MEASUREMENTS ON SUPERSONIC TURBINE CASCADES - METHODICAL APPROACH

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ABSTRACT

The paper deals with methodical approach to investigation of turbine profile blade cascades at supersonic inlet and outlet flow conditions. It is difficult to model such conditions experimentally mainly due to problems with starting supersonic flow, achieving periodic inlet and outlet flow field, and finding appropriate wind tunnel setting. The paper revives supersonic flow theory to explain unique incidence rule, to show basic concept of shock reflection cancellation devices, and to describe wind tunnel settings. Modelling supersonic inlet and outlet conditions into a blade cascade is presented and discussed. Discussions are backed by experiments performed in a high-speed wind tunnel for blade cascades which provided data for further analyses. Experimental facility is described and selected results are presented.

1 INTRODUCTION

The present development in turbomachinery opens complex problems namely for high-speed aerodynamic research. Transonic and supersonic flows occur in highly-loaded stages of machines. Aerodynamic modelling in wind tunnels has to answer questions on reliable performance of transonic blading, on keeping of design parameters, on prosperousness of a particular flow part design. There are limits in physical understanding of effects of complex phenomena occurring in high-speed flows. Therefore, need of information and data supporting development of design procedures is urgent.

One of the topical problems is to ensure supersonic and possibly periodic flow conditions at inlet of a profile blade cascade. In the past, this problem was investigated at design and operation of transonic compressor stages. See, for example [1], [2], [3], [4], and others. Data on transonic compressor blade cascades has also been obtained and compared to CFD, e.g. [5], [6], [7]. However, the new task concerns tips of turbine blades of the last stages of large output steam turbines. Relatively narrow space for aerodynamic design of these long blades often results in blades with tips operating at supersonic regimes. This brings number of problems. Main of these are additional sources of loss and rather thin border between sections operating under turbine and compressor regimes one adding to performance and the other deteriorating it. Therefore, aerodynamic cascade data on turbine blading for a range of supersonic inlet and outlet conditions are needed. However, ensuring specified supersonic conditions in a high-speed wind tunnel for blade cascade tests is not an easy task and thus it deserves attention.

Papers on tests of supersonic turbine blade tip cascades are not so numerous. Different experimental setups used for aerodynamic tests of supersonic turbine blade tip cascades can be found in [8], [9], [10]. Authors in [8] tested aerodynamic performance of a supersonic turbine cascade in a high speed wind tunnel using dry steam. Acceleration to supersonic speeds at inlet with parallel upper and lower walls was achieved by converging insert mounted on the lower wall of the inlet channel (Fig. 1). Wall of this insert was perforated and suction was applied to i) accelerate flow to low supersonic velocities and to ii) avoid reflections of inlet shock waves. No devices were used to control outlet flow quality. Nevertheless, no signs of shocks reflected from free jet boundary were apparent in the presented area of investigation at two investigated exit Mach numbers.

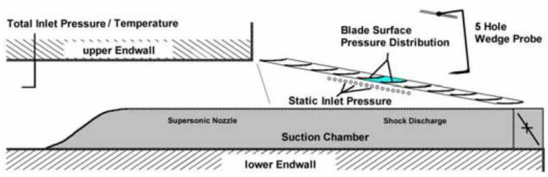

Fig. 1: Test section used for turbine blade tip aerodynamic tests by Parvizinia et al. [8]

In [9], current authors tested tip section featuring supersonic inlet velocity at negative off-design incidences. In these measurements, quality of the flow field was assessed based on optical measurements (interferograms and schlieren pictures). The used test section was equipped with ventilated upper wall at inlet. Apart from that, no special devices improving supersonic inlet or outlet flow were used. Therefore, outlet flow field was at certain exit Mach numbers influenced by waves reflected from the free jet boundary. Also at regimes with the highest inlet velocities (about M_1 = 1.2), reflected compression waves appeared at the inlet. One of the latest tests of turbine blade tip cascades was published in [10]. Inlet flow was accelerated using convergent divergent nozzle (Fig. 2). Back pressure was set by adjusting angle of the upper tailboard. This special type of shaped perforated tailboard was also used for cancellation of exit shock wave reflections. Wavy shape of this tailboard's wall was designed to follow a wake line predicted by CFD calculations of infinite cascade.

