)

In all cases considered the flow at the cascade inlet is assumed to be uniform, axial, with a constant total pressure and temperature. At the cascade outlet the static pressure is specified (according to (5) it is equivalent to specification of isentropic exit Mach number). The no-slip conditions are applied on the solid walls. The walls are considered as adiabatic in calculations without injection and with cold air injection. In cases with injection the total temperature and mass flow of injected air are prescribed at the inlet of slot.
UNGUIDED TURNING ANGLE

The influence of the unguided turning angle (δ) on the base pressure and losses was studied using 4 vane cascades corresponded to real cooled gas turbine profiles with about the same basic geometric parameters except for the angle δ. These basic parameters of the investigated cascades are presented in Table 1. It shows that all cascades have exactly the same pitch to chord ratio (0.734) and approximately the same ratio of trailing edge thickness to throat width (0.115).

<table>
<thead>
<tr>
<th>Vane</th>
<th>β₁m [deg.]</th>
<th>β₂e [deg.]</th>
<th>t/l [-]</th>
<th>δ [deg.]</th>
<th>d₂/a₂ [-]</th>
<th>Φ [deg.]</th>
<th>ϑ/d₂</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cascade 1</td>
<td>86.2</td>
<td>21.91</td>
<td>0.734</td>
<td>2.3</td>
<td>0.115</td>
<td>50</td>
<td>0</td>
</tr>
<tr>
<td>Cascade 2</td>
<td>84.4</td>
<td>21.82</td>
<td>0.734</td>
<td>8.0</td>
<td>0.114</td>
<td>50</td>
<td>0</td>
</tr>
<tr>
<td>Cascade 3</td>
<td>84.6</td>
<td>21.42</td>
<td>0.734</td>
<td>11.0</td>
<td>0.115</td>
<td>50</td>
<td>0</td>
</tr>
<tr>
<td>Cascade 4</td>
<td>85.2</td>
<td>21.6</td>
<td>0.734</td>
<td>15.9</td>
<td>0.116</td>
<td>50</td>
<td>0.36</td>
</tr>
</tbody>
</table>

Fig. 4 shows the measured profile loss and base pressure coefficient as functions of the exit Mach number for Cascade 1 and Cascade 4. There is a correlation between the base pressure coefficient and profile losses. In Cascade 1 (see Fig. 5-a) with practically a flat suction surface after the throat (δ = 2.3°) the base pressure coefficient decreases from $C_{pb} = -0.1$ to $C_{pb} = -0.49$ over the exit Mach number range 0.6 - 0.98. At the same time the profile loss increases from Loss = 0.043 to Loss = 0.075. At $M_{2is} = 0.98$ the base pressure coefficient reaches the minimum and the profile loss its maximum. Such behaviour of these parameters is related to the flow structure on the suction side. The thickness of boundary layer on the suction surface increases after the shock in the throat area, and on the flat surface this thickness remains practically unchanged up to the trailing edge. The boundary layer on the surface influences on the wake thickness, and therefore on loss. The size of recirculation zone behind the trailing edge is also affected by the boundary layer, and this size is correlated with the base pressure. With increase of exit Mach number up to $M_{2is}=0.98$ the shock at the throat became stronger, and therefore leads to loss increase and base pressure drop.

At supersonic exit flow the acceleration near the trailing edge partially compensates the boundary layer growth, and therefore loss is somewhat reduces and base pressure increases.

![Fig. 4 Measured losses and base pressure coefficient](image-url)

