DETERMINATION OF NON-LINEAR AERODYNAMIC CHARACTERISTICS OF AN AIRCRAFT USING A POTENTIAL FLOW MODEL AND VISCOUS AIRFOIL CHARACTERISTICS

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Abstract

The article presents a hybrid method of determination aerodynamic characteristics of an aircraft at high angles of attack, consisting of a composition of a low-order panel method and modified Vortex Lattice Method. The modifications include determination of the position of control point for boundary condition based on two-dimensional lift slope of a wing section and an iterative procedure of simulating decrease of velocity circulation in a wing section due to flow separation through reduction of sectional angle of attack. The input data include two-dimensional aerodynamic viscous characteristics of wing sections along the wingspan. Since two-dimensional viscous airfoil characteristics can be computed with relatively low cost, or may be known from earlier wind-tunnel investigations, the presented method is very efficient at early design stages. The method is capable of analysing configurations with high-lift devices such as flaps or slats. The results of it's application for the tailless configuration of PZL M-18 aircraft show good agreement of computed c_{Lmax} and α_{CLmax} for cruise configuration. For the landing configuration the c_{Lmax} coefficient is slightly underpredicted, while α_{CLmax} is predicted correctly.

1. INTRODUCTION

The region of aerodynamic characteristics at high angles of attack is very important for the succesful design, since the value of $c_{L\,max}$ determinines stall speed of an aircraft, which is dictated by airworthiness requirements. This makes it important to be able to predict values of $c_{L\,max}$ and $\alpha_{CL\,max}$ reliably, on early phase of the design process, and for all configurations with different settings of high lift devices. Determination of aircraft aerodynamic characteristics at high angles of attack is a demanding task for numerical methods. Solutions of Reynolds-Averaged Navier Stokes solvers in this range of angles of attack depend strongly on the selected turbulence models and require a high-quality computational mesh which requires high workload and experience of the user of the software. A more economic option in terms of license costs and human workload is to use panel methods applying viscous-inviscid interaction, with boundary layer flow modeled by Prandtl equations.[1]. These methods allow for determination of friction drag, position of laminar-turbulent transition and separation of flow on airfoil and wing surface, but their aacuracy in the region near c_{Lmax} decreases (Fig. 1) due to quasi-two dimensional model of the boundary layer flow (neglecting spanwise flow and its influence on the flow separation), effects of surface roughness and surface imperfections (rivets, surface segmentation etc.)

2. DESCRIPTION OF THE COMPUTATIONAL METHOD

An alternative method for determination of nonlinear wing characteristics at early design stage was proposed by van Dam [2]. In this method the spanwise distribution of circulation is determined using lifting line theory and modified using two-dimensional α-c_L characteristics of local cross-sections. The idea of modifying spanwise distribution of circulation presented in [2] is a basis for a present method merging panel method approach for modeling thick bodies[3] (fuselage, engine cowling, etc) with modified vortex lattice method adopting solutions presented in[2]. Combining of panel method approach with vortex lattice method makes it possible to determine pressure distribution on thick bodies and chordwise and spanwise load distribution on wings – an improvement over the original method[2], with the same capabilities at high angles of attack (Fig. 2). The main assumption of the present method, as in[2] is, that by taking advantage of two-dimensional viscous characteristics of local airfoils it is possible to determine the spanwise distribution of circulation also for the range of angle of attack, for which there exist regions of separated flow on wing. The effects of high lift devices is taken into account in two-dimensional local characteristics which serve as a basis for modification of the spanwise distribution of circulation, determined for inviscid flow. An advantage of this approach is the possibility of using viscous characteristics obtained from wind-tunnel tests. The decrease of circulation, due to flow separation is modeled in the inviscid method by reduction of local angle of attack of an wing strip.