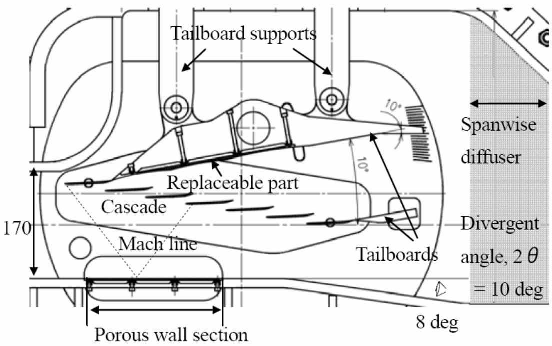

Fig. 2: Test section used for turbine blade tip aerodynamic tests by Shibata et al. [10]

Problems of wave reflection cancellation using perforated or slotted tailboards were investigated and discussed, e.g. in [11], [12], [13], [14]. All these publications confirm functionality of slotted or perforated tailboards in canceling reflections or reducing intensity of reflected shock waves in relatively wide range of backpressures.

This paper brings together basic information and methodology necessary for successful testing of supersonic turbine blade cascades in high speed wind tunnels. Aim is also to share experience of the authors with tests of supersonic turbine blade tip cascades.

NOMENCLATURE

A^*	throat area, m^2
f	ratio of lower wall suction to inlet mass flow
h	tunnel throat opening
H	tunnel inlet channel height
M	Mach number, -
p_0	total pressure, Pa
p	static pressure, Pa
q	dynamic pressure
r	gas constant, J.kg^{-1}·K^{-1}
T	temperature, K
v	velocity, m.s^{-1}
ε	perforation ratio
θ	flow angle
ι	incidence angle, °
κ	ratio of specific heats
μ	Mach angle
ρ	density

Subscripts

1	inlet
2	outlet
is	isentropic
N	nozzle

2 UNIQUE INCIDENCE RULE

Theoretical derivation of unique incidence is based on balance of mass flow and on mathematical description of compressible fluid flow by second order partial differential equation. The solution of this equation is based on method of characteristics. The neutral characteristic for inlet supersonic Mach number (exactly for the inlet velocity $v_1 = M_1 \cdot \sqrt{\kappa r T}$) and inlet flow angle α_1 is shown in the (Fig. 3). For simplicity, blade cascade with sharp leading edge and thus with no subsonic region is assumed.

The Mach angle (angle between this characteristic and the inlet velocity vector) is given by $\mu_1 = arcsin\frac{1}{M_1}$. The mass flux through the part of neutral characteristic having the length of $t{\cdot}\cos\alpha_1$ has to be equal to the mass flux in the section $\overline{KL}$ of the characteristic line, which goes through the leading edge in the case of simple expansion occurring on the profile surface between points *N* and *K*.

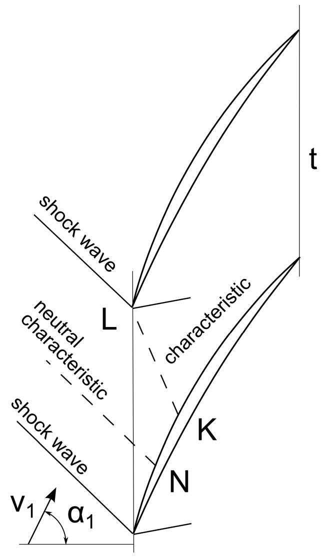

Fig. 3: Cascade supersonic inlet

It is possible to express the balance of mass flow as

$$f_N\left(M_1, t\cos\alpha_1\right) = f_K\left(M_K, \overline{KL}\sin\mu_K\right), \qquad (2.1)$$

where M_K is Mach number in the point K. M_K is solved applying invariant condition

$$\alpha_1 + \omega\left(M_1\right) = \alpha_K + \omega\left(M_K\right) \qquad (2.2)$$

together with the geometrical configuration of profile and its stagger angle in the cascade. At the same time, the length of the $\overline{KL}$ is determined. The expression $\omega(M)$ is the Prandtl-Meyer function, which is derived and described in text books of gas dynamics.

