ON RESULTS' ACCURACY AT TWO-DIMENSIONAL WIND TUNNEL TESTING

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Abstract

The principal factors which influence the accuracy of two-dimensional wind tunnel test results are analyzed. The influences of Reynolds number, Mach number and wall interference with reference to solid and flow blockage (blockage of wake) as well as the influence of side-wall boundary layer control are analyzed.

1 Introduction

For the successful aerodynamic designing of a new modern aircraft it is necessary to know the accurate aerodynamic characteristics of the whole aircraft, as well as of its individual constituent parts. Since there is no adequate mathematical model of turbulent flows, we cannot solve completely the problem of aerodynamic designing by computer simulation and calculation. We still have to solve many problems related to aerodynamic designing by making tests in wind tunnels, see Figure 1. However, wind tunnel simulation is connected with many problems which cause many distortions of flow conditions around the tested models, which finally results in inaccuracy of the measured aerodynamic values. There are many reasons for that, but it is quite understandable that even the best wind tunnels cannot provide conditions for the simulation of the flows around the model which would be identical to the flows in the free air. Therefore, the resolving of the problem related to the definition and elimination of the wind tunnel wall interference is a lasting task to be solved through experimental and theoretical research, either during the construction of new wind tunnels or during their exploitation.

Since serious airplane development in wind tunnels started, aerodynamicists struggled with the Reynolds number problem, so called "Reynolds number gap". The wind tunnels became bigger and bigger but also the airplanes became bigger and bigger and faster. So at all times of wind tunnel utilization the Reynolds number achieved in wind tunnel was far below the full scale Reynolds number. In Figure 2 the maximum Reynolds number envelope achieved in all existing European and US wind tunnels is plotted against Mach number. The cruise and take-off and lending Reynolds number regions of transport airplanes are far outside of all wind capabilities, besides the NASA's cryogenic wind tunnel at Langley (NTF), USA and the European cryogenic transonic wind tunnel (ETW) [1-6].

The original motivation to build these facilities for flight Reynolds number tests on aircraft models was based on significant differences between wind tunnel tests and real flight, often leading to costly design changes after the first flights of a new aircraft.

The importance of wind tunnel simulation can be perceived from a comparative analysis of the time spent on aerodynamic tests carried out on particular airplanes in the development phase, the results of which are shown in Figure 1 [1, 3, 8-11]. According to this analysis, the development of the famous DC-3 in the 30s took only 100 hours of aerodynamic testing; in the 70s it took Lockheed over 25.000 hours to develop the wide-bodied Tristar L-1011; at the beginning of the 90s, the development of the Airbus-340 passenger carrier exceeded 50.000 hours, which is the equivalent to over five years of wind tunnel testing. A grand total of 43.889 wind tunnel test hours have been accumulated

