FORCED OSCILLATIONS OF A SUPERCRITICAL SWEPT WING IN TRANSONIC FLOW

P. C. Steimle, W. Schröder, M. Klaas

Aerodynamisches Institut,
Rheinisch-Westfälische Technische Hochschule Aachen
Wüllnerstrasse zw. 5 u. 7, 52062 Aachen, Germany

Introduction

Wind tunnel experiments have been performed with an oscillating swept wing in transonic flow to investigate unsteady aerodynamic phenomena as initiating mechanisms for aeroelastic instabilities occurring at modern transport type wing configurations. The dynamic shock – boundary layer interaction in the trailing edge region can produce strong pressure fluctuations acting on the wing structure as excitory load distribution. Modern transport wings with optimized light weight designs exhibit a distinctive structural response to unsteady loads, i. e. the eigen frequency of the first bending mode is in the region of the dominant frequencies in the aerodynamic pressure field. The dynamic fluid – structure interaction acts as a triggering mechanism for an unstable aeroelastic flutter response of the wing structure most often developing a metastable limit cycle oscillation (LCO) due to the inherently non-linear character of the transonic flow regime.

Aeroelastic instabilities are subject to various recent experimental (refs. 1-3, 6) and numerical research activities ref. (refs. 4-6). The objective of the current investigation is to deliver new steady and unsteady flow data from a three-dimensional flow case to assess the accuracy and validity of new computational models regarding the dynamic fluid – structure interaction and thus to contribute to the further development of numerical simulations. Furthermore, fluid dynamical mechanisms enabling aeroelastic instabilities and forming the nonlinear character of the transonic flow regime should be analyzed. A basic understanding of the instability phenomena on the aerodynamic part requires a reduction of mechanisms in the system of action and reaction. Therefore, the experiments only consider the flow around a wing in forced harmonic oscillatory motion separately in the pitch and heave degree of freedom, while suppressing the structural response to the aerodynamic forces with the use of a highly rigid wing model (ref. 7).

The test campaign presented here in this study investigates a two segment wing with the supercritical BAC 3-11/RES/30/21 airfoil geometry (ref. 8), which has a relative thickness of 11% at 39% of the chord length. Similar experiments with this airfoil in the two-dimensional flow have been published in refs. 9, 10. The wing has a leading-edge sweep of 34°, half span of 280 mm, aspect ratio of $\Lambda = 10.3$, and mean aerodynamic chord $c = 74.3$ mm, the largest chord length is 104.35 mm at the wing root (Fig. 1). To measure dynamic pressure distributions, the wing is equipped with 27 pressure transducers located at a relative half span $y/s = 0.286$. The pressure measuring area is limited to $0.051 \leq x/c \leq 0.785$ on the

Fig. 1 Swept wing model with supercritical BAC 3-11/RES/30/21 wing section
upper and $0.085 \leq x/c \leq 0.648$ on the lower side due to the diameter of the sensors. The laminar – turbulent transition of the boundary layer is fixed at $x/c = 0.05$ with a 0.1 mm transition strip. The wing model features an ultra high modulus carbon fiber shell to stiffen the structure in the bending degree of freedom. The maximum deformation is smaller than 1.65% of the wing span (ref. 7). The investigation was carried out in the Trisonic Wind Tunnel of the RWTH Aachen University, an intermittent vacuum storage tunnel with a 400 x 400 mm test section with adaptive upper and lower wall. The tunnel total pressure and temperature are equal to the ambient conditions, thus the Reynolds number depends on the Mach number and ambient conditions and ranges from 1.3 to $1.6 \times 10^7$ m$^{-1}$ in transonic flow.

**Dynamic shock wave boundary-layer interaction**

The unsteady aerodynamic phenomena have to be analyzed on the basis of a steady flow investigation (ref. 11). The time-averaged pressure distribution around the steady wing at $y/s = 0.286$ for a fixed angle of incidence of $\alpha_0 = 0^\circ$ and Mach numbers between $M_\infty = 0.80$ and 0.94 shows a compression shock immediately behind the pressure minimum on the upper and lower wing surface (Fig. 2).

A coating of the shear sensitive chiral-nematic liquid crystal mixture Hallcrest CN/R2 was used to qualitatively visualize the wall shear stress distribution on the upper wing surface (ref. 13). A description of the visualization setup can be found in ref. 14. Images of the liquid crystal coating show a growing trailing edge separation with increasing Mach numbers, which is already present at sub-critical incident flows. If a shock wave occurs, the separation immediately grows upstream to coincide with the shock footprint line (Fig. 3). This behavior is Mach number and incidence angle dependent and can be very well observed at a fixed $M_\infty = 0.84$ and changing $\alpha_0$ from $-1^\circ$ to $0^\circ$ (Fig. 4). Stronger shock waves developing at higher $M_\infty$ most likely cause a shock induced separation merging with the trailing edge separation. Pearcey et al. classify this behavior as shock induced separation type B3b (ref. 15).

