RECORDS OF UNSTEADY TRANSONIC FLOW PAST BLADE CASCADES BY MEANS OF OPTICAL METHODS

J. Ulrych, J. Benetka/Aeronautical Research and Test Institute, Prague

P. Šafarík/Czech Technical University in Prague

ABSTRACT

The contribution deals with optical methods and results of aerodynamic research of transonic blade cascades and a single profile. The unsteady transonic flow is recorded by means of the schlieren method and the flow field structure is described and analysed. The possibilities of the colour streak schlieren method are presented and discussed.

INTRODUCTION

Optical methods represent an inseparable part of flow field investigation at the Department of High-Speed Aerodynamics (HSA), Aeronautical Research and Test Institute. During the last three decades a large number of experiments focused on the flow visualization using shadowgraph, schlieren and interferometric methods have been carried out. Although the most of pictures provide us with only qualitative information, the colour schlieren video records enable us to obtain a helpful insight into the real transonic flow fields under the unsteady conditions, namely in the cases of oscillating blades and profiles. Moreover, the electronic data processing makes possible to refine the records, to reveal hardly observable phenomena and finally the new approach to the flow field analysis allows us to acquire more information from schlieren pictures.

The aim of this paper is to describe the optical facility, and to present and discuss some of the results.

NOMENCLATURE

c	[mm]	…	chord
f	[Hz]	…	frequency of oscillation
M_1	[1]	…	inlet Mach number
M_{2is}	[1]	…	outlet Mach number
M_∞	[1]	…	Mach number of the incoming flow
t	[mm]	…	pitch
α	[deg]	…	angle of attack
α_1	[deg]	…	inlet angle
γ	[deg]	…	angle of stagger
Seq.		…	movie sequence

THE OPTICAL MEASUREMENT SYSTEM

The experiments discussed in this article were carried out by means of two schlieren optics devices, which were applied on two continuous wind tunnels.

The video sequences introducing an oscillating profile were taken using the portable optical measurement system, which had been developed at the Department of High-Speed Aerodynamics. The optical arrangement for schlieren method is depicted schematically at Fig.1.

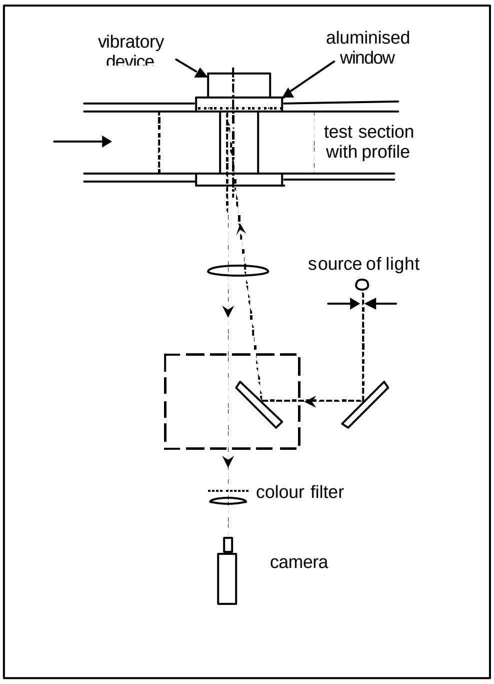

Fig.1: Optical arrangement for schlieren method – oscillating profile configuration.

A halogen lamp was used as a source of light. The beam of rays was deflected by mirrors and then went through a lens and an optical window into the test section. On the opposite wall of the wind tunnel the light was reflected from the aluminised surface of the window and passed the test section and lens one more time. After passing through the colour filter the condensed rays were recorded by high-speed camera.

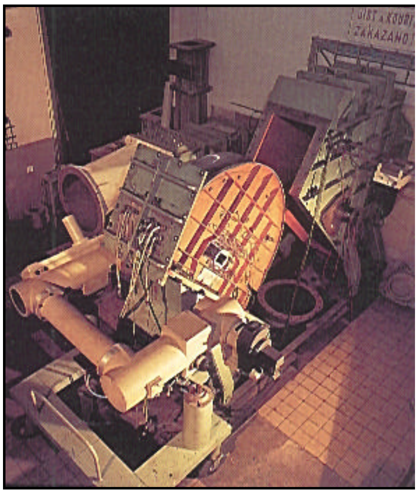

Fig.2: Continuous wind tunnel for blade cascades with the ZEISS 80 schlieren device.

