31st Congress of the International Council of the Aeronautical Sciences
Belo Horizonte, Brazil; September 09-14, 2018

APPLICATION OF THE COMBINED BOUNDARIES TO REDUCE WALL INTERFERENCE FOR NACA 0012 AIRFOIL TESTS

S.L. Chernyshev[1], A.I. Ivanov[2], E.V. Streltsov[3], A.O. Volkova[4]
[1-4]Central Aerohydrodynamic Institute (TsAGI), 1 Zhukovsky St., Zhukovsky, Moscow Reg., 140180, Russian Federation, [2] ivanov_a_i@list.ru, [3]evg.streltsov@gmail.com, [4]a.volkova.mipt@gmail.com

Keywords: *wind tunnel, wall interference, boundary layer*

Abstract

A new method for wall interference reduction is considered that is a combination of perforated panels and a controlled boundary layer, the so-called combined boundaries. The study of this concept was carried out using the well-known model of the airfoil NACA-0012 in the transonic wind tunnel TsAGI T-112. To obtain the unbounded-flow characteristics of the tested model, numerical simulation was performed. The analysis of the results made it possible to draw a number of conclusions concerning the possibility of reducing the interference of flow boundaries and to choose the optimal combinations of parameters characterizing these boundaries.

1 General Introduction

The creation of first wind tunnels was accompanied by the appearance of the walls interference problem, because of which the balance characteristics of tested models were different from those obtained in free flight. Consequently, the question was raised of developing theoretical and experimental methods for correcting the data obtained excluding the influence of the wind tunnel walls.

The effect of wall interference on the flow over tested models occurs in the case of non-fulfillment of certain laws along the boundary of the test section of wind tunnel. Thus, if the test section has solid walls, then the velocity component normal to the wall surface must be zero; if the test section is a free jet, then on the boundary of the jet the pressure must be equal to the pressure in the chamber surrounding the test section. In both cases, the flow near the model placed in the test section of wind tunnel turns out to be different from the flow existing around the same model in the unbounded flow. It is evident, for example, that the effect of solid walls is displayed in the fact that they limit the possibility of free deflection of the streamlines at the location of the model. As a result, if the flow, for example, is subsonic, this will lead to an increase in the axial velocity in the region of the model in comparison with the unbounded flow velocity [1].

Particularly topical is the problem of reducing the wall interference of the test section became when creating transonic wind tunnels. The using of solid walls became unacceptable due to the phenomenon of the test section blockage in transonic wind tunnels that is connected with the fact that in the narrowest section, usually at the location of the model, sonic velocity is achieved. In this case, any further increase in capacity of compressors does not lead to increase in flow velocity. All additional energy is absorbed by the shock wave and leads only to an increase in its intensity.

An effective approach for wall interference reduction is the method based on the use of a controlled boundary layer [2]. This approach proved to be quite effective and technically easy to implement, which was shown both experimentally and numerically. Practical implementation of the new approach was carried out basing on the facility of transonic and supersonic speeds T-112 TsAGI.

The effectiveness of using a controlled boundary layer on smooth walls of wind tunnel to reduce wall interference was demonstrated for a standard experiment with models of small relative sizes [2]. However, in a number of cases, the tested models are oversized, and then the thickness of the boundary layer is insufficient to eliminate the solid wall interference. In order to overcome this obstacle, it was suggested to test a new kind of boundary conditions that is a combination of a controlled boundary layer and perforated walls. In order to study the new boundary condition, a set of experiments were performed in the TsAGI T-112 facility.

2 Experimental Investigations

2.1 Wind Tunnel Description

The experiment was carried out in the wind tunnel T-112 of TsAGI. This wind tunnel of transonic and supersonic speeds is a periodic ejector-type facility with a semi-closed circuit and a closed test section with the dimensions of 0.6×0.6× 2.55m (Figure 1).

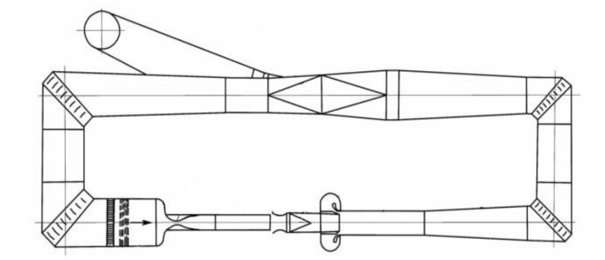

Fig. 1 - Wind tunnel T-112 layout.