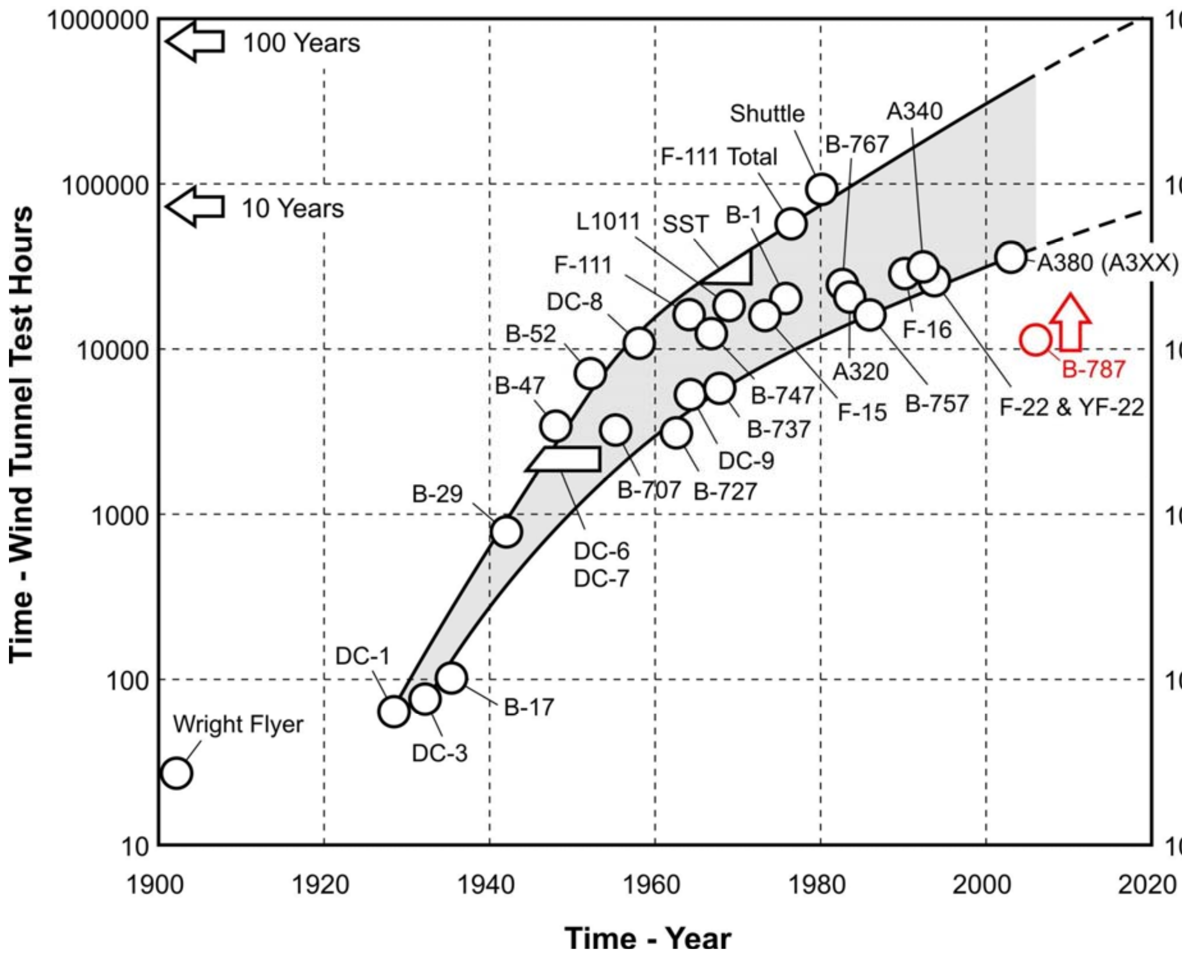

Fig. 1: The participation of wind tunnel tests in design, research and development of new airplanes - current status.

on the YF -22 and F- 22 configurations at the mid of the 90s. So far, most time has been spent on wind tunnel tests on Space Shuttle - a total of 10 years. It has been estimated that at the beginning of the new millennium the development of new aircraft will consume as much as incredible 10^6 hours of wind tunnel time, which is the equivalent to 100 years?

The new wind tunnel testing results show, that this trend has been disturbed for the first time and that for development of Boeing's Dreamliner, it was necessary 15.000 hours wind tunnel testing, only (see Figure 1).

It is interesting fact also, that design time computers 800.000 hours of computing time on Cray computers was needed versus 15.000 hours of wind tunnel testing for the development of Boeing 787.

The fields containing problems that contribute to inaccuracy in defining wind tunnel corrections can be arranged into four groups: (1) Nonlinearity of the referent equation in the condition of supercritical flow, (2) Nonlinearity of the boundary conditions of crossflow through ventilated walls and difficulties in predicting or measuring them, (3) Geometric characteristics of the wind tunnel (finite length of the ventilated walls), the entrance to the diffuser and the presence of the testing wake rake and its support, and (4) Boundary layer at the sides of wind tunnel walls, which produces flow deviations as regards the conditions of two-dimensional flow [12-19].

In the case of the simulation of transonic flow, the situation becomes even more complex when defining the aerodynamic flow

