J.Woisetschläger, H.Jericha, W.Sanz, H.P.Pirker, A.Seyr, T.Ruckenbauer (1997)
Experimental Investigation of Transonic Jet Film Cooling in a Linear Cascade, Proc. 2nd European Conference on Turbomachinery - Fluid Dynamics and Thermodynamics, Antwerpen, pp 447-451

EXPERIMENTAL INVESTIGATION OF TRANSONIC WALL-JET FILM COOLING IN A LINEAR CASCADE

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ABSTRACT

The evolution of increasing turbine inlet temperature has led to the necessity of full-coverage film cooling for the first turbine vane and blade. A new approach using high pressure steam and high speed wall jets for blade cooling has been proposed by the authors. In this paper a first experimental analysis of the aerodynamic behaviour of these surface coolant jets in a linear cascade is presented. Special emphasis was put on the investigation of the coolant flow around the leading edge. Pressure ratios, Reynolds numbers, blowing ratios and surface Mach number changes are given together with a Schlieren visualisation of the flow.

NOMENCLATURE

BR	blowing ratio = $(\rho v)_c / (\rho v)_m$
L	chord length [m]
M	Mach number
$\dot{m}$	mass flow [kg/s]
P	total pressure [Pa]
p	static pressure [Pa]
Re	Reynolds number with respect to chord length
Re_s	Reynolds number with respect to cooling slit height
ρ	density [kg/m^3]
T	Temperature [°C]
Tu	Turbulence intensity [%]
v	magnitude of velocity vector

Subscripts

1,2	inlet, outlet condition
$()_c$	coolant flow
$()_m$	main flow
$()_{tot}$	stagnation condition

INTRODUCTION

The objective of this investigation was to test underexpanded jets for a novel turbine blade film cooling system in a linear cascade, especially their aerodynamic behaviour around the leading edge. Since these underexpanded jets have a strong tendency to bend towards the surface a higher cooling efficiency is expected due to the improved cooling film attachment.

The effect of air flows bending around curved surfaces is generally connected with the name "Coanda-effect" and originally describes subsonic flows attaching to the surface by its turbulent structure (e.g. Fernholz and Wille, 1971; Ameria and Dybbs, 1993).

Several experiments with underexpanded jets bending towards a convex surface have been performed and published (Gregory-Smith and Gilchrist, 1987; Gilchrist and Gregory-Smith, 1988; Gregory-Smith and Hawkins, 1991; Gregory-Smith and Senior, 1994). An underexpanded jet means a jet from a choked convergent nozzle where the flow subsequently expands supersonically because the external pressure is below the critical value.

In many of these experiments, a two-dimensional underexpanded jet has been blown over a convex surface and studied experimentally. These curved supersonic jet have a complex shock cell structure with compression and expansion waves in the underexpanded core of the jet, with an outer free shear layer of the jet and a boundary layer towards the surface. This complex structure in the core of the jet is caused by the Mach waves from the initial Prandtl-Meyer expansion which are `reflected` at the surface thus forming the cell like structure all together bending the jet towards the surface.

Since these underexpanded jets bend towards convex surfaces it has been suggested (Jericha et al., 1995) that its use might be considered for high temperature turbine blade film cooling. As a result a first investigation was performed with an acrylic glass model of a cooling slot to determine a slot geometry well suited for the production of a uniform underexpanded surface jet (Woisetschläger et. al. 1995). This geometry (given in Fig.1. together with the result of an interferometric investigation of the density distribution of the flow) provided a uniform film over a large surface area without a jet `breakaway´ or `lift-off´ from the surface even at high total pressures inside the slot chamber (versus atmospheric pressure). The optimization was performed with and without main air flow and provided the basis for the cooling slot design inside the

