Numerical modeling of the buffet onset in 2-D and 3-D flows.

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Abstract

Presented are buffet onset in 2-D and 3-D numerical investigation. Programme ANSYS was used to provide computations. 2-D Navier-Stokes equations solutions with the turbulence model have been obtained for the profile NACA 0012, solutions for three-D Navier-Stokes equations have been obtained for the wing ONERA M6. For 2-D flows boundaries of regimes characterized by shock selfoscillations have been determined for different Mach numbers and angles of attack .

Presented is quantative buffet onset physical mechanisms analysis. Investigated are some regimes of the boundary layer flow control like blowing, suction, surface heating and cooling to provide buffet onset boundaries change.

New methods of selfoscillated flow regimes control with the thing films application are discussed.

In some cases depending on Mach number and angle of attack shock wave selfoscillations can arise. Corresponding flow regimes are investigated numerically for 3-D flows on basis of SST turbulence model.

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1 Introduction

For transonic flow regime shock wave may be formed near upper wing surface of an airplane. The shock wave may lead to the formation of the boundary layer separation and as a result to the shock wave selfoscillacions (buffer onset). It was shown that Navier-Stokes equations with the appropriate turbulence model can be used to describe such flows. . Numerical experiments [1] and comparison with experiments have allowed to identify some (mainly differential) turbulence models adequately reflecting characteristics of unsteady separated flows. As well in some papers LES approach was used [2] and DNS modelling

2. 2D Flows

Flight of an aeroplane with the transonic cruise regime may be associated with selfoscillations origination due to the boundary layer flow interaction with the shock wave.

This flow regime is often called as a buffet onset. This phenomenon investigation is a subject of the report. Example of previous investigations were published in AGARD series [3] as well as in other papers and books. In the paper quantative analysis was done but main results were obtained numerically on the basis of Navier-Stokes diest numerical solution along with the Navier-Stokes equations with the appropriate turbulence model solution.

Supposed flow structure allowed to develop appropriate similarity parameter to calculate buffet onset arising. It may be supposed that significant similarity parameter is associated with the ratio of shock wave pressure rise and the pressure rise leading to the laminar or turbulent boundary layer separation

$$N = \frac{\Delta p_1}{\Delta p_2}.$$

The first pressure rise depends on the Mach number upstream of the shock depending on profile geometry and angle of attack. The second one may be calculated on the basis of approximate relations describing boundary layer separation known in the literature.

Most of numerical results have been obtained numerically for Naca0012 profile. Computational grid was created with the help of ICEM-CFD code.

For computations CFX code was used with the SST turbulence model. Main results were obtained for Mach number of undisturbed flow: $0.2 \le M \le 1.0$, angles of attack: $0^0 \le \alpha \le 40^0$ and Reynolds number calculated on the basis of profile chord length: $\mathrm{Re} = 3.91\,10^6$.

On the basis of computations next function was determined describing limiting Mach number values as function of the angle of attack when buffet onset will arise.

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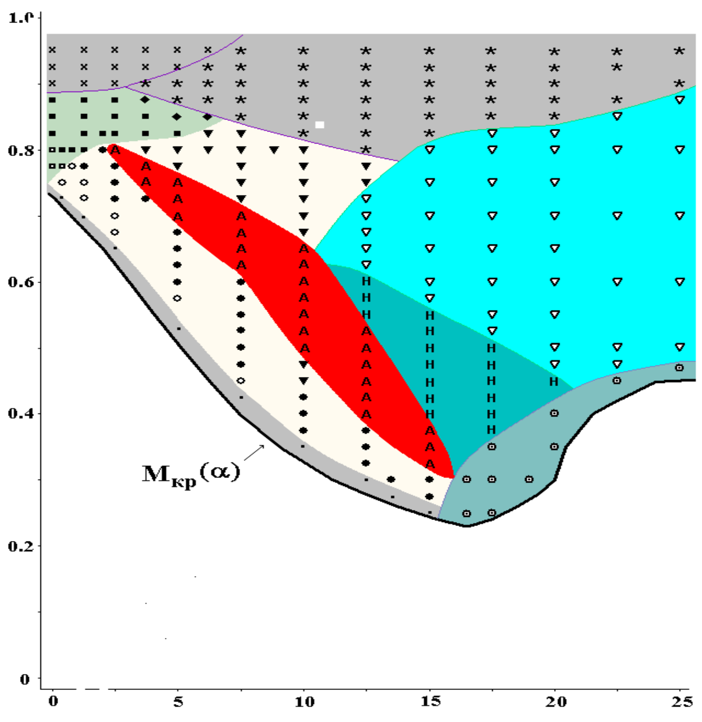