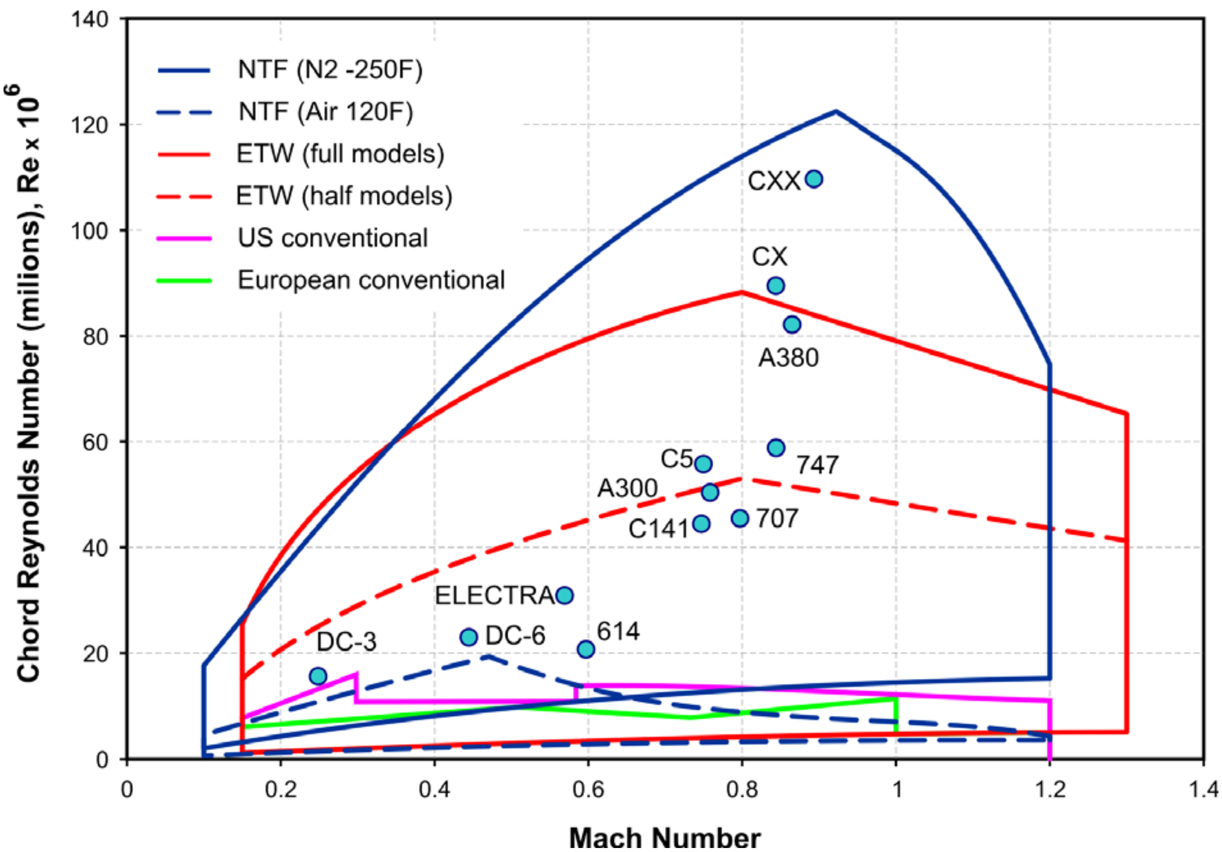

Fig. 2: The representative flight Reynolds numbers for several vehicles in the function of the Mach numbers, as compared to some European and US wind tunnels (Mach No. vs. Reynolds Numbers).

parameters. The effects of solid and flow blockage are even more evident, the side-wall boundary layer becomes thicker, the areas of separated flow and shock waves are created, which cannot be eliminated even by the full presence of the ventilated transonic walls. All this makes it even more difficult to define the exact aerodynamic parameters measured in wind tunnels. All controversy and uncertainty of the achieved results can be seen in Figures 3 and 4 [3, 12-19].

2 Analysis of Problem

The purpose of this paper is to point out the principal factors which contribute to the greatest extent to the inaccuracy and diversity of results of measuring aerodynamic values expressed through lift-curve slope ($a=dC_L/d\alpha$) of conventional symmetrical NACA 0012 airfoil. On the basis of the results of this study, an attempt has been made to give an answer to the question: What is the actual lift-curve slope of the conventional symmetrical NACA 0012 airfoil according to the Mach and Reynolds flow numbers?

In order to give an answer to this question an analysis should be made of the available results of wind tunnel tests which are published in international literature about such a subtle premature as lift-curve slope of airfoil [3, and 20-35].

First, in order to exclude from the analysis the effect of the Mach number, the range of subsonic flow (up to March number 0.55) has been analyzed at small angles of attack only, because of which the possibility of creating and separating the flows and shock waves have been eliminated. Then the Mach number effects have been included in the analysis. In both cases the effect of the Reynolds numbers to the models and wind tunnels has been also analyzed.

