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SELECTED ASPECTS OF INTRODUCTION OF A LAMINAR-FLOW WING INTO TRANSPORT AVIATION

Abstract

The paper presents results of numerical simulation of turbulization of boundary layer interacting with a shockwave on laminar airfoil in transonic flow. Three configurations were tested using a "2.5-dimensional" flow model: baseline configuration of clean airfoil and two configurations using micro-vortex generators submerged in the airfoil boundary layer in front of the shockwave. Unsteady Reynolds-Averaged Navier-Stokes equations were solved using as a closure a four-equation turbulence model capable of resolving laminar-turbulent transition. It was shown that micro-vortex generators submerged in the boundary layer are capable of eliminating flow separation zone at the foot of the shockwave and in effect, of achieving lower aerodynamic drag at the same lift force that occuring at natural laminar-turbulent transition at the shockwave

INTRODUCTION

Laminar wing is characterized by stable presence of laminar flow up to, approximately, the middle region of wing chord resulting in low friction drag. Low-friction-drag properties make this solution attractive for applications beyond the low-speed aircraft design range where this technology originated and in the recent years the sector of large transport aircraft, moving at transonic speed, is a new target for application of the laminar flow technology. Presence of supersonic flow velocity over a large part of wing at transonic flight-speed range may lead to the rise of strong shockwaves detrimental to performance and aircraft safety. Alleviation of harmful phenomena such as high drag rise or unsteady shockwave-boundary layer interaction, caused by shockwave-laminar-boundary-layer interactions leading to oscillation of shockwave position known as buffet phenomenon is a subject of work in European FP7 Project TFAST. The paper presents approach to alleviation of these problems by application of flow control in the boundary layer.

1. SHOCKWAVE-BOUNDARY LAYER INTERACTION ON BASELINE AIRFOIL

A case study for shockwave-boundary layer interaction occurring for laminar flow in the TFAST project is flow over a laminar airfoil specially designed for numerical and experimental investigations. The numerical investigations presented here were conducted by solution of Unsteady Reynolds-Averaged Navier-Stokes equations using ANSYS Fluent 14.5 solver with four-equation Transition SST turbulence model capable to simulate laminar-turbulent transition phenomenon [1]. The side-view of computational domain is shown in Fig. 1. The basic version of computational domain included constant-chord section of wing model of width allowing for subsequent application of a single vortex generator in the modified configuration with distance of 5% of its span on each side to the side walls of the domain where symmetry boundary conditions were implemented. By this choice of boundary conditions the computational model represents conditions of three-dimensional flow over a section of a infinite-span-wing of constant chord. This kind of model is often called a two-and-a-half-dimensional model because of three-dimensional flow model and constant-chord wing model based on two-dimensional airfoil. The top and bottom sides of computational domain represented solid walls for consistency with the results of subsequent wind-tunnel investigations.

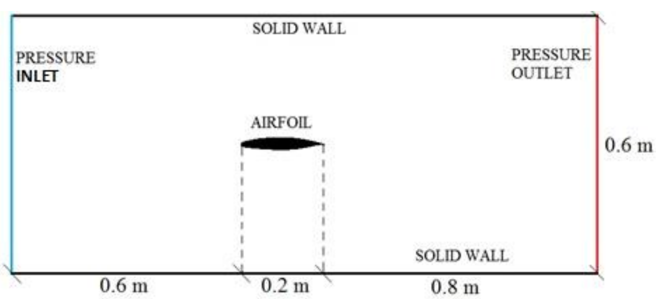

Fig. 1. *Side view of computational domain for investigations of shockwave-boundary layer interactions for V2C airfoil.*

At free-stream Mach number of 0.7 the flow is characterized by the presence of a strong shockwave at approximately 45% of airfoil chord, seen in the Mach contour in Fig. 2 and rise of pressure coefficient in Fig. 3 and in Fig. 4.