The hydraulic crankshaft vibratory device was placed on the outer side of the tunnel behind the aluminised window. The apparatus was used to generate harmonic oscillation of the model within the range of frequencies f = 1 – 43 Hz.

The experiments focused on recording the transonic flow in blade cascades were accomplished in the test section of the special wind tunnel equipped with the optical windows and the standard ZEISS 80 schlieren device (see Fig.2). When the case of oscillating blade in the cascade was investigated the experimental device was in principle the same as the system at Fig.1. While the portable schlieren system was fitted to the particular wind tunnel (Fig.2), the test section was again equipped with one simple and one aluminised optical windows, and with the vibratory device.

THE MODELS

For the very first attempt to record the transonic flow around a body in the test section the symmetric NACA 0012 profile, which is commonly used as an etalon aerofoil, was chosen.

In the experiment focused on the flow past blade cascades the DCA- S compressor blade cascade was used. The scheme of the cascade is at Fig.3.

The third series of experiments, concentrating on the flow field around an oscillating blade in the turbine cascade, was carried out using the model of experimental stator blade cascade determined for transonic regimes.

THE OPTICAL MEASUREMENTS RESULTS

The first aim during the investigation of an unsteady transonic flow was to observe and record the flow field development around a single oscillating profile under the conditions of increasing Mach number $M_\infty = 0{,}75 - 1$, $\alpha = 0$ deg, $\Delta\alpha = \pm 3$ deg, f = 1 Hz.

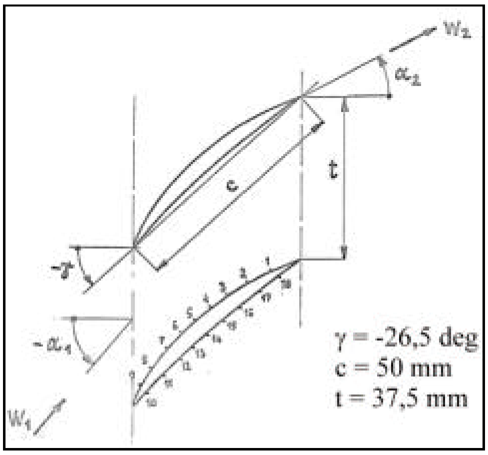

Fig.3: The scheme of the DCA-S compressor blade cascade.

The result is presented in Seq.1 and some characteristic images are at Figs.4a- f.

Figure 4a represents the starting position. Mach number of an incoming flow is slightly over the critical value. In the vicinity of the leading edge it is possible to see the red region of deceleration. Behind the stagnation point the speed of flow increases up to the supersonic value (green area) but due to the M close to unity no terminate shock wave is visible. Downstream the trailing edge is the straight wake. The first terminate shock wave appears on the suction side when the profile reaches the top dead centre (Fig.4b). The light area of compression, which is connected to the transition point, is upstream the wave. With the increasing M_∞ the terminate waves are observable on the both suction and pressure sides during the whole oscillation. The shift of the waves to the trailing edge and the separation point are remarkable (Figs.4d-e).