Substituting (2.2) to (2.1) gives definite relation between inlet Mach number and inlet flow.

$$\alpha_1 = \arccos\left[\frac{\left(1+\frac{\kappa-1}{2}M_K^2\right)^{-\frac{\kappa+1}{2(\kappa-1)}}}{M_1\left(1+\frac{\kappa-1}{2}M_1^2\right)^{-\frac{\kappa-1}{2(\kappa-1)}}}\,\frac{\overline{KL}}{t}\right] \qquad (2.3)$$

The unique incidence rule claims, it is impossible to choose any inlet condition. Thus, the inlet angle is unambiguous function of inlet Mach number. Inlet angle far upstream of an infinite cascade corresponds to direction tangent to profile suction surface (compressor cascades) or pressure surface (turbine cascades) at point N where the neutral characteristic emanates form the surface. This follows form the fact, that all expansion characteristics coming form the surface before and after the neutral characteristic weakens the front shock wave so that it soon becomes Mach wave identical to the neutral characteristic.

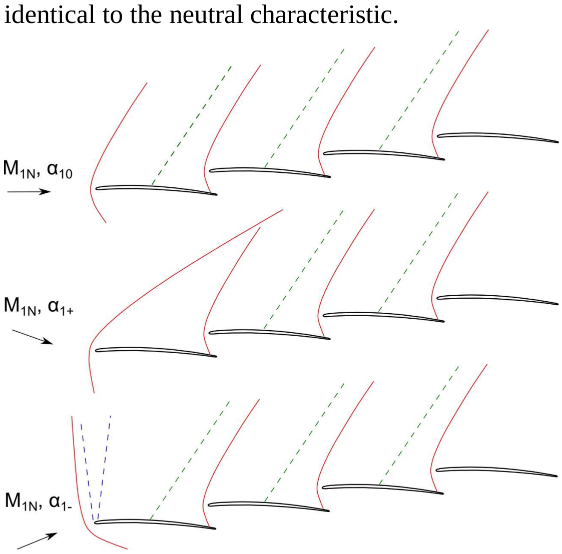

Fig. 4: Self adjustment of unique incidence conditions at inlet when incidence angle of the flow coming form the nozzle is equal, higher or lower than the unique value.

When a finite blade cascade is placed in the test section of a wind tunnel and a supersonic flow with subsonic axial component is incident on the blade cascade, conditions satisfying unique incidence rule are set by either a shock wave or expansion formed in front of the first upstream blade. See Fig. 4 where differences in inlet angles are purposely exaggerated.

Fortunately, sensitivity of the inlet angle to the inlet Mach number is in case of little curved tip section profiles very low (Fig. 5). Therefore, during experiments, the blade cascade is set in the position corresponding to the design inlet angle and all changes of the inlet flow field are realized through changes of inlet Mach number without great differences in shape or intensity of the first shock wave and those following.

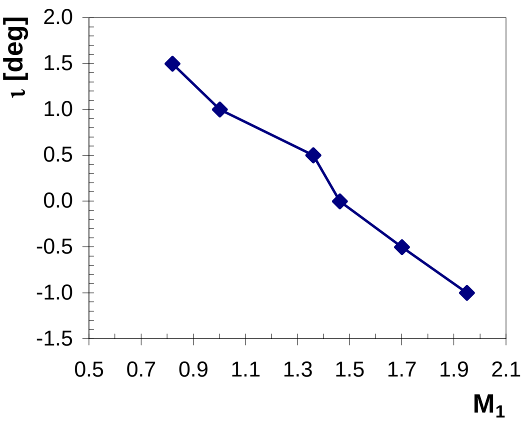

Fig. 5: Dependence of inlet angle on inlet Mach number for typical tip section of a long rotor blade (Fig. 4). Result of CFD.

3 CANCELLATION OF SHOCK REFLECTIONS

In supersonic turbine cascades, devices cancelling shock wave reflections off tunnel walls need to be applied in both inlet and outlet flow in order to keep the flow as periodic as possible. Following is theoretical derivation of suitable perforation ratio for cancelling of incident shock waves. It is based on linearized theory of supersonic flow. Scheme of the situation is shown in Fig. 6. Flow is considered to be isentropic and the wall is infinitely thin.

