ON A SLIDING DATA TECHNIQUE OF TRAVERSING PROBE MEASUREMENTS

Milan Matejka
Czech Technical University in Prague, Faculty of Mechanical Engineering

Pavel Safarik
Czech Technical University in Prague, Faculty of Mechanical Engineering

Martin Luxa
Academy of Science of the Czech Republic, Institute of Thermomechanics, v.v.i.

David Simurda
Academy of Science of the Czech Republic, Institute of Thermomechanics, v.v.i.

ABSTRACT

This paper deals with application of the data reduction method at a sliding data technique. It offers to study the influence of non-periodic processes in the flow field of blade cascade namely on the value of loss coefficients. The operational regime of the blade cascade is characterized by transonic and supersonic exit velocity. The experimental data analysis concerns pneumatic measurements with aim to calculate the total pressure and the kinetic energy loss coefficients, and to evaluate these coefficients with respect to the aperiodicity of the flow field. The flow structure is also studied on the basis of results obtained by means of optical methods, such as schlieren technique and interferometry.

INTRODUCTION

Research and design in the field of high-speed aerodynamics are uneasy due to the complexity of both experimental work and numerical solutions. It is difficult to obtain reliable experimental data, but also to evaluate it properly. It is therefore necessary to seek further evaluation procedures and detailed analysis of the measured data and used methodology. An attempt to give new perspectives on the evaluation of the data from the experimental measurements is given in this paper.

In an axial flow turbomachine a simplified approach is accepted by introducing the model of the plane (or straight) cascade, which is an unfolded cylindrical section through a blade row at certain radius. According to this definition the plane cascade is represented by a two-dimensional, periodic flow field with a linear arrangement of an unlimited number of blades. The correlations between the plane cascade parameters (profile shape, pitch-chord ratio, stagger angle), the upstream and downstream values of certain quantities (velocities, flow angles) and the losses are the basic data for the design and assessment of operation of turbomachine.

For experimental investigation in a wind tunnel the blade cascade is constructed as a limited number of straight blades. It is usual to install the blade cascade between a pair of sidewalls. The flow is guided up to the blade cascade by block walls to define inlet flow angle. The flow field downstream of the blade cascade is usually free and the local flow parameters must be recorded by traversing a probe parallel to the plane of trailing edges of profiles of the cascade. Consequently the flow data are function of the pitchwise coordinate.

The substitution of the measured data by derived homogeneous values is called a reduction of the measured data. Reduced data are a basis for the determination of blade forces, loss coefficients, entropy increase, etc. The algorithm for the reduction of the measured data has been developed. The data reduction method belongs to the software feature at aerodynamic research facilities.

The first realized data reduction method for blade cascade tests in compressible fluid flow [1] was limited to the subsonic region of velocities. The equations for solution of reduced data of compressible fluid flow in full range of velocities were derived in [2]. The Symposia on Measuring Techniques in Turbomachinery are the forums where achievements in development of data reduction method are to be presented. The extension of data reduction system to flow fields with non-constant total temperature was shown in [3] and further extension to flow fields with injection of another gas was derived in [4]. An attempt to derive basic equations for solution of reduced data of three-dimensional compressible fluid flow field was performed in [5].

As mentioned above the basic assumption in definition of the plane cascade is two-dimensionality and periodicity of the flow field. Often it is difficult in an experiment to ensure perfect periodicity of the flow field behind the blade cascade, and this can have an appreciable effect on the values of the aerodynamic characteristics of the blade cascade. Especially at transonic and supersonic velocities, complex physical phenomena may occur, that influence the flow field at the exit from the blade cascade. For example, a common problem is the complex reflection of exit shock waves from the tunnel wall shear layers downstream of the final blades of the cascade. The assumption of two-dimensional flow can be satisfied scarcely due to development of boundary layers and vortex structures on the side walls namely for cascades with low value of blade aspect ratio.

