Aerospace Science and Technology ••• (••••) •••–•••

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

Pressure wave damping in transonic airfoil flow by means of micro vortex generators

M. Gageik [a,*], J. Nies [a], I. Klioutchnikov [a], H. Olivier [a,b]

[a] Shock Wave Laboratory, RWTH Aachen University, 52056 Aachen, Germany
[b] JARA – High-Performance Computing, 52425 Jülich, Germany

ARTICLE INFO

Article history:
Received 4 August 2015
Received in revised form 11 December 2016
Accepted 9 May 2018
Available online xxxx

ABSTRACT

In transonic airfoil flow, pressure waves are generated mainly at the trailing edge and in the case of a shock in the region of the shock/boundary layer interaction. Depending on the Mach number, these waves lead to oscillating shock waves and an unsteady pressure distribution. For a free steam Mach number of $M = 0.76$ and a chord length based Reynolds number of $Re = 10^6$, micro vortex generators (µVG) are applied to dampen pressure waves. This is studied experimentally in a shock tube and numerically by using a high-order finite difference scheme (under-resolved Direct Numerical Simulation). The agreement of the pressure distribution and Schlieren pictures between simulation and experiment is good. By means of numerical visualizations, instability waves are identified within the separated boundary layer above a marginal boundary layer separation bubble. The applicability of µVG for dampening the pressure waves and stabilizing the flow field is possible and is studied in this paper. By numerical Schlieren pictures and further visualizations, the flow around the VG is characterized. The spanwise oriented instability waves are partly disintegrated which is also confirmed by the analysis of the vorticity. Finally, the nonlinear wave propagation is investigated and an explanation for the typical 1 to 2 kHz pressure oscillation is given.

1. Introduction

In transonic airfoil flow, pressure waves are generated mainly at the trailing edge and in the case of a shock in the region of the shock/boundary layer interaction. In the present paper, the term pressure wave includes two kinds of waves. Firstly, sound waves that have very low or in the limit vanishing amplitudes, moving with the speed of sound and showing a purely linear behavior. Secondly, already steepened waves with higher amplitude which transform smoothly from sound waves into shock waves, due to the nonlinear effect of wave steepening [1]. Shock waves show arbitrarily high amplitudes, propagate faster than the speed of sound, behave nonlinearly and increase the entropy. This nomenclature is pragmatically motivated to simplify the description of waves in the transonic flow.

In the context of pressure wave analysis, three different characteristic flow regimes are identified by Nies and Olivier [2]. In the pure subsonic regime ($M_\infty < 0.72$) the whole airfoil flow is subsonic. Here, pressure waves, which are generated at the trailing edge, move freely upstream. In the intermediate region ($0.72 < M_\infty < 0.78$) a supersonic region occurs which prevents pressure waves from a further upstream movement. Consequently, pressure waves gather between the trailing edge and downstream of the supersonic region and merge. The merged waves show an increased amplitude. Additionally, the nonlinear effect of wave steepening transforms the pressure waves into weak shock waves. These shock waves increase their amplitude by the described merging processes until their propagation velocity is larger than the local flow velocity and consequently, the waves start moving upstream. The Mach number of the shock wave (SW) is determined by the ratio of the specific heats γ and the pressure jump over the shock p_2/p_1:

$$M_{SW} = \sqrt{\frac{\gamma+1}{2\gamma}\left(\frac{p_2}{p_1}-1\right)+1} \tag{1}$$

As long as the waves form a merged wave with sufficient strength, the shock wave can move upstream. But when the waves split, the shock Mach number $M_{SW} = f(p_2/p_1)$ decreases and the shock waves propagate downstream where they merge with new pressure waves. In a numerical analysis and visualization, this loop is confirmed and presented in this paper.

* Corresponding author.
E-mail addresses: gageik@swl.rwth-aachen.de (M. Gageik), nies@swl.rwth-aachen.de (J. Nies), klioutchnikov@swl.rwth-aachen.de (I. Klioutchnikov), olivier@swl.rwth-aachen.de (H. Olivier).

https://doi.org/10.1016/j.ast.2018.05.011
1270-9638/

Finally, in the transonic regime ($M_\infty > 0.78$) the supersonic region ends with a stationary shock. Here, the waves pass the stationary shock in the subsonic region above the shock. It is to be noted, that the transition from one to another regime is smooth and the Mach number ranges mentioned above were found for the considered BAC 3-11 airfoil for a chord length based Reynolds number of $Re_c = 10^6$.

The pressure waves cannot be considered as sound waves since they interact with the flow field and have nonlinear characteristics. Lee [3] studied theoretically their propagation in transonic flows using the nonlinear transonic small disturbance equation. At the Shock Wave Laboratory (RWTH Aachen University, Germany) the generation and propagation mechanisms were analyzed numerically and experimentally by several authors who presented their results in multiple publications. In the following, only a selection is given. First, Alshabu [4] and Hermes [5] studied the pressure waves at the BAC 3-11 airfoil. Then, a detailed study of the Mach number influence was performed by Nies [2]. Furthermore, Hermes and Nies [6] studied the pressure wave generation mechanism at a blunt trailing edge of a generic airfoil and the influence of its thickness. Hermes and Nies results are similar to those of Babucke [7] studying the mixing layer past serrated nozzle ends. Beside the aspect of generation and propagation of pressure waves, it is demonstrated that trailing edge modifications (serrated trailing edge) can decompose the spanwise oriented vortex-street of the wake into weak, fully three-dimensional vortices resulting in a reduction of the pressure wave amplitude. Additionally, Nies studied the influence of brushes attached to the trailing edge of the BAC 3-11 airfoil [8]. In the low transonic regime ($0.68 < M_\infty < 0.71$), a reduction of the pressure wave amplitude could be shown.

