High-speed tomographic PIV measurements of buffet flow over a supercritical airfoil

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Abstract In the present study, time-resolved tomographic particle-image velocimetry measurements are applied to investigate the transonic buffet flow over a supercritical DRA 2303 airfoil at a Mach number \( M_\infty = 0.73 \) and an angle of attack \( \alpha = 3.5° \). The freestream Reynolds number based on the chord length is \( 2 \times 10^6 \). To confirm the buffet model of Lee [12] and its extension [10], the analysis of the data is focused on the detection of vortical structures and their sound production in the region in the direct vicinity of the airfoil suction side. The vortical structures are detected by a two-point correlation of the velocity field and the sound source is captured by the three-dimensional Lamb being the major sound source. The obtained results are in good agreement with formulations of the buffet model.

1. Introduction

The transonic flow over supercritical airfoils is characterized by a local supersonic flow field on the upper suction surface that is terminated by a shock wave at its downstream end. At increasing Mach numbers the shock moves downstream and becomes stronger with a larger supersonic pocket. When the pressure rise across the shock is sufficiently large, shock induced separation of the boundary layer occurs. A highly complex and time-dependent flow pattern develops and for certain ranges of the Mach number or the angle of attack self-sustained shock wave oscillations, i.e., buffet, occur. Those low-frequency large-amplitude shock motions may cause severe loads acting on the wing structure and are therefore of great concern regarding future, highly flexible, light-weight wing structures for the transonic flight regime. Among other studies, a comprehensive review on the buffet issue was given by Tijdeman and Sebass [20] and Lee [12]. However, all studies show that the details of the mechanisms leading to buffet are not fully understood, yet. Furthermore, the numerical simulation of transonic buffet flow is still extremely challenging when the high Reynolds number range is considered.

Lee [12] proposed an explanation model for self-sustained shock wave oscillations in transonic flows with a fully separated boundary layer downstream of the shock wave. According to this model, small disturbances are formed due to the shock movement that propagate downstream in the separated flow region at a velocity \( u_d \) (figure 1):

![Figure 1: Transonic buffet flow over a supercritical airfoil according to Lee.](image-url)
When the pressure waves reach the trailing edge, new disturbances are generated that propagate upstream outside of the separated flow regime at a velocity $u_u$ as described by Voss [21]. On reaching the shock wave, the disturbances impart energy to the shock to maintain its oscillation. Thus, a feedback loop is formed and the frequency of the shock wave oscillation $f_s$ is given by the time it takes the shock induced pressure waves to reach the trailing edge plus the time it takes the disturbances generated at the trailing edge to propagate upstream to the shock wave [10]

$$f_s = \left( \frac{c-x_d}{u_u} + \frac{\sqrt{(c-x_d)^2 + x_u^2}}{u_u} \right)^{-1}.$$

According to the theory recently proposed by Crouch [4], buffet is driven by global instabilities of the flow field. It can be assumed, however, that the aforementioned feedback loop represents the physical mechanism of this instability.

Experiments conducted by Hartmann et al. [8, 9] showed that sound waves are generated due to the interaction of the shock induced disturbances with the trailing edge and that their varying sound pressure level leads to the shock movement. Depending on the sound pressure level of the waves reaching it, the shock is forced to move upstream or is able to move downstream again to its former position. The sound pressure level of the upstream propagating waves depends on the strength of the downstream propagating vortices that generate these waves by their interaction with the trailing edge. Furthermore, the sound waves undergo different refractions depending on the extension of the flow separation. Sound waves are more deflected when the separation is more pronounced, yielding a weakened sound pressure level which influences the shock wave [10]. Optical studies [6] showed a coupling between the shock position and the extension of the separation downstream. The separation oscillates synchronously with the shock such that it reaches its maximum extension when the shock is its most upstream position. Investigations by Stanewsky and Basler [17] revealed a small phase lag between the shock movement and the growth of the boundary layer separation. Based on these findings, Hartmann et al. [10] gave a detailed description of the mechanisms of the feedback loop.
Figure 2 shows the schematic of four phases of a shock oscillation cycle. The cycle starts with the shock in its most downstream position (figure 2a). The separation of the flow occurs furthest downstream. In the following, the shock moves upstream (figure 2b) and the separation area grows until the shock reaches its most upstream position (figure 2c). Then, it travels downstream again (figure 2d) and the separation region diminishes. During one oscillation cycle the shock wave moves at different velocities with respect to the mean flow. While moving upstream, the shock travels at its maximum relative velocity and during its downstream motion it travels at lower velocities [17]. The change of the shock velocity is indicated by the dashed ellipsoid arrows which also indicate the various shock positions in figure 2. The faster the shock relatively moves, the stronger is its strength. A stronger shock wave triggers a stronger shear and leads to a more pronounced vortex roll-up. Therefore, stronger vortices are generated, when the shock moves upstream. In figure 2b), this is indicated by the single strong vortex $\omega_{II}$. Note that the vortices are sketched larger than the separation for reasons of illustrations although they are located inside the separation region. In the following time-steps, this vortex propagates downstream and reaches the trailing edge when the shock is in its most downstream position (figure 2a), generating noise with a higher sound pressure level by passing over the trailing edge. The sound waves with an elevated sound pressure level will force the shock to move upstream. In summary, it can be said that the noise source in the trailing-edge region changes during one oscillation cycle of the shock, i.e., the sound pressure level of the generated sound waves becomes stronger and weaker. Additionally, it is supposed that the varying sound pressure level of the sound waves reaching the shock wave leads to buffet.