This paper presents the data reduction method as a tool for solution of parameters of flow containing mentioned shortcomings. On the purpose to obtain data for analysis and discussion a sliding data technique is developed and applied [6]. The aim is to focus on capturing the impact of dissipative processes related to the flow field setup in a blade cascade placed in a wind tunnel test section, and on highlighting possible shortcomings of linear blade cascade testing.

EXPERIMENTAL DATA

The experimental results were obtained by means of measurements in the suction-type high-speed wind tunnel at the Institute of Thermomechanics AS CR. The turbine blade cascade with relatively large values of angle of stagger and pitch/chord ratio was used. The blade cascade was mounted in the rotatable test section of the wind tunnel, which allows the desired angle of incidence ι to be set. A Mach-Zehnder interferometer, was used for interferometric and schlieren optical measurements, which were performed under almost the same conditions as pneumatic measurements in the range of isentropic exit Mach numbers $M_{2is} \in \langle 0.5;1.5 \rangle$ and incidence angles $\iota \in \langle\ 30°;+30° \rangle$; the Reynolds number related to profile chord b and inlet Mach number M_1 was in the range from $Re = 5 \bullet 10^6$ to $Re = 10 \bullet 10^6$. The schlieren picture in Fig.1 was taken at incidence angle $\iota = -20°$ and isentropic exit Mach number $M_{2is} = 1.335$.

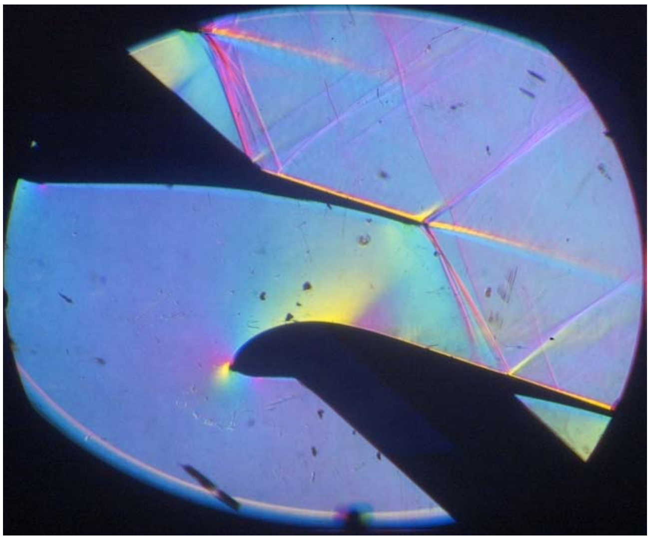

Fig. 1 Schlieren photo, incidence angle $\iota = -20°$, isentropic exit Mach number $M_{2is} = 1.335$

In the schlieren picture color filter is set up at zero level and flow structure is well apparent. The wake behind the trailing edge of the blade is split up in colour approximately into two symmetric parts – yellow upper part and dark blue lower part. Flow angle β2 can be roughly estimated from the direction of wake downstream. In schlieren pictures are seen two types of shock waves. First type – right running are red, and the second type - left running are violet (purple). Flow field near the stagnation point on the leading edge of the blade is also split symmetrically in color. Expansion on the suction side of the blades is visible as a yellow region and supersonic compression accompanying the transonic expansion as a purple region. The internal branch of exit shock waves is reflected on the suction side of the neighbouring blade regardless of incidence angle ι in the same position. Interaction of the shock wave has a significant effect on both the nature and the thickness of the boundary layer, which is highlighted as the turbulent boundary layer in yellow on the suction side near the trailing edge of the blade.

Interferogram in Fig.2 shows distribution of parameters in transonic flow fields under the same parameters as schlieren picture in Fig.1. Fringes represent lines with constant values of air density. It is possible to identify the stagnation point in the vicinity of leading edge. For negative incidence angles the stagnation point is shifted on the suction side of the profile. Acceleration of flow in the channel between blades takes place and velocity exceeds the speed of sound. Supersonic compression accompanying transonic expansion occurs. The exit shock waves arising downstream of the trailing edge are typical for regime of supersonic exit velocities. Interaction of the internal branch of exit shock waves with laminar boundary layer takes place on the suction side of the profile. It is possible to study development of the wake downstream of the trailing edge.