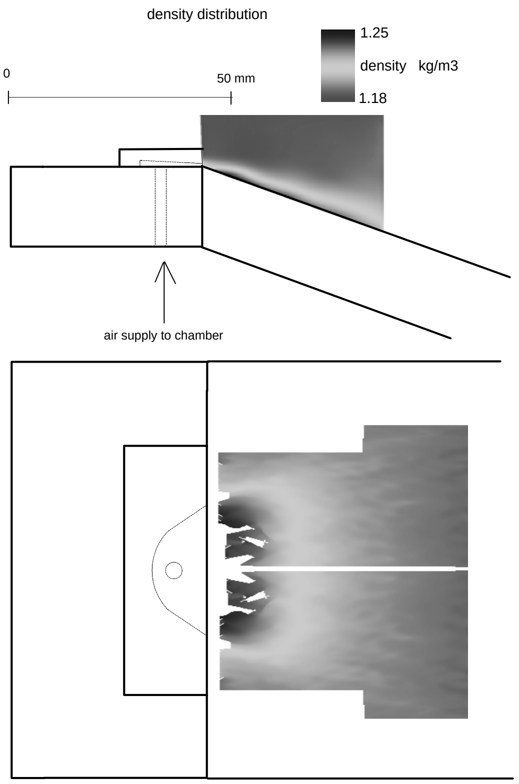

Fig.1 Test model for the investigation of the principle geometry - experimental result. The objective was to produce a uniform film in front of the exit slit using an underexpanded jet (see Woisetschläger et al., 1995)

linear blade cascade at the Institute for Thermal Turbomachinery and Machine Dynamics, TU Graz.

EXPERIMENTAL SETUP - TEST SECTION DESIGN

Test Facility

This experiment was performed in the Institute's cascade test rig being supplied from a continuously running compressor station located at the second basement of the institute. From the three compressors available (total engine power 3MW, volume flow 3000-50.000 m^3/h) the screw compressor (ATLAS COPCO, 2.6 kg/s maximum mass flow, 3.1 maximum pressure ratio at 400 kW engine power) had been used for the experiments presented here (For more details on the compressor station itself see Pirker et al.; 1995).

The test section design was based on a cycle proposed by Jericha et al. (1995) using high pressure steam as a cooling medium for the high temperature gas turbine in a combined cycle. Since it was not possible to run the linear cascade test rig with the same flow temperatures as a real turbine an aerodynamically similar test section was designed using CO_2 as cooling flow medium. For purposes of comparison the thermodynamical data are given in Table 1 together with the data for the steam cooled gas turbine and for a possible gas turbine cycle using air as cooling medium. In this case this air has to be provided by an additional compressor to ensure underexpanded cooling jets. For the CO_2 coolant the critical pressure ratio is 1.83, for air it is 1.89.

TABLE 1: Design data of the experiment, comparison of thermodynamical data

	gas turbine steam cooled	**gas turbine air cooled**	**wind tunnel**
medium main flow	fumes	fumes	air
medium coolant	steam	air	CO_2
temperature $T_{m,1}$ main flow	1500°C	1500°C	40°C
temperature $T_{c,1}$ coolant	310°C	650°C	20°C
pressure ratio $P_{c,1}/P_{m,1}$	2	2	2
density ratio at cooling slit* ρ_c/ρ_m	2.5	2.32	2.14
Re_S	39,500	27,000	15,900

*based on polytropic expansion of the cooling flow in the cooling slit exit nozzle

TABLE 2: Geometric data of the blade and cooling slits

blade chord length	143 mm	**cooling slit length**	23.5 mm
blade pitch	99.5 mm	**cooling slit height**	0.2 mm
blade height	100 mm	**wind tunnel cross section**	230 x 100 mm

Fig.2 shows the test section with the linear arrangement of four blades (not shown are the tail boards for adjustment of periodicity). The central blade channel was optically fully accessible through window sections on both sides. Pressure taps were placed on the suction as well as pressure side along the mid blade channel. Additionally pressure, total pressure and total temperature were measured before and after the blade cascade. A detailed view of the lower blade is given in Fig.3. Here the pressure taps and the cooling slits are numbered in the way used in this paper. The principle geometry of the exit slit and the cooling chamber were the same as in Fig.1, the actual length and width are given in Table 2 together with the other design data. Within the linear blade cascade the