Fig.4a

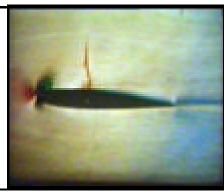

Fig.4b

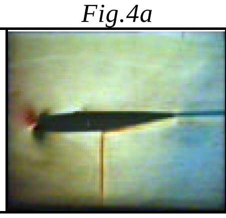

Fig.4c

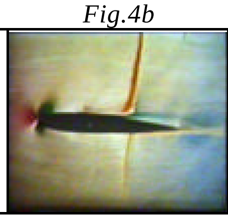

Fig.4d

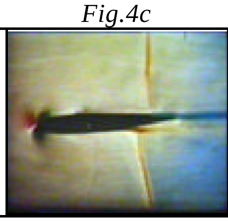

Fig.4e

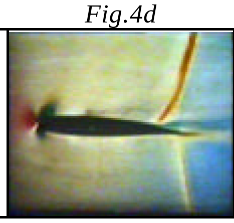

Fig.4f

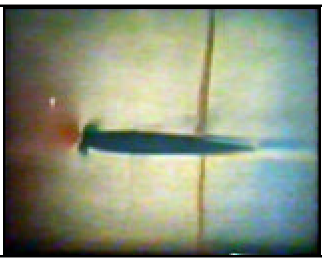

Fig.5a

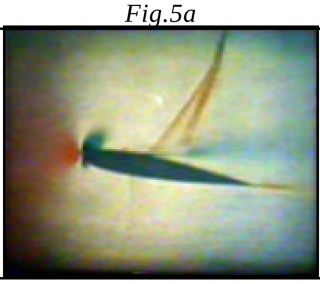

Fig.5b

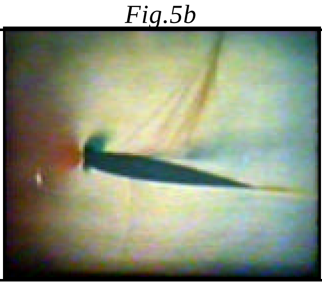

Fig.5c

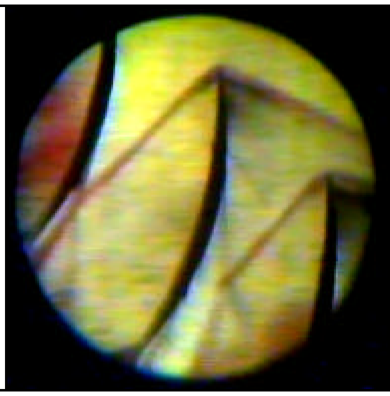

Fig.6a

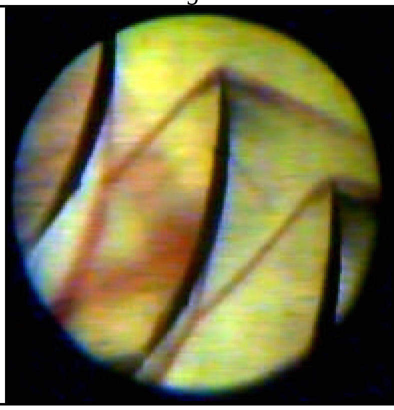

Fig.6b

Another set up of the experiment was defined by the constant $M_\infty = 0{,}9$ and increasing frequency of oscillation f = 1 – 30 Hz. The initial angle of incidence and amplitude remained $\alpha = 0$ deg, $\Delta\alpha = \pm 3$ deg.

The record is introduced in Seq.2. The terminate shock waves on the suction and pressure sides are situated at 75% of the profile length and are almost perpendicular to the chord. The region of compression, the boundary layer separation and the wake are obvious. Although the separation of the flow at higher frequencies seems to be on both sides of the aerofoil, the single movie frames show the separation only on the suction side.

When the initial angle of attack was changed to $\alpha = +4$ deg and the frequency was kept at the value of 30 Hz – while $\Delta\alpha$ and M_∞ were not changed –, the curious effect appeared. The phenomenon is recoded in Seq.3 as well as at Figs.5a- c.

Figure 5a shows the profile at the bottom dead centre ($\alpha = +1$ deg) where the terminate shock wave on the suction side is perpendicular to the chord and the separation point is at the spot of interaction of the shock wave and boundary layer. As the angle of attack increases the compression area is more intensive and a λ shock is formed (Fig.5b). Close to the top dead centre the shock wave starts to fray out and oscillate with significantly higher frequency (Fig.5c, Seq.3). This strange effect is further analysed in the following chapter.

After the basic investigation of the possibilities of recording the characteristic flow phenomena by means of schlieren video images the series of experiments with compressor and turbine blade cascades were carried out.