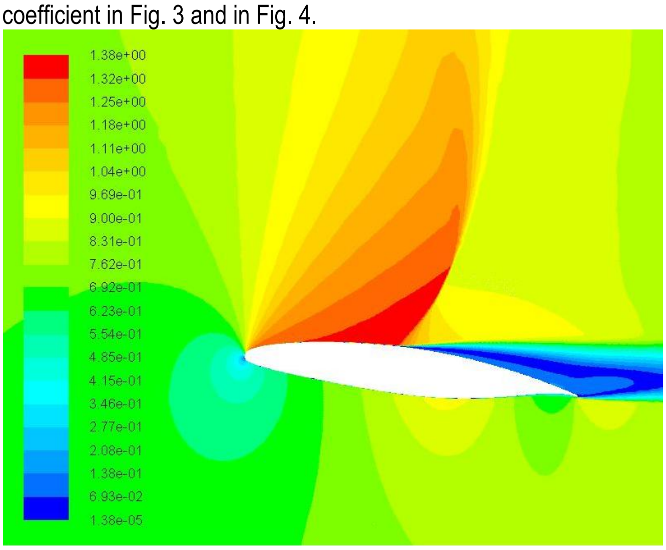

Fig. 2. *Mach number contour in the boundary layer in the vicinity of the shockwave. Mach number M=0.7 and angle of attack α=7°.*

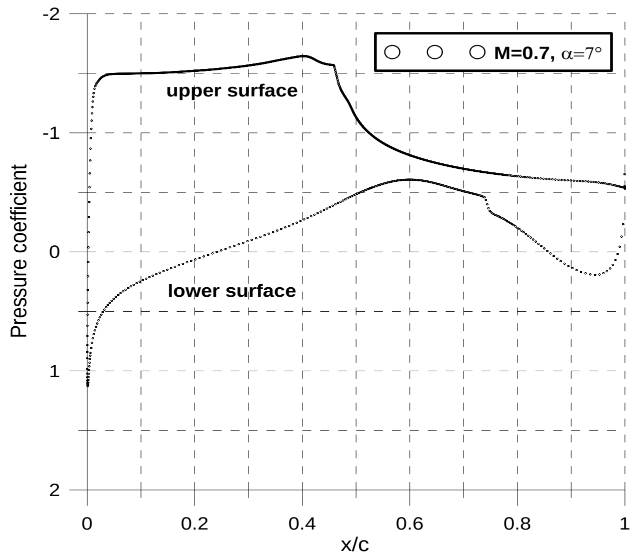

Fig. 3. *Pressure coefficient distribution in the symmetry plane of the V2C airfoil at Mach number M=0.7 and angle of attack α=7°.*

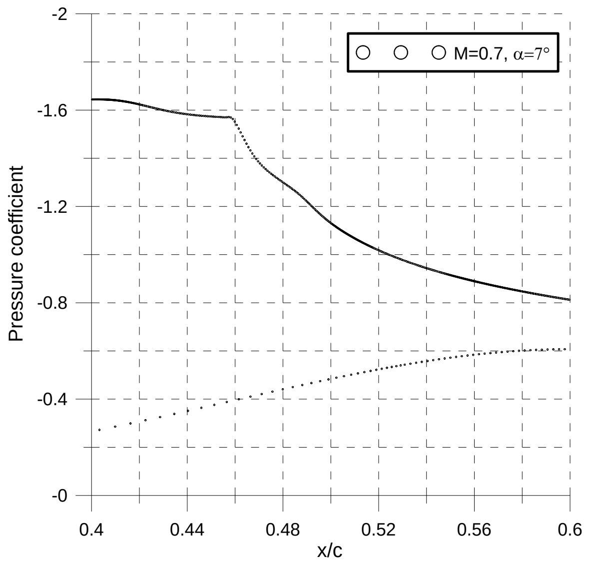

Fig. 4. *Enlarged fragment of distribution of Pressure coefficient distribution in the shockwave region.*