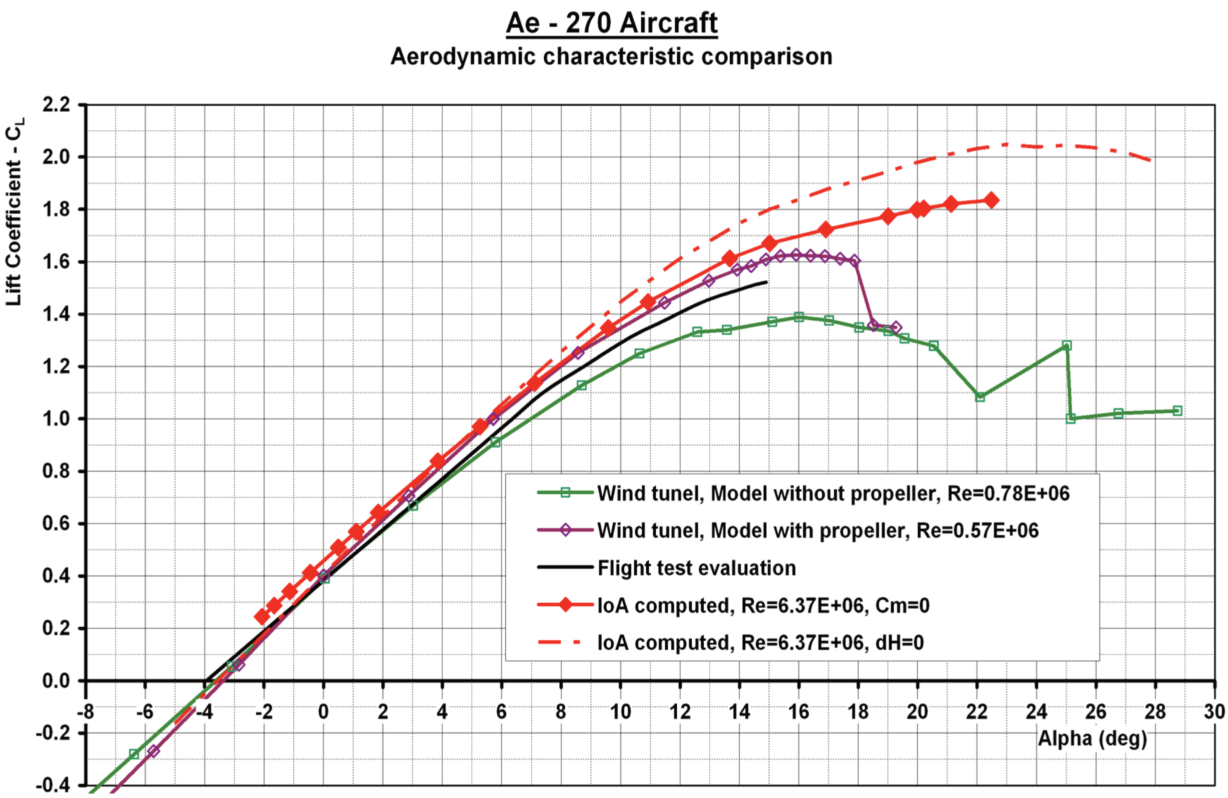

Fig.1. Comparison of α-c_L curves of a single-engine aircraft computed with panel method applying viscous-inviscid interaction with results of wind tunnel experiment and flight test [1].

2a. MATHEMATICAL MODEL

It is assumed, that flow is inviscid, irrotational, compressible and may be modeled with Prandtl-Glauert equation:

$$(1-M_\infty^2)\frac{\partial^2\Phi}{\partial X^2}+\frac{\partial^2\Phi}{\partial Y^2}+\frac{\partial^2\Phi}{\partial Z^2}=0 \quad , \quad (1)$$

which may be reduced to Laplace equation

$$\nabla^2\Phi = 0 \quad (2)$$

through a transformation of the reference system.
where φ is velocity potential.

The flow is modeled through a small disturbances of the free-stream velocity potential produced by surface distributions of singularities – sources and doublets. A constant-intensity doublet distribution is an equivalent of vortex ring placed on the boundaries of surface panel.

