Experimental and Numerical Investigation of Unsteady Shock Wave-Boundary Layer Interaction and Shock Control Techniques for Wings

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Overview

• Introduction

• Current numerical simulations and wind-tunnel tests considering the unsteady shock wave-boundary layer interaction at the AIA

• Numerical Methods used at the VZLU

• Previous studies of shock control techniques for wings using bumps by the Universities of Sheffield and Manchester

• Joint proposal
Supercritical airfoils are characterized by a local supersonic region.

- The flow structure is determined by the classical problem of shock-boundary-layer interaction.
- This interaction results in an oscillating shock generating time-dependent load (buffeting phenomenon).
- Due to the light-weight design a distinctive structural response occurs which has to be understood for a more efficient design process.
- Experimental data close to free-flight condition is essential to validate new computational approaches for Reynolds numbers in the order of $10^7$. 

Introduction 1/2
The presence of shock-waves during transonic flight leads to an increase of drag, determining drag rise Mach number and limiting operation

- Shock control can extend the operational range for transonic flight by extending the drag rise Mach Number
- Shock control is also crucial for natural laminar flow wings as they require a significant part of favourable pressure gradient
- Previous studies in shock control show substantial drag reduction
Unsteady Shock Wave-Boundary Layer Interaction

Transonic flow around a BAC 3-11 airfoil

Results from 2C-PIV Measurements

$M_\infty = 0.86$, $\alpha_0 = 0^\circ$, $Re_\infty = 1.09 \times 10^6$, $\Delta t = 0.67$ms
Numerical approach: Zonal LES-RANS

- Flow regions where complex flow structures appear are simulated by an embedded LES
- Flow regions are coupled via LES- and RANS boundary conditions where flow variables are exchanged
- Reduction of number of grid points by 50% compared to full LES
- Exponential time window function was adjusted to buffet frequency to satisfy sufficient averaging time of turbulent flow features and proper transmission of pressure signal
Results: Transonic flow around a DRA 2303 airfoil

\[ Re_c = 2.6 \cdot 10^6, \quad M = 0.7, \quad \alpha = 3^\circ, \quad \frac{30.0}{10^6} \text{ gp (full LES)} \]
Power spectral density of pressure fluctuations for full LES and zonal RANS-LES at three locations

- Power spectral density of zonal LES part and full LES are in good agreement regarding buffet and turbulence spectra.
- Zonal RANS part captures buffet frequency and amplitude compared to full LES and experimental results (whereas $\omega_{\text{full RANS}} \approx 1.2$).

$Re_c = 2.6 \cdot 10^6$, $M = 0.72$, $\alpha = 3^\circ$, $14.0 \cdot 10^6$ gp (zonal), $30.0 \cdot 10^6$ gp (full LES)
Unsteady Shock Wave-Boundary Layer Interaction

Experiments: Test facility

**Trisonic wind tunnel**
Intermittent suction-type wind tunnel
Mach number range 0.4 - 3.0
Reynolds number range $10 - 15 \times 10^6$ m$^{-1}$
Measurement time 2 - 5 s
Adaptive test section 0.4 m x 0.4 m
**Experiment: Airfoil model**

Airfoil model comprises the supercritical type profile DRA 2303 which spans the entire test section. Relative thickness: 14%

Chord length: \( c = 200 \text{ mm} \)

Transition fixed at 5% chord.

Conventional pressure tabs and 11 in-situ pressure sensors Kulite XCQ-080 at upper surface at \( x/c = 0.45 - 0.7 \) (dist. 0.05c).

Flush mounted aluminum tape installed to withstand laser light exposure.
Experiment: Particle-Image Velocimetry

Illumination

High-Speed Laser Quantronix Darwin Duo 40M double-Pulsed Nd:YLF laser (527 nm)
Repetition rate: 1.5 kHz
Approx. 15 mJ per pulse
Light sheet thickness: 1 mm

$$f_s^{PIV} = 1.5 \text{ kHz}, \text{ 6 times higher than the highest expected fluctuation frequency}$$

Image recording

2 SA3 at 3 kHz in Scheimpflug condition
Measurement plane: 94mm x 40mm
Pulse distance: $$\Delta t = 3.4 \mu s$$
Mean displacement: 10 px (≈ 1 mm in the flow)

Synchronization of TR-PIV time-resolved pressure measurements with DAQ system
Maximum time deviation of 25 µs
Shock motion present with an amplitude of 5% chord.

Frequency analysis evidence a peak at \( \omega^* = 2\pi f \bar{c}/u_\infty = 0.68 \).

\[ M_\infty = 0.72 \]
\[ \alpha = 3^\circ \]
\[ Re_\infty = 2.63 \times 10^6 \]
Time sequence of pressure distribution and velocity field

\( M_\infty = 0.72 \)
\( \alpha = 3^\circ \)
\( Re_\infty = 2.63 \times 10^6 \)
\( \Delta t = 0.67 \text{ ms} \) (2\( \Delta t \) shown)

Sinusoidal shock motion resembles Tijdeman type A \([7]\)

Amplitude of 5% chord

Extension of the separation varies with the position of the shock wave \( \rightarrow \) influence on trailing-edge noise
Conclusions

• A fully coupled zonal RANS-LES approach was applied to a shock-boundary-layer interaction case. Results show good agreement with full LES results.

• TR-PIV and dynamic pressure measurements were simultaneously applied to unsteady flows at self sustained shock motion.

• Large-scale shock motion and boundary layer separation were resolved.

