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"USSAERO" PROGRAM MODIFICATIONS FOR ASYMMETRIC FLOW ANALYSIS

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1. Introduction.

Based upon the elementary theoretical postulations of Woodward [1], this paper will try to give approach for asymmetrical flow analysis. Therefore, effects of sideslipping, rolling and command device deflections on aerodynamic load of aircraft will be obtained.

Influencing aerodynamic coefficients, representing induced velocity components over the normal outward direction in panel control points, in case of antisymmetric flow will be obtained by proper consideration of panels symmetricaly placed in respect to the plane of airplane symmetry.

Original concept of constant and linearly varying source and vortex distribution over the body and wing panels, will be the same as defined in [1].

2. Theoretical postulation.

2.0. Analysis of the system of linear equations. The system of linear equations representing a link between the boundary conditions of the tangential flow over the body and the unknown singularities in general (asymmetric) case may be given in the form:

$$\begin{aligned}
a_{1,1}\gamma_1 + \cdots + a_{1,N}\gamma_N + a_{1,N+1}\gamma_{N+1} + \cdots + a_{1,2N}\gamma_{2N} &= b_1,\\
a_{2,1}\gamma_1 + \cdots + a_{2,N}\gamma_N + a_{2,N+1}\gamma_{N+1} + \cdots + a_{2,2N}\gamma_{2N} &= b_2,\\
&\vdots\\
a_{N,1}\gamma_1 + \cdots + a_{N,N}\gamma_N + a_{N,N+1}\gamma_{N+1} + \cdots + a_{N,2N}\gamma_{2N} &= b_N,\\
a_{N+1,1}\gamma_1 + \cdots + a_{N+1,N}\gamma_N + a_{N+1,N+1}\gamma_{N+1} + \cdots + a_{N+1,2N}\gamma_{2N} &= b_{N+1},\\
&\vdots\\
a_{2N,1}\gamma_1 + \cdots + a_{2N,N}\gamma_N + a_{2N,N+1}\gamma_{N+1} + \cdots + a_{2N,2N}\gamma_{2N} &= b_{2N},
\end{aligned} \tag{2.1}$$

or in the matrix form

$$\begin{bmatrix} a_{1,1} & \cdots & a_{1,N} & a_{1,N+1} & \cdots & a_{1,2N} \\ \vdots & \ddots & \vdots & \vdots & \ddots & \vdots \\ a_{N,1} & \cdots & a_{N,N} & a_{N,N+1} & \cdots & a_{N,2N} \\ a_{N+1,1} & \cdots & a_{N+1,N} & a_{N+1,N+1} & \cdots & a_{N+1,2N} \\ \vdots & \ddots & \vdots & \vdots & \ddots & \vdots \\ a_{2N,1} & \cdots & a_{2N,N} & a_{2N,N+1} & \cdots & a_{2N,2N} \end{bmatrix} \begin{bmatrix} \gamma_1 \\ \vdots \\ \gamma_N \\ \gamma_{N+1} \\ \vdots \\ \gamma_{2N} \end{bmatrix} = \begin{bmatrix} b_1 \\ \vdots \\ b_N \\ b_{N+1} \\ \vdots \\ b_{2N} \end{bmatrix}, \tag{2.2}$$

or

$$[\mathbf{A}]\{\gamma\} = \{\mathbf{b}\}. \tag{2.3}$$

In the system of equations (2.1) the coefficients $a_{i,j}$ represent induced velocity components projected over the normal outward direction in panel control points; γ_i are unknown values of intensities of singularities to be determined (source and vortex density) and b_i corresponding boundary conditions, i.e. velocity pojections of undisturbed stream over the outward surface normal in corresponding control points. The total number of panels at a half of configuration symmetrical to the xOz plane is N, or $2N$ on the entire configuration.

In case of the symmetrical flow the following relations are:

$$b_1^s = b_{N+1}^s, \quad b_2^s = b_{N+2}^s, \quad \ldots, \quad b_N^s = b_{2N}^s \tag{2.4}$$

and accordingly:

$$\gamma_1^s = \gamma_{N+1}^s, \quad \gamma_2^s = \gamma_{N+2}^s, \quad \ldots, \quad \gamma_N^s = \gamma_{2N}^s. \tag{2.5}$$

Relations (2.4) represent equality of the boundary conditions on symmetrically positioned configuration panels (in respect to the symmetry plane xOz).

