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# Forced oscillations of airfoil flows

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**Abstract** The effect of background flow oscillations on a transonic airfoil (NACA 0012) flow was investigated experimentally. The oscillations were generated by means of a rotating plate placed downstream of the airfoil. Owing to the expansion and compression waves generated at the plate, the flow over the airfoil flow was drastically disturbed. This resulted in the presence of high intensity oscillations of a shock wave and a separation bubble on the suction surface of the airfoil. For relatively large values of the airfoil angle of attack, weak shock waves (transonic sound waves) were periodically shed upstream of the airfoil.

## 1 Introduction

Airfoil flows at high subsonic Mach numbers at relatively large angle of attack are accompanied by shock waves and flow separation. These phenomena lead in most cases to self-excited instabilities (buffeting), which result in unsteady airfoil loading and considerable noise generation. Self-excited airfoil flow oscillations were the subject of interest of many authors: Naumann (1965), Karashima (1961), Meier (1974), Finke (1977), Basler (1987), Mabey (1989), Geissler (1993). Their comprehensive experimental and theoretical investigations allowed them to elucidate oscillation mechanism of this type. In both experimental and theoretical studies only the steady background flow has been taken into account by the authors mentioned above. However, in helicopter flight, turbine and compressor blade flows and other similar cases, the background flow is oscillatory. This appears as a result of periodical changes of the relative flow velocity or disturbances generated by the downstream blade rows.

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The present work is aimed at the investigation of the effect of background flow oscillation in separated and non-separated airfoil flows, which correspond to large and small angle of attack, respectively.

## 2 Apparatus

The transonic wind tunnel, having a test section of 330 mm height and 100 mm width, was used for the experiments (Fig. 1). The side walls of the test section were provided with two glass windows for flow observations. The wind tunnel operated on a vacuum downstream. The flow velocity of the air was controlled by adjusting the opening of a precisely adjustable valve.

A NACA 0012 airfoil profile of 120 mm chord was used in the experiments. A plate 30 × 100 mm was placed downstream of the profile to generate the background flow oscillations. It was rotated in the frequency range of 55–116 Hz by means of an electric motor (see Fig. 1).

The airfoil flow was visualised by means of a Mach-Zehnder interferometer. A pulsating light source controlled by a trigger conditioner, which was developed at Max-Planck-Institut für Strömungsforschung (Stasicki et al. 1990), was used for illumination. High speed recording of flow interferograms with a frame rate of 5 kHz was achieved by means of a drum camera.

Piezoresistive pressure transducers (Kulite) were used for transient pressure measurements. Four transducers were placed on a disk to which the airfoil was attached. The disk could be installed on the side wall of the test section in place of the glass window. An additional transducer was fixed to the test section wall upstream of the airfoil (Fig. 2).

## 3 Results

### 3.1 Background flow oscillations

In high subsonic flows the rotating plate induces flow disturbances in its upstream region. These disturbances consist of compression and expansion waves which move alternately. However, according to well known characteristics of non-linear waves, the gradients of flow properties increase across the compression wave and decrease across the expansion region as the waves propagate away from their source. One can observe it as the steepening and smoothing of pressure signals for compression and expansion waves (Fig. 3), respectively.