The interaction between shock and separation is one of the most essential features of the steady as well as unsteady flow field leading to a distinct oscillation of the flow field due to shock buffet. This is independent from any forced excitation of the wing model. The flow at $M_\infty = 0.88$ and $\alpha_0 = 0^\circ$ with a well-developed shock wave on both the upper and lower wing surface exhibits a high pressure fluctuation level (Fig. 5a) peaking in the time-averaged shock locations (Fig. 5b). From the spectral analysis of the pressure fluctuations in the shock region on the upper surface (Fig. 5c) the shock buffet reduced frequency can be identified as $\omega^* = 2\pi f c/\bar{u}_\infty = 0.49$. This frequency clearly dominates the entire flow field and is also conveyed to the lower surface (Fig. 5e, Fig. 5f) around the trailing edge. A significant damping of the buffet frequency can be observed in the trailing edge separation, where the higher harmonics contain most of the spectral power density (Fig. 5d). This unsteady flow behavior resembles the type 3 flow described by Mundell and Mabey with high frequency fluctuations in the trailing edge region (ref. 16).

Finke and Lee also investigated shock buffet in two-dimensional flow around similar supercritical airfoils and measured shock buffet frequencies of $\omega^* = 0.5 \div 2.0$ on the 12 % thick NACA 63-012 and $\omega^* \approx 0.5$ on the 11.8 % BGK No. 1 airfoil (ref. 17, 18). The reduced buffet frequency on the BAC 3-11 wing seems to be typical for supercritical airfoils with this thickness ratio and comparable shock wave position. The close resemblance of the frequencies may also be a sign of a wide similarity between the buffet mechanisms in two and three-dimensional flow, at least for the steady wing flow.
Fig. 2 Mach number effect on time averaged pressure distributions on the upper surface (filled symbols) and lower wing surface (open symbols) at $y/s = 0.286$ for a fixed angle of incidence $\alpha_0 = 0.01^\circ$; $0.8 \leq M_\infty \leq 0.94$

Fig. 3 Chiral-nematic liquid crystal coating on the upper wing side, $\alpha_0 = 0^\circ$, $M_\infty = 0.881$, $Re_\infty = 3.89 \times 10^6$

Fig. 4 Time averaged position of the shock wave and the separation line at $y/s = 0.286$ for $-2^\circ \leq \alpha_0 \leq 3^\circ$ and $M_\infty = 0.80, 0.84, 0.88$ determined from pressure distributions and liquid crystal coating images.

Fig. 5 Analysis of time-resolved pressure distributions, $\alpha_0 = 0^\circ$, $M_\infty = 0.881$, $Re_\infty = 3.89 \times 10^6$ with pressure distributions $c_p(t)$ and time-averaged pressure distribution $c_p$ in (a), fluctuation level $\text{RMS}(c_p')$ of the pressure coefficient in (b) and power spectral density $|c_p'|^2/f$ of the pressure fluctuations with respect to the reduced fluctuation frequency $\omega^*$ in two chord locations on the wing upper side (c, d) and lower side (e, f).
Harmonic pitch oscillation experiments

First experiments with the unsteady wing model were conducted as forced harmonic rotary oscillations around an axis perpendicular to the incident flow (Fig. 1). Unsteady pressure distributions were measured synchronously to the model motion at the wing root for mean angles of attack $\alpha_0 = 0^\circ$ and $2^\circ$, amplitudes $\alpha_1 = 0.7^\circ$ and $1.4^\circ$, and excitation frequencies $f_\alpha = 45$ Hz and $60$ Hz. The Mach number range was $0.80 \leq M_\infty \leq 0.92$ corresponding to reduced oscillation frequencies between $0.07$ and $0.12$ (ref. 19). While the time averaged pressure distribution in the tap section remains almost unaffected by the harmonic pitch oscillation, the pressure fluctuations in the supersonic flow region are dominated by the fundamental oscillation frequency $f_\alpha$.