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3 3D Flows

3.1 Geometry, grid design and numerical method

Onera M6 wing geometry is presented on the Figure 1 [4]

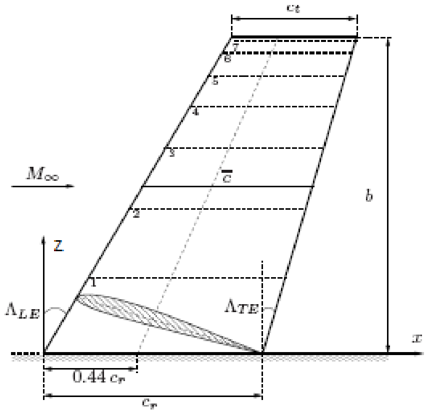
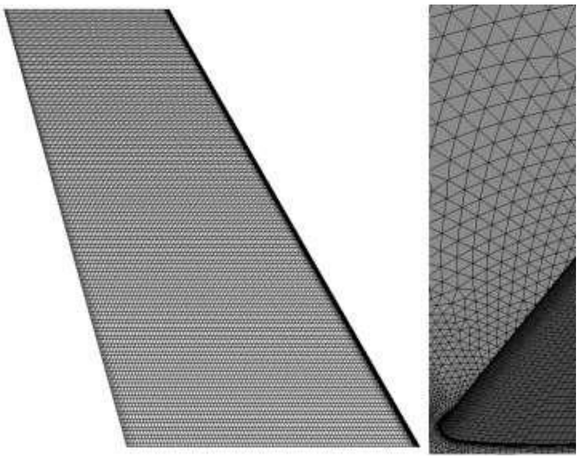


Figure. 1. the geometry of the wing

For calculations in the space around the model 3D grid
was generated using ICEM-CFD approach with more than 3 million cells (see Figure 2a and 2b).



2 (a)

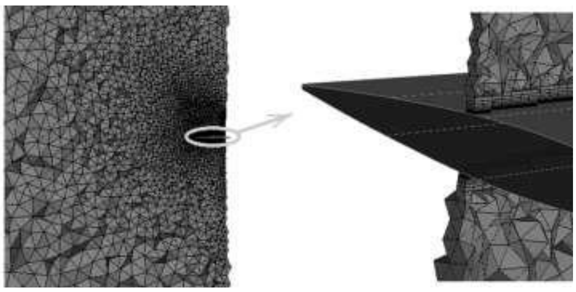


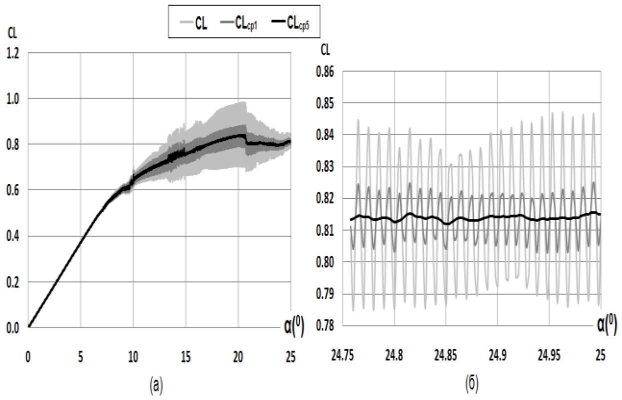
Figure. 2. Surface mesh and grid around the wing (b)

To solve the problem commercial program CFX (MIPT) was used. The flow was modeled using three-dimensional unsteady Navier-Stokes equations written in arbitrary curvilinear coordinates. SST turbulence model was used for perfect gas. Calculations were performed for ONERA M6 wing in the range of the angle of attack: $0_0 \leq \alpha_0 \leq 30$ and Reynolds numbers $Re = 3.9110_6$.
Let us consider unsteady flow regimes for different angles

of attack at fixed value of Mach number $M_\infty = 0.2 \div 0.9$ For each number of Mach, the angle of attack changes from 0^0 up to 30^0 under the law: $\alpha(t) = 0.5*t = 0.5*Nt*\Delta t_{process}$, where Nt time-step iteration number, Δt -time step.