The laminar-to-turbulent transition process in the presence of a shockwave is shown in Fig. 5 and in Fig. 6. At the foot of the shockwave there occurs a separation region over which the flow becomes turbulent, which can be seen as rapid rise of intermittency (probability of boundary layer being at any moment turbulent) in Fig. 5. It is typical for shockwaves occurring in laminar flow that at the foot of the shockwave there appears a separation bubble and a subsequent reattachment of flow with turbulent boundary layer. However, for the investigated airfoil at Mach number M=0.7 and angle of attack α=7 degrees the shockwave is very strong and the flow, which is separated at the shockwave foot, remains separated over the rest portion of the airfoil, which can be seen as negative values of tangential stress exerted by fluid on the airfoil surface in the X-direction (chordwise, towards trailing edge) in Fig. 6.

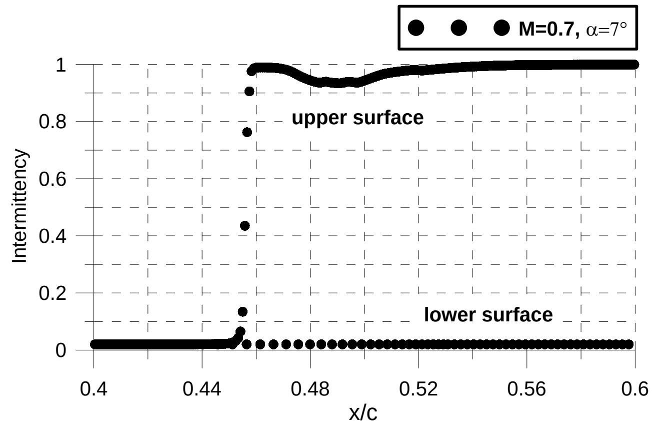

Fig. 5. *Intermittency profile in the boundary layer in the vicinity of the shockwave. Mach number M=0.7 and angle of attack α=7°.*

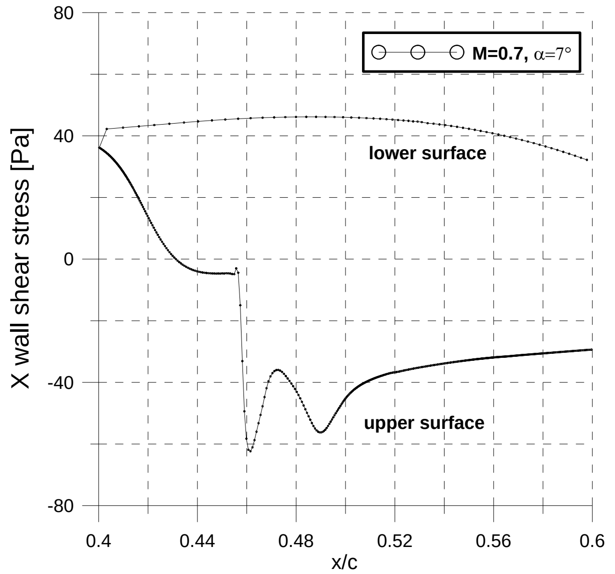

Fig. 6. *Wall shear stress in the symmetry plane of the V2C airfoil at Mach number M=0.7 and angle of attack α=7°.*

2. MODIFICATION OF SHOCKWAVE-BOUNDARY LAYER INTERACTION

Shockwave-boundary-layer-interaction effects are usually much weaker in case of turbulent boundary layer than in a case of laminar flow. Therefore it is expected that enforcing laminar-to-turbulent transition at some distance in front of the shockwave should lead to reduction of the separation region and improvement of airfoil characteristics.