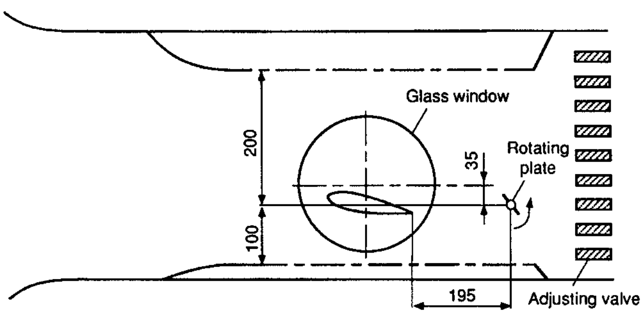


**Fig. 1.** Test section of transonic wind tunnel

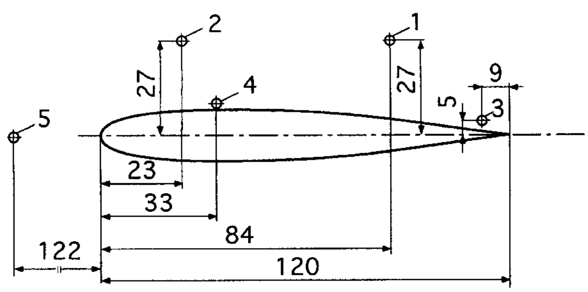


**Fig. 2.** Airfoil and pressure transducers location

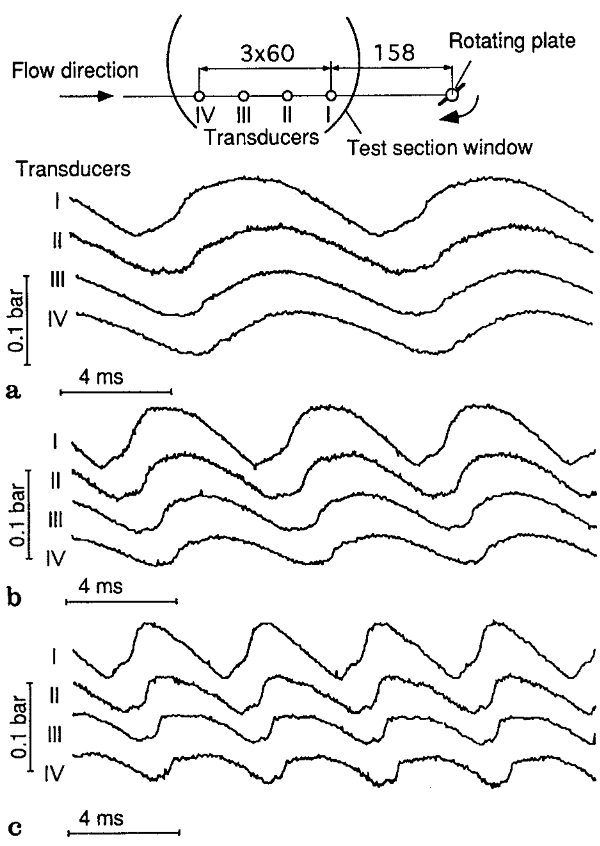


**Fig. 3a–c.** Pressure signals at points i–iv upstream of the rotating plate for three oscillation frequencies of background flow. **a** 115 Hz, **b** 170 Hz and **c** 232 Hz. Transducers are distributed along the line corresponding to the airfoil position

This effect is stronger in the high frequency range and can lead to the formation of shock waves. The existence of weak shock waves can be deduced from the pressure signal for $f=232$ Hz (Fig. 3c).

The pressure signals presented in Fig. 3 show that the oscillation amplitude decreases at consecutive points upstream of the rotating plate. The decrement of the oscillation amplitude is larger for higher oscillation frequencies; for $f=111$ Hz the amplitude decreases from 0.14 bar (peak to peak) at point (i) to 0.1 bar at point (iv), whereas for $f=232$ Hz it drops from 0.14 bar to 0.06 bar, respectively. This is an effect of slitted upper and bottom walls of the test section, which to some extent reflects the disturbances generated by the rotating plate. It appears that the walls of 8% slitting rate used in the experiments effectively reflect disturbances of rather low frequencies.

A mean Mach number $M=0.7$ was kept constant during the experiments. Assuming the linear relationship $\Delta u=\Delta p/\rho c$ across the waves ($\Delta u, \Delta p, c$ and $\rho$ mean flow velocity and pressure oscillation amplitude, speed of sound and gas density, respectively), one obtains the following values of the Mach number amplitudes (peak to peak) $\sim 0.125$ and $\sim 0.1$ for $f=111$ Hz and 232 Hz, respectively. The above data were valid in the test region where the airfoil was attached.

## 3.2 Airfoil flow

### 3.2.1 Steady background flow

Figure 4 shows two interferograms corresponding to extremal positions of the oscillating shock wave during the buffeting (without rotating plate in the test section). In both cases, the boundary layer is separated at the foot of the shock. The separation bubble, however, exhibits various positive pressure gradients in each case (This feature can be deduced considering the number of interferometric fringes through the separation region). The pressure gradient is larger for the left hand side interferogram which corresponds to lower pressure behind the shock wave. The shock is at a greater distance downstream of the leading edge in this case. The oscillation frequency is about 170 Hz, which corresponds to the Strouhal number equal to 11.