For thick bodies an internal Dirichlet boundary condition of constant, zero potential is applied [3]:

$$\phi_i = -\frac{1}{4\pi}\int_{S_{AC}}\left[\sigma\left(\frac{1}{r}\right)-\mu\vec{n}\circ\nabla\frac{1}{r}\right]dS+\frac{1}{4\pi}\int_{S_{SW}}\mu\vec{n}\circ\nabla\frac{1}{r}dS=0 \quad (3)$$

On wing surface the Neuman boundary condition of zero normal velocity is applied at control points:

$$\vec{V}\cdot\vec{n}=0 \quad (4)$$

The Kutta-Joukowski condition of finite velocities at trailing edge is fulfilled by the addition of vortex wake elements of circulation intensity equal to the wing vortex ring intensity at trailing edge.

The modification of the standard vortex lattice method is done in two steps. In the first step the slope of the linear part of the ‘ inviscid’ $c_L(\alpha)$ characteristic of local cross-section of the wing is corrected in order to be the same as the slope of the local airfoil viscous characteristic. This is done by the shift of the control point from standard position in 0.75 panel chord to a new position, which may be obtained by expressing the boundary condition (4) with induced velocity generated by the panel’s attached vortex of circulation intensity related to local $c_L(\alpha)$ slope through Joukowsky theorem[2,3].

The second step concerns modeling of the effects of strong separation and extending the applicability of the method up to the negative slope of the $c_L(\alpha)$ curve. The effect of flow separation is the reduction of velocity circulation at the local spanwise position. This is modeled with an iterative procedure of reduction of local angle of attack in wing strips where the computed value of $c_{L\_inviscid}$ exceeds the local airfoil’s $c_{L\_viscous}$ for the local angle of attack, determined as $\alpha_{loc}= c_{L\_inviscid}/a_{\_viscous,}$ where $a_{\_viscous}$ equals the local slope of $c_{L\_}(\alpha)$ curve applied in the first step for the modification of the position of wing strip control points. In such situation the local angle of attack is reduced to a value, for which the new ‘inviscid’ c_L equals the ‘viscous’ c_L determined in the first step. The iterative procedure is necessary, since change of vortex circulation on one panel modifies values of induced angle of attack in all wing strips. In case when α_{loc} determined in the firs step exceeds $c_{L\ max}$, the new α_{loc} is selected in the region of positive $c_L(\alpha)$ slope. The new, lower value of local angle of attack leads to lower intensity of circulation in the next iteration, determined based on the boundary conditions. The algoritm of the iterative procedure of modification of circulation intensity in wing strips is shown in Fig. 2.