2.2 Test section perforation

The perforation of the walls of the wind tunnels is a system of holes of round, elliptical, slit or other shape located on the walls. In this case, the test section is surrounded by a chamber and mass exchange takes place through the perforation between the main gas flow in the test section and the gas in the plenum chamber.

Perforation of the walls is used to carry out a continuous transition of the flow velocity through the sonic speed and to reduce the influence of the test section boundaries in aerodynamic testing of models.

The adjustable perforation of the T-112 test section had horizontal panels with the open-area ratios of 0%; 2%; 10% and 23%. In these tests, all top and bottom panels had the same permeability.

2.3 Controlled Boundary Layer

In this paper, the concept of using a controlled boundary layer as an analog of jet boundaries was used to reduce wall interference, since the boundary layer can be considered as a near-wall jet with a lower dynamic pressure (in comparison with the main flow). Such a choice seems to be correct, because according to the conclusions of [3], the optimal jet near a solid wall should always have a lower velocity than the main flow.

In all experiments, the control of the boundary layer was carried out by extending the wedge-shaped spoilers installed at the entrance to the test section and intended to create additional resistance at the walls (Figure 2).

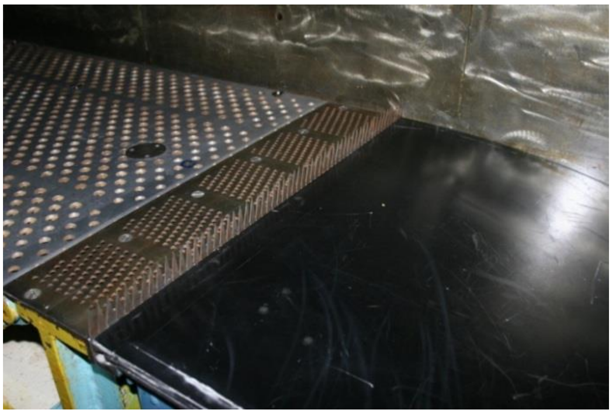

Fig. 2 - Spoilers at the entrance of the T-112 test section.

2.4 Airfoil NACA-0012 model

As the tested model, the drained symmetrical airfoil NACA-0012 with chord c = 150 mm was chosen (Figure 3).

Tests were carried out in a wind tunnel both without fixing the laminar-turbulent transition on the model surface and with turbulence strips located at the 11% of the airfoil model chord, for artificial fixation of the transition at a given line.

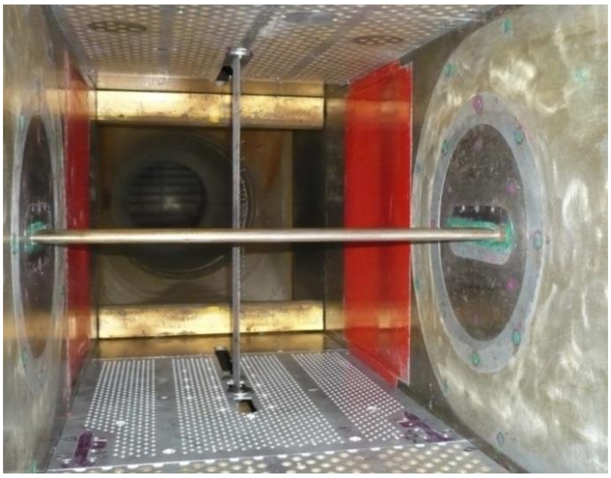

Figure 3 - NACA 0012 airfoil in T-112 wind tunnel

The free-transition tests of the airfoil model NACA-0012 were performed under different boundary conditions, which were perforated panels with zero open-area ratio, as well as 2%, 10% and 23%. In addition, an experiment was carried out using a new promising approach to reduce the wall interference of the WT test section – the use of combined boundaries, which are a combination of perforated panels and a controlled boundary layer. Experimental Mach numbers were 0.6, 0.65, 0.7, and 0.74. The angle of attack varied from -4 ° to 6 °. The height of the spoiler extension was 30 mm (10% half-height of the wind tunnel T-112 test section).