Seq.4 introduce the development of flow field around the DCA-S compressor blade cascade under the conditions of extremely large incidence angle $\alpha_1 = -60$ deg. The inlet Mach number rises up to $M_1 = 1{,}15$.

During the increase of the inlet velocity it is possible to observe the progress of typical subsonic and transonic flow features. Nevertheless, the most interesting phenomenon comes into view when the inlet Mach number reaches the maximum value. Although the inlet flow parameters remain constant, the response of the flow field in the inter-blade channel is unsteady. The situation is demonstrated in Figs.6a, b. Figure 6a illustrates the bow- shock wave close to the leading edge of the blade. The inner branch of the wave interacts with the boundary layer on the suction side of the previous blade at approximately 60% of the chord length, causes its separation, and reflects as the compression structure. Subsequently, the backpressure from the downstream region induces the shift of the separation point upstream to 25% of chord length.

At the same time the volume of the area of separated flow inside the channel grows dramatically and the effective inter-blade channel contracts (see Fig.6b). The system of compression waves, which is formed on the suction side in the vicinity of leading edge, extends into the channel and reflects from the pressure side of the next blade keeping its compress character. Due to velocity and pressure changes accompanying the effects described above, the flow is reattached again and the situation is similar to that at Fig.6a.

As the movie sequence and photographs suggest, the instability in one inter-blade channel probably bears on the actual situation in the neighbouring channels. Therefore, the investigation of a single channel could be insufficient and misguided.

The oscillation of the separation point in the inter-blade channels under constant inlet conditions is called the transonic instability. With respect to its negative impact on compressor and turbine wheels it is important to define the conditions under which it occurs and describe its mechanism.

The changes of flow field in the DCA- S blade cascade as a reaction on the continuous change of the incidence angle are visualised in Seq.5. As the starting angle of attack the α_1 = - 60 deg was chosen. The inlet Mach number was increased up to M = 1,1 and then α_1 was changed to the value of –38 deg. Figs.7a-e show flow field characterised by α_1 = - 60, - 55, - 50, - 45, - 40 deg. respectively. Due to the wind tunnel design the change of incidence angle is accompanied by the decrease of M_1. For that reason the inlet bow-shock detaches off the leading edge and then disappears. In the inter- blade channel the interaction between the inner branch of the shock wave and the shear layer over previous blade is visible (Figs.7a, b). The reflected wave has the expansion character and the deflection of the boundary of the separated flow is evident. According to decreasing α_1 the separation point moves downstream. At Fig.7c the detached inlet wave, the expansion area on both suction and pressure sides, the lip shock on the pressure side as well as the terminate shock wave are easy to detect. While the expansion to supersonic velocities on the pressure side becomes more distinguish at Fig.7d owing to decreasing α_1, the terminate shock wave within the channel is less intensive because of the decreasing inlet Mach number.

The case of oscillating single blade in the cascade was examined using the experimental turbine stator blade cascade. The record of the flow past the cascade at subsonic regime with a transonic region due to blade oscillations is in Seq.6. The initial angle of attack of the oscillating blade does not correspond with the other blades in the cascade. Figures 8a-c introduce the situation defined by the smallest (γ = 40 deg), the middle

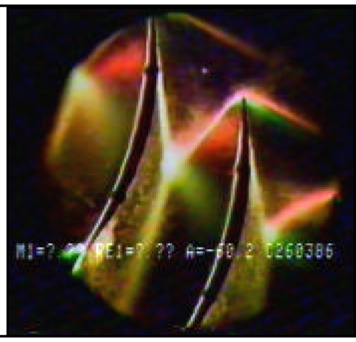

Fig.7a: a_1 = - 60 deg.

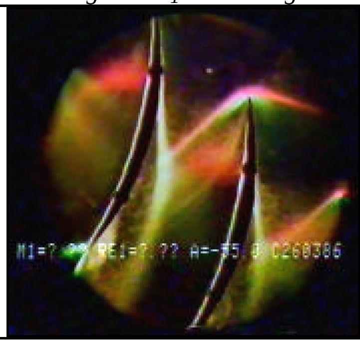

Fig.7b: a_1 = - 55 deg.