Apart from this it can be shown that enforcing turbulization of boundary layer interacting with shockwave may lead to less intensive buffet oscillations than occurring for laminar boundary layer reaching the shockwave position [2]. The effects of turbulization of boundary layer in front of the shockwave on airfoil aerodynamic characteristics were investigated through application of a vortex generators in chosen locations in front of the shockwave. The geometry of the investigated vortex generator (VG) is shown in Fig. 7. Its height was chosen such that it is fully submerged in the boundary layer and at the same time Reynolds number based on its height exceeds 600, which, according to experimental practice should trip laminar boundary layer [3].

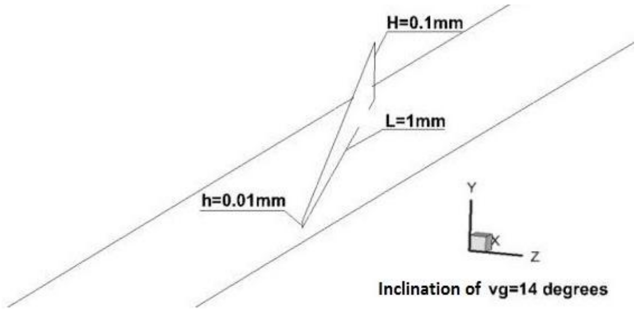

Fig. 7. Geometry of a delta-shaped vortex generator used for modification of of shockwave-boundary layer interactions.

Two locations of VG were investigated: 30% of airfoil chord and 40% of chord. The corresponding distributions of pressure coefficient are compared with distribution of pressure coefficient of baseline clean V2C airfoil in Fig. 8. It can be seen that a portion of the pressure distribution of baseline airfoil corresponding to the laminar separation region at 40-50% chord is replaced by a stronger peak of suction (negative values of pressure coefficient) which is caused by the removing of a separation bubble. Similar type of pressure distribution near the shockwave was obtained for V2C airfoil in [4] assuming fully-turbulent flow analyzed using two-equation turbulence models. This is confirmed by a velocity distribution in 43% chord shown in Fig. 9 and by distribution of tangential stress in X direction shown in Fig. 10, where it can be seen, that the small separation region in front of the shockwave (negative values before x/c=0.45) present in the baseline V2C case is removed in configurations using VGs.

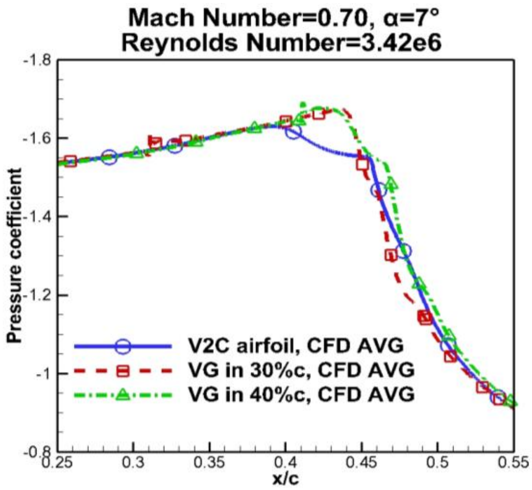

Fig. 8. Effects of application of a vortex generator in front of the schockwave on the presure coefficient distribution in the region of the shockwave.

Turbulization of the boundary layer by the VGs is shown in terms of rise of intermittency and turbulent kinetic energy in Fig. 11 and in Fig. 12. Both these quantities were detrmined at the distance from the surface of 0.15mm, that is, above the VG. It can be seen that for the VG located at 30% of chord intermittency rises to the level of 0.9 which means that the boundary layer is transitional, but almost fully turbulent. It can be seen also, that the position of 30% chord is close to optimum for the VG, as the effects of its action are fully developed several percent of airfoil chord behind the VG. The effect of VG located in 40% chord on intermittency is much weaker. However, the rise of turbulent kinetic energy presented in Fig. 12 is in both cases rapid and similar in value.