### 3.2.2 Unsteady background flow

It is well known that the characteristics of separated flow are very sensitive to internal and external excitation. Acceleration

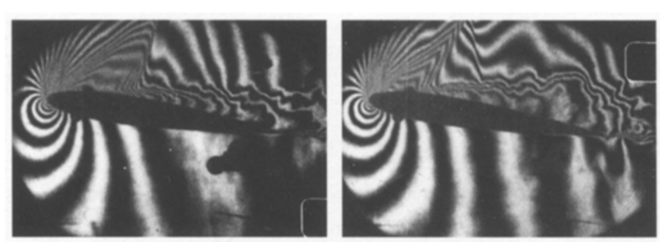

**Fig. 4.** Extremal flow patterns during buffeting, $\alpha=8.5°$

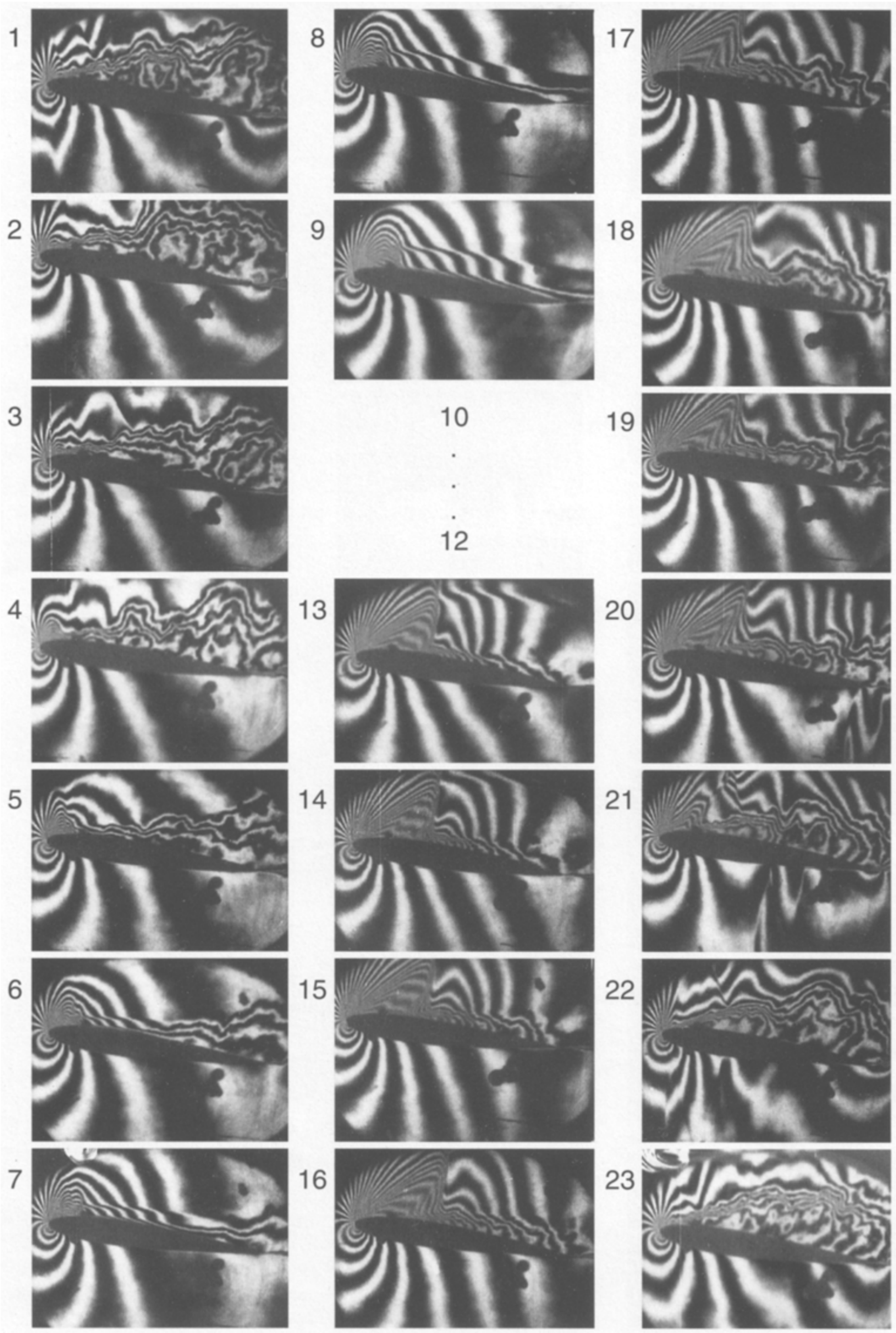


**Fig. 5.** Interferometric photographs of excited airfoil flow, $\alpha = 8.5°$; Excitation frequency $f = 114$ Hz. Time intervals between two exposes $\Delta t = 0.4$ ms