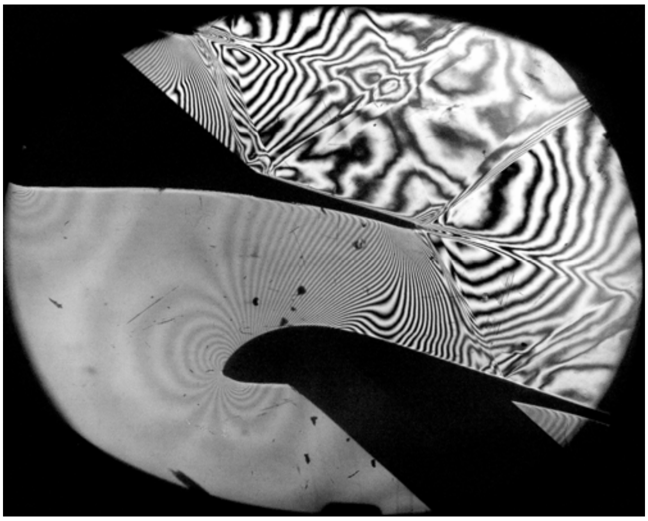

Fig. 2 Interferogram, incidence angle ι = - 20°, isentropic exit Mach number M_{2is} =1.335

Scheme of the measurement setup is shown in the Fig. 2. Distances given in the scheme are in mm. In the Fig.2 positions of inlet pressure measurements are shown and traversing probe path downstream of the cascade parallel to the plane of trailing edges is marked.

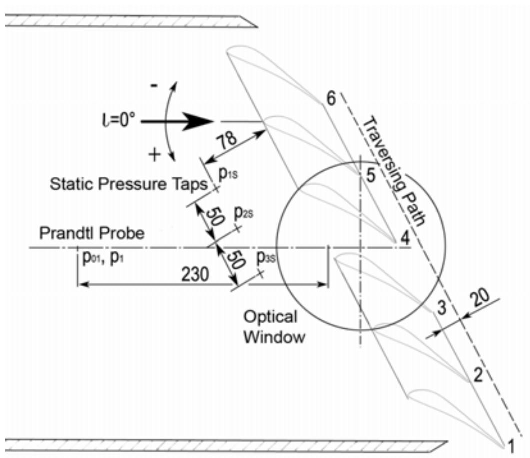

Fig. 3 Scheme of the measurement setup. Distances are given in mm.

Distributions of values of total pressure p_{02y}, static pressure p_{2y} and local angle of velocity vector β_{2y} (related to the normal line to the plane of trailing edges) along distance of two pitches (downstream of trailing edges of blades No. 3 and 4) were obtained from the traversing measurements and the data was processed to be prepared for application of the data reduction method, i.e. appropriate distributions of dimensionless velocity λ_{2y} and density ρ_{2y} according to equations

$$\lambda_{2y} = \sqrt{\frac{\kappa+1}{\kappa-1}\left[1-\left(\frac{p_{2y}}{p_{02y}}\right)^{\frac{\kappa-1}{\kappa}}\right]} \tag{1}$$

and

$$\rho_{2y} = p_{2y} r T_0 \left(1 - \frac{\kappa - 1}{\kappa + 1}\lambda_{2y}^2\right), \tag{2}$$

respectively. κ is ratio of heat capacities (for air κ = 1.4), r is specific gas constant (for air r = 287.06 J/(kgK)) and T_0 is total temperature measured in the inlet of the wind tunnel and according to adiabatic flow conditions it is supposed to be equal in the whole flow field.