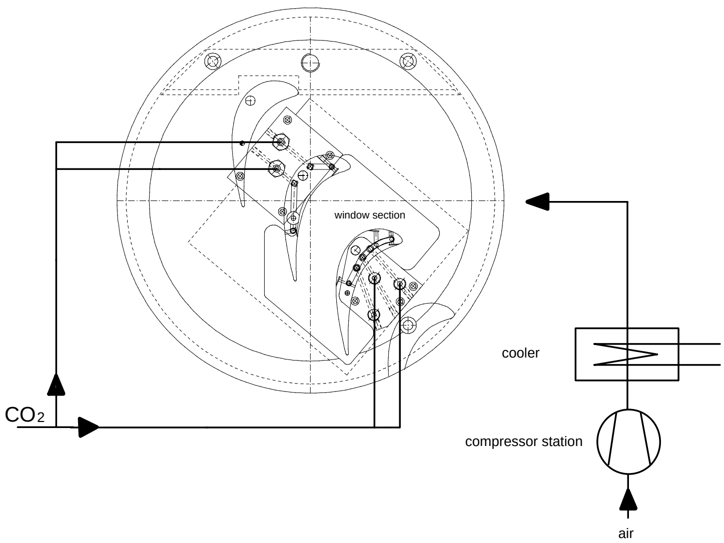

Fig. 2 Linear cascade - Test section

cooling slits were arranged in the cascades axis, i.e. at blade height 50 mm. The blade material was aluminium with the cooling slots manufactured separately to ensure appropriate chamber geometry. The shape of the turbine blade was similar to the one we currently design for our rotating test turbine, since in a next step a test of this cooling film in a rotating test rig is planned.

In this experimental setup (given in Fig.2) every cooling slot can be fed separately. For the one cooling slit used in this work a calibration for pressure losses inside the tubings had been performed, so that all pressures and temperatures (incoming cooling flow) given in this paper are cooling slot chamber data.

Measurement techniques

The total and static pressure of the incoming flow had been measured at several points upstream of the blade cascade at a distance of 150 - 200 mm at various spanwise locations as well as downstream the blade cascade. While the pressure distribution along the blade surface was measured in groups of ten, the incoming and outgoing flow condition was monitored continuously by a Scanivalve ZOC 14NP/16Px-50 with an electronic multiplex interface and a 486 PC using a National Instruments AT-MIO-16H-9 AD/DA converter board together with Lab View software.

Temperature was measured at several points upstream and downstream by K-type thermoelements together with an amplifier board and was also recorded by the PC based system.

The cooling mass flow was detected using a Fischer & Porter mass flow meter and the corresponding Fischer and Porter software for meter reading correction due to actual pressure and temperature of the CO_2. The main mass flow was measured using a mass flow nozzle in the main line

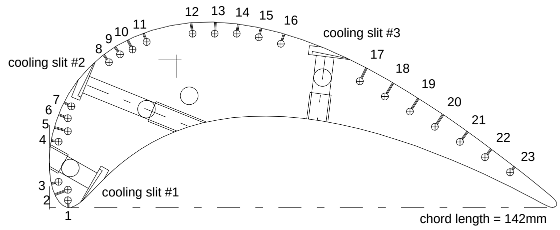

Fig.3 Lower blade with the position of the cooling slits and pressure taps

and with velocity and density measurements upstream of the cascade.

Beside pressure and temperature measurements, flow visualisation was performed by dark field transmission Schlieren technique using a circular Schlieren filter (see e.g. Woisetschläger et al., 1995). Due to the circular Schlieren filter density gradients in different directions within the flow field are visualized simultanously, no direction is missed. As light source a He-Ne laser as well as a white light source (halogen lamp) had been used.

For the velocity and turbulence data published in this paper a two channel Laser Doppler Velocimeter (LDV; DANTEC fiber flow, Burst Spectrum Analyser) has been used together with DEHS as seeding material. The mean droplet diameter of this material (produced by a PALLAS AGF 5.0 particle generator) was 0.7 μm.

RESULTS AND DISCUSSION

The objective of the measurements presented here was the investigation of the aerodynamic behaviour of underexpanded surface coolant jets in a linear cascade, especially the investigation of the coolant flow around the leading edge. For this reason Schlieren visualisation together with a recording of the surface pressure distribution has been performed.