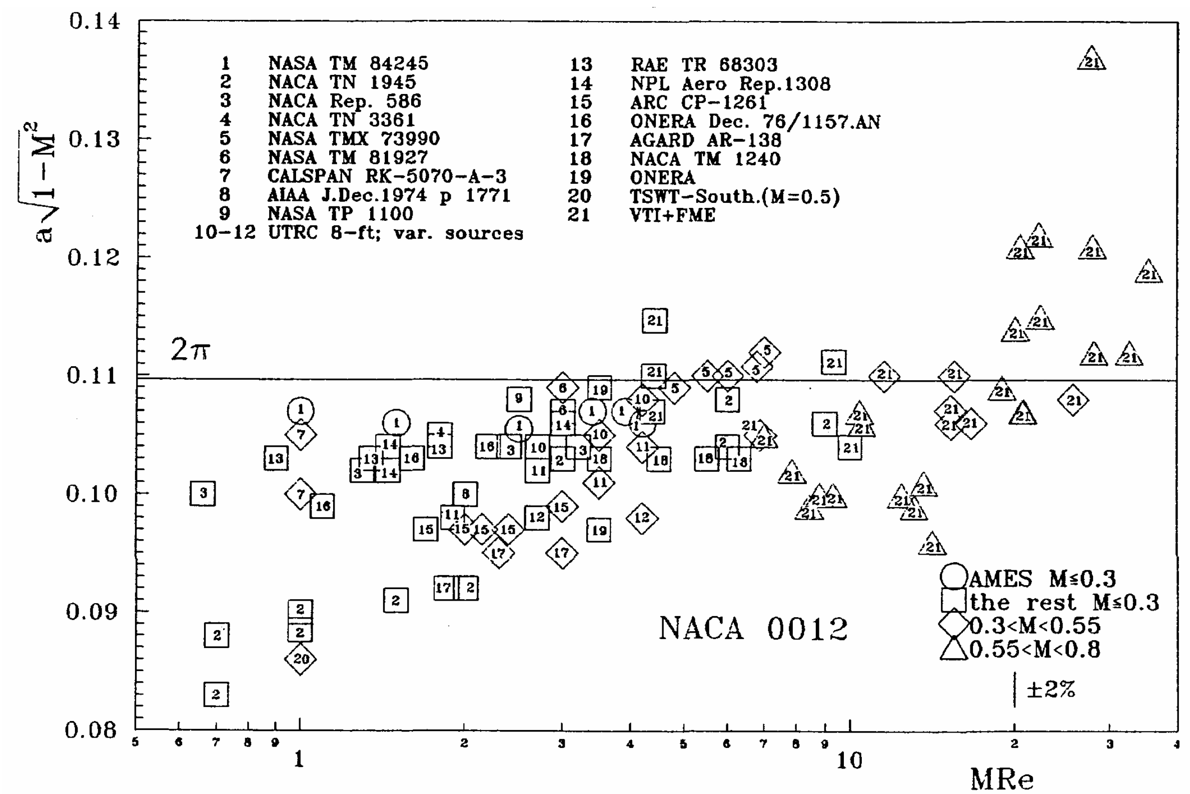

Fig. 3: Illustration of the collected results of the tests of lift-curve slope in the function of the Reynolds number.

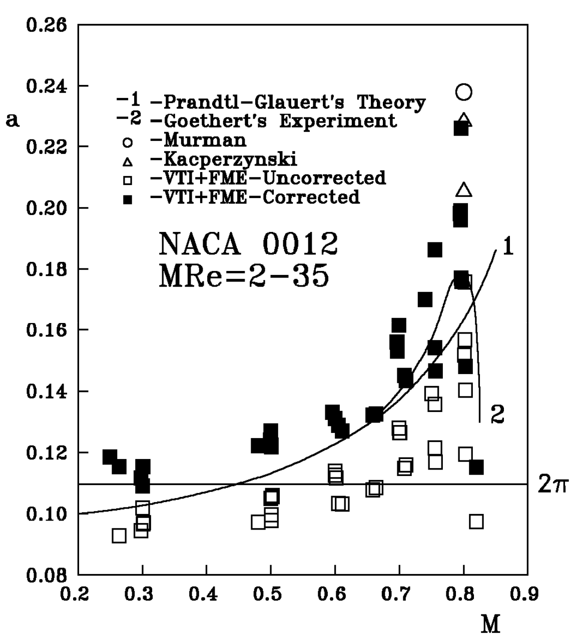

Fig. 4: Results of the test of the dependence of the lift-curve slope from Mach number.

The results of this analysis are presented in Figure 3 for NACA 0012 airfoil. They are grouped according to 21 sources of quotation. Many of these results have been achieved by the outstanding and widely known international aerodynamic institutions. For example, an analysis has been made of some old wind tunnel low speed tests made by NACA Institute (symbols 2-4), contemporary results of the NASA (1,5 and 6), the results achieved in the very good industrial facilities (10-12), detailed studies of the NPL and RAE (13-15), the results achieved by AGARD working group 04 DATA BASE (17), the results of ONERA (16-19), of the VTI and the Faculty of Mechanical Engineering (21), etc.

According to this illustration there is a great diversity in the achieved results, as a consequence of the strong influence of the Reynolds numbers effects on the test models and wind tunnels, of inadequate conditions of two-dimensional flows in the test section and the wall interference in the test section of wind tunnel. Wishing to complete this study, the analysis has been extended to the transonic speed range and it has incorporated new tests