Acoustic analogies exist that allow to determine the acoustic pressure from flow quantities that can be easily measured or computed. In this context the Lamb vector, which is the cross product of vorticity and velocity, i.e., $L = \omega \times u$, and the Lamb vector divergence play a crucial role, since they appear as an acoustic source,
In the past, acoustic analogies have been applied to analyze trailing-edge noise. Ewert and Schröder [5] and Manoha et al. [14] investigated numerically the unsteady incompressible flow around the blunt trailing edge of a thick flat plate via large-eddy simulation. Curle's formulation [14], which presents a solution of Lighthill's wave equation for flows embedding solid bodies, and the acoustic perturbation equations [5a] were applied to compute the far-field radiated noise. It was shown that, although the Reynolds numbers didn't match, the noise prediction was consistent with the measurement results of the noise radiated by an airfoil with a blunt trailing edge in a quiet flow facility. Schröder et al. [16] investigated experimentally the trailing-edge noise on a flat plate model for chord based Reynolds numbers of $Re = 5.3 \times 10^6$ and $Re = 6.6 \times 10^6$. They applied time-resolved particle-image velocimetry simultaneously with trailing-edge noise emission measurement. The perturbed Lamb vector that presents the major vortex sound source term was derived directly from the TR-PIV velocity field quantities. The compressible flow past an 18% thick circular-arc airfoil for a Mach number of 0.76 and a Reynolds number of $1.1 \times 10^6$ was investigated numerically with detached-eddy simulation by Chen et al. [3]. The flow exhibits shock wave motions of type C as described by Tijdeman [19]. A shock is formed by coalescing weak compression waves, moves upstream, and finally exits from the leading edge. This phenomenon repeats itself and no downstream motion of the shock wave is observed. Due to the symmetry of the airfoil, this process is repeated alternatively between the upper and the lower surfaces. The acoustic character of the flow downstream the shock wave is analyzed based on the Lamb vector divergence distribution. It has been observed that the moving shock wave strengthens the turbulent shear layers and therefore, affects the acoustic character which can be seen in the changes of the Lamb vector divergence distribution.

To determine the Lamb vector three dimensional velocity data are required to analyze the sound pressure level in the trailing-edge region. Moreover, according to the feedback loop buffet model described above, it is expected that the sound pressure level varies. Therefore, time-resolved volumetric velocity data are needed. Hence, in this study time-resolved tomographic particle-image velocimetry measurements are performed to analyze the transonic flow field of a supercritical airfoil under buffet conditions to confirm the buffet models discussed in [12, 10].