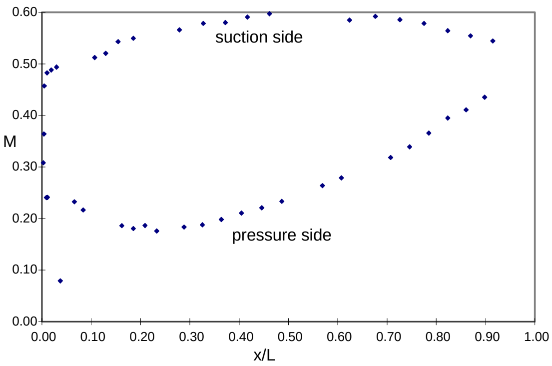

Fig.4 Isentropic Mach number distribution for exit Mach number M_2 = 0.55 and exit Reynolds number Re_2 = 1.4 .10^6 .

Fig.4 gives the isentropic Mach number distribution on the blade as measured with the pressure taps. The exit Mach number was M_2 = 0.55 and the exit Reynolds number Re_2 = 1.4.10^6. The inlet turbulence level recorded with the LDV system was Tu_1 = 3%, the main mass flow $\dot{m}_m$ = 0.99 kg/s. The isentropic Mach number indicates a strong flow acceleration at the beginning of the suction side followed by a slighter increase up to 2/3 of the chord length.

Three Schlieren recordings of the leading edge area are then compared in Fig.5. (The main flow direction is from the right to the left). The first (Fig.5a) gives the density gradient without any coolant flow, clearly showing the stagnation point area followed by the acceleration zone along the suction side (upper side) with a clearly observable density change. In this zone the density changes were showing up as light areas, the intensity is an approximate measure for the size of the change. In Fig.5b, an underexpanded jet left the cooling slit and bent towards the blade surface by the Mach waves forming the cell like density structure visible in this picture. This curved supersonic jet had a complex shock cell structure with compression and expansion waves in the underexpanded core of the jet. Even at high pressure ratios this inner structure kept the jet on the blade's leading edge.

The final picture (Fig.5c) gives an idea what happens to a standard cooling flow ejected in this area of the leading edge. Here the pressure in the cooling slot chamber was adjusted to a very small value so that the pressure ratio $P_{c,1}/P_{m,1}$ was under its critical value. In this case a low speed cooling jet was ejected and clearly lifted off the blade surface by the main flow.

A further indication of the strong binding between this type of cooling film and the surface can be seen from the pressure tap recordings. In Fig.6 the Mach number distribution with respect to the main flow's total pressure is given for three different total pressure ratios between incoming coolant and main flow $P_{c,1}/P_{m,1}$. In this graph the pressure taps are numbered according to the numbering in Fig. 3 above. The influence of the cooling

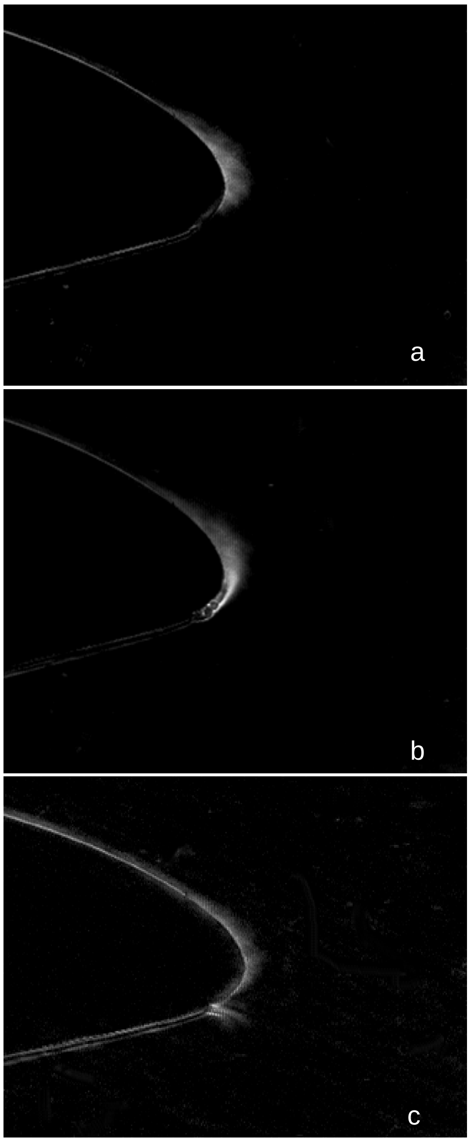

Fig.5 Schlieren visualisation of the leading edge flow. a) shows the main flow without cooling film, b) the main flow with the underexpanded cooling film at a pressure ratio of $P_{c,1}/P_{m,1}$= 2.5 and c) a low speed cooling film ($P_{c,1}/P_{m,1}$< 1.4) for purpose of comparison.

film is clearly seen for all pressure taps along the leading edge.