The second series of experimental studies were carried out with a fixed laminar-turbulent transition. The model with turbulence strips was tested in conditions of smooth walls with spoilers, as well as using perforated panels with open-area ratios of 0% and 2% with spoilers and without them. The height of spoilers was 30mm. As a result, the pressure distributions were obtained over the NACA-0012 airfoil surface in the central section.

2.5 Experimental results

Consider the differences between the obtained pressure distribution results under the boundary conditions, which are a smooth wall with spoilers and open-area ratio of perforated walls 0%. (Figure 4)

It is worth noting that the perforated wall with 0% open-area ratio is not a smooth wall, since it was obtained by shifting one panel relative to the other and as a result, the walls were a surface with blind holes 6 mm deep, which also caused additional flow disturbances.

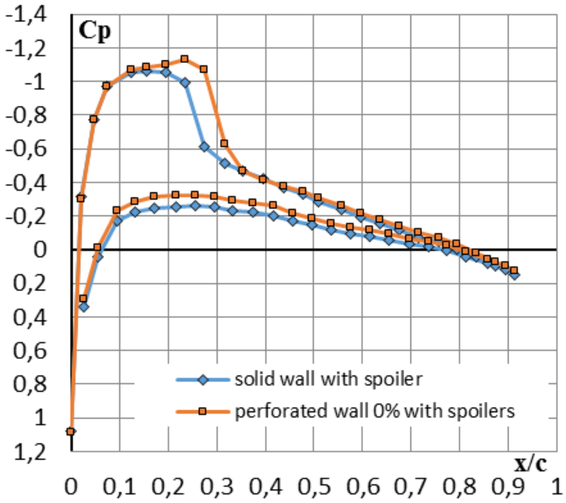

Fig. 4 - Pressure coefficient distribution along the airfoil NACA-0012 at the Mach number of 0.74 and an angle of attack of 2º with fixed transition on the model.

Thus, in Figure 4, one can see the differences in flow over the model in combined boundaries with a 0% open-area ratio of perforated walls, as well as in the conditions of controlled boundary layer on the solid walls. The choice of this or that boundary condition has a special effect on the location of the shock wave on the surface of the model. In the case of solid walls, the shock was located at 23% of the airfoil chord, and in the 0% perforation conditions, it moves to 27%. This is explained by the presence of a thicker boundary layer on the wall of the wind tunnel.

In addition, the lift force coefficient was calculated based on the results of pressure distribution along the airfoil. For instance, a $C_L(\alpha)$ curve is given for the lift force at Mach number of 0.65 at the different boundary condition (Figure 5). As can be seen on the graph in the conditions of perforated walls with 0% open-area ratios, the coefficient of lift is too high, and presence of spoilers affects the flow over the tested model in the best way. In this case, lift coefficient is about 0.76 at the 6° angle of attack when boundary is a 0% perforated wall; when it is 0% perforated wall with spoiler, the lift coefficient is about 0.705. Moreover, the lift coefficient decreases to the value of 0.622 for 23% perforated wall; and for the perforated wall 23% with spoiler the lift coefficient is about 0.538.

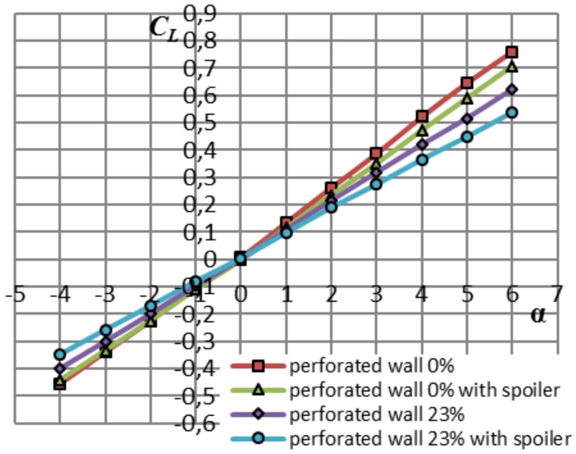

Fig. 5 - $C_l(\alpha)$ NACA-0012 at the Mach number of 0.65.

An important aspect of this experimental study was the effect of fixed transition. As a result, in the presence of the turbulence strips on the airfoil model, some changes were observed in the region of the location of the shock wave on the tested model, which can be seen in Figure 6. Thus, when the NACA-0012 airfoil model was tested at a Mach number of 0.74 and an angle of attack of 2° without transition fixing on the model, the pressure rise began at approximately 28% of the chord, while when using turbulence strips, its location shifted closer to 31% of the chord. This effect of changing the location of the shock wave on the model is also observed under other boundary conditions, at different positions of the model.