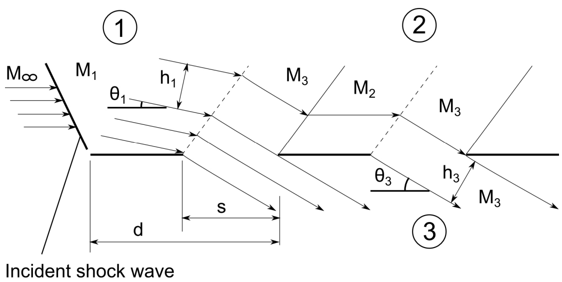

Fig. 6: Schematic of flow past perforated wall with incident shock wave [11].

Continuity equation

$$h_1(\rho_1 v_1) = h_3(\rho_3 v_3) \qquad (3.1)$$

and supersonic wave relationship

$$\rho_3 v_3 = \rho_1 v_1\left[1-\sqrt{(M_1^2-1)}(\theta_3-\theta_1)\right] \qquad (3.2)$$

yields

$$h_1 = h_3\left[1-\sqrt{(M_1^2-1)}(\theta_3-\theta_1)\right]. \qquad (3.3)$$

When we assume that

$$h_1 = d\sin\theta_1 \cong d\theta_1 \qquad (3.4)$$

and

$$h_3 = s\sin\theta_3 \cong s\theta_3 = Rd\theta_3 \qquad (3.5)$$

we arrive at relation

$$\theta_1 = R\theta_3\left[1-\sqrt{(M_1^2-1)}(\theta_3-\theta_1)\right]. \qquad (3.6)$$

Solving this for $\theta_3-\theta_1$ by expansion in a series and further linearization we get

$$\theta_3-\theta_1 = \left(\frac{1}{\varepsilon}-1\right)\theta_1 \qquad (3.7)$$

Assuming linearized supersonic flow theory and flow with waves of one kind only, pressure change expressed as a simple function of direction change. Written for expansion between regions 1 and 3 this gives

$$\frac{p_3-p_1}{q_1} = \frac{-2}{\sqrt{(M_1^2-1)}}\left(\frac{1}{\varepsilon}-1\right)\theta_1 = K\theta_1. \qquad (3.8)$$

and for the incident shock wave

$$\frac{p_1-p_\infty}{q_1} = \frac{2}{\sqrt{(M_1^2-1)}}\theta_1. \qquad (3.9)$$

To prevent the incident shock wave from reflection, pressure p_3 below the perforated wall has to be equal to the static pressure of the undisturbed flow p_∞ which gives relation

$$\frac{p_1-p_\infty}{q_1}+\frac{p_3-p_1}{q_1} = \frac{p_3-p_\infty}{q_1} = 0. \qquad (3.10)$$

Substituting equations (3.8) and (3.9) to(3.10) yields

$$\frac{2}{\sqrt{(M_1^2-1)}}\theta_1 - \frac{2}{\sqrt{(M_1^2-1)}}\left(\frac{1}{\varepsilon}-1\right)\theta_1 = 0. \qquad (3.11)$$

Thus the sought perforation ratio is

$$\varepsilon = 0.5.$$

This theory is not valid for M_1 close to 1 since K in equation (3.8) goes to infinity. Experiments [1] have proved that in range of Mach numbers between 1.5 and 2.0, walls with perforation ratio 0.5 performs well when influence of boundary layer is minimized.

4 TEST SECTION FOR SUPERSONIC TURBINE BLADE TIP CASCADES

The authors use for their experiments suction type high-speed wind tunnel of the institute of Thermomechanics CAS, v.v.i., scheme of which is shown in Figure 7. This wind tunnel is equipped with variable geometry inlet nozzle (Fig. 7 #4). Shape of the inlet nozzle can be continuously changed and in case of typical turbine blade tip cascades inlet Mach numbers up to M_1 = 2.0 can be reached. At the exit of a blade cascade, working medium (air) is discharged to a spacious settling chamber (Fig. 7 #7). Backpressure is controlled by variable throat control nozzle (diffuser) (Fig. 7 #8).