In this paper, a pressure wave reduction method for the secondly mentioned generation mechanism (shock/boundary layer interaction) is presented. Depending on the Mach and Reynolds numbers, for transonic airfoil flow, the boundary layer separates in the vicinity of the shock. For a laminar boundary layer and a sufficiently high Reynolds number upstream of the separation, a transition process starts. The boundary layer instabilities propagate into the shear layer (separated boundary layer). The shear layer rolls up and forms spanwise oriented, two-dimensional vortices. Since they are mainly developed in a shear layer, in the following they are simply called instability waves in order to distinguish them from the classical Tollmien-Schlichting waves. These vortices interact with the shock causing pressure waves.

Experimental and numerical investigations of the vortex/shock-interaction [9] show that due to this interaction the foot of the shock deforms and the vortices remain nearly unaltered. A Direct Numerical Simulation (DNS) of the vortex/shock-interactions at a free shear layer confirms the production of an acoustic field due to this interaction. Manning [10] identified the resulting upstream movement of the shock as source for pressure waves. In the present study, micro vortex generators (μVG) are utilized to optimize the shock/boundary layer interaction aiming at a reduction of the pressure waves. To damp the unsteady shock/boundary layer interaction, μVG are placed upstream of the shock. By stabilizing the shock, the generation of pressure waves is also dampened.

The μVGs are known to reduce the risk of separation and to modify the shock/boundary layer interaction [11,12]. They establish streamwise oriented wake vortices which transport energy-rich fluid into the boundary sublayer. Consequently, the boundary layer profile becomes fuller and therefore can withstand higher positive pressure gradients without separation. When applied to the presented problem, the vortex system induced by the μVG interact with the spanwise oriented, two-dimensional vortices. This interaction results in the disintegration of the two-dimensional structures into small three-dimensional ones. It is expected that the resulting shock/boundary layer interaction is more steady and the pressure waves are dampened.

The present investigation was performed numerically (under-resolved DNS (UDNS)) and experimentally (modified shock tube). By conducting experiments and performing UDNS, the airfoil flow of the BAC 3-11 with and without μVG is investigated experimentally and numerically. Configurations with and without μVG are compared for the same Mach and Reynolds numbers.

In the present study, the free stream Mach number is $M_\infty = 0.76$ for experiments and simulations. A variation of the Mach number is not presented because of two reasons. First, a variation of the Mach number would require an adaption of the position of the μVG since the position of the spanwise oriented vortices would change. The position has been optimized for this Mach number and multiple variations would require an enormous effort for the experiments. Second, although only numerical results of UDNS and not fully-resolved DNS are presented, these are very expensive and more than two simulations were not possible to perform with the available computer resources.

Because no strong dependence of the pressure wave behavior on the Reynolds number is found in the range of $1 \cdot 10^6 < Re_c < 5 \cdot 10^6$ [13], for the following experiments and simulations the Reynolds number is set to $Re_c = 10^6$. Furthermore, the angle of attack is zero for all investigations.

In the following the pressure waves are analyzed by comparing the standard deviation of the pressure histories at various positions along the airfoil. The standard deviation of the pressure histories is an appropriate physical parameter since all kinds of fluctuations are included. Especially, the simultaneously occurring sinusoidal (sound wave) and saw tooth (shock wave) wave forms and changing wave frequencies complicate the use of many other statistical data processing methods. The standard deviation σ of the pressure fluctuations is calculated as,

$$\sigma = \sqrt{\frac{1}{N-1}\sum_{n=1}^{N}(p-\bar{p})^2} \tag{2}$$

with N being the number of samples. The instantaneous and time averaged pressures are denoted as p and $\bar{p}$. Both are related to the free stream pressure p_∞ and therefore nondimensional. Nies and Olivier [2] demonstrated that the qualitative and quantitative agreement of the standard deviation with the amplitude of the dominating pressure waves (1–2 kHz) found by FFT of the experimental data is very good. This confirms the usefulness of the standard deviation for this study. Additionally, time averaged pressure and skin friction coefficients and a couple of flow visualizations (e.g. Schlieren images) are presented.

The numerical method and the experimental setup are described in Section 2, the geometry and further characteristics of the μVG are presented in Section 3 and the achieved results in Section 4, where numerical results are discussed in detail and compared to experimental ones.

2. Numerical method and experimental setup

2.1. Experimental setup

2.1.1. Shock tube

Experiments have been conducted in a modified shock tube (STK) for the testing of transonic airfoils. The shock tube consists of five main parts, the high pressure part, followed by the double membrane chamber, which is connected to the low pressure part and the following test section with a vessel attached at its end. The rectangular test section holds cookie cutters to remove the shock tube boundary layer on the top, bottom and side walls