2. Experimental setup

Experiments are carried out in the closed test section of the trisonic wind tunnel of the Institute of Aerodynamics. This wind tunnel is an intermittently working vacuum-storage tunnel providing flows at Mach numbers ranging from 0.4 to 3.0 with measurement times between two and three seconds depending on the Mach number. The turbulence intensity of the flow is below 1% [7]. Figure 3 shows a schematic of the tunnel facility. Prior to each test, the tanks are evacuated and the air is led through a silica gel based drier into the settling balloon. The relative humidity of the air is kept below 4% at a temperature of about 293 K such that any influence on the shock-wave position is avoided [2]. When the main valve of the tunnel is opened, the air is sucked out of the settling balloon, passes the test section and enters the tanks again. The total temperature and the total pressure of the flow are determined by the ambient conditions inside the settling balloon. Hence, the Reynolds number depends on the ambient conditions and the Mach number and ranges from $12 \times 10^6$ m$^{-1}$ to $14 \times 10^6$ m$^{-1}$. The square test section with cross section of 0.4 m $\times$ 0.4 m is equipped with flexible upper and lower walls to simulate unconfined flow conditions [15]. The wall contours are calculated by the one-step method solving the Cauchy integral based on the time-averaged pressure distribution measured along the centerline of the flexible walls [1]. In standard configuration, the wall mounted dynamic pressure transducers in the test section do not show any distinct resonance frequencies and the RMS value of the pressure fluctuations is below 1% [8].
The wind tunnel model is a two-dimensional supercritical laminar-type DRA-2303 profile which was one of the reference geometries in the Euroshock project [18]. The wing structure is made of an orthotropic ultrahigh modulus carbon fiber laminate sandwich shell which is lightweight, stiff, and may be assumed to be rigid. A beam, which is wider than the model itself, is incorporated inside the model and serves as a mounting for the airfoil inside the test section. The chord length of the wing measures $c = 150$ mm and the relative thickness to chord ratio is 14%. In the present investigation the incidence $\alpha$ with respect to the profile is 3.5°. Boundary layer transition is triggered on the suction side as well as on the pressure side at 5% chord by a 117 $\mu$m zigzag-shaped transition strip.

An overview of the tomographic PIV setup is shown in figure 4. DEHS droplets are used as seeding. The particles are introduced into the dry air in the settling balloon prior to a test run in order to achieve a homogeneous seeding distribution. Two Nd:YLF high speed lasers (Quantronix Darwin Duo 527-40-M and Quantronix Darwin Duo 527-100-M) are arranged on an extra mounting next to the tunnel to protect them from tunnel vibrations. Both lasers emit light with a wavelength of 527 nm at a frequency of 1000 Hz and a pulse separation time of 4 $\mu$s. The beam of the upper laser is reflected two times off a 45° mirror and is thereby overlaid with the beam of the underneath laser such that a higher energy level of the light beam is achieved. The conjunction of the light beams is shaped into a 6 mm thick sheet using light sheet optics. The light can enter the test section through a small slot in the upper wall of the freestream chamber that also serves as a knife-edge slit filter and cuts the low-energy fringes on the sides to obtain top-hat like light intensity distribution. A system of four cameras is used to record the forward Mie scattering images of the seeding particles in the trailing-edge region from different directions. Two Photon Fastcam SA5 and two Photon Fastcam SA3 (model 120K) CMOS cameras are installed symmetrically to the channel in Scheimpflug arrangement with a viewing angle of 28° and 44°, respectively. All cameras operated with a 13.5° angle with respect to the horizontal plane to achieve a viewing direction parallel to the profile surface in the trailing-edge region. Four 180 mm Tamron 1:1 lenses were mounted to the cameras. The images are taken at a frame rate of 2000 fps and a resolution of 1024 x 1024 pixels for the measurement time of 2.72 seconds.
The evaluation of the particle images is performed by the DaVis 8.20 software from LaVision. In a first step, a 3 x 3 sliding minimum was subtracted from the images to remove background noise. The viewing planes of all four cameras were matched using the images of a one-level target that was shifted within the volume and applying a transformation matrix (3rd order polynomial approach). This calibration was improved by a volumetric self-calibration as described by Wieneke [23]. The illuminated volume was reconstructed by an iterative multiplicative algebraic reconstruction technique (fastMART). The cross correlation was performed by a multi-pass algorithm with decreasing window size. The final window size contained 96 x 96 x 96 pixels with an overlap of 75%. A universal outlier detection [22] and a 3 x 3 x 3 Gaussian smoothing filter have been applied to the calculated vector field to detect spurious vectors. The coordinate system is chosen such that x and z are pointing along the chord and upward direction of the airfoil. The coordinate y points in the spanwise direction. Furthermore, the measurement volume was centered within the test section. The extension of the volume measures 0.753 ≤ x/c ≤ 1.273, -0.02 ≤ y/c ≤ 0.02, and -0.036 ≤ z/c ≤ 0.342 and the evaluation results in a vector spacing of 1.637 mm in the x-direction, 1.1817 mm in the y-direction and 1.1281 mm in the z-direction. The side view of the three-dimensional measurement volume is sketched in figure 5.
Figure 6a shows a photograph of the wing. To investigate the time-averaged pressure distribution on the airfoil surfaces, 45 pressure orifices on the suction side and 22 pressure orifices on the pressure side are used. The pressure orifices are arranged in the midspan region at $0.0267 \leq x/c \leq 0.9067$ in a distance of $\Delta x/c = 0.02$ on the suction side and at $0.0367 \leq x/c \leq 0.9$ in a distance of $\Delta x/c = 0.04$ on the pressure side. Figure 6b shows a sketch of the inner structure of the wing. The pressure data are acquired by a Scanivalve RAD 3200 pressure measurement system that is capable to measure 128 time-averaged pressure signals. To determine the fluctuating surface pressure, the airfoil model is instrumented with Kulite XCQ-080 subminiature pressure transducers. 15 transducers are inserted on the suction side in a range of $x/c = 0.1667$ to 0.9 and 4 transducers are inserted on the pressure side at $x/c = 0.2, 0.47, 0.73$ and 0.9.