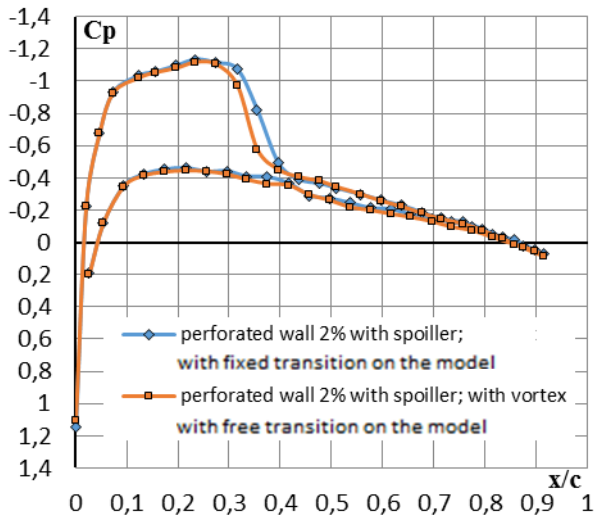

Fig. 6 - Pressure coefficient distribution along the airfoil NACA-0012 at the Mach number of 0.74 and an angle of attack of 2° with free and fixed transition on the model.

3 Numerical investigations of the flow over NACA-0012

3.1 CFD code and technology

When developing the technique of reducing the interference of flow boundaries in transonic wind tunnel, an important step is to obtain the unbounded flow field around the model. In the future, the characteristics obtained in these conditions should serve as reference ones for selecting and optimizing the parameters of the boundaries.

In general, “unbounded” characteristics might be obtained by testing comparatively small models in wind tunnel with a large test section, when the shading of the cross-section by the model is less than 1% (thus the model "does not feel" the presence of boundaries). In addition, in the case of relatively simple geometry, it is economically more advantageous to obtain unbounded flow over a model using numerical methods using software packages for aerodynamic calculation.

In this paper, the calculation of the characteristics of the unbounded flow over the test model was carried out using the ANSYS CFX software package, by numerically solving the Reynolds equations system (Reynolds-averaged Navier-Stokes, RANS) with SST-model of turbulence.

Numerical calculations of flow over the NACA 0012 airfoil with a 150-mm chord were performed under conditions corresponding to the experimental ones (Mach number: 0.6, 0.65, 0.7 and 0.74, angle of attack: 0 °, 1 °, 2 °, 3 ° and 4 °).

The distance from the surface of the airfoil to the outer boundary of the calculated area for all grids was 20 chords. The nets were thickened near the nose and the trailing edge of the airfoil and along the normal to the airfoil surface (Figure 7). Based on preliminary calculations, at least 32 cells are placed in the boundary layer; the size of the y_1^+ cell is less than 0.5. The grid capacity was about 0.7 million cells.

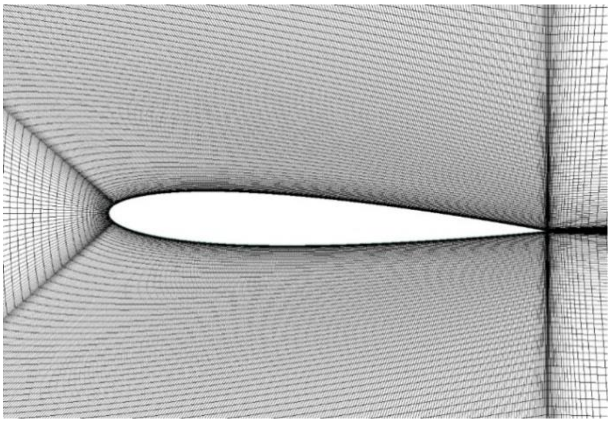

Fig. 7 – Grid over the airfoil NACA 0012.

3.2 Validation

For the validation of the numerical method, the experimental data presented in [4] were chosen. The results of the calculation are in good agreement with the experiment (Figure 8) and allow one to obtain the unbounded flow over the reference model and to define its characteristics in the framework of numerical simulation.