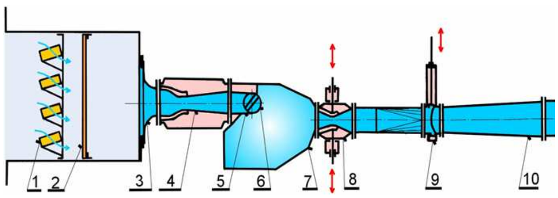

Fig. 7: Scheme of the suction type high-speed wind tunnel. 1 silicagel dryer, 2 filters, 3 entrance nozzle, 4 inlet nozzle, 5 transient insert, 6 rotable test section, 7 settling chamber, 8 control nozzle, 9 quick-acting valve, 10 diffuser, 11 main duct

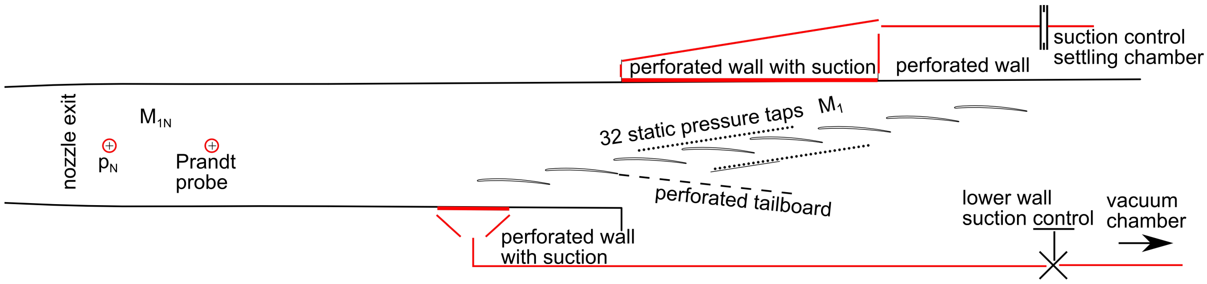

Fig. 8: Test section for measurements on supersonic turbine blade tip cascades

Test section of the wind tunnel has been modified in order to improve modeling of supersonic flow past cascades. (Fig. 8) Both upper and lower wall of the inlet channel are equipped with perforated inserts with perforation ratio $\varepsilon = 0.5$. Both inserts are connected to low pressure parts of the wind tunnel which drive the suction. The upper wall insert is connected to the settling chamber and the lower wall insert to the vacuum chamber. Thus, portion of mass flow from the inlet channel is bypassed to either settling chamber or vacuum chamber. Suction mass flow is controlled by valves. Positive effects of upper wall suction are shown in Figs. 9 and 10. Periodicity of the outlet flow field is improved by perforated tailboard with perforation ratio $\varepsilon = 0.5$. This value of the ratio should ensure functionality of the tailboard in a wide range of regimes. Influence of perforated tailboard can be seen in Figs. 11 and 12.

Nozzle Mach number M_{1N} and cascade inlet Mach number M_1 are evaluated based on total pressure measured by the Prandtl probe and static pressures from taps p_N and 32 static pressure taps before the blade cascade, respectively.

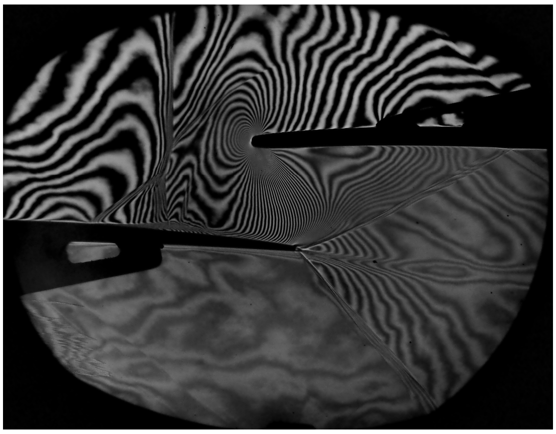

Fig. 9: Interferogram taken at M_{2is} = 1.674 and M_1 = 1.066. Upper wall suction off.