DATA REDUCTION METHOD

Method of evaluation of experimental data obtained from pneumatic measurements was based on data reduction method defined by [7]. Method is based on the consequential conservation equations for mass, momentum and energy flux. In this case of two-dimensional flow of an ideal unary gas only three balance equations are needed, one for mass flow and two for momentum under assumption of constant total temperature. The integral of mass flow I_M can be formulated after certain algebraic operations as

$$I_M = \frac{1}{t}\int_0^t p_{02y}\cdot\Theta_{2y}\cdot\cos\beta_{2y}\cdot dy \tag{3}$$

and two integrals of momentum fluxes – one axial I_A and second peripheral I_C - can be defined as

$$\mathrm{I_A} = \frac{1}{\mathrm{t}}\int_0^{\mathrm{t}} \mathrm{p_{02y}}\cdot\left(2\cdot\frac{\mathrm{q_{2y}}}{\mathrm{p_{02y}}}\cdot\cos^2\beta_{2y} + \frac{\mathrm{p_{2y}}}{\mathrm{p_{02y}}}\right)\cdot \mathrm{dy}, \tag{4}$$

$$I_C = \frac{1}{t}\int_0^t p_{02y}\cdot 2\cdot\frac{q_{2y}}{p_{02y}}\cdot\cos\beta_{2y}\cdot\sin\beta_{2y}\cdot dy, \tag{5}$$

where

$$\Theta_{2y} = \frac{\rho\cdot w}{\rho^*\cdot w^*} = \left(1-\frac{\kappa+1}{\kappa-1}\cdot\lambda_{2y}^2\right)^{\frac{1}{\kappa-1}}\cdot\lambda_{2y}\cdot\left(\frac{\kappa+1}{2}\right)^{\frac{1}{\kappa-1}} \tag{6}$$

is distribution of local dimensionless mass flow rate,

$$\frac{q_{2y}}{p_{02y}} = \frac{\kappa}{\kappa-1}\left(\frac{p_{2y}}{p_{02y}}\right)^{\frac{1}{\kappa}}\left[1-\left(\frac{p_{2y}}{p_{02y}}\right)^{\frac{\kappa-1}{\kappa}}\right] \tag{7}$$

is distribution of local dimensionless dynamic pressure and $\frac{p_{2y}}{p_{02y}}$ is distribution of local dimensionless static pressure.

If we express and modify the integrals of mass flow Eq. (3) and both momentum flows Eq. (4), Eq. (5) using reduced quantities we obtain expressions as follows.

$$p_{02}\cdot\Theta_2\cdot\cos\beta_2 = I_M, \tag{8}$$

$$p_{02}\cdot\left(2\cdot\frac{q_2}{p_{02}}\cdot\cos^2\beta_2 + \frac{p_2}{p_{02}}\right) = I_A, \tag{9}$$

$$p_{02}\cdot 2\cdot\frac{q_2}{p_{02}}\cdot\cos\beta_2\cdot\sin\beta_2 = I_C\cdot \tag{10}$$

Θ_2, $\frac{q_2}{p_{02}}$ and $\frac{p_2}{p_{02}}$ are aerodynamic functions of reduced dimensionless velocity λ_2 and ratio heat capacities κ. Equations (8) to (10) commonly with equation of energy, Eq. (11), representing the assumption of equal total temperature

$$T_0 = T_2\left(1-\frac{\kappa-1}{\kappa+1}\lambda_2^2\right)^{-1}, \tag{11}$$

and equation of state, Eq. (12),

$$\rho_2 = p_2 r T_2 \tag{12}$$

form the system of five equations containing five unknown quantities. A biquadratic equation can be derived for the reduced dimensionless velocity λ_2 as a function of ratio of specific heat capacities κ, integrals of mass flow I_M, Eq.(3), and momentum flows I_A, Eq.(4), and I_C, Eq.(5), and can be solved. The solution is given by Eq. (13)