of the excited flow counteracts separation, whereas deceleration strengthens it. Figure 5–7 show three sets of interferometric photographs of airfoil flow for three frequencies of background flow oscillations and constant airfoil angle of attack of $\alpha = 8.5°$. The characteristic feature of this flow is that the boundary layer remains attached during a relatively long portion of the oscillation period. This feature, which appears for each of the frequencies studied, constitutes the essential difference when comparing the excited and non-excited background flow cases for $\alpha = 8.5°$; in the last case the airfoil flow remains permanently separated.

The airfoil flow oscillation period is composed of two phases: the acceleration and the deceleration phase. At the beginning of the acceleration phase the flow for $\alpha = 8.5°$ is separated close to the leading edge of the profile (frame 1 in Fig. 5). The separation region is very wide. The flow in this region shows high turbulence dominated by large vortices. As a result of the background flow acceleration the separation

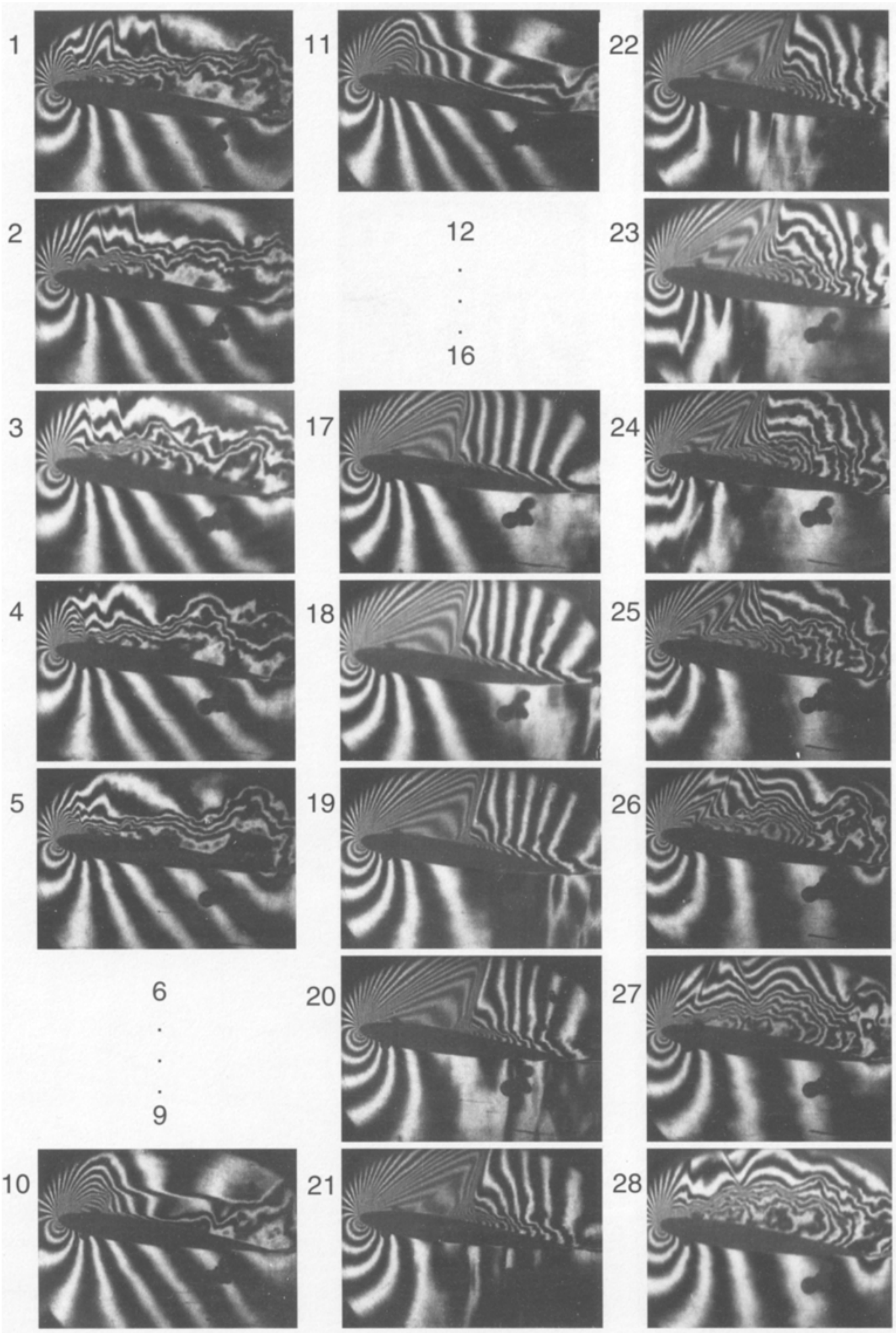


**Fig. 6.** Interferometric photographs of excited airfoil flow, $\alpha = 8.5°$, $f = 168$ Hz, $\Delta t = 0.2$ ms

point shifts downstream (frame 2 and the following). Simultaneously, the supersonic flow region terminated by a very weak shock wave repeats itself (frame 6). The height of the shock wave is indicated by the spacing between the airfoil and the density discontinuity plane parallel to the upper surface of the airfoil. The appearance of this plane is due to the increase of temperature of particles as they cross the shock wave. As the separation point is quickly "washed out" downstream, the attached, relatively thin boundary layer repeats itself (frame 7). The smooth interferometric fringes in the region close to the profile surface indicate that the flow in this region is of low turbulence. In the following phase, the supersonic flow region increases its volume; this corresponds to the shock wave gaining in strength (frame 9). This process is accompanied by the increase of boundary layer thickness which now shows larger turbulence intensity (frame 13). Frame 14 presents an extreme flow pattern, characterized by the strongest shock wave, more distant from the leading edge. However, this