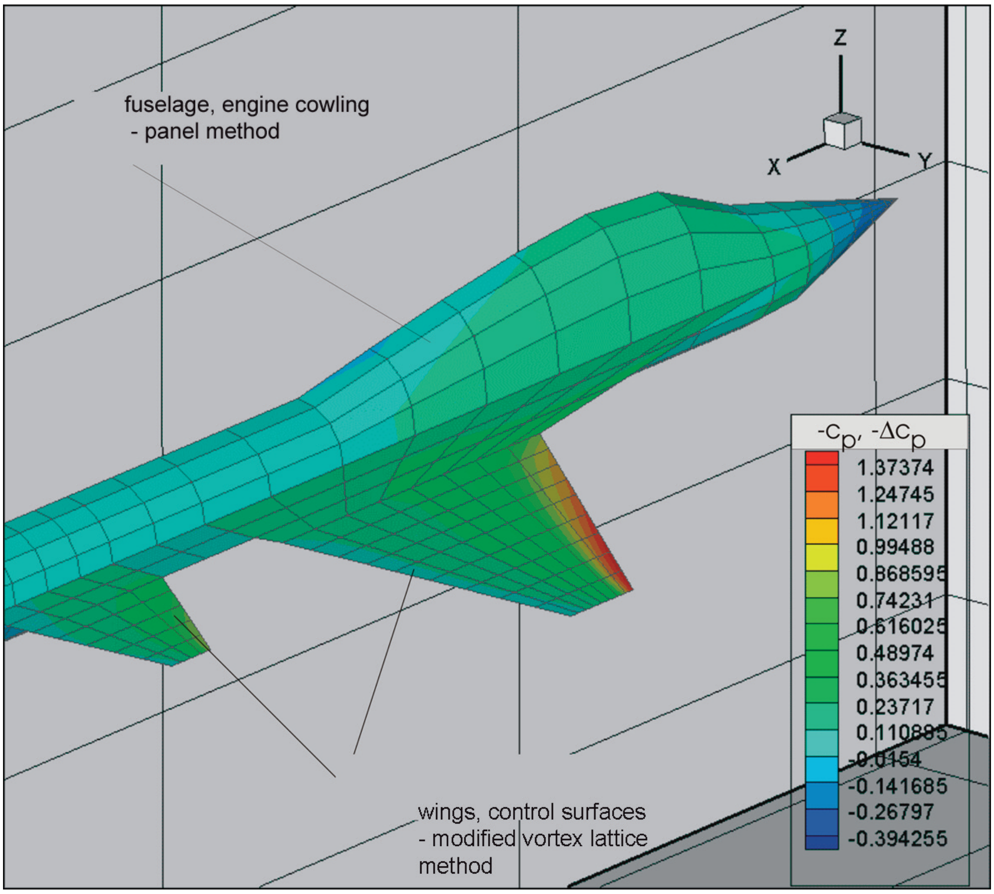

Fig. 2. Application of a hybrid panel/VLM method on aircraft surface

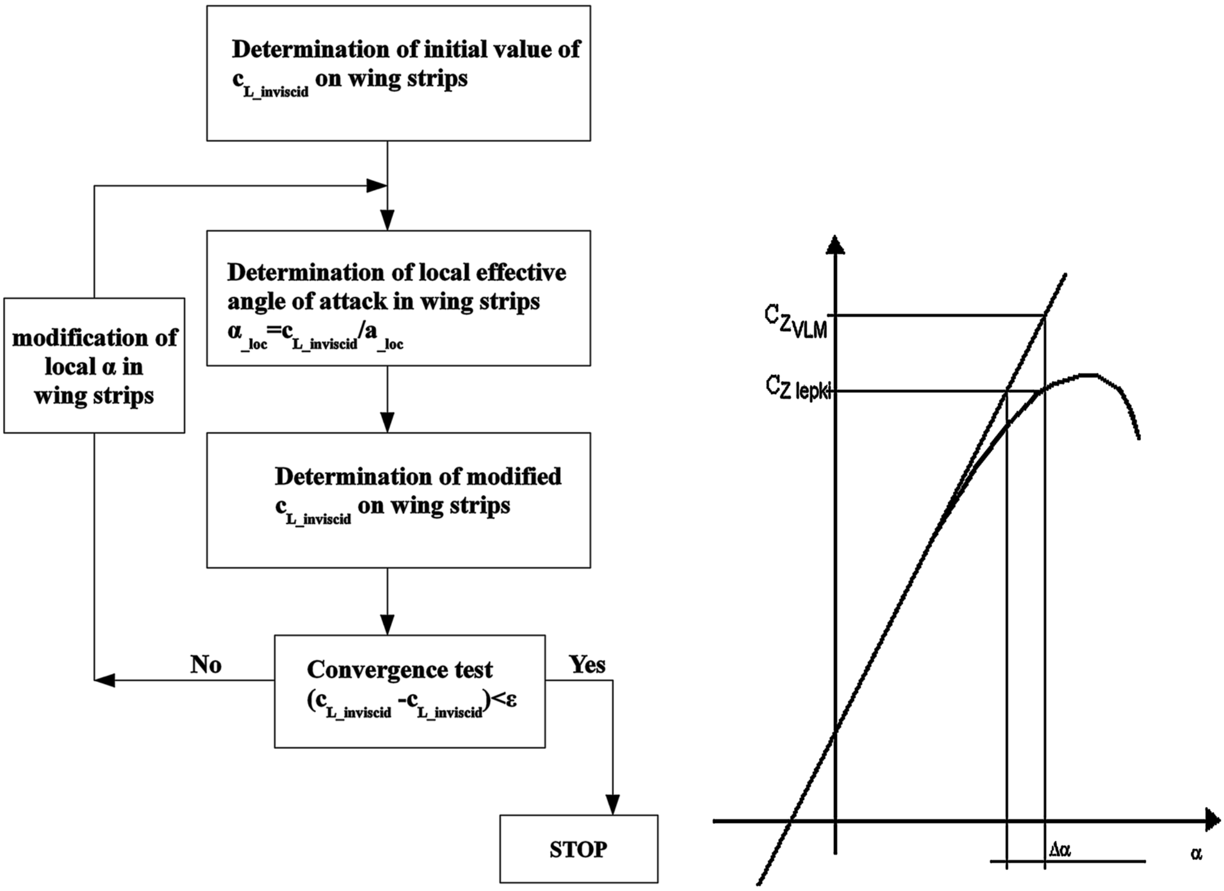

Fig. 3. Ilustration of the algorithm of modification of angle of attack on wing strips

In order to test the method, it was applied for the determination of the aerodynamic characteristics of the PZL-M18 aircraft in tailless configuration including chacracteristics at high angles of attack. The model of PZL-M18 was particularly convenient for the test, because in wind tunnel test the two-dimensional airfoil characteristics and three-dimensional aircraft characteristics were determined for the same Reynolds number (this aircraft has a constant-chord wing).[4,5] In this case the method's capablities of accounting for different settings of high-lift devices could also be tested.

2b. RESULTS OF THE COMPUTATIONS

The airfoil characteristics with flap retracted and extended 30^o in landing configuration were determined in wind-tunnel investigations conducted for M=0.2 and Re=1.4 *mln*. The wind-tunnel investigations were element of the program of improving the aircraft's high lift system effectiveness at low flight speed[6].

In the numerical test of the present method the three-dimensional flow around fuselage was not modeled. It was justified by the good agreement of the wing c_L at low angles of attack obtained with VLM method with the results of the wind-tunnel test for the wing-fuselage configuration. (In engineering methods of determination of aircraft characteristics it is often assumed, that c_L of central section of isolated wing is approximately equal to fuselage c_L). The central wing strip representing fuselage was excluded from the iterative procedure of modification of local angle of attack. The aircraft geometry and the geometry of the computational model is shown in Fig. 4.