$$\lambda_2 = \sqrt{\left(\frac{\kappa+1}{2}\right)^{\frac{2}{\kappa-1}}\cdot\frac{I_A^2}{I_M^2}\cdot\left[\frac{1}{2}-\left(\frac{2}{\kappa+1}\right)^{\frac{2}{\kappa-1}}\cdot\frac{I_M^2}{I_A^2}+\frac{\kappa+2}{2\kappa}\cdot\frac{I_C^2}{I_A^2}\pm\sqrt{D}\right]} \tag{13}$$

where the discriminant D is

$$D = \frac{1}{4} - \left(\frac{2}{\kappa+1}\right)^{\frac{2}{\kappa-1}} \cdot \frac{I_M^2}{I_A^2} + \frac{\kappa-1}{4\kappa} \cdot \frac{I_C^2}{I_A^2} \ . \tag{14}$$

Analysis of the solution is given in [7]. If we select the appropriate root of Eq. (13) reference value of the downstream flow angle can be calculated as

$$\beta_2 = arcisn\left[\frac{1}{\kappa \cdot \lambda_2}\left(\frac{\kappa+1}{2}\right)^{\frac{\kappa}{\kappa-1}} \frac{I_C}{I_M}\right]. \tag{15}$$

Total pressure p_{02} and static pressure p_2 can be solved from Eqs. (16) and (17), respectively

$$p_{02} = \frac{I_M}{\Theta_2 \cdot \cos\beta_2}, \tag{16}$$

$$p_2 = p_{02}\left(1 - \frac{\kappa-1}{\kappa+1} \cdot \lambda_2^2\right)^{\frac{\kappa}{\kappa-1}}. \tag{17}$$

Total pressure loss coefficient can be expressed as

$$\xi = 1 - \frac{p_{02}}{p_{01}}, \tag{18}$$

where p_{01} is total pressure of inlet flow. Kinetic energy loss coefficient is solved by means of the relation

$$\zeta = 1 - \frac{\lambda_{02}^2}{\lambda_{02IS}^2}, \tag{19}$$

where reference dimensionless isentropic exit velocity λ_{2is} is solved from the relation

$$\lambda_{2IS} = \sqrt{\frac{\kappa+1}{\kappa-1}\left[1 - \left(\frac{p_2}{p_{01}}\right)^{\frac{\kappa-1}{\kappa}}\right]}. \tag{20}$$

THE SLIDING DATA TECHNIQUE

Calculation described in the Section Data Reduction Method must be based on a set of data obtained by traversing along one pitch at least. When the traversing is performed along more than one pitch, the distributions of reduced quantities can be evaluated. The area of one pitch length, data from which are used for the solution of integrals Eqs. (3), (4), and (5), can be moved by one step (one step of the traversing pneumatic probe) and the data reduction method is applied again giving new reduced values corresponding to the shifted pitch. The gradual shifting of the integration area, Fig. 4, enables to obtain the distribution of reduced quantities. This is the principle of the sliding data technique.

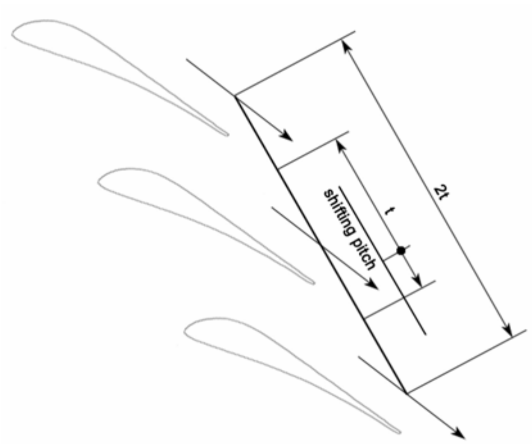

Fig. 4 Scheme of gradual shifting of the integration area along the traversing path.

In the case of theoretical ideal conditions of performance of experiments the distributions of reduced parameters obtained by means of the sliding data technique should be constant values. But real test data are, of course, under influence of three-dimensional effects, non-periodic conditions, flow perturbances, and others.