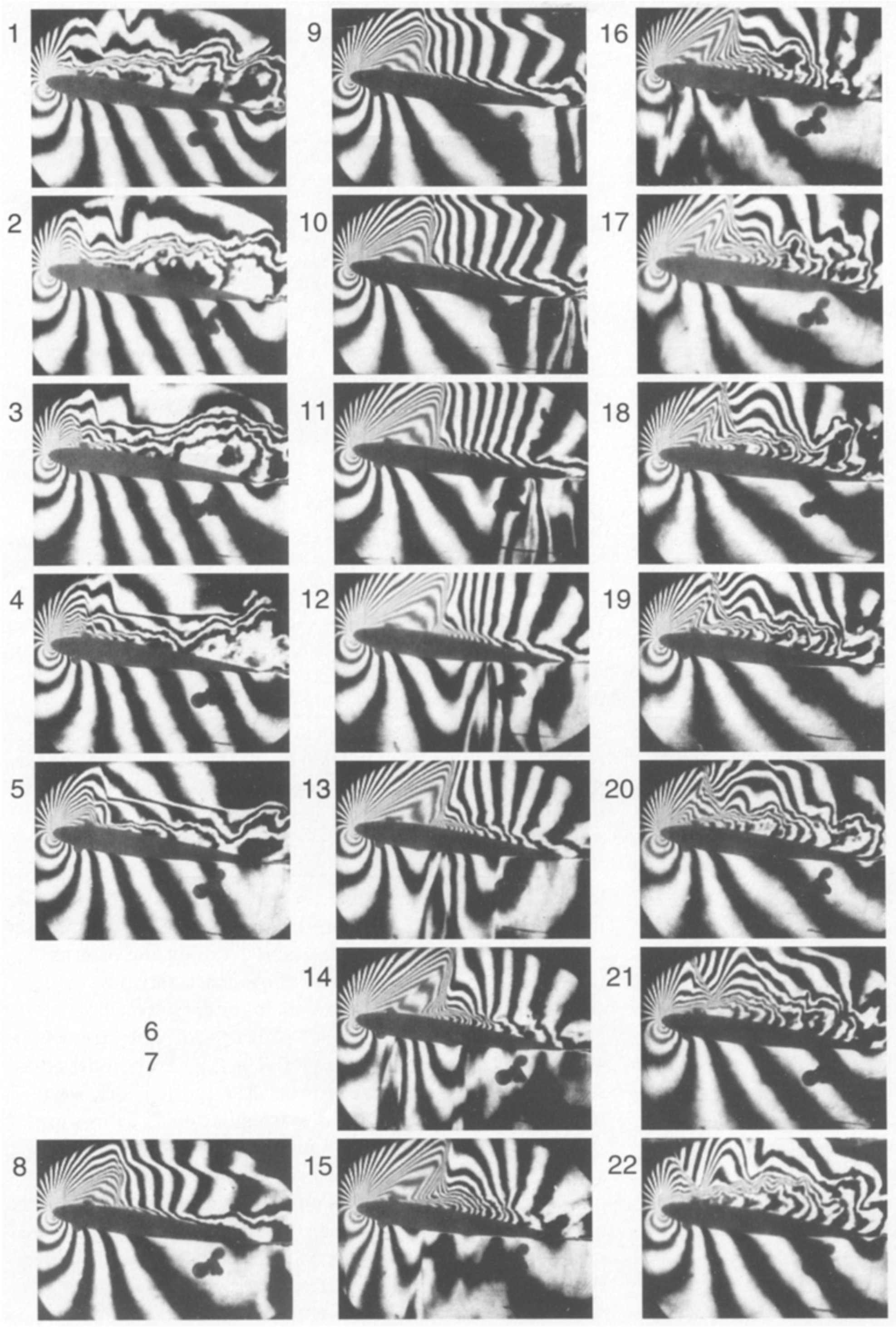


**Fig. 7.** Interferometric photographs of excited airfoil flow, $\alpha = 8.5°$, $f = 232$ Hz, $\Delta t = 0.2$ ms

situation is not stable. Due to high positive pressure gradient through the shock, the separation emerges again (frame 15). In the following phase the separation point moves upstream and finally reaches the leading edge (frame 22). The compression wave, which corresponds to the decelerated background flow, can be seen in frame 20 and subsequent frames. Frame 20 shows distinct disturbances at the bottom side and condensed interferometric fringes at the upper side of the airfoil. The compression wave steepens stronger in the high subsonic flow region close to the airfoil rather than in the background flow in its absence.

The airfoil flow behaviour described above shows that the background oscillations are apparently the controlling factor of the phenomenon studied. The eigen-oscillation mode, which is characteristic for the non-excited background flow case (see Fig. 4), seems to be of second order significance. The case when the forced and eigen-oscillation mode frequency are very close each other, is displayed in Fig. 6. The difference, which could