From this graph (Fig.6) one can also see the optimum range of pressure ratios. While the Mach number is steadily increasing in position tap 1 for all pressure ratios the change in position 2 between the higher pressure ratios $P_{c,1}/P_{m,1}$ = 3 and 4 is no longer significant. This is an indication of a low speed recirculation zone forming between jet and surface in the boundary layer. This recirculation will lead to a jet "breakaway" from the surface with too high chamber pressure (compare also

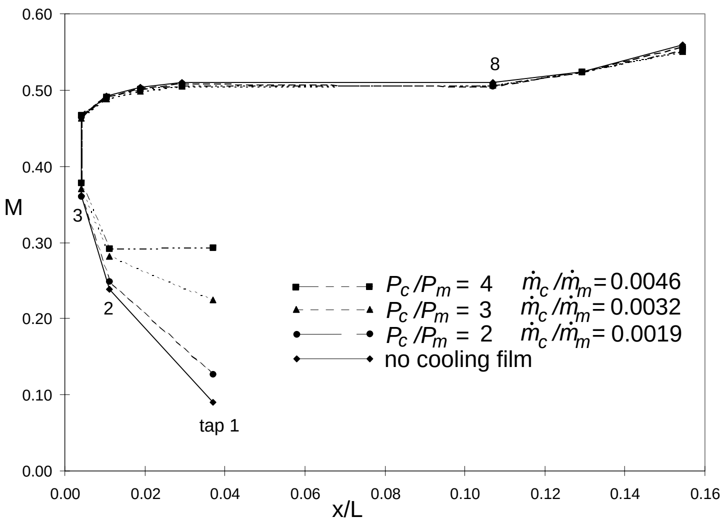

Fig.6 Mach number distribution around the leading edge with and without cooling flow at different pressure ratios. The numbers correspond to the pressure taps in Fig.3 above.

Gilchrist and Gregory-Smith, 1988 and Woisetschläger et al., 1995).

In this way both Schlieren pictures and pressure recordings show a strong tendency for the coolant film to stick towards the leading edge surface with an optimum total pressure ratios between incoming coolant and main flow $P_{c,1}/P_{m,1}$ = 2 to 3.

The mass flow ratios for the different pressure ratios is also given in Fig.6. and was between 0.2 and 0.3 % in the optimum pressure ratio range $P_{c,1}/P_{m,1}$ = 2 to 3, respectively. In the experiment presented here the cooling film covered about one quarter of the full blade height. With this type of cooling the mass flow rate strongly depends on the size of the cooling slit due to the supercritical conditions in this type of cooling. This is also the reason for a slightly higher blowing ratio than in standard subsonic film cooling . (The blowing rate BR in the optimum pressure ratio range $P_{c,1}/P_{m,1}$ = 2 to 3 was *BR* = 3.3 to 5.6, respectively.)

All together these first results of the aerodynamical behaviour of these coolant films show a type of film with a strong adhesion to the surface. Beside the supercritical condition the slit design is important since it influences mass flow and blowing ratio significantly.

SUMMARY

Underexpanded cooling films have a strong tendency to bend towards the surface which makes them especially interesting for film cooling studies around the leading edge area. In this work a pressure ratio between $P_{m,1}/P_{c,1}$ = 2 to 3 turned out to provide an optimum cooling film around the leading edge area.

Ongoing investigations will deal with velocity, turbulence and density changes due to underexpanded surface coolant films with and without upstream flow distortions.

ACKNOWLEDGEMENT

The authors gratefully acknowledge the support by the Austrian Science Foundation (FWF) supporting this research as well as ongoing research in efficiency improvement of thermal energy production.

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