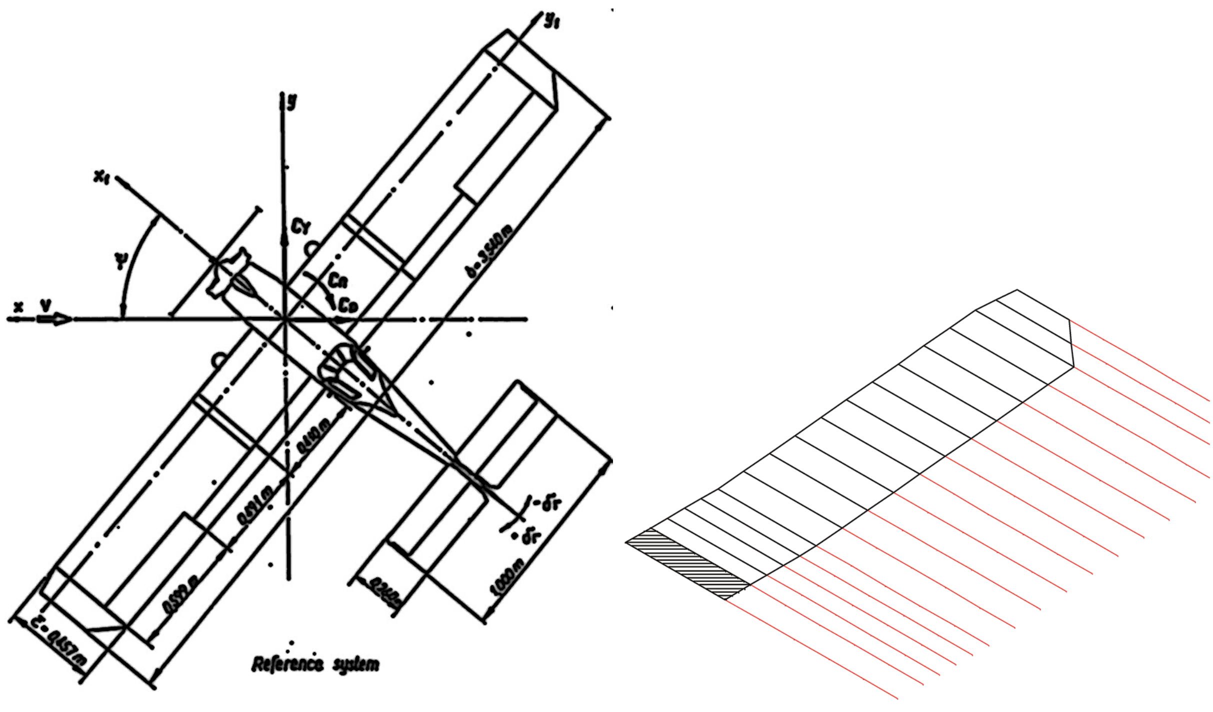

Fig. 4. Geometry of the PZL-M-18 aircraft model in wind-tunnel investigations and model of wing surface applied in the present method

The evaluated values of $c_L(\alpha)$, $c_m\alpha)$ i $c_D(\alpha)$compared with the results of wind-tunnel invetigations are shown in Figs. 5, 6, 7.

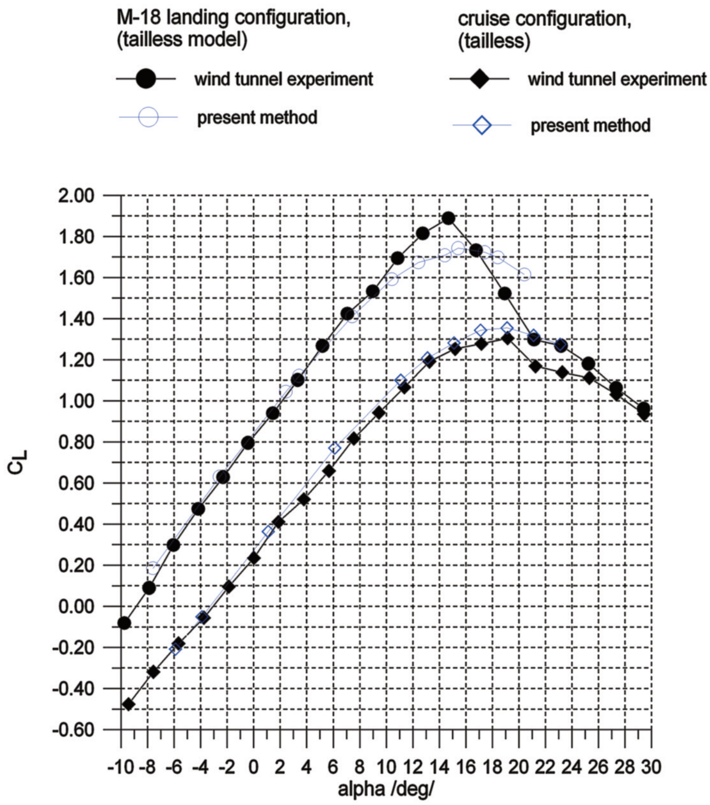

Fig. 5. Comparison of $c_L(\alpha)$ characteristics computed with the present method computed for two flap settings with results of wind-tunnel investigations [5]