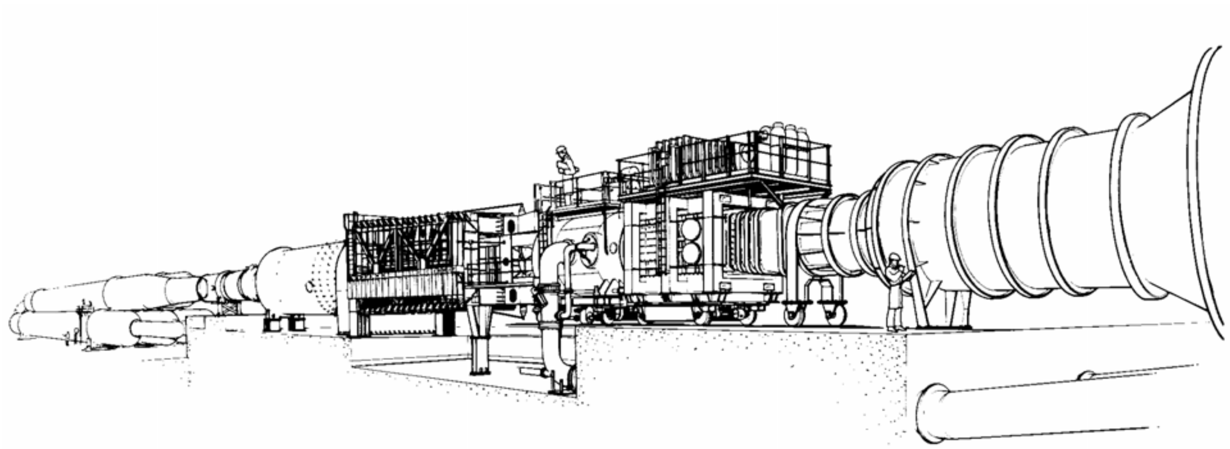

Fig. 5: Pictorial drawing of the VTI - 38 trisonic wind tunnel.

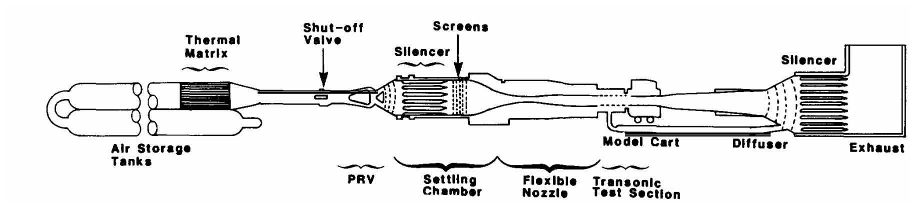

Fig. 6: Schematic of wind tunnel. *(PRV - Pressure Regulating Valve).*

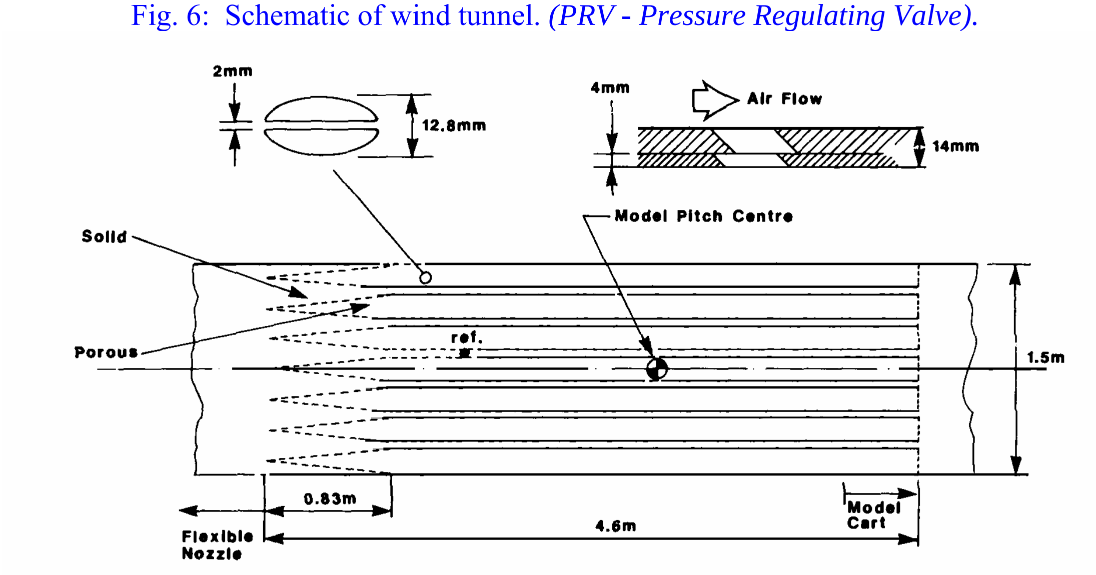

Fig. 7: Schematic of test-section walls.

made by the VTI as well as the calculation of wall corrections made at the Faculty of Mechanical Engineering (see Figures 3, 4 and 10) [3, 22-26 and 30-35].

3 Facility Description and Experimental Results

The VTI-Aeronautical Institute trisonic blowdown wind tunnel has a transonic test section with two- and three-dimensional inserts (Figures 5-8). The inserts have 60^o inclined-hole porous walls with variable porosity adjustment capability. Mach number is nominally set using either the second throat or flexible nozzle contour, depending on whether the flow is to be subsonic or supersonic. Final Mach number trimming is done using a blowoff system (with ejector assist if required) in which air reenters