An overview of the pressure measurement locations on the upper and lower surface is given in figure 8. The transducers are installed in closest proximity to the corresponding pressure orifice to minimize the damping and phase shift of the measured pressure signal. An illustration of the steady pressure orifices and the unsteady pressure transducers after mounting is shown in figure 7. The unsteady pressure data are recorded by a data acquisition system consisting of five Imtec T-112 data acquisition boards with simultaneous analog-to-digital conversion of 40 channels at 12 bit resolution with a sampling rate of up to 1.25 MHz per channel. In the present experiments, a sampling frequency of $f_{s,DAQ} = 20$ kHz is chosen. The signals are conditioned by a 4-pole Butter-worth low-pass filter with a 10 kHz corner frequency and are hundredfold amplified with a bandwidth of 100 kHz by Endevco 136 DC-amplifiers.
3. Results

From former experiments, e.g., [8], it is well known that the flow around the DRA 2303 airfoil exhibits distinct shock oscillations at a Mach number of $M_\infty = 0.73$ and an angle of attack of $\alpha = 3.5^\circ$. Hence, this parameter combination is chosen for the present study. The freestream Reynolds number based on the chord length is $2 \times 10^6$. Figure 9a shows the time-averaged chordwise pressure distribution on the two sides of the DRA 2303 airfoil. The horizontal line marks the critical $c_p$ value, such that the region where the pressure drops below this value represents the supersonic flow pocket on the suction side of the airfoil. The shock is located at the second intersection point of the $c_p$ line with the line that marks the critical value which is at $x/c = 0.45$. The steep pressure rise due to the shock wave is somewhat smoothed due to the shock oscillations during the measurement time. The pressure plateau downstream of the shock wave on the suction side indicates that the shock is strong enough to induce boundary layer separation. Figure 9b shows the frequency analysis of the pressure fluctuations at $x/c = 0.42$ on the suction side of the airfoil. The frequency $f_S$ is given as reduced frequency $\omega^* = 2\pi f_S c/u_\infty$ based on the chord length $c$ and the freestream velocity $u_\infty$. The amplitude spectrum of the FFT-analysis reveals an increased fluctuation level around $\omega^* = 0.68$ corresponding to the frequency $f_S = 170$ Hz. According to Tijdeman [19], this corresponds to an almost sinusoidal shock motion of type A at an oscillation frequency of 170 Hz.
In the following, the results obtained by the time-resolved tomographic PIV measurements are presented. The data are sorted depending on the shock wave position and presented in phase-averaged contour plots, i.e., on the one hand, the data are time-averaged over all time steps, where the shock is located most downstream, and on the other hand, the data are time-averaged over all time steps, where the shock is located most upstream. It was stated before that the extension of the boundary-layer separation is directly connected to the shock position, such that the extension becomes maximum when the shock is in its most upstream position and minimum when the shock is in its most downstream position. Since the shock wave is not included in the measurement volume, this criterion represents a tool for the phase-averaging in this study. Figure 10 depicts the contour plots of the phase-averaged velocity magnitude $|\overline{V}_{abs}| = \sqrt{U^2 + V^2 + W^2}$.