a) - Cascade 1 with δ = 2.3°

b) - Cascade 4 with δ = 15.9°

(symbols are measured values, curves – trend lines)
In cases with high unguided turning angle the shock at throat is less intensive, and therefore the boundary layer on the suction surface is thinner than in cascades with low $\delta$. The separation from the trailing edge is also delayed, which may lead to higher base pressure than averaged static pressure $P_{2av}$ and positive base pressure coefficient ($C_{pb}>0$). Fig. 4-b shows the measured profile loss and base pressure coefficient as functions of the exit Mach number for Cascade 4 with maximal unguided turning angle $\delta =15.9^\circ$. In this case the base pressure coefficient increases from values close to zero to positive values $C_{pb}>0$ and the profile loss goes down over the exit Mach number range 0.72-0.94. Over the range of $M_{2is} = 0.94-1.1$ the base pressure coefficient decreases and the profile loss grows. The behaviour in the first range of exit Mach number can be explained by the above mentioned effect of suction surface curvature, and in the second range the behaviour of loss and base pressure is similar to Cascade 1 because in this range the throat shock strength is more affected by increase of Mach number level then by suction side curvature. The results of calculations presented below (Fig.7) illustrate the effect of parameter $\delta$ on the flow structure. The maximal level of profile losses in the Cascade 4 at $M_{2is} = 1.1$ equals to Loss = 0.061. As in Cascade 1 there is a correlation between the base pressure coefficient and the profile loss throughout the whole investigated range.

The results of the base pressure coefficient and profile loss measurements for all vane cascades listed in Table 1 are summarised in Fig.5. In order to make this chart easier to read, only the trend lines are presented and the measured points are not indicated in these figures.

Fig.5-a demonstrates that the base pressure coefficient is a non-monotonic function of the exit Mach number and in each variant it has a minimum. The point of minimum moves from $M_{2is} = 0.98$ in Cascade 1 ($\delta =2.3^\circ$) to $M_{2is} = 1.1$ in Var. 4 ($\delta =15.9^\circ$). The comparison of the base pressure coefficient for different variants shows that in the transonic range ($M_{2is} = 0.77 – 1.12$) and with a constant value of the relative trailing edge thickness ($d_2/a_2 = 0.115$) the base pressure coefficient $C_{pb}$ strongly depends on the unguided turning angle (i.e. it depends on the curvature of SS after throat).

The increase in the unguided turning angle results in the increase of the base pressure coefficient and decrease of the profile losses. In Cascade 4 with relatively high unguided turning angle $\delta =15.9^\circ$ the base pressure coefficient became positive $C_{pb}>0$.

Thus the test data shows that in the transonic range ($M_{2is}=0.77 – 1.0$) the increase of the vane unguided turning angle causes the increase of the base pressure coefficient and reduction of the profile losses by 3% - 4%.

The losses in Cascades 1- 4 were also numerically calculated at the exit Mach numbers equal $M_{2is} = 0.77, 0.88, 1.06$. Fig.6 shows the comparison between measured and calculated profile loss over a range of unguided turning angles from $2^\circ$ to $16^\circ$. There is reasonable agreement between the numerical and measured data. Both
the experimental and numerical data show that as the unguided turning angle increases, the profile losses in Vane decreases by 2-4% over the exit Mach number range 0.77 - 0.94.

The calculated flow structure (isolines of constant Mach number) and the surface isentropic Mach number distributions for Cascade 1 and Cascade 4 are presented in Fig.7. This data, which corresponds to the same exit Mach number ($M_{2is} = 0.88$), illustrates the change of flow structure with variation of unguided turning angle. The Mach distributions in Fig.7-a and 7-b show that with increase of $\delta$ the throat shock strength is significantly reduced, which is also seen in Fig.7-c and 7-d. As the result the thickness of boundary layer on the suction surface is reduced as well. In Cascade 1 on the SS big flow acceleration is closed by strength shock. After this shock the flow starts acceleration again up to TE where the flow goes on acceleration around thick TE. Thus the flow leaves profile with a high velocity and this fact leads to decrease of base pressure up to $C_p = -0.4$ and the increase of profile losses up to loss $= 0.072$ in Cascade 1. The other flow pattern is observed in Cascade 4. On the SS after weak second shock the flow goes on the deceleration up to TE and pressure downstream of the TE becomes higher than averaged static pressure downstream of Cascade 4. Therefore base pressure coefficient becomes $C_p > 0$ and loss has minimum at $M_{2is} = 0.9$. Moreover in Cascade 1 the interaction of the shock on the PS with wake leads to much thicker wake than in Cascade 4 (this feature can be seen in Fig.7-a and 7-b). The base pressure in Cascade 4 increases due to delay of separation in shock free flow. At the same exit Mach number similar changes in flow structure were observed also in medium range of unguided turning angle (Cascades 2 & 3), although they are less pronounced. Similar variations of flow structure are also observed at the lower transonic Mach number ($M_{2is} = 0.8$). At the supersonic exit Mach number the influence of unguided turning angle is reduced, and flow structures in all variants are similar to Cascade 1.
TRAILING EDGE INJECTION

To evaluate the effect of injection the delta between losses with and without injection was calculated. The results of this evaluation for investigated Cascade 4 are presented in Fig. 8.