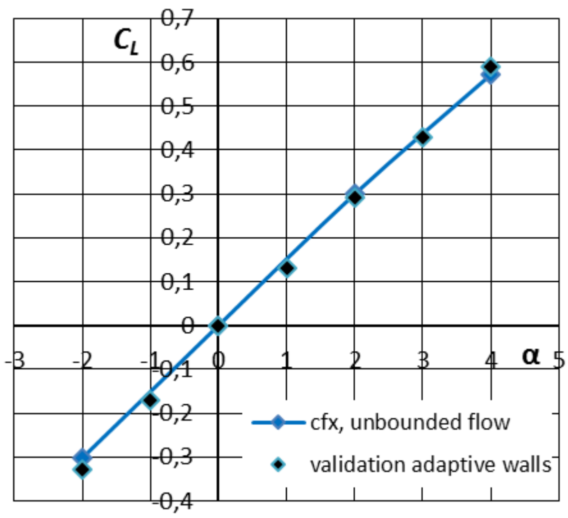

Fig. 8 - C_L (α) NACA-0012 at M=0.7

4 Experimental and numerical data comparison

Figure 9 shows the effect of boundaries on the flow around the model. It can be noted that for the same parameters of the oncoming flow and the position of the tested model, the flow patterns are different. Thus, under solid wall conditions, the shock wave on the model is much more pronounced than in the case of unbounded flow, which demonstrates well the wall interference.

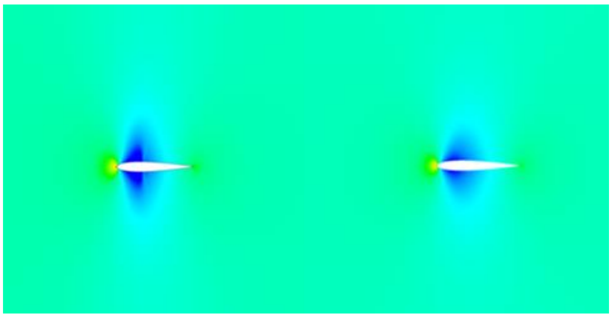

Fig. 9 – Numerical simulation of the flow over the airfoil NACA-0012 at M=0.74, α=0° in wind tunnel with solid walls and porous wall boundaries.

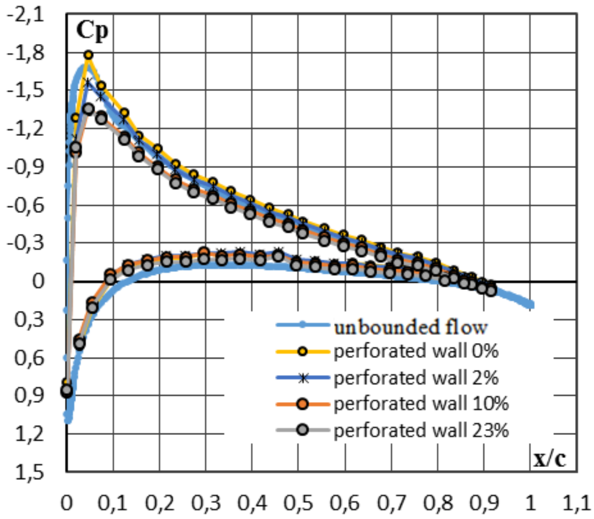

Fig. 10 - Pressure coefficient distribution along the airfoil NACA-0012 at the Mach number of 0.6 and an angle of attack of 4º.

Figure 10 shows the distribution of pressure over the airfoil surface in the conditions of perforated boundaries with different permeability. It is important to note that with an open-area ratio of 10% and 23%, the area between the curves is much smaller, compared to the results of numerical calculations with the unbounded flow. Accordingly, the experimental lift coefficient may be considered as underestimated in tests where 10% and 23% perforatied walls were used. A similar effect is observed when the test section of the wind tunnel is open, therefore it can be concluded that these open-area ratios of the perforated panels are too high and give a negative effect. Consequently, further analysis of the results

obtained was based on the study of combined boundaries at 0% and 2% open-area ratios.
With an increase in the angle of attack to 6°, a shock wave appears, which is clearly visible both in the experimental data and in numerical simulation (Figure 11). It should be noted that for moderate subsonic Mach numbers, when the perturbations introduced into the flow by the model are small, as for example at M = 0.6, the results obtained for all four boundary conditions are in good agreement with the pressure distribution curve corresponding to the unbounded flow predicted by numerical modeling. However, the best coincidence with the calculated pressure distribution for unbounded flow was achieved in the tests in combined boundaries with a 2% open-area ratio perforation.