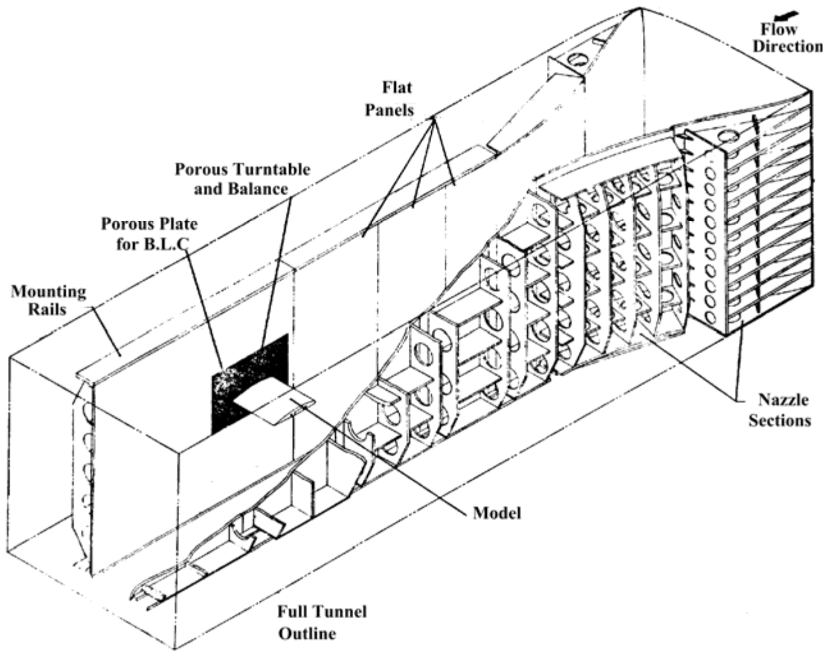

Fig. 8: 0.38x1.5 m Two-Dimensional Insert.

the circuit in the wide-angle diffuser just before the exhaust stack. Figure 6 shows a schematic of the circuit airline [3, 36 and 37].

Each of the four parallel walls of two-dimensional insert are 4.6 m long: side-walls are 1.5 m wide and the upper and lower wall are 0.38 m. Upper and lower wall consists of a pair of perforated plates with holes inclined 60 deg. to the vertical. Variable porosity is achieved by sliding the backplate to throttle the hole opening, the range being 1.5-8 %. Motion of the throttle plate is forward from full-open; i.e. cutoff is from the down-stream edge of each hole. A hole size is of 12.8 mm, and the combined two-plate thickness 14 mm. A splitter plate 2 mm thick is integral with each hole in the main plate-splitters are not incorporated into the throttle plate. Figure 7 shows the hole geometry and "finger" region where the porosity is gradually developed on a wall. A reference static hole ("ref." in Figure 7) located on one wall is used for control of nominal Mach number during a test run. The NACA 0012 model has a chord of 0.254 m.

PERFORMANCE ENVELOPE: Mach number range: 0.2 to 4, Reynolds number range: up to 140 million/m, Run time: 6 to 60 seconds, Stagnation pressure: 1.8 to 14 bar, Run frequency: average 1 run/hour, Blowing pressure regulation: +/-0.3%, Mach number regulation: +/-0.5% and Flow uniformity: LEHRT requirements.

MODEL SUPPORTS: Straight and bent-sting pitch/roll 3D model support, Half-model sidewall support and Wing-section (2D) sidewalls model support.

TEST SECTIONS: Subsonic/supersonic solid-walls 3D test section 1.5 x 1.5 m; Transonic perforated-walls 3D/half-model test section 1.5 x 1.5 m with controlled blow-off; Subsonic/transonic 2D (wing section) test section 0.38 x 1.5 m with controlled blow off and sidewall boundary layer removal (Figure 8).

INSTRUMENTATION: Teledyne data acquisition system: 64 analog and 8 digital input channels, 16-bit resolution, 120 KHz total sampling rate; Data acquisition computer: Compaq 440 MHz PII; Pressure scanning system: five S3 or D9 Scanivalves (230 pressure taps), expandable; Flow visualization system: Parallel-beam Schlieren; A range of five- and six-component force balances is available and Diverse flowfield probes are available.

Experimental tests have been made in blowdown trisonic wind tunnel T-38 with transonic two-dimensional working section of dimensions 0.38 x 1.5 m with changeable perforation of walls from 1.5 to 8 % (see

Figures 5 - 8). Aerodynamic coefficients have been calculated by measuring the distribution of the static pressure in 80 equally distributed tested points along the upper and lower side of NACA 0012 model with a chord of 0.254 m. For this measuring, the complete most modern equipment for aerodynamic measuring has been used. Figure 9 presents the selected results of the measurement of the distribution of the static pressure along the upper side and lower side of the airfoil at angle of attack of 2.0^o at Mach number of 0.8.