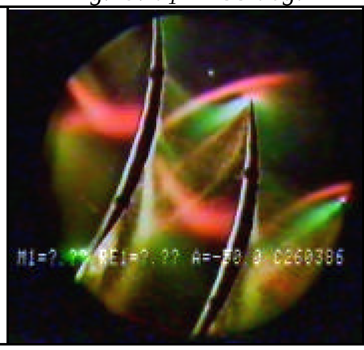

Fig.7c: a_1 = - 50 deg.

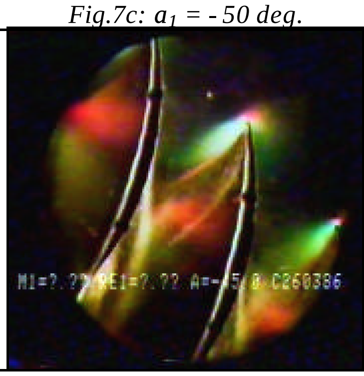

Fig.7d: a_1 = - 45 deg.

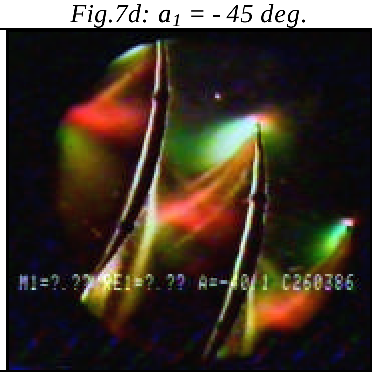

Fig.7e: a_1 = - 40 deg.

(γ = 42,5 deg), and the largest (γ = 45 deg) value of the stagger angle. On the pressure side of oscillating blade it is possible to observe the flow separation at the leading edge (Fig.8a). The subsonic flow accelerates in the inter-blade channel to low supersonic velocities. The curved inlet branch of the exit shock wave terminates local supersonic region in the vicinity of the channel throat. Shear layer downstream in the exit part is evident. Around the middle phase (Fig.8b) the very weak internal branch of the exit shock wave terminates the local supersonic region. Figure 8c shows the maximum value of the stagger angle γ during oscillation. The flow separation on the pressure side of the leading edge disappears and the local supersonic region is not observed.

THE OPTICAL MEASUREMENTS ANALYSIS - THE COLOUR STREAK SCHLIEREN METHOD

The colour streak schlieren method was developed on the basis of the simple streak schlieren method, which was first applied at the Institute of Thermomechanics, Academy of Sciences of the Czech Republic (see [3]). The principle of the method is to observe the flow development in the specified area (streak) in order to examine the time dependence of the flow phenomena position.

In this case the flow field around the NACA 0012 aerofoil under the transonic conditions M_∞ = 0,9, α = +4 deg, $\Delta\alpha$ = ± 3 deg, f = 30 Hz was investigated.

The basic record is presented in Seq.3 and described in the previous chapter. The movie sequence was adapted to the colour streak schlieren method using the Adobe Premiere, Corel PHOTO-PAINT, and Corel DRAW software subsequently.

At the beginning the area of interest above the aerofoil was defined. The dimensions and position of the streak were determined with respect to the intelligibility of the final picture. Then the chosen part of the schlieren picture was extracted from every movie frame and lined up according to the time succession. The whole procedure is depicted at Fig.9.

In this particular case the colour streak schlieren image of the flow field in the vicinity of the NACA 0012 profile (Fig.10) enable us to observe the time oscillation of the terminate shock wave as the response on the profile oscillation, the periodic changes of the supersonic region development, and to assess the oscillation of the shock wave with considerably higher frequency after the flow separation close to the leading edge. The phase shift between the aerofoil and the terminate shock wave oscillations could be examined as well.