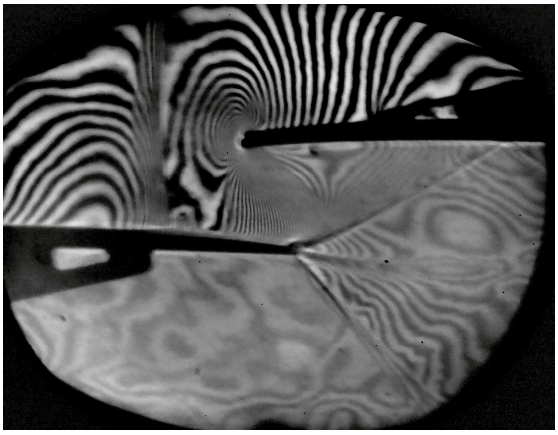

Fig. 10: Interferogram taken at M_{2is} = 1.689 and M = 1.077. Upper wall suction on.

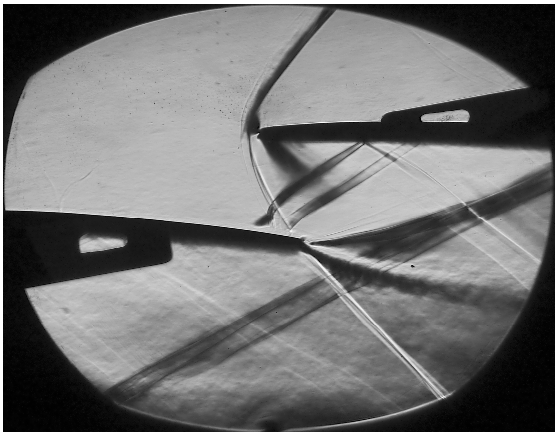

Fig. 11: Schlieren picture taken at M_{2is} = 1.838 and M_1 = 1.29 showing shock wave reflected from the free jet boundary when no perforated tailboard is applied.

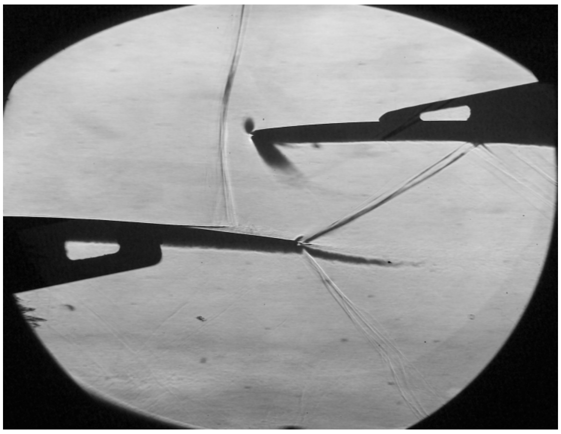

Fig. 12: Schlieren picture taken at M_{2is} = 1.798 and M_1 = 1.257. Perforated tailboard applied and set to an angle α_{pt} = 15.75°

The tailboard not only prevents exit shock waves from reflecting back to the exit flow field, it also prevents expansion taking place at the lower wall block edge from influencing the exit flow field. Due to very limited access to these parts of the wind tunnel and thus limited possibilities of investigating the flow field, CFD has been used [15] to reveal configuration of the flow field there. Origin of both undesirable features is apparent from CFD results in Figure 13.

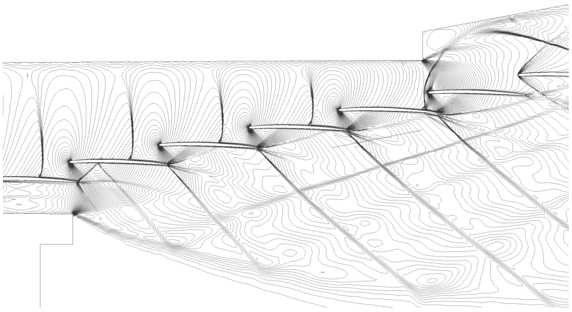

Fig. 13: Numerical simulation of the flow past turbine blade tip cascade model in the wind tunnel with test section consisting of no devices improving supersonic inlet and outlet flow. Reprinted from [15]

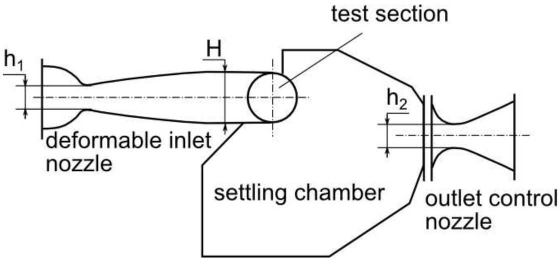

Fig. 14: Scheme presenting the wind tunnel as a channel with two aerodynamic throats.