Therefore results of the sliding data technique show non-constant distributions of reduced flow parameters. Hence new source materials for analysis of flow modelling are available.

The sliding data technique was applied in the case of the blade cascade presented in this paper. The source database of results of traversing is given in [8]. Distributions of total pressure loss coefficient ξ for incidence angles $\iota = 0°$, $-20°$, and $-30°$ were obtained. They are shown in Figs.5 to 7.

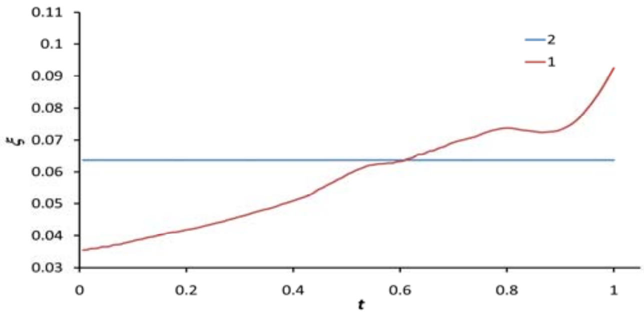

Fig. 5 Total pressure loss coefficient, blue line – 2 span, red line – 1 span, incidence angle $\iota = 0°$

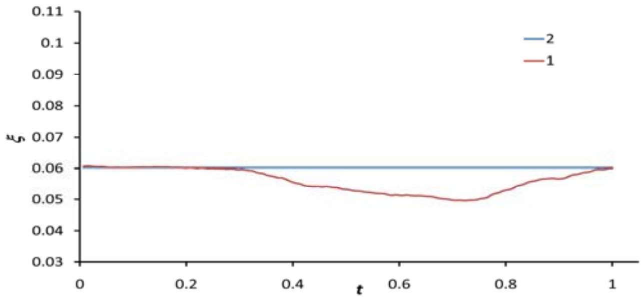

Fig. 6 Total pressure loss coefficient, blue line – 2 span, red line – 1 span, incidence angle $\iota = -20°$

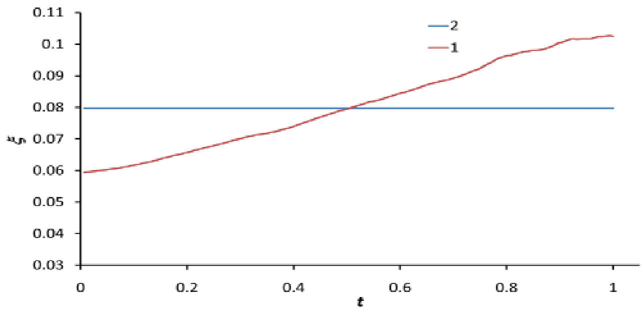

Fig. 7 Total pressure loss coefficient, blue line – 2 span, red line – 1 span, incidence angle $\iota = -30°$

In Figs. 8 to 10 the pitchwise distributions of values of the dimensionless exit velocity λ_2 obtained from the sliding data technique are shown. The blue line corresponds to calculating the value of dimensionless exit velocity λ_2 of integration area of two pitches. The red line shows the distribution of dimensionless exit velocity λ_2 calculated using sliding data technique.

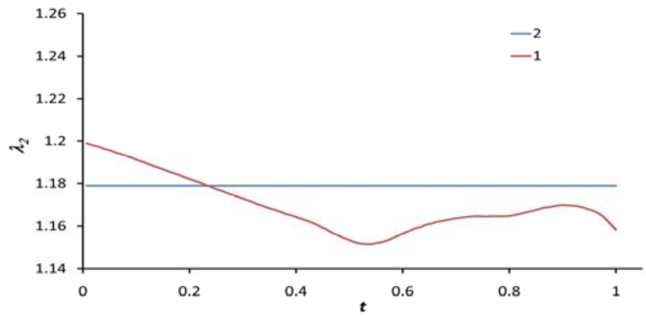

Fig. 8 Dimensionless exit velocity λ_2, blue line – 2 span, red line – 1 span, incidence angle ι = 0°