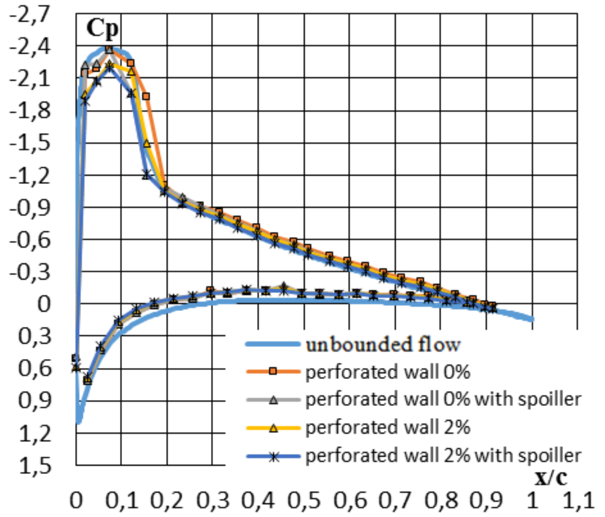

Fig. 11 – Pressure coefficient distribution along the airfoil NACA-0012 at the Mach number of 0.6 and an angle of attack of 6º, boundary condition – perforated walls with the open-area ratios 0% and 2%, and combined boundaries at 0% and 2% open-area ratios.

At a relatively moderate level of perturbations generated by the model (for Mach numbers up to 0.74 and angles of attack from - 2° to + 2 °), it can be seen that the selected boundary conditions significantly affect the aerodynamic characteristics of the tested model (see Figure 12) However, in this case, the smallest discrepancy with the curve corresponding to the unbounded flow was obtained in combined boundaries with a 2% perforation and artificially thickened boundary layer.

When comparing the numerical calculation prediction and the experimental data obtained in the wind tunnel with fixed transition on the model surface, one can observe a good convergence between the CFD results and pressure coefficient distributions measured on the model in the combined boundaries with the perforation open-area ratio of 2% for different angles of attack (Figure 12).

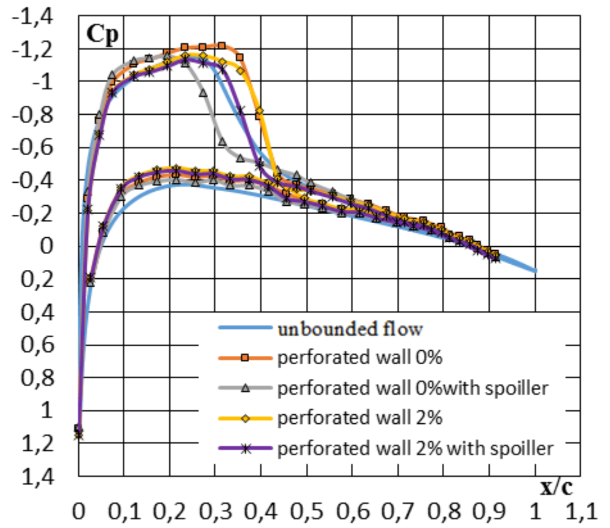

Fig. 12 - Pressure coefficient over the airfoil NACA-0012 at the Mach number of 0.74 and an angle of attack of 2º, boundary condition – perforated walls with the open-area ratios of 0% and 2%, and combined boundaries at 0% and 2% open-area ratios.

For the lift coefficient, a comparison was also made of the experimental data and the results of numerical calculations. The results of analysis for regimes with different angles of attack of the airfoil model depended on the boundary conditions in the experiment. For example, for the angle of attack of 1º, the smallest discrepancy in C_L was observed between the CFD results and tests in perforated walls with 0% open-area ratio (of the order of 0.01). At the angle of attack of 2º, the best convergence between the CFD prediction and experimental data was in the case of combined boundaries with the open-area ratio of 0% (≈0.015); at the angle of attack of 3º, the minimum discrepancy was for 2% perforated walls (≈0.012); and finally, at the angle of attack of 4º, the best coincidence between calculation and experiment was in combined boundaries with 2% perforation (≈0.011) (Figure 13).