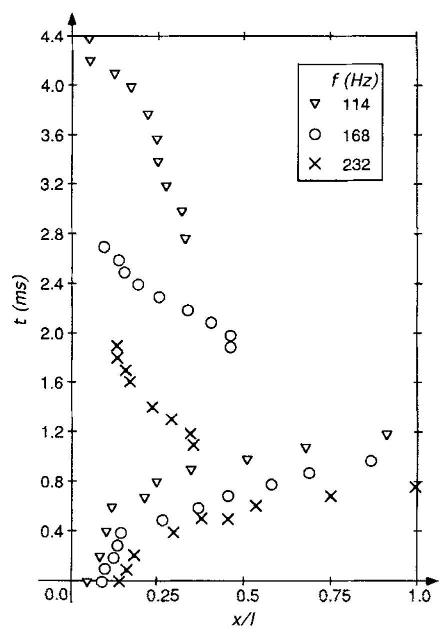


**Fig. 8.** Positions of the separation point for three oscillation frequencies

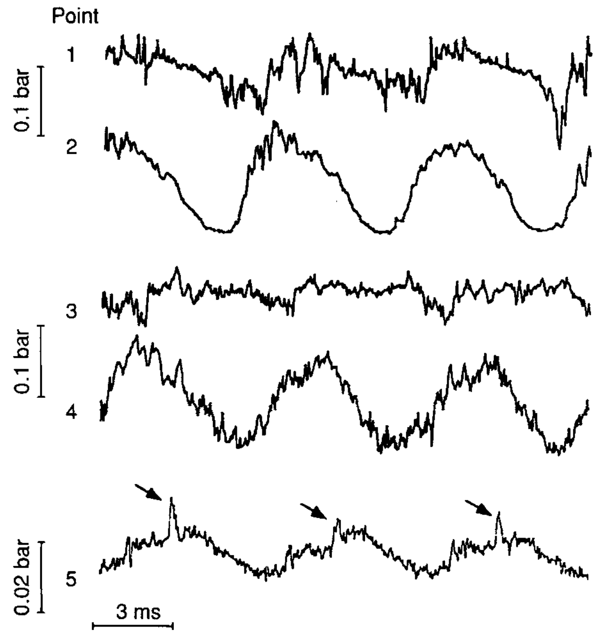


**Fig. 9.** Pressure signals at points shown in Fig. 3, $\alpha = 8.5°$, $f = 170$ Hz

be seen comparing the photographs in Figs. 5 and 6, exhibits a larger oscillation amplitude for $f = 168$ Hz than for $f = 114$ Hz. It is expressed by the larger spacing between the leading edge and the extreme downstream shock wave position, which is equal to about 0.5 (frame 19 in Fig. 6) and 0.4 (frame 14 in Fig. 5) of the airfoil chord length for $f = 168$ Hz and $f = 114$ Hz, respectively. This spacing for transonic airfoil flow is proportional to the shock wave strength.

For the forced oscillation frequency $f = 232$ Hz, which is now larger than that for resonance, the airfoil flow oscillation amplitude decreases again. In this case the shock wave reaches its extreme downstream position at 0.36 of chord length (frame 13 in Fig. 7).

Considering the flow photographs for all three excitation frequencies discussed above one can note that separation starts when the shock wave is of maximum strength during the oscillation cycle (frame 15 in Fig. 5, frame 20 in Fig. 6 and frame 14 in Fig. 7). The separation bubble, initially closed, expands rapidly in the downstream direction while the background flow acceleration decreases. As the downstream end of the separation bubble approaches the trailing edge, the back flow along the upper surface of the airfoil gains in strength. It results in faster upstream motion of the separation point (see frame 19 in Fig. 5, frame 24 in Fig. 6 and frame 17 in Fig. 7). In the case of $f = 114$ Hz, the separation appears (see frame 15 in Fig. 5) before the compression wave approaches the airfoil (frame 20 and the following). For higher frequencies (170 Hz and more), however, the separation begins as the shock is overtaken by the upstream propagating disturbance compression wave (frame 21 in Fig. 6). In these cases, the compression wave strengthens the shock wave and thus indirectly causes the boundary layer separation.