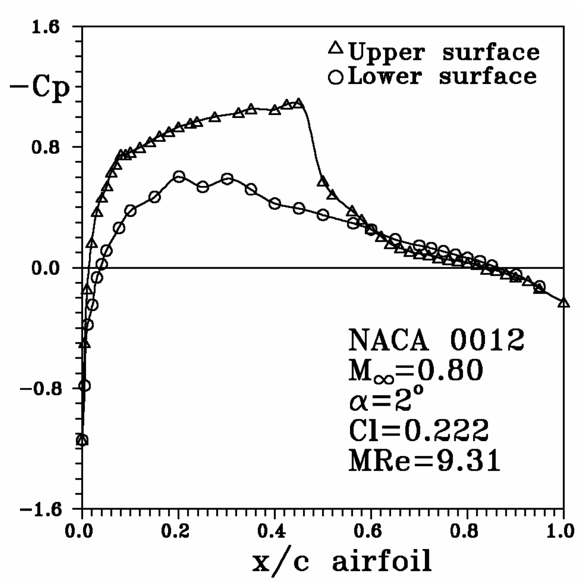

Fig. 9: Results of measurement of the distribution of the static pressure along the upper and lower side of NACA 0012 airfoil at angle of attack of 2^o at Mach number of 0.8.

This additional experimental study has included the Mach test number from 0.25 to 0.8 and the Reynolds model numbers from 2 to 35 MRe. It has corroborated the conclusions made at the beginning about the influences of the Reynolds number in the subsonic speed range and at the same time it has expended them to the transonic range, i.e. to the Mach number effects to the results of the wind tunnel tests [20].

The experiments and theoretical studies carried out recently by Murman [38], Kacperzynski [39], Chan, Jones and Catherall [40-42] and the latest tests made in NASA, Canada, by the VTI and the Faculty of Mechanical Engineering [23-27, 43] illustrate an exceptionally great interdependence of the Mach and Reynolds number effects, side-wall suction and the influence of the wind tunnel walls on test results in transonic wind tunnels. These conclusions are completely evident in the results of the lift-curve slopes tests made by the VTI which are presented in Figure 10, as well as in the corresponding results achieved in the world and presented in Figures 3 and 10 [3, 23-27].

In all analyses of tests results achieved in wind tunnels, the question of wall tunnel interference has been always raised. It has been manifested that, irrespective of the increased dimensions of the test section, i.e. of the Reynolds number effects on the wind tunnel, the effects of solid and flow blockage, i.e. the wind tunnel wall interference cannot be eliminated. If we look at the results of the tests carried made by the VTI, with high Reynolds numbers and different Mach numbers which are presented in Figures 3 and 10, we can establish that these results, if not corrected, are completely useless from the point of view of an engineer. Only when the wall tunnel influence is calculated, for example by the methods presented in the papers [3, 23-27], these test results could be accepted as real results which are achieved in the world today and which could be expected in the conditions of free air flow.

During all tests made by the VTI, the calculation of the perforated wall interference of transonic T-38 wind tunnel has been made by the Fourier's method used to solve the Dirihlet's problem in the rectangle of the wind tunnel test section.

During this calculation, in order to preserve in the computer analysis the reality of flows at the test section boundaries of transonic wind tunnel, the boundary conditions which are necessary to know for the solution of this type of boundary problems, have been experimentally defined by measuring the distribution of static pressure along the upper and lower wall of the test section in 46 equally distributed tested points. Figure 11 illustrates the distribution of measured pressure coefficients along the upper and lower wall of working section at angle of attack of 2.0^o at Mach number of 0.8.

For the solution of the problem of wall interference, the concept of local linearization of external flow outside and around the model has been applied, which have been replaced by the singularities of adequate strength.

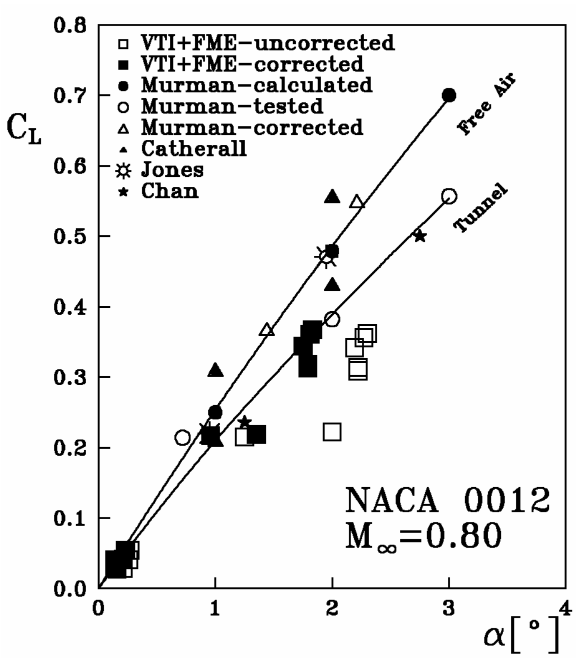

Fig. 10: Results of the test of lift coefficient in the function of the angle of attack.