In principle, all of the phenomena mentioned above can be observed and investigated in the more

Fig.8a

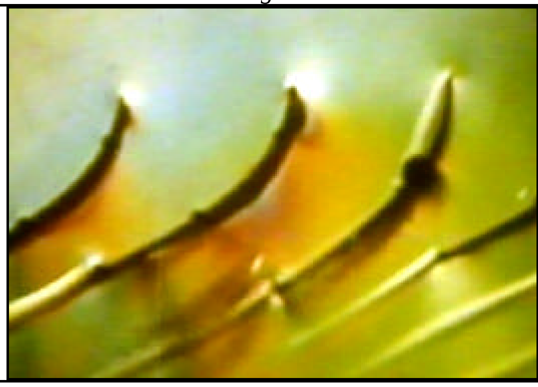

Fig.8b

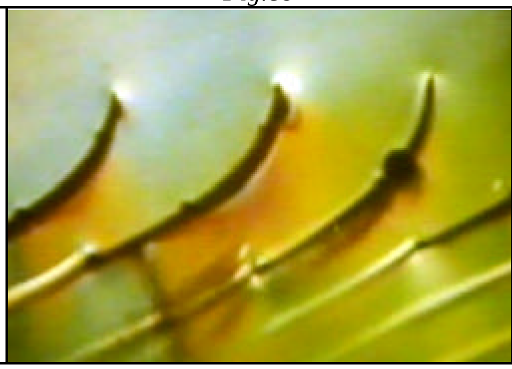

Fig.8c

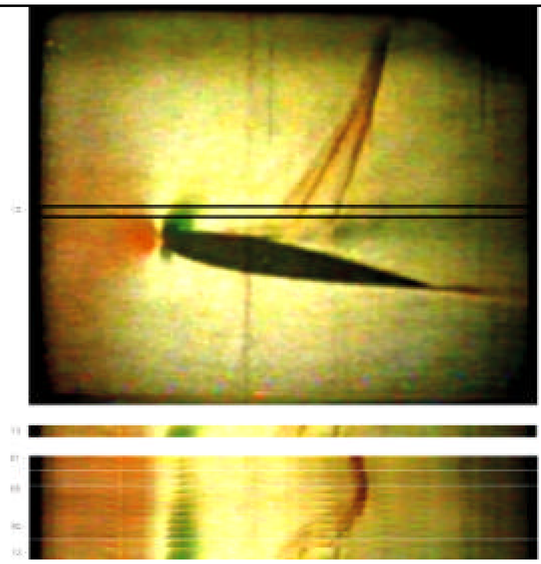

Fig.9: The principle of the colour streak schlieren method.

complex flow past blade cascades. The transonic instability, as it was presented in Seq.4., Figs.6a, b, could be analysed using the same method.

CONCLUSIONS

The advantage of application of the schlieren measurement records enables us to obtain an insight into the real transonic flow fields under the unsteady conditions, namely in the cases of oscillating blades or profiles. According to the shock wave structure development and boundaries of the expansion regions the response of the basic transonic flow on unsteady external conditions could be observed from records.

The experimental data analysis supported by optical measurement records is a powerful tool for research and design of turbomachines operating at transonic regimes.

The further development of the colour streak schlieren method and its application in the research of unsteady flow past blade cascades could be of a significant importance for research workers in the field of turbomachinery, aerodynamics and aeroelasticity.

ACKNOWLEDGMENTS

All the experiments were carried out at the Department of High-Speed Aerodynamics, Aeronautical Research and Test Institute. The electronic data processing was accomplished in cooperation with Czech Technical University in Prague, Faculty of Mechanical Engineering. The support of this research from the Aerospace Research Centre and the Grand Agency of the Czech Republic under the Grand No. 101/01/0450 is gratefully acknowledged.

RELATED MOVIE SEQUENCES

Seq.1 – NACA 0012 aerofoil, M_∞ = 0,75 – 1, α = 0 deg, $\Delta\alpha = \pm 3$ deg, f = 1 Hz (1 min.)

Seq.2 – NACA 0012 aerofoil, M_∞ = 0,9, α = 0 deg, $\Delta\alpha = \pm 3$ deg, f = 1 – 30 Hz (1 min.)

Seq.3 – NACA 0012 aerofoil, M_∞ = 0,9, α = +4 deg, $\Delta\alpha = \pm 3$ deg, f = 30 Hz (35 s)

Seq.4 – DCA-S compressor blade cascade, M_1 = 1,15, α_1 = - 60 deg, transonic instability (2 min.)