Based upon these relations the system of equations (2.1) degenerates into two identical systems where the first one, in the case of symmetrical flow is:

$$\begin{aligned} (a_{1,1} + a_{1,N+1})\gamma_1^s + (a_{1,2} + a_{1,N+2})\gamma_2^s + \cdots + (a_{1,N} + a_{1,2N})\gamma_N^s &= b_1^s, \\ (a_{2,1} + a_{2,N+1})\gamma_1^s + (a_{2,2} + a_{2,N+2})\gamma_2^s + \cdots + (a_{2,N} + a_{2,2N})\gamma_N^s &= b_2^s, \\ &\vdots \\ (a_{N,1} + a_{N,N+1})\gamma_1^s + (a_{N,2} + a_{N,N+2})\gamma_2^s + \cdots + (a_{N,N} + a_{N,2N})\gamma_N^s &= b_N^s, \end{aligned} \tag{2.6}$$

which written in the form of matrix becomes

$$\begin{bmatrix} (a_{1,1} + a_{1,N+1}) & (a_{1,2} + a_{1,N+2}) & \cdots & (a_{1,N} + a_{1,2N}) \\ (a_{2,1} + a_{2,N+1}) & (a_{2,2} + a_{2,N+2}) & \cdots & (a_{2,N} + a_{2,2N}) \\ \vdots & \vdots & \ddots & \vdots \\ (a_{N,1} + a_{N,N+1}) & (a_{N,2} + a_{N,N+2}) & \cdots & (a_{N,N} + a_{N,2N}) \end{bmatrix} \times$$

$$\times \begin{bmatrix} \gamma_1^s \\ \gamma_2^s \\ \vdots \\ \gamma_N^s \end{bmatrix} = \begin{bmatrix} b_1^s \\ b_2^s \\ \vdots \\ b_N^s \end{bmatrix}. \tag{2.7}$$

When the matrix of the influencing system coefficients (2.6) becomes diagonally dominant with non zero determinant, the problem can be solved by the selection of the suitable numerical procedure [3].

In case of the asymmetric flow around the symmetric configuration (in literature [4] the concept of the antimetric flow was introduced) it can be expressed as folows:

$$b_1^a = -b_{N+1}^a, \quad b_2^a = -b_{N+2}^a, \quad \ldots, \quad b_N^a = -b_{2N}^a. \tag{2.8}$$

Having in mind the relations (2.8) by addition of the initial N equations of the system (2.1) to the remaining ones of the same system, a new system of homogenous equations for antisymmetrical flow is obtained:

$$\begin{aligned}
&(a_{1,1}+a_{1,N+1})(\gamma_1^a+\gamma_{N+1}^a)+\cdots+(a_{1,N}+a_{1,2N})(\gamma_N^a+\gamma_{2N}^a)=0,\\
&(a_{2,1}+a_{2,N+1})(\gamma_1^a+\gamma_{N+1}^a)+\cdots+(a_{2,N}+a_{2,2N})(\gamma_N^a+\gamma_{2N}^a)=0,\\
&\qquad\vdots\\
&(a_{N,1}+a_{N,N+1})(\gamma_1^a+\gamma_{N+1}^a)+\cdots+(a_{N,N}+a_{N,2N})(\gamma_N^a+\gamma_{2N}^a)=0,
\end{aligned} \tag{2.9}$$

taking care about equations $a_{i,j} = a_{N+i,N+j}$, i.e. $a_{N+i,j} = a_{i,N+j}$. As the matrix of the system coefficients (2.9) is identical to the matrix system which is not singular (2.6), the system (2.9) can not have other solutions than trivial ones, i.e.:

$$\gamma_1^a + \gamma_{N+1}^a = 0, \quad \gamma_2^a + \gamma_{N+2}^a = 0, \quad \ldots, \quad \gamma_N^a + \gamma_{2N}^a = 0, \tag{2.10}$$

or

$$\gamma_i^a = -\gamma_{N+1}^a, \qquad i = 1,(1),N. \tag{2.11}$$

As in each case of flow around the symmetrical configuration may be observed as a sum of symmetrical and antisymmetrical flow, i.e.