• At $Ma = 0.72$, $Re = 2.6 \cdot 10^6$, and $\alpha = 3^\circ$ numerical and experimental data showed a convincing agreement as to amplitude and frequency of shock oscillations.
EDGE – introduction 1/2

• CFD solver for 2D/3D viscous/inviscid, compressible flow problems on unstructured grids with arbitrary elements developed at FOI in Sweden
• EDGE adopts an edge-based formulation for arbitrary elements
• Uses a node-based finite volume technique to solve governing equations which are integrated explicitly toward steady state with Runge-Kutta time integration
• Convergence is accelerated with agglomeration multigrid and implicit residual smoothing
• A central spatial discretization is used for the convection of the mean flow and a second-order upwind scheme is used for the turbulence
EDGE - introduction 2/2

- Can be used in RANS, LES or DES mode
- Quantity of turbulence models are implemented
- Number of boundary conditions are implemented
- RANS mode with SST $k$-$\omega$ turbulence model is usually used at VZLU for external aerodynamics
- Main CFD solver at VZLU
Airfoil optimization by means of hybrid method GA+NM in MSES program

Results from Investigations at the VZLU

More information:
Shock Control Techniques for Wings

Shock Control Methods

• Passive Control (e.g. a perforated plate in the shock region without applying suction.)

• Active Control (e.g. ventilation through blowing or suction.)

• Shock control bumps (can be either active or passive)

Mechanism of a 2D bump
The RAE5243 NLF Aerofoil

- Natural laminar flow airfoil at transonic condition
- Design condition at $M = 0.68$ and $CL = 0.82$
The 2D bump design

- 4 design variables: bump height, position, length and crest position

- Bump added to the aerofoil shape

- Tangent at connection points and at the crest
Mechanisms of 2D shock control bumps

- Pressure contour lines of the datum case without bumps.

- The crest of the optimised 2D bump is placed ~3%c downstream from the strong normal shock wave.
Effects of the 2D bump on the shock wave

- The shock wave is converted into a weaker “knee”-shape shock.
- Its position has also displaced slightly downstream close to the bump crest.
Streamwise pressure distribution
3D shock control bump design

- Studies have shown that 3D bumps are potentially beneficial over a wider range of operating conditions than 2D bump, which can be a significant advantage.
Shock Control Bumps on a 2D Wing

Both 2D & 3D bumps are generated on a 2D wing with a constant Natural Laminar Flow aerofoil (RAE 5243) sections.
Streamwise pressure distribution

- The optimised crest position is ~5%c downstream of the original normal shock.
- Pressure variations across different spanwise locations.
Optimization of 3D bump shape

<table>
<thead>
<tr>
<th>Bump Design</th>
<th>Total Drag</th>
<th>Pressure Drag</th>
<th>Skin Friction Drag</th>
<th>Total Drag Reduction</th>
</tr>
</thead>
<tbody>
<tr>
<td>Datum wing</td>
<td>0.01622</td>
<td>0.01063</td>
<td>0.005586</td>
<td>-</td>
</tr>
<tr>
<td>Optimized 2D Bump</td>
<td>0.01326</td>
<td>0.007563</td>
<td>0.005700</td>
<td>18.2%</td>
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<tr>
<td>Optimized 3D Bump</td>
<td>0.01296</td>
<td>0.007208</td>
<td>0.005756</td>
<td>20.1%</td>
</tr>
</tbody>
</table>

Comparisons of optimized 2D and 3D bumps performances
Bump Operation Range

- Operation range is crucial for applications
- 3D bump offers a better off-design performance
Potential off-design problems

- Flow separation near the trailing edge of the 3D bump due to spanwise flow
- Performance at varying Mach number, altitude (Reynolds number), and lift conditions
- The robustness of the design - more robust design needs to be investigated
Proposed design

• A vortex generating shock control bump has been proposed (patent pending)
• It combines the shock weakening property of the 3D bump with its vortex generating capability
• The vortices suppress streamwise separation at the rear part of the bump at off-design conditions and also reduce the potential wing trailing edge separation.
• The robustness of the design can therefore be substantially improved
Joint Proposal 1/3

Experimental and Numerical Investigation of Unsteady Shock Wave-Boundary Layer Interaction

Measurements in ETW to continue current investigations at the AIA with a 3D model close to free-flight conditions and to validate the corresponding numerical 3D Simulation by the VZLU

• Higher Reynolds numbers and extended measurement time
• Greater model scales and smaller ratio of model/tunnel increase the accuracy and resolution of measurements and decrease tunnel wall interference
• Measurement of airfoil deformation resulting from buffeting (acceleration sensors, optical methods)
Joint Proposal 2/3

Experimental and Numerical Investigation of Shock Control Techniques for Wings

The University of Manchester will conduct the experiments in ETW while the numerical simulations are performed by the University of Sheffield

- Conduct detailed shock control vortex generating bump design for the experimental investigation
- In-depth numerical investigation of the flow physics at design and off-design conditions to prepare and provide guidance for the extensive experimental study
- Experimental study in ETW at flight Reynolds number to extract flow physics and validate the numerical simulation
- Experimental tests at a small range around the design point to test the robustness of the device
Joint Proposal 3/3

Existing BAC 3-11 Wing model of the SFB 401 to investigate the unsteady shock wave-boundary layer interaction and bump shock control technique.
Mechanisms of 3D bumps
Shock Control Techniques for Wings

Optimization of BWB with 2D bumps

- Larger number of design variables: 34 sections + 13 bumps = 657 design variables
- Optimized at a given lift condition with volume and pitching moment constraints
- Combine aerofoil profile optimization at master sections with additional shock control bumps for transonic performance
- Detailed study of shock bumps and winglet in the context of the whole aircraft