Seq.5 – DCA-S compressor blade cascade, initial M_1 = 1,1, α_1 = - 60 – - 38 deg (2,5 min.)

Seq.6 – oscillating turbine blade in the cascade at transonic regimes, γ = 40 – 45 deg (45 s)

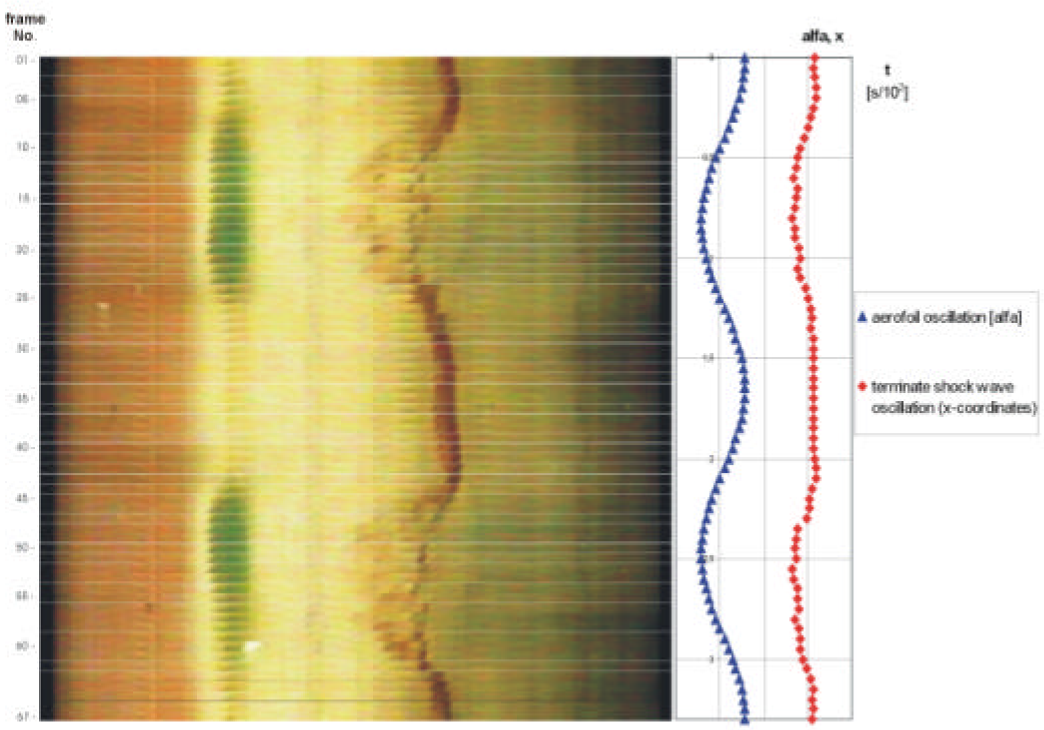

Fig.10: The colour streak schlieren method - NACA 0012 aerofoil, M¥ = 0,9, a = +4 deg, Da = ± 3 deg, f = 30 Hz.

REFERENCES

1) Šafarík, P., Luxa, M.: *Using Optical Methods in High-Speed Aerodynamic Research*, pp. 21 – 27, in: *Measuring Techniques in Turbomachinery*, Proceedings of the XVth Symposium on Measuring Techniques for Transonic and Supersonic Flow in Cascades and Turbomachines, Florence, 2000
2) Ulrych, J.: *Development of Flow Visualisation – Optical Methods*, separate issue from 2002 Conference on Student Creativity and Invention, CTU in Prague, Faculty of Mechanical Engineering, Prague, 2002
3) Šafarík, P., Vlcek, V.: *Using Interferometric Measurements in Calculation of Aerodynamic Forces*, in: *Optical Methods in Dynamics of Fluids and Solids*, Proceedings of an International Symposium held at Institute of Thermomechanics, Czechoslovak Academy of Sciences, Liblice Castle, 1984