The history of the separation point positions for the three oscillation frequencies considered, found from the interferograms discussed above, are shown in Fig. 8. The series of three bottom and upper lines in this figure correspond to the acceleration and deceleration phase of the background flow, respectively. It can be noted that during the deceleration phase, the separation begins approximately at the half of the chord length for the oscillation frequency close to the eigenfrequency of buffeting (170 Hz).

Additional features of the considered oscillating flow behaviour can be deduced from the pressure signals measured at the five points (see Fig. 3) on the side wall of the test section (Fig. 9). The signals at points 1 and 2 exhibit the external (external to the boundary layer) flow characteristics, the signals at points 3 and 4 show the boundary layer flow behaviour, and the signal at point 5 shows the disturbances which are radiated upstream of the airfoil. The large amplitudes of the signals at point 2 and 4 appear because to the shock wave passes these points periodically. Irregular disturbances are superimposed on the distinct fundamental harmonic of these signals. The disturbances are very strong in regions corresponding to the separated flow phase. In this phase the flow over the low pressure side of the airfoil is characterized by large turbulence intensity (see Fig. 5). This effect is also visible in signals recorded at point 1. Significant increase of other disturbance amplitudes in this signal takes place when the separation bubble reaches this point (see Fig. 5).

The pressure signals recorded at points 1–4 are dominated by airfoil flow oscillations. The effect of the background flow disturbances alone seems to have only second order contribution. However, the pressure signal measured at point 5 is a different case. In this signal, the effect of airfoil flow oscillations exhibits distinct peaks (marked by arrows) superimposed on the fundamental harmonic of background flow oscillations. They appear as a result of the collapses of supersonic flow regions which occur periodically during the deceleration phases of the airfoil flow oscillations. In this case the sound wave, being a trace of the shock wave, (which