The angle of a streamline positioned above the separated area serves as a criterion for the phase-averaging.

As illustrated in figure 2, it is assumed that stronger vortical structures can be found within the separation region near the airfoil surface when the shock is in its downstream position. To detect the vortical structures, a two-point correlation of the velocity field is performed using the following equation:

$$R_{x_a,x_b} = \frac{(u'w')_{x_a,x_b}(u'w')_{x_i,x_i}}{\sqrt{(u'w')^2_{x_i,x_i}}}.$$
The contour plots in figure 11 show the phase-averaged distribution of the two-point correlation of the velocity, i.e., $R_{\text{vav}}$ contours are shown. The point $(x_a,z_a)$ is the fixed correlation point. Figure 11a reveals strong vortical structures within the separation region near the airfoil surface for the shock being in its most downstream position. The strong vorticity in the near wall region is the driving noise mechanism. In the contours in figure 11b, however, only weak structures are found in the shear layer of the separation and no strong vortical structures are found within the separation close to the airfoil surface.

In the following, the Lamb vector, which represents the vorticity-velocity cross product, is analyzed. The contour plots in figure 12 present the root mean square values of the perturbed Lamb vector: $L' = \omega' \times \bar{u} + \omega \times u'$. The results are phase-averaged depending on the shock wave position. In the contour plots of figure 12 the shock is located most downstream and in the contour plots of figure 13 the shock is located most upstream. First, the rms values of the absolute total Lamb vector perturbation $|L'_{\text{rms}}|$ are presented, followed by the decomposition into the components $L'_{x,\text{rms}}, L'_{y,\text{rms}}$ and $L'_{z,\text{rms}}$ in the x, y, and z direction. The analysis of the plots focuses on the region in the direct vicinity to the upper airfoil surface. Regarding the rms values of the absolute total Lamb vector perturbation in the first row of the contour plots, higher values can be found in the direct vicinity of the upper airfoil surface, i.e., when the shock is positioned downstream. The same also applies for the single components of the Lamb vector, where the component in the z-direction contributes the strongest to the high values of the absolute total Lamb vector perturbation when the shock is located in the most downstream location.

**Fig. 11a** Phase-averaged two-point correlation of the velocity field, at $y = 0, x_a/c = 1.0072, z_a/c = 4.1152 \times 10^{-4}$, shock in the most downstream position.

**Fig. 11b** Phase-averaged two-point correlation of the velocity field, at $y = 0, x_a/c = 1.02, z_a/c = 0.2465$, shock in the most upstream position.
4. Conclusion

The buffet flow over a supercritical DRA 2303 airfoil at a Mach number of $M_\infty = 0.73$ and an angle of attack of $\alpha = 3.5^\circ$ has been investigated experimentally by applying time-resolved tomographic particle-image
velocimetry to the trailing-edge region as well as steady and unsteady pressure measurements on the airfoil surfaces. The PIV data are presented phase-averaged, i.e., on the one hand, the data are averaged over the time steps, where the shock is in its most upstream position and on the other hand, the data are time-averaged over the time steps, where the shock is in its most downstream position. The two-point correlation of the velocity field reveals strong vortical structures in the near wall region of the suction side for the shock being in its most downstream position, whereas only weak structures can be found for the shock being in its most upstream position. The stronger vorticity in the near wall region is the driving noise production mechanism. In a further step, the rms values of the perturbed Lamb vector are analyzed. It has been shown that higher rms values of the perturbed Lamb vector can be found in the near wall region when the shock is located furthest downstream. These results support the feedback buffet model presented by Lee [12] and redefined by Hartmann et al.[13], where it is assumed that the shock movement is forced by the change of the sound pressure level of sound waves originating at the trailing edge. Furthermore, it is assumed that the change of the sound pressure level of the sound waves originating at the trailing edge is caused by the change in the strength of the vortical structures that are generated by the moving shock wave and propagate downstream where they produce sound waves when they pass over the trailing edge. It can be therefore summarized that when the shock is positioned downstream, stronger vortical structures are located in the near wall region of the suction side that will produce sound waves with a higher sound pressure level. These sound waves propagate upstream and will force the shock to move into the upstream direction.

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6. References