The spectral analysis shows a significant maximum at the reduced fundamental frequency $\omega_\alpha^* = 0.076$ (Fig. 6). The buffet frequency, which was identified in the steady wing flow, can also be found in the unsteady pressure field, allowing the assumption that the shock buffet mechanism itself remains unaffected by the unsteadiness of the wing. The fundamental frequency vanishes from the spectral analysis in the interaction region between shock wave and boundary layer separation. Despite of this observation, the strength of pressure fluctuations in the shock region are strongly influenced by the amplitude of the rotary oscillation. Fig. 7 compares the spectral analysis in the shock for the steady wing with the pitching wing with amplitudes of $\alpha_1 = 0.7^\circ$ and $1.2^\circ$. The power spectral density of the pressure fluctuations reduces with the smaller amplitude and is greatly amplified at the higher pitch amplitude.

The chord position of the separation line plays a key role in this interaction process. While amplitudes of $\alpha_1 = 0.7^\circ$ lead to an earlier separation, thus enabling the damping of fluctuations by the separated boundary layer, the flow tends to stay attached in the shock foot due to a higher amount of kinetic energy carried into the turbulent boundary layer by the oscillating wing surface. The damping effect of the trailing edge separation is independent from the frequencies in the boundary layer.

Fig. 8 displays the time averaged pressure distribution together with the root mean square of the fluctuating quantities and the first harmonic unsteady pressure distribution for small pitch amplitudes at $\alpha_0 = 0^\circ$, $M_\infty = 0.880$, where the shock strength on the upper and lower surface is almost equal. The damping can be observed in the comparison between the upper wing surface, where the separation interacts with the shock wave, and the lower surface with the attached turbulent boundary layer interacting with the shock. The fluctuating quantities are significantly lower on the upper side.
In the unsteady pressure distribution, the lower shock is represented by a significant peak of $c'_{p, \alpha}$ indicating a strong periodic flow motion in this area. This behavior coincides with high values of the phase angle represented by $c''_{p, \alpha}$. The damping effect of the trailing edge separation leads to much smaller real and imaginary parts downstream of the shock wave on the upper surface.

**Harmonic heave oscillation experiments**

The investigation of the rotary degree of freedom was expanded to the bending degree of freedom reducing the kinematically coupled bending and torsional swept wing deformation to a pure heave motion (ref. 0). The fluid mechanical parameters were the same as in the previous experiments, while the angle of incidence was fixed to $\alpha = \alpha_0 = 0^\circ$. The heave amplitude was $h = 0.1$, $0.5$, and $1$ mm corresponding to relative heave amplitudes of $h^* = h/c = 0.1\%$, $0.7\%$, and $1.5\%$ with heave frequencies of $f_h = 15$, $30$, $45$, and $60$ Hz. Unlike the pitch oscillations the heave oscillation frequency content in the pressure fluctuations is negligible. Consequently, it was impossible to influence the unsteady flow field directly with the heave motion. However, the pressure fluctuation strength in the boundary layer demonstrates to be very sensitive to the heave motion, even more sensitive than to the previously described pitch motion. The comparison between pressure fluctuations in the shock foot region for the steady and for the heaving wing with the three different heave amplitudes and two reduced heave frequencies (Fig. 9) indicates a dramatic change in the power spectral density of the reduced buffet frequency. Spectral powers are amplified by a factor of 10 when applying the smallest heave amplitude.

![Fig. 8](image_url) Time averaged $c_p$ (a), RMS$(c'_p)$ (b), real part $c'_{p, \alpha}$ (c) and imaginary part $c''_{p, \alpha}$ (d) of the first harmonic unsteady pressure distribution, $\omega^*_h = 0.076$, $\alpha_0 = 0^\circ$, $\alpha_1 = 0.7^\circ$, $M_\infty = 0.880$, $Re_\infty = 3.87 \times 10^6$

![Fig. 9](image_url) Power spectral density $|c'_{p, \alpha}|^2/f$ of the pressure fluctuations with respect to the reduced fluctuation frequency $\omega^*$ in the shock region $x/c = 0.652$ on the wing upper side for different reduced heave oscillation frequencies $\omega_h^*$ and relative heave amplitudes $h^*$. $\alpha_0 = 0^\circ$, $M_\infty = 0.881$, $Re_\infty = 3.89 \times 10^6$
This amplification reduces significantly for larger amplitudes, a behavior that is even better displayed by the pressure fluctuation PSD peaks related to the PSD of the heave oscillation in Fig. 10. It is also interesting that the buffet frequency changes from $\omega^* = 0.49$ to 0.43 in some oscillation parameter combinations, which may be an indication of a change in the buffet mechanism. The excitation of higher harmonic frequencies was described to be a sign of the presence of the trailing edge separation, which can be observed in the shock region at $\omega_0^* = 0.074$ at an amplitude of $h^* = 1.5\%$. Obviously, the separation again plays a key role in the damping effect of pressure fluctuations induced by the wing oscillation.

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