In Cascade 4 the injection lead to additional losses at supersonic exit Mach number, and at transonic $M_{2s}=1.05$ injection lead to the small reduction of loss. The latter can be explained by the additional impulse,
which injection of air with high velocity introduces in the main flow. At supersonic condition the same amount of injected air has lower impulse than main flow, and therefore creates losses.

Thus one can conclude that for cascades with moderate unguided turning angle ($\delta <16^\circ$) and TE thickness in range $d_2/a_2 <0.2$ the injection through the TE slot parallel to the main flow leads to neutral loss impact and even loss reduction in subsonic range and loss increase in the supersonic range of exit Mach numbers.

All calculations were performed for the isothermal conditions, where boundary conditions correspond to the experimental conditions.

The blade surface in these cases is assumed adiabatic without heat transfer. The pressure drop over the blade is maintained, and the mass flow at the outlet is increased according to the amount of injected air. The gas properties of injected air are the same as in the main flow.

The simulation of flow with trailing edge injection was performed in case $M_{2is}=1.43$. The results of additional loss calculations in comparison with measurements are presented in Fig. 9 (the measured data is the same as shown in Fig. 8). In the range of moderate rate of injection the agreement is quite well. Some deviation is observed at the low level of injection. This deviation partially might be related to difference between 2D model and injection conditions in the test rig, which are 3D (i.e. different at midspan and near the endwalls). The measurements with low injection rate are also not detailed enough. Nevertheless the calculations capture the trend in loss variation over the investigated range of injection rate.

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**Fig. 9. Losses caused by injection in Cascade 4 in isothermal conditions**

($M_{2is}=1.43$, measurements and calculations correspond to axial position $z = 0.65a_2$)

**Fig. 10. Velocity distribution behind Cascade 4 in isothermal conditions**

($M_{2is}=1.43$, $\bar{m} =1.94\%$, axial position $z = 0.65a_2$).
The velocity distribution presented in Fig.10 shows that significant contribution of injection to loss is related to the wake, which is in case with injection wider and with lower velocity. However, there is also another effect of injection related to change of shock position with some contribution to the loss.

CONCLUDING REMARKS

Experimental and numerical methods were used to investigate the effect of the unguided turning angle and trailing edge injection on the profile loss of transonic high pressure cooled turbine blades. Four cascades with the same main geometric parameters but different unguided turning angles were studied. Numerical data was validated on the basis of test data for these cascades. In general, a good agreement between the test and the computed data was observed. The investigations showed that the unguided turning angle has a significant effect on the base pressure coefficient and profile losses over the transonic operating range.

The increase of the unguided turning angle results in the increase of the base pressure coefficient from large negative values to positive values, and decrease of the profile losses. For example, in variant with relatively high unguided turning angle $\delta = 15.9^\circ$ the profile loss goes down by about 4% compare to the variant with low angle $\delta = 2.3^\circ$ at the exit Mach number $M_{in} = 0.88$.

These investigations show that for cascades with moderate unguided turning angle ($\delta < 16^\circ$) and trailing edge thickness in the range $d_2/a_2 < 0.2$ the injection through the trailing edge slot parallel to the main flow leads to neutral impact on loss and even loss reduction in subsonic range and loss increase in the supersonic range of exit Mach numbers. Cooling air injection leads to loss variation in the range of -2/+2%, which is significant and must be taken into account in the blade design process.

The numerical simulations of injection impact on losses are in reasonable agreement with measurements, which means that numerical tool can be used for this purpose in design practice.

Thus this study has demonstrated how to achieve a reasonable level of losses in transonic turbine blades with a significant trailing edge thickness. The options considered provide more degrees of freedom for blades designer, reduce restrictions on tolerances, improve manufacturability and reduce costs.

REFERENCES