Thus, the results for the lift coefficient curves behaviour demonstrates, that when the level of perturbations generated by the model increases, it is also necessary to increase either the permeability of the walls or the thickness of the boundary layer in the combined boundaries, thereby achieving a low-interference flow.

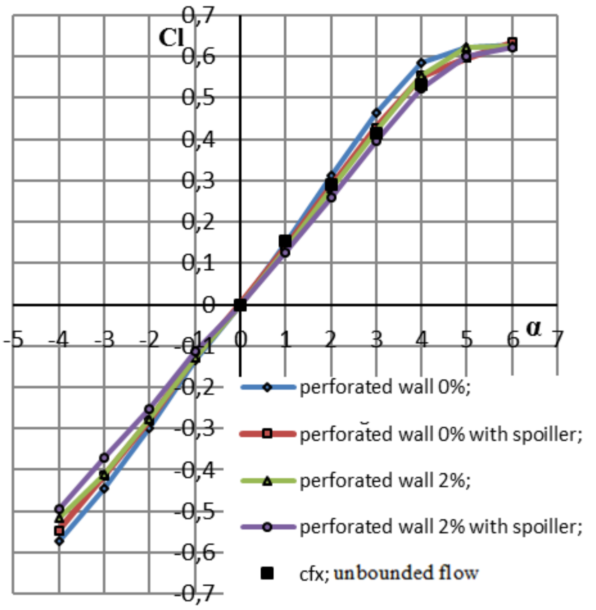

Fig. 13 - $C_l(\alpha)$ NACA-0012 at the Mach number of 0.74 with fixed laminar-turbulent transition on the model surface.

5 Conclusions

This research paper describes the cycle of experimental and computational researches aimed at continuing the development and implementation of a new methodical approach to reducing the wall interference of flow boundaries in the test sections of wind tunnels at subsonic and transonic flow velocities. The created technique is based on the previously proposed concept of using a controlled boundary layer as an analog of jet boundaries.

In the wind tunnel T-112, a series of studies of the aerodynamic characteristics of the NACA-0012 symmetrical airfoil model was performed. The measured pressure distributions over the model surface as well as the wake survay results made it possible to obtain all the basic aerodynamic characteristics of the model in different boundary conditions.

In the paper, experimental data for the airfoil model were obtained for the boundaries of various types: smooth walls, perforated walls with variable permeability, smooth walls with a controlled boundary layer, and combined jet-perforated boundaries. The studies were carried out at a Mach number of the flow, varying in the range from 0.6 to 0.74.

The experimental data obtained for the classical symmetrical airfoil NACA-0012 were compared with the results of computer modeling of flow around the model based on the software package ANSYS CFX under conditions of the unbounded flow (in a two-dimensional setting) and in solid walls. Since the calculation of two-dimensional flows in the absence of flow boundaries is an approved procedure that provides reliable data, these results were used as benchmarks for estimating residual wall interference.

The analysis of the results made it possible to draw a number of conclusions concerning the possibility of reducing the wall interference of flow boundaries and to choose the optimal combinations of parameters characterizing these boundaries. It is shown that at a relatively moderate level of perturbations introduced into the flow by the model (at Mach numbers up to 0.74 and angles of attack from -2 ° to + 2 °), the combination of a perforated wall with the open-area ratio no more than 2% and a controlled boundary layer, generated by 30 mm height wedge spoilers (10% half-height of the T=112 wind tunnel). The chosen combination of parameters allows practically eliminating all types of wall interference of the walls for a model with the chord does not exceeding 25% of the height of the test section. With the increase in the model lift force or its dimensions, the optimum open-area ratio of the perforation or boundary layer thickness should also increase.

References

[1] Pankhurst R., Holder D. *Technique of experiment in wind tunnels.* Foreign Literature Publishing House, Moscow, 1955.

[2] Ivanov A.I., Streltsov E.V. *Controlled boundary layer on the solid walls of wind tunnels: new approach to the boundary interference problem*. ICAS 2014.

[3] Neiland V.M. *Subsonic and transonic gas flows in adaptive boundaries*. Thesis for a doctor's degree in mathematics. Sciences, Moscow, 1989.

[4] Green L.L., Newman P.A. W*all-interference assessment and corrections for transonic NACA 0012 airfoil data from various wind tunnels*. // NASA Technical Paper 3070. April 1991.

Copyright Statement

Acknowledgement

The reported study was funded by RFBR according to the research project № 18-38-00735.