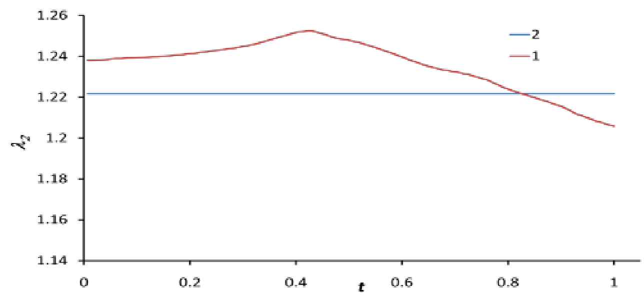

Fig. 9 Dimensionless exit velocity λ_2, blue line – 2 span, red line – 1 span, incidence angle ι = - 20°

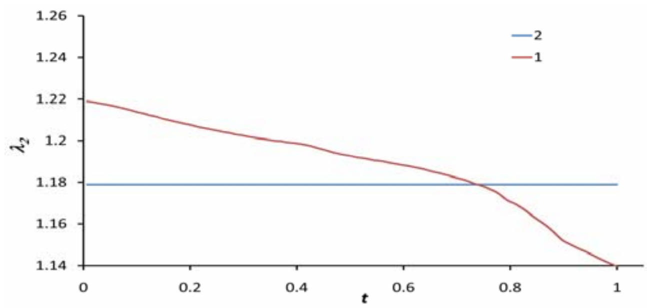

Fig. 10 Dimensionless exit velocity λ_2, blue line – 2 span, red line – 1 span, incidence angle ι = - 30°

ANALYSIS AND DISCUSSIONS

High-speed aerodynamic flow fields are characterized by a considerable sensitivity on small disturbances or differences of parameters on boundaries and by an appearance of new flow structures. Mathematical theory gives argument for the reason of this - a nonlinear description of flow fields near boundaries of areas described by elliptic and hyperbolic equations. In real flow, transonic effects occur. They are shock waves, expansions over sonic conditions, aerodynamic choking, development of boundary layers at interaction with shock waves, possible flow separations, wakes, supersonic compression accompanying transonic expansion, the flow past trailing edges, etc. The effects can be observed by flow visualisation techniques.

It would be overidealistic to insist on a perfect periodic, two dimensional, steady behaviour of high-speed real flow. Experimental results evaluated by the sliding data technique of traversing probe measurements pointed to new possibilities to understand high-speed aerodynamic modeling in wind tunnels. A comparison with results by means of optical methods, Figs.2 and 3, confirms a complex flow structure developing in flow field in exit part of blade cascade at transonic and supersonic velocities. Shock waves, namely internal branch of exit shock waves interacting with boundary layer on the suction side of the neighbouring profile seems to be important dissipative phenomena. Their occurrence in the flow field downstream of the blade cascade is rightful and their affect on the flow parameters has to be taken in account. Due to finite number of blades in experimental arrangement of the blade cascade in test section of wind tunnel additional compression effects appear. If they are strong, they are called “tunnel shock waves” or “parasite shock waves” and their occurrence was described by Starken and Lichtfuss in [9] and can be observed not only by means of optical methods but also by means of the sliding data technique.

Differences in distributions of total pressure loss coefficient ξ in Figs. 5 to 7 seems to be relatively large but analysis of sensitivity of evaluated values on uncertainties of measured parameters should be performed. From this point of view, distributions of dimensionless exit velocity λ_2 in Figs. 8 to 10 show maximum relative differences about ± 4% which can be satisfying result for common experimental accuracy at high-speed velocities. Of course, analysis of uncertainties of measured and evaluated parameters should be performed

Further study of flow field should be performed to obtain more detailed information on flow structure and influence of dissipative phenomena. For instance, information on influence of incidence angle on flow parameters is valuable for design and research of turbomachines. Aerodynamic modeling of off-design performance of turbomachines can detect limits of economic and reliable operation of turbomachines. Figures 5 and 8 show results obtained during tests at nominal operation conditions. Figures 2, 4, 6, 7, 9, 10 present results achieved at part load regimes. Determination of non-periodic effects at off-design performance of turbomachines can be a suggestion for the next detailed investigations.