5 SETTING THE WIND TUNNEL FLOW

Supersonic regimes of turbine blade tip cascades are defined by inlet Mach number M_1 (determines also inlet angle) and isentropic exit

Mach number M_{2is}. Since from the design point of view, inlet angle is more convenient parameter, range of investigated inlet Mach numbers with respect to inlet flow angle has to be determined by CFD first. Any combination of the two may be set within operational range of the wind tunnel, that is for supersonic values $M_1 \in \langle 1.0; 2.0\rangle$ and $M_{2is} \in \langle 1.0; 2.0\rangle$. The desired regime is reached by correct setting of throat openings h_1 and h_2 in the inlet nozzle and outlet control nozzle, respectively (Fig. 14). For the desired nozzle Mach number M_{1N}, the throat h_1 is given by H and mass flow sucked out through the lower wall perforated insert. Theoretically, set up of the wind tunnel at cascade supersonic regimes corresponds to the ideal case of flow past channel with two aerodynamic throats. In which case, the second throat opening h_2 and corresponding throat area is given by Donaldson law

$$A_2^* = A_1^* \frac{p_{01}}{p_{02}}. \tag{5.1}$$

This relation is valid in the ideal case when there is standing normal shock wave in the second throat and the wind tunnel operates at the highest efficiency. However, since part of the mass flow is bypassed through the inlet lower wall perforation behind the second throat into vacuum chamber, this relation must be corrected. If we express lower wall suction mass flow as a fraction f of the mass flow passing the first throat we get

$$A_2^* = A_1^* \frac{p_{01}(1-f)}{p_{02}}. \tag{5.2}$$

If this held true in reality (with all assumptions fulfilled) and providing the total pressure loss in between both tunnel throats, and nozzle Mach number M_{1N} required for particular cascade inlet Mach number M_1 are known prior to measurements, opening of the second throat h_2 can be determined. However, experience shows that the problem is more complicated and setting the wind tunnel is not so simple. Unless the whole configuration is not modeled by CFD first, neither total pressure loss nor required nozzle Mach number M_{1N} is known. Results of measurements show (Fig. 15), that in reality, strict conditions for validity of equation (5.2) are not fulfilled. Fig. 15 shows that ratio of throat areas is much higher than ratio of total pressures. If (5.2) was to be valid than the total pressure loss taking place between cascade exit and the second throat would have to be very high (p_{02} was evaluated from traversing) or the second throat area A_2^* would have to be much smaller. This suggests either that the tunnel is not started, i.e. Mach number in the second throat is not unity (although investigated portion of the cascade exit flow field exhibits parameters of the required regime) or that the second throat effective opening is smaller due to flow separation at the throat entrance.

Figure 16 shows dependencies of measured (full symbols) and theoretical (empty symbols) values of nozzle inlet Mach number required for achieving desired cascade inlet Mach number (solid lines). It is obvious, that not even theoretical determination of inlet nozzle Mach number M_{1N} based on ratio h_1/H is not correct. This is due to development of boundary layer in the inlet channel which is relatively long and also due to suction through the perforated inserts.

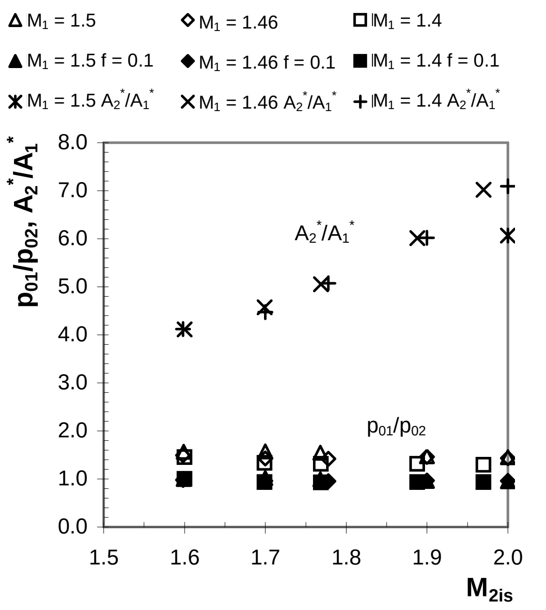

Fig. 15: Dependence of ratios A_2^*/A_1^* (according to (5.1) and (5.2)) and p_{01}/p_{02} on isentropic exit Mach number M_{2is}.