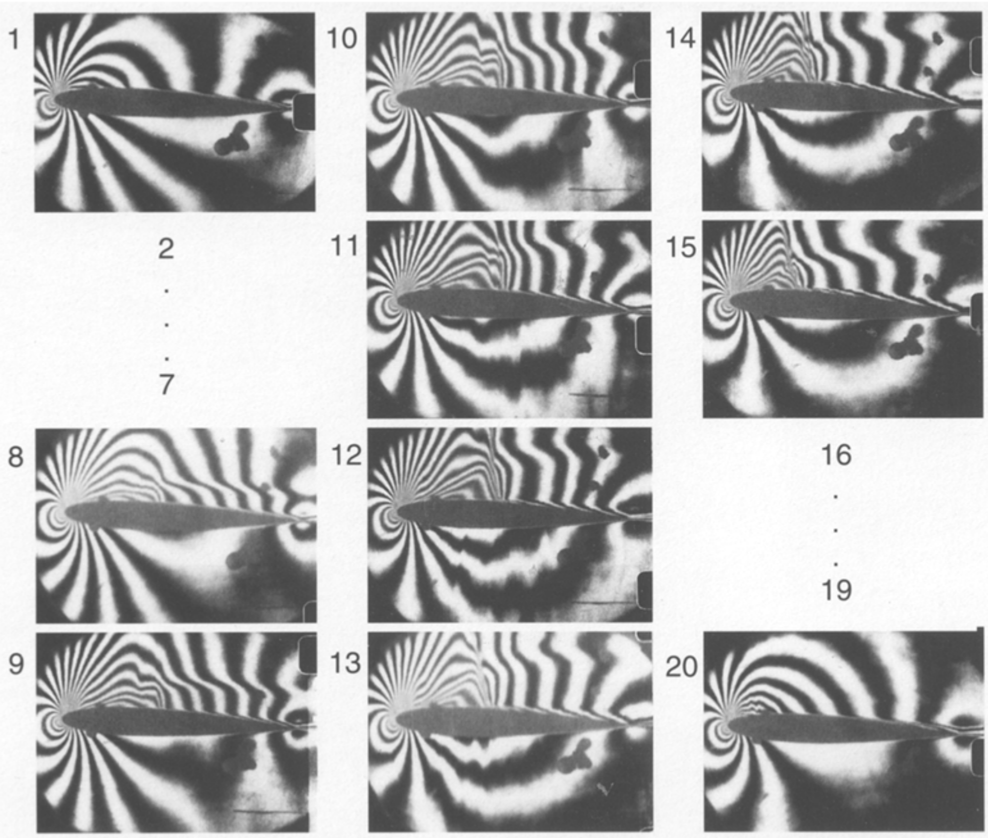


**Fig. 10.** Interferometric photographs of excited airfoil flow, $\alpha=5°$, $f=115$ Hz, $\Delta t=0.4$ ms

initially terminated the supersonic flow region) propagates upstream. A similar phenomenon is observed during the vortex-airfoil interaction in the transonic flow range (Lent et al. 1990).

Figure 10 shows the set of interferograms of the airfoil flow oscillations for the lowest airfoil angle of attack $\alpha=3°$ and $f=115$ Hz. In this case, the boundary layer remains attached during each oscillation phase. The characteristic feature of the flow is that the flow velocity increase through the expansion wave in the background flow does not lead to the supersonic region formation. Nevertheless, the shock wave also appears in this case. This is possible only due to the upstream propagating disturbance compression wave which steepens when it approaches the large velocity region on the airfoil surface (frame 9 and the following). However, the shock wave weakens again and finally disappears during its upstream motion through the region of decreasing flow velocity. The acoustic disturbances upstream of the airfoil does not occur in this case.

## 4
## Concluding remarks

The transonic airfoil flow appears to be very sensitive to external excitation. It exhibits strong shock wave and the boundary layer instabilities. The effect of subsonic flow excitation downstream of the airfoil is that the background flow is superimposed by upstream propagating expansion and compression waves. If the background flow acceleration through the expansion wave is strong enough, the flow which was initially separated attaches to the low pressure surface along its entire length. This effect was observed in the present experiments also for relatively high airfoil angle of attack ($\alpha=8.5°$).

The compression wave can affect the airfoil flow in two ways depending on $\alpha$. For relatively large $\alpha$, for which the flow is separated, the disturbance compression wave (which corresponds to the flow deceleration phase) causes the upstream displacement of both the separation point and the shock wave attached to it. In contrast, for low $\alpha$ corresponding to non-separated flow, the shock wave appears due to the steepening of the disturbance compression wave as it approaches the high velocity region on the suction surface of the airfoil. In this case, the shock wave is usually not accompanied by flow separation. For large airfoil angle of attack the weak shock waves, which are shed periodically, move upstream like a transonic sound wave.

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