The sliding data technique can be applied at numerical modeling of flow in blade cascades and experience on this should be undoubtedly valuable.

Another question appears with application of sliding data technique. The basic assumption of evaluation procedure is two-dimensional formulation balance equations – balance of mass, momentum and energy. Possible 3D effects in experimental modeling and measurements should be analyzed.

CONCLUSIONS

The performed study of total pressure loss coefficient ξ of transonic and supersonic flow in the turbine blade cascade uses experimental data from tests in the wind tunnel. Results obtained by means of optical methods give an insight into flow structure and are a good basis for description of flow development. The sliding data technique can be applied on data from traversing measurements in order to evaluate distributions of reduced parameters especially of total loss coefficient ξ and dimensionless exit velocity λ_{2is}. The results open questions about influence of uncertainty of measured quantities, occurrence of non-periodic phenomena, and possible 3D effects in flow field at traversing plane.

The sliding data technique gives an impetus to further investigations of blade cascade flows. Giving information on the periodicity of the exit flow field the Sliding Data Technique can become standard technique applied when evaluating traversing data from tests of linear blade cascades.

ACKNOWLEDGEMNTS

The work was enabled thanks to support and collaboration with the SKODA POWER and has been supported by the project of Progressive Technology System for Power Engineering 1M06059, by the Ministry of Education, Youth and Sports of the Czech Republic No. 1M06031. Grant support from Grant Agency of the Academy of Sciences of the Czech Republic No. IAA200760801, and from the Grant Agency of the Czech Republic No. GAP101/10/1329 is also gratefully acknowledged.

REFERENCES

[1] G.Kynast : Über die Auswertung von Nachlaufmessungen an ebenen Schaufelgittern bei kompressibler Strömung in Unterschallbereich, Zeitschrift für Flugwissenschaft, 12 (1964), pp.69-71

[2] J.Amecke : Auswertung von Nachlaufmessungen an ebenen Schaufelgittern, AVA-Report 67 A 49, 1967

[3] M.L.G.Oldifield, D.L.Schultz, J.H.Nicholson : Loss Measurements Using a Fast Traverse in an ILPT Transient Cascade, In : Proceedings of 6th Symposium on Measuring Techniques in Transonic and Supersonic Flows in Cascades and Turbomachines, Lyon, 1981

[4] J.Amecke, P.Safarik : Data Reduction Method of Wake Measurements with Injection of Another Gas, In : Proceedings of 12th Symposium on Measuring Techniques in Transonic and Supersonic Flows in Cascades and Turbomachines, Prague, 1994

[5] P.Safarik, M.Luxa, J.Benetka, J.Michalek : Data Reduction Method for Three-Dimensional Transonic Flow Measurements, In : Proceedings of 17th Symposium on Measuring Techniques in Transonic and Supersonic Flows in Cascades and Turbomachines, Stockholm, 2004

[6] M.Matejka, M.Luxa, D.Simurda, J.Synac, P.Safarik : Loss Coefficient Dependence of Midsection of 12200 mm Long Rotor Turbine Blade, ASME Paper No. GT2010-22740, New York, 2010

[7] J.Amecke, P.Safarik : Data Reduction of Wake Measurements with Injection of an Other Gas, DLR Report No.95-32, Köln, 1995

[8] D.Simurda, M.Luxa : Pneumatic Measurements on the TR-Z-1 Cascade, Report of IT CAS No.Z 1452/10, Prague, 2010 (in Czech)

[9] Lichtfuss H.J., Starken H.: Supersonic Cascade Flow, pp.37-149 In : Progress in Aerospace Science, Pergamon Press, Oxford, 1974