4. Results

Results of calculations have shown that at low angles of attack lift force and drag force coefficients increase linearly. Further increasing the angle of attack leads to the unsteady flow regimes. Figures 3 and 4 present lift and drag coefficients as a functions of an angle of attack. Oscillatory behavior of aerodynamic characteristics is presented on figures 3B and 4B.



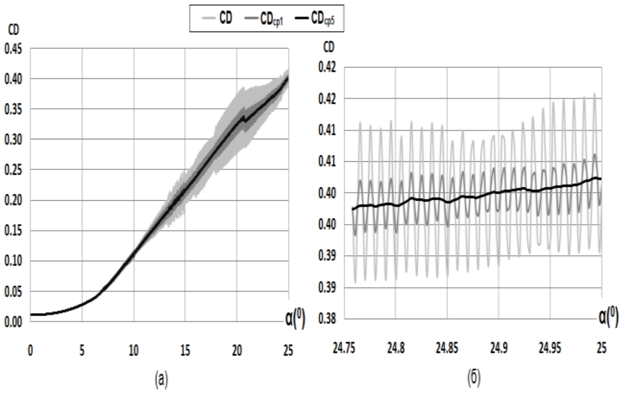


Figure. 4. Drag coefficient as function of an angle of attack

As a result of computations it was shown that for the case $M_\infty = 0.7$ unsteady regimes start when angle of attack reached critical value $\alpha = 7$. This angle of attack depend on the Much number (fig.5).

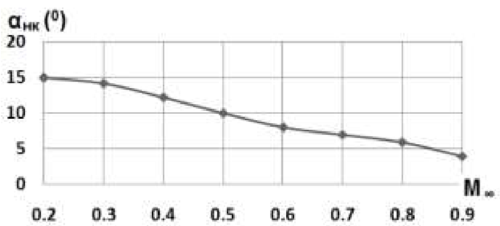


Fig 5 Critical angle of attack as a function of Mach Number

It should be noted that for relatively large Mach numbers M_∞ (for example $M_\infty = 0.7$, 0.8 and 0.9), after oscillatory regime the flow may reach steady regime again. Let us consider some flow regimes. For fixed Mach number value we will analyze two regimes corresponding to the angle of attack values $\alpha = 5^0$ and $\alpha = 13^0$ for Mach number $M_\infty = 0.7$. Lift and drag coefficients distributions depicted on the fig. 6 show significant difference.

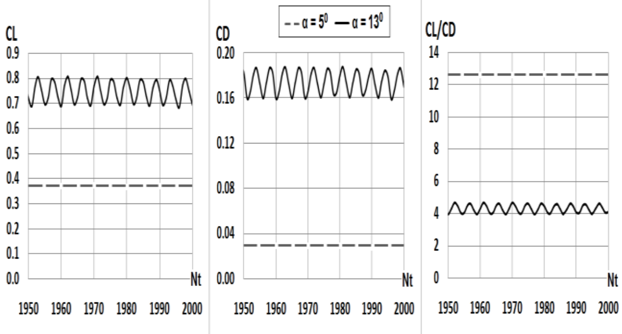


Fig. 6 Coefficients CL and CD time dependence for $\alpha = 5^0$ and $\alpha = 13$ for fixed Mach number $M = 0.7$

Let us consider supersonic flow region transformation during one period of oscillations (fig. 7). It was shown that supersonic flow region located nearby corner wing section is relatively stable. We can suppose that supersonic flow region may be considered as a rigid body. Then at the time moment $t = (1/6)T$ supersonic flow region reaches minimal volume. For larger time values this region expands along wing leading edge $t = (2/6)T$ and $t = (3/6)T$). At this moment supersonic flow region separates into two subregions $t = (4/6)T$, $t = (5/6)T$ and $t = T$)). The first subregion is located near corner wing section. The second one is not fixed and moves to the opposite wing section and eventually disappears.