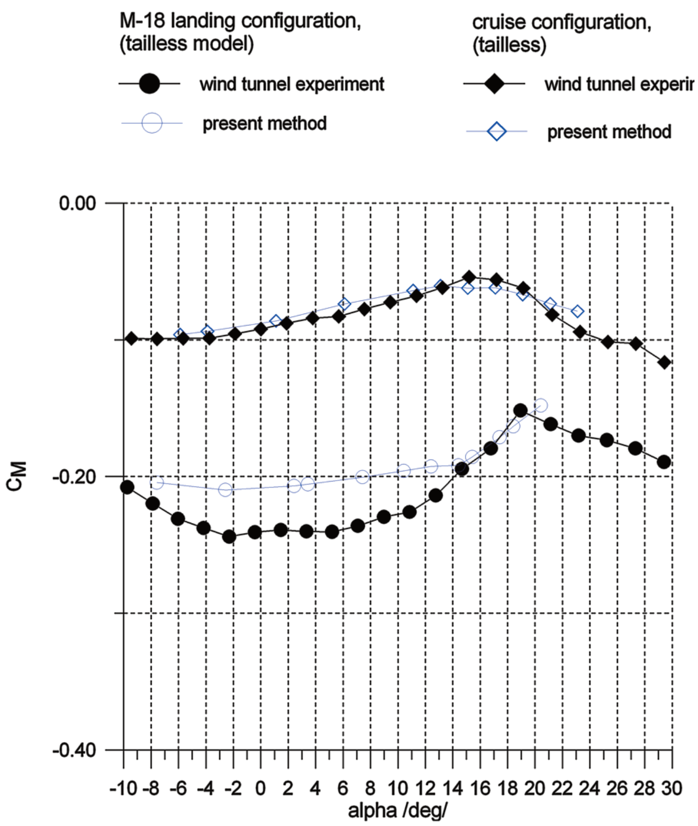

Fig. 6. Comparison of $c_m(\alpha)$ characteristics computed with the present method computed for two flap settings with results of wind-tunnel investigations

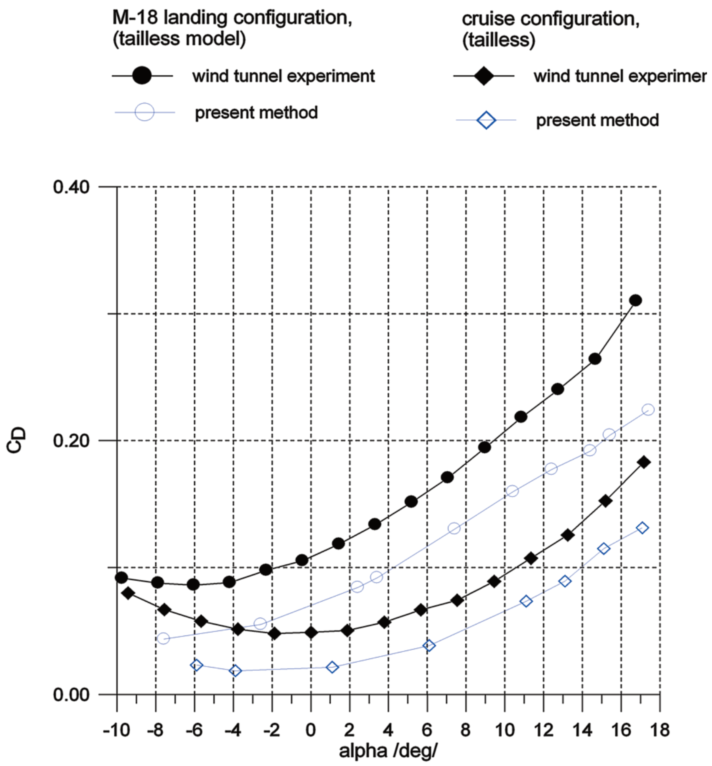

Fig. 7. Comparison of c_D (α) characteristics computed with the present method computed for two flap settings with results of wind-tunnel investigations

The comparison of numerical and experimental results shown in Figs 4-7 shows good agreement of the computed and experimental $c_L(\alpha)$ and $c_L(\alpha)$ characteristics for the cruise configuration. For this configuration only the drag values differ significantly, but this is expected, since the fuselage was not modeled realistically enough for the drag evaluation. For the configuration with flap deflected by 30 the wing $c_L(\alpha)$'experimental' curve has different shape at high angles of attack than the computed curve. This indicates that the flow separation on three-dimensional wing with part-span flap develops in a different way than on two-dimensional model. However, the c_L *max* was predicted with high accuracy in both cases.