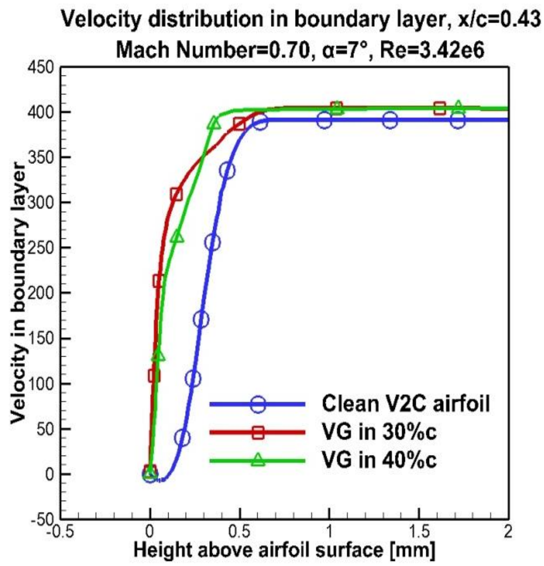

Fig. 9. Velocity distribution in boundary layer in 43% of airfoil chord for three investigated cases.

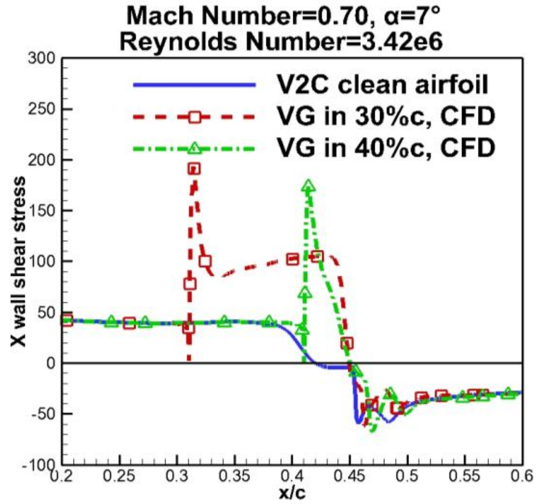

Fig. 10. Tangential stress in X direction for three investigated cases.

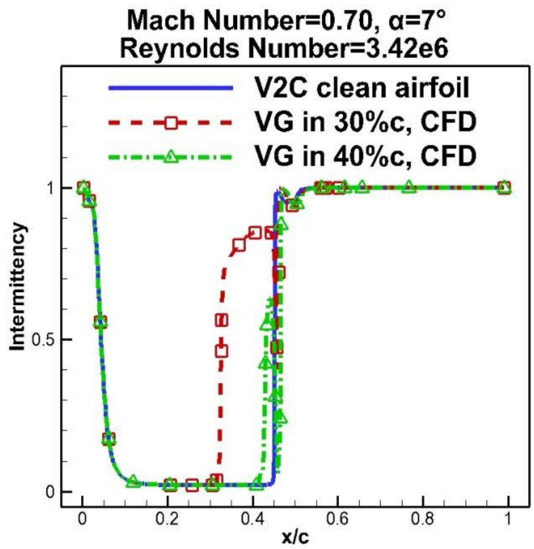

Fig. 11. Rise of intermittency in the boundary layer at a distance from surface of 0.15mm for three investigated cases.

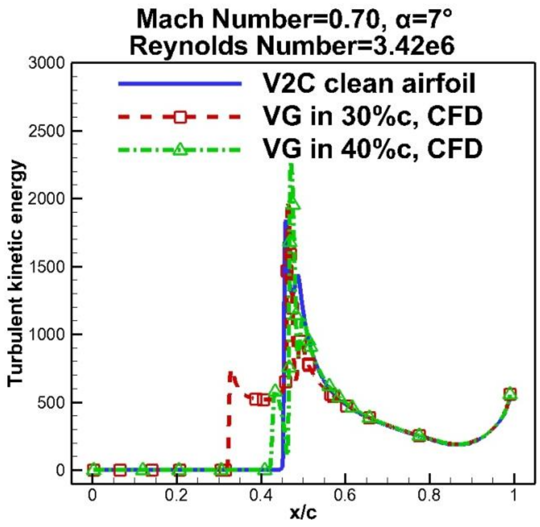

Fig. 12. *Rise of turbulent kinetic energy in the boundary layer at distance from surface of 0.15mm for three investigated cases.*