This process is periodic. It can be mentioned that for the case $\alpha_0 = 5$ supersonic flow regions doesn't separate into two subregions

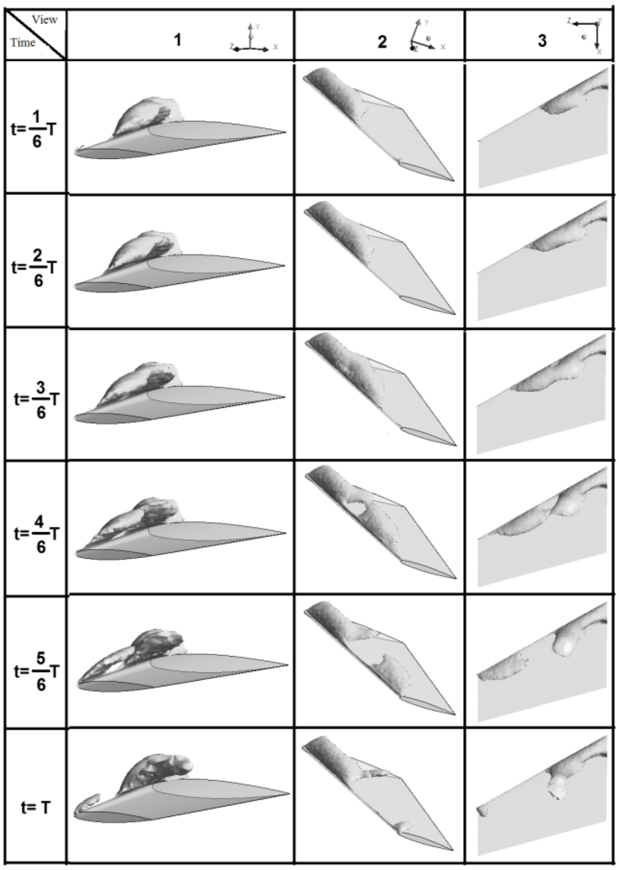
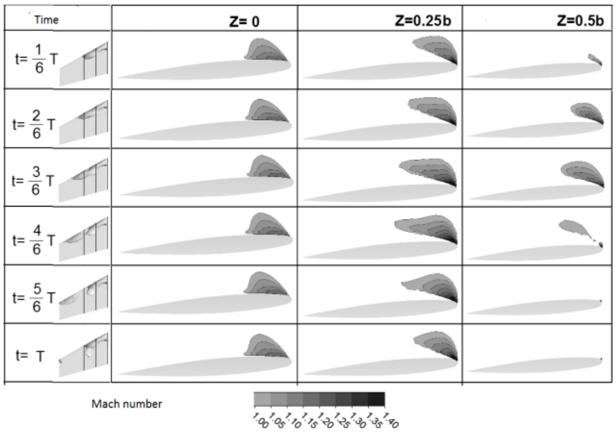


Fig 7. Supersonic flow region configuration for one time period for the case $M_\infty = 0.7$, $\alpha_0 = 13$

In the other wing sections supersonic flow region diminishes. On the fig.8 changes in the Mach number during one period are depicted.

Supersonic flow region for Z=0 doesn't change during one time period. At the same time we can see changes of supersonic flow region for the wing sections $Z = 0.25b$ and $Z = 0.5b$.



Conclusions

1. In the interval $M_\infty \in [0.2 \div 0.9]$, for each Mach number corresponding angle of attack α was found when selfoscillations start.

2. Supersonic flow structure for the regime of 3-D selfoscillations was analyzed. Effect of this region evolution with two subregions formation was found.

References

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[3] D.G.Mabey. Prediction of the severite of buffeting. – In: Three Dimensional and Unsteady Separation at High Deynolds Numbers, 1978, p. 143–172, (AGARD LS-94).

[4]. On website http://www.aero.polimi.it/freecase/ONERA_M6_wing