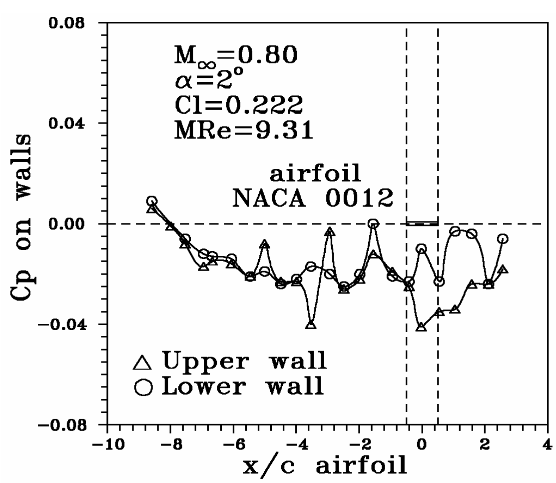

Fig. 11: Results of the test of lift coefficient in the function of the angle of attack.

The problem of boundary value has been analyzed, while the solution has been adapted for the application of the Fourier transformation and the Fourier coefficients have been calculated by the application of the fast Fourier transformation [3, 23-27].

4 Suction of the Boundary Layer from the Side Walls of Wind Tunnel

In order to create correct two dimensional flow conditions and uniform spanwise loading of the airfoil model, it is necessary to apply side-wall suction, i.e. the control over the boundary layer along the side walls of the wind tunnel. In the case that the control of boundary layer along the side walls is not ensured, this will certainly result in a loss of lift (and difference in drag) caused by the two basic effects of the complex flow. First, the loss of lift is caused by the decreased speed near the wall (by the decreased circulation). This effect can be significantly diminished if the side-wall boundary layer is reduced to the value which is very small in comparison with the spanwise of the model. Second, the influence of the airfoil pressure range will cause nonuniform increase of boundary layer along the side walls which will result in the creation of some tree-dimensional effects in the flow around the airfoil. The separation along the side walls is also quite normal. For example, it usually occurs near a rounded leading edge (in the vicinity stagnation point), approaching the trailing edge and during the subcritical and supercritical flow, as well as in the zone of the maximum local value of pressure.

It is desirable that the quantity of the removed volume of the air through porous side walls of the wind tunnel is minimal as required for creating satisfactory conditions for two-dimensional flow. If the too much quantity of air is removed from the working section this will cause an extensive axial gradient of pressure in the wind tunnel, which will result in (buoyancy) defect in drag and in the Mach number.

The importance of the correct definition of the quantity of the removed air is evident from the ONERA tests presented in Figure 3 for its results given under point 19. The lower point is the case with inadequate suction and the upper point with right quantity of the removed air. Most frequently the removed quantity of air is expressed through the ratio of normal component of flow velocity through the wall, to the velocity of undisturbed flow (far upstream from the model) V_n/V_∞. In all tests made by the VTI which are presented in Figures 3, 4, 9-12, the velocity ratio has been within the limits: $V_n/V_\infty = 0.0050 - 0.0054$.

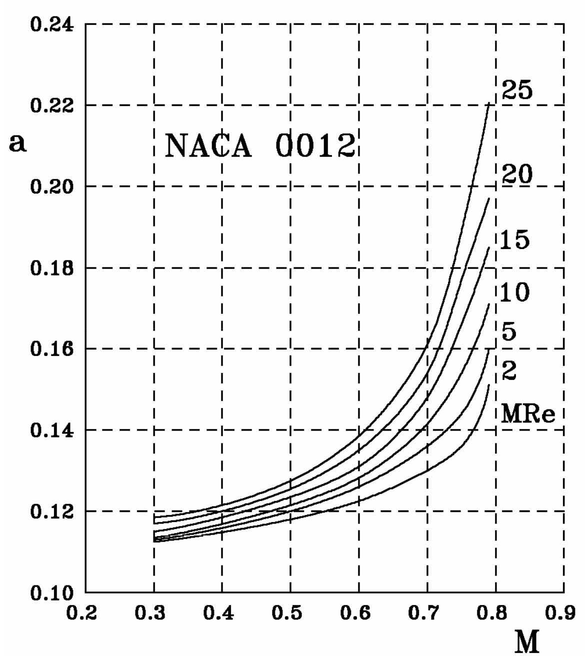

Fig. 12: Results of the test of the dependence of the lift-curve slope from Mach number.

5 Conclusion

This rather pessimistic picture which one could get on the basis of the presented results can be partially balanced by the new development of corrections of walls and calculation methods which are published and used in the world today, and which, when applied in practice, should increase the confidence in the results of wind tunnel tests. In this context, it is more precise to take the definition of the correction of walls as "adaptation of walls" which shall incorporate all mentioned factors which have an impact on the quality and accuracy of the flow area of the wind tunnel test section and thereby contribute to the increased accuracy of the measured aerodynamic values. The results achieved in this way (see Figure 12) could satisfy the users of "accurate" results of two-dimensional aerodynamic tests during the design and fundamental research or the testing of validity of the numerical methods of calculation.

The wind tunnels are becoming more and more the instrument for supporting computational tool validation role and less and less are used directly for designing the aircraft. This can be seen the best from the Dreamliner-Boeing 787 development and change the wind tunnel testing participation in designing process in relation to the present trends (see Figure1).

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