The effects of the VGs on distributions of pressure and velocity in the boundary layer, as small and local as they are, have nevertheless impact on the aerodynamic characteristics of the airfoil. The increase of upper-surface suction shown in Fig. 8 is responsible for increase of lift coefficient C_L from 0.9560 to 0.9596, by 0.38% in case of VG located in 30% chord and to 0.9659, by 1.04% for VG located in 40%chord. The change of drag is also positive, mainly due to the introduction of a VG - an obstacle to the flow. For VG located in 30% chord drag coefficient C_D rises from 0.1081 to 0.1084, by 0.26% and for VG located in 40% chord to 0.1092, by 1.05%. It must be remembered, however, that from the view of practical applications the most important is change of the lift-to-drag coefficient which is +0.12% for VG located in 30% chord and -0.011% for VG located in 40% chord. Positive change of lift-to-drag coefficient means that an aircraft of fixed mass needs, after modification of wing, lower lift and produces lower drag than the baseline configuration, which is an advantage that can be transferred to economic gains by lower fuel costs or increased range.

CONCLUSIONS

It can be concluded that modification of baseline shockwave-boundary layer interaction by application of vortex generators submerged in the boundary layer leads to qualitative change of the interaction between a boundary layer and a shockwave. Through the application of micro-vortex generators placed in front of the shockwave the laminar boundary layer was turbulized and laminar separation region at the foot of the shockwave was eliminated. The changed pressure distribution in the shockwave zone had higher local suction and in consequence airfoil lift was increased. The total drag change was positive due to the addition of a flow disturbance in the form of a vortex generator, but for one of the investigated configurations the lift-to-drag ratio was increased which means that for fixed lift force, equal to lift of the baseline configuration the drag force is decreased.

BIBLIOGRAPHY

1. *Ansys Fluent Theory Guide*, ANSYS Inc. Southpointe 275 Technology Drive Canonsburg, PA 1537.
2. Sznajder J, Kwiatkowski T. ,*Effects of turbulence induced by micro vortex generators on shock-wave – boundary layer interactions*, Journal of KONES Powertrain and Transport, Vol 22, No.2, pp. 241-248, 2015, ISSN: 1231-4005.
3. Braslow A.L., Knox E.C., *Simplified Method for Determination of Critical Height of Distributed Roughness Particles for Boundary Layer Transition at Mach Numbers From 0 to 5*, National Advisory Comitee for Aeronautics, Technical Note 4363, Washington, September 1958.
4. Grossi F. *Physics and modeling of unsteady shock wave/boundary layer interactions over transonic airfoils by numerical simulation.* Ph.D. dissertation , Institut National Polytechnique de Toulouse, 2014.

WYBRANE ASPEKTY WPROWADZANIA SKRZYDŁA O OPŁYWIE LAMINARNYM DO LOTNICTWA TRANSPORTOWEGO

Streszczenie

W artykule przedstawiono wyniki numerycznej symulacji turbulizacji warstwy przyściennej oddziałującej z falą uderzeniową w przypadku laminarnego profilu lotniczego w opływie transonicznym. Badano trzy konfiguracje stosując „2.5 wymiarowy" model opływu: konfigurację podstawową obejmującą gładki profil i dwie konfiguracje w których zastosowano miniaturowe generatory wirów w warstwie przyściennej przed falą uderzeniową. Wykazano że miniaturowe generatory wirów o kształcie "delta" umieszczone w warstwie przyściennej są w stanie wyeliminować strefę oderwania opływu pod falą uderzeniową i w efekcie uzyskać mniejszy opór aerodynamiczny przy tej samej sile nośnej co w przypadku swobodnego przejścia laminarno-turbulentnego na fali uderzeniowej.

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