$$b_i = b_i^s + b_i^a, \qquad i = 1,(1),N \tag{2.12}$$

therefore, the intensities of the corresponding singularities may be obtained by the relation

$$\gamma_i = \gamma_i^s + \gamma_i^a, \qquad i = 1,(1),N. \tag{2.13}$$

However, the calculation precision is obtained by fine geometrical discretization, i.e. by increasing the number of panels, for the more accurate calculations a relatively large number of panels should be used, which means that a large system of linear equations is to be solved, the order of which in the most general case (the system of equations (2.1)) becomes doubled. On the other hand this demands a large computer system storage and requires a long processing time. A convenient way to avoid such obstacles can be provided by the separation of the flow into symmetric and antisymmetric ones as well as by the use of relations (2.5) and (2.11) so that the unknown singularities over the entire configuration can be obtained by combining the solutions of one half of the aircraft only, i.e.:

$$
\begin{aligned}
\gamma_1 &= \gamma_1^s + \gamma_1^a, \\
\gamma_2 &= \gamma_2^s + \gamma_2^a, \\
&\vdots \\
\gamma_N &= \gamma_N^s + \gamma_N^a, \\
\gamma_{N+1} &= \gamma_1^s - \gamma_1^a, \\
\gamma_{N+2} &= \gamma_2^s - \gamma_2^a, \\
&\vdots \\
\gamma_{2N} &= \gamma_N^s - \gamma_N^a.
\end{aligned}
\tag{2.14}
$$

With the source density and circulation intensities thus determined velocity distribution per aircraft configuration can be also defined and according to it's pressure distribution as well as local and global and global aerodynamical coefficients [1].

2.1 Calculation of coefficients in case of asymmetrical flow. In the symmetrical flow the induced velocity components at control point i in respect to the axis of the local coordinate system of the panel i are

$$
\begin{aligned}
w_i''^s = \sum_{j=1}^{N} [& w_{ij}' \cos(\theta_j - \theta_i) + v_{ij}' \sin(\theta_j - \theta_i) \\
& + \overline{w'}_{ij} \cos(\theta_j + \theta_i) + \overline{v'}_{ij} \sin(\theta_j + \theta_i)] \gamma_j^s,
\end{aligned}
\tag{2.1.1}
$$

$$
\begin{aligned}
v_i''^s = \sum_{j=1}^{N} [& v_{ij}' \cos(\theta_j - \theta_i) - w_{ij}' \sin(\theta_j - \theta_i) \\
& - \overline{v'}_{ij} \cos(\theta_j + \theta_i) + \overline{w'}_{ij} \sin(\theta_j + \theta_i)] \gamma_j^s.
\end{aligned}
\tag{2.1.2}
$$

In the equations (2.1.1) and (2.1.2) w_{ij}' and v_{ij}' are velocity components induced at the control point i by the panel j and expressed in the local coordinate system of the panel j, while $\overline{w'}_{ij}$ and $\overline{v'}_{ij}$ represent velocity components induced at the control point symmetrical to the point i in respect to the symmetry plane xOz and expressed in the local coordinate system of the panel j.

In case of the antisymmetric flow, induced velocity components at the control point of panel i, taking care about the panel opposite to the symmetry plane [2], take the form of

$$
\begin{aligned}
w_i''^a = \sum_{j=1}^{N} [& w_{ij}' \cos(\theta_j - \theta_i) + v_{ij}' \sin(\theta_j - \theta_i) \\
& - \overline{w'}_{ij} \cos(\theta_j + \theta_i) - \overline{v'}_{ij} \sin(\theta_j + \theta_i)] \gamma_j^a,
\end{aligned}
\tag{2.1.3}
$$

$$
\begin{aligned}
v_i''^a = \sum_{j=1}^{N} [& v_{ij}' \cos(\theta_j - \theta_i) - w_{ij}' \sin(\theta_j - \theta_i) \\
& + \overline{v'}_{ij} \cos(\theta_j + \theta_i) - \overline{w'}_{ij} \sin(\theta_j + \theta_i)] \gamma_j^a.
\end{aligned}
\tag{2.1.4}
$$