The differences in $c_m(\alpha)$ characteristics for the high-lift configuration are difficult to explain; it is possible, that the chordwise load distribution on the flapped wing segment in the presence of fuselage differs from the distribution on two-dimensional wing model. This could be verified by CFD investigation with RANS solver and a proper turbulence model.

Apart from the determination of lift and moment coefficient the present may be helpful in the assessing of the total drag force of the configuration.

Having obtained the spanwise distribution of circulation intensity on wing, the local wing-strip c_D may be obtained from the airfoil $c_D(c_L)$ characteristic for given value of c_L.

The induced drag may be obtained through the evaluations of downwash in the Trefftz plane. Having computed the distribution of local flow velocity on the surface of fuselage with panel method the fuselage drag may be assessed using friction coefficient for given $Re_{fuselage}$[7].

3. CONCLUSIONS

A method for determination of aircraft non-linear, high-lift characteristics, based on modified Vortex-Lattice Method and Panel Method has been proposed and tested on a case of wind-tunnel model of a typical general-aviation low-wing aircraft using results of wind-tunnel experiment.

The results show a very good agreement of lift and moment characteristics at high angles of attack for the cruise configuration and slightly worse for a landing configuration with large flap deflection, but with proper prediction of $\Delta\alpha_{Clmax}$ due toflap deflection and $\frac{\partial cm}{\partial \alpha}$ for positive values of α which is an operational range of angle attack.

The advantage of the present method is, that prediction of high-angle-of-attack aerodynamic characteristics of complex, three-dimenional configuration may be conducted using reliable two-dimensional characteristics which may come from experiment, or from easier to conduct, two-dimensial computations of viscous flow. Possible direction of development of this method is application for wings with moderate sweep, taking advantage of simple sweep theory.

LITERATURE

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Streszczenie

Artykuł prezentuje hybrydową metodę wyznaczania charakterystyk aerodynamicznych samolotu na dużych kątach natarcia. Omawiana metoda jest złożeniem metody panelowej niskiego rzędu i zmodyfikowanej metody siatki wirowej. Modyfikacje polegają na wyznaczaniu położenia punktu kontrolnego warunku brzegowego metody siatki wirowej w zależności od nachylenia dwuwymiarowej zależności współczynnika siły nośnej w danym przekroju skrzydła od kątu natarcia oraz na zastosowaniu iteracyjnej procedury symulacji redukcji cyrkulacji w przekroju skrzydła będącej skutkiem oderwania opływu przez redukcję lokalnego kąta natarcia. Dane wejściowe zawierają dwuwymiarowe lepkie charakterystyki profili skrzydła wzdłuż rozpiętości. Ponieważ dwuwymiarowe charakterystyki mogą zostać wyznaczone numerycznie przy relatywnie niskim koszcie obliczeniowym lub też znane z badań tunelowych, przedstawiana metoda jest bardzo przydatna na wczesnym etapie projektowania. Metoda może służyć do analizy konfiguracji z urządzeniami zwiększającymi siłę nośną takimi jak klapy lub sloty. Wyniki uzyskane przy jej zastosowaniu do analizy konfiguracji bez usterzenia samolotu PZL M-18 wykazują dobrą zgodność wyznaczonych wartości c_{Lmax} i $\alpha\ c_{Lmax}$ konfiguracji przelotowej. Dla konfiguracji do lądowania wyznaczony współczynnik c_{Lmax} jest lekko zawyżony, podczas gdy c_{Lmax} jest wyznaczone poprawnie.