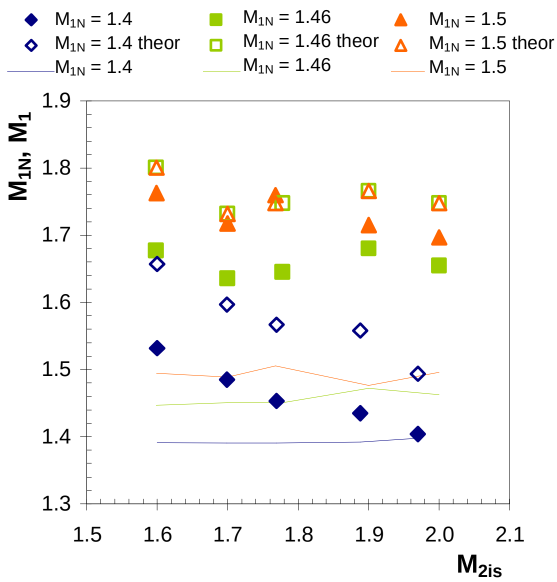

Fig. 16: Dependencies of nozzle inlet Mach numbers (measured and theoretical) and cascade inlet Mach numbers on isentropic exit Mach number

Despite these discrepancies between theory and reality, used wind tunnel characteristics (with respect to desired cascade inlet and exit Mach numbers) still exhibit clearly defined dependencies of throat opening ratio and relative second throat opening upon isentropic exit Mach number (Fig. 17). h_{2min} in Fig. 17 is the second throat opening at the lowest investigated M_{2is} (~ 1.6 in Fig. 17) for particular M_1 (particular curve). It can be seen, that curves are only little dependent on cascade inlet Mach number. Based on these dependencies it is possible to find correct wind tunnel setting (h_1, h_2) for the desired regime (M_1, M_{2is}). Once a regime on a curve of particular M_1 is found (based on experience), other regimes with the same M_1 are easily determined.

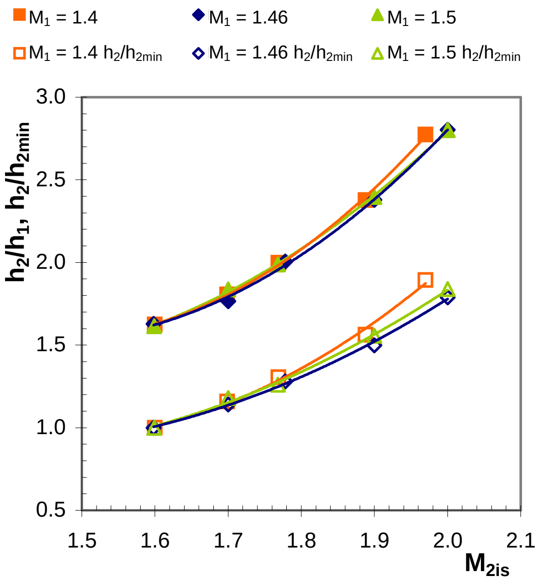

Fig. 17: Wind tunnel characteristics - dependencies of the wind tunnel setting upon isentropic exit Mach numbers M_{2is} for several cascade inlet Mach numbers M_1

6 CONCLUSIONS

Experimental investigation of flow past supersonic turbine blade tip cascades poses complex problem where different interesting fluid flow phenomena take place. Nature of the flow past supersonic cascades requires additional knowledge in order to perform successful aerodynamic experiments as compared to subsonic cases. Main features are unique incidence rule and presence of shock waves in inlet and outlet flow field. The first introduces complications to the wind tunnel settings. The latter may significantly disturb flow field periodicity and therefore devices for shock wave reflection cancellation have to be used.

The paper revives supersonic flow theory to explain unique incidence rule, to show basic concept of shock reflection cancellation devices, and to describe wind tunnel settings. Experiments show that although flow past the presented wind tunnel does not fulfil assumptions of the basic theory (flow past channel with two throats), well defined character of the wind tunnel characteristics makes it possible to predict correct wind tunnel settings.

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