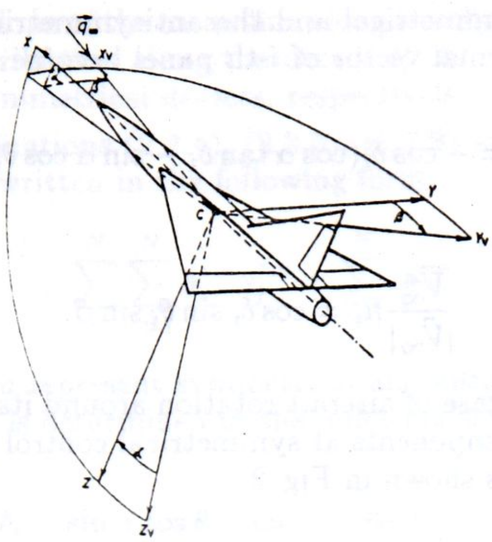

Fig. 1. Definition of α and β angles in respect to the global coordinate system

and the three components of induced velocity at i control point, projected over the axis of global (referent) coordinate system become

$$\Delta u_i^a = \sum_{i=1}^{N} (u'_{ij} - \overline{u'}_{ij})\gamma_j^a, \tag{2.1.5}$$

$$\Delta v_i^a = v_i''^a \cos\theta_i - w_i''^a \sin\theta_i, \tag{2.1.6}$$

$$\Delta w_i^a = w_i''^a \cos\theta_i + v_i''^a \sin\theta_i, \tag{2.1.7}$$

so that the normal component $\Delta\omega_i^a$ of induced velocity becomes

$$\Delta\omega_i^a = w_i''^a \cos\delta_i - \Delta u_i^a \sin\delta_i, \tag{2.1.8}$$

where δ_i is the panel inclination angle defined in [1].

2.2. Definition of the boundary conditions in cases of sliding, rolling and mechanisation deflection. Freestream velocity vector $\vec{V}_\infty$ according to Fig. 1, can be presented by the following relation:

$$\vec{V}_\infty = |\vec{V}_\infty|(\sin\alpha\cos\beta\vec{k} + \cos\alpha\cos\beta\vec{i} - \sin\beta\vec{j}), \tag{2.2.1}$$

where the symmetrical component is given by the first two terms within brackets of the equation (2.2.1), i.e.

$$\frac{\vec{V}^s_\infty}{|\vec{V}_\infty|} = (\sin\alpha\cos\beta\vec{k} + \cos\alpha\cos\beta\vec{i}) \tag{2.2.2.a}$$

and the antisymmetrical component by the third addend

$$\frac{\vec{V}^a_\infty}{|\vec{V}_\infty|} = -\sin\beta\vec{j}. \tag{2.2.2.b}$$

On the basis of [1], unit vector of the outward normal n_i to the panel i of the configuration observed is defined as

$$\vec{n}_i = -\sin\delta_i\vec{i} - \cos\delta_i\sin\theta_i\vec{j} + \cos\delta_i\cos\theta_i\vec{k}. \tag{2.2.3}$$

Projections of the symmetrical and the antisymmetrical velocity components over the unit outward normal vector of i-th panel become

$$\frac{\vec{V}_\infty^s}{|\vec{V}_\infty|}\vec{n}_i = -\cos\delta_i(\cos\alpha\tan\delta_i - \sin\alpha\cos\theta_i)\cos\beta \tag{2.2.4}$$

and

$$\frac{\vec{V}_\infty^a}{|\vec{V}_\infty|}\vec{n}_i = \cos\delta_i\sin\theta_i\sin\beta. \tag{2.2.5}$$

In rolling, i.e. in the case of aircraft rotation around its axis Ox, there are only antisymmetric velocity components at symmetrical control points in respect to the symmetrical plane xOz as shown in Fig. 2.

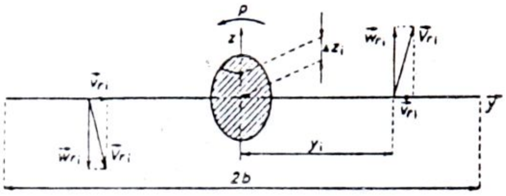

Fig. 2. Aircraft rotation around axis Ox (rolling)

It can be easily shown that velocity vector $\vec{V}_r$ components, due to rotation with angular velocity p around aircraft Ox axis, are equal to the following expressions:

$$V_{z_i} = py_i, \tag{2.2.6}$$
$$V_{y_i} = p\Delta z_i, \tag{2.2.7}$$

in which y_i and Δz_i represent the projections of control point i position vector upon the axis of the global coordinate system and thus velocity vector can be given in the form of

$$\vec{V}_r = -p\Delta z_i\vec{j} + py_i\vec{k}, \tag{2.2.8}$$

so the projection over the surface normal vector becomes

$$\frac{\vec{V}_r}{|\vec{V}_\infty|}\vec{n}_i = \cos\delta_i(\sin\theta_i\Delta z_i + \cos\theta_i y_i)\frac{2\omega}{b}, \tag{2.2.9}$$

where helix angle ω is determined by the expression

$$\omega = \frac{pb}{2V_\infty}. \tag{2.2.10}$$

In cases of mechanical deflection (ailerons, flaps, leading edge flaps) $\tan\delta_i$ from the equation (2.2.4) in the planar boundary condition option of the lifting surfaces can be written in the following form:

$$\tan\delta_i \approx \left(\frac{\partial z_c}{\partial x}\right)_i \pm \left(\frac{\partial z_t}{\partial x}\right)_i - \tan\delta_{M_i}^s - \tan\delta_{M_i}^a, \tag{2.2.11}$$

where $(\partial z_c/\partial x)_i$ is the slope of the mean line along the chord; $(\partial z_t/\partial x)_i$ — variation of profile thickness distribution along the chord; δ_i^s, δ_i^a — deflection angles of the symmetrical and antisymmetrical devices, respectively.

Considering the equations (2.2.4), (2.2.5), (2.2.9) and (2.2.11), the system of equations (2.1) can be written in the following form:

$$\sum_{i=1}^{N}\sum_{j=1}^{N} a_{ij}^{s,a}\gamma_j^{s,a} = \sum_{i=1}^{N}\omega_i^{s,a}, \tag{2.2.12}$$

in which indices s and a represent symmetrical and antisymmetrical flows, respectively. The value of ω_i^s is determined in the following way:

$$\omega_i^s = \cos\delta_i(\cos\alpha\tan\delta_i - \sin\alpha\cos\theta_i)\cos\beta - \cos\alpha\cos\beta\sum_{j=1}^{NW} b_{ij}\left(\frac{\partial z_t}{\partial x}\right)_j. \tag{2.2.13}$$

In the equation (2.2.13) NW indicates the number of panels on the wing, while the value of b_{ij} is defined in [1]. For the panels lying in the wing's plane in the planar boundary condition option of lifing surfaces, the following relation can be applied:

$$\left(\frac{\partial z_t}{\partial x}\right)_i \cos\delta_i = \sum_{j=1}^{NW} b_{ij}\left(\frac{\partial z_t}{\partial x}\right)_j, \tag{2.2.14}$$

so that the equation (2.2.13) becomes

$$\omega_i^s = \cos\delta_i\left[\cos\alpha\left(\frac{\partial z_t}{\partial x}\right)_i - \sin\alpha\cos\theta_i - \cos\alpha\tan\delta_{M_i}^s\right]\cos\beta. \tag{2.2.15}$$

The value of ω_i^a is defined by expression

$$\omega_i^a = -\cos\delta_i\left(\omega\frac{2\Delta z_i}{b}\sin\theta_i + \omega\frac{2y_i}{b}\cos\theta_i - \cos\alpha\tan\delta_{M_i}^a\cos\beta - \sin\beta\sin\theta_i\right). \tag{2.2.16}$$

2.3. Calculation of induced velocities and pressure distribution in the case of general flow. Solution of the equation system (2.2.12) determines vortex intensities and sources distributed over the wings and body panels in cases of symmetrical and antisymmetrical flows. Since the strengths of these singlularities are determined, three components of induced velocities in respect to the axis of the global coordinate system can be calculated. In the general case:

$$\begin{aligned} \Delta u_i &= \Delta u_i^s + \Delta u_i^a, \\ \Delta v_i &= \Delta v_i^s + \Delta v_i^a, \qquad i = 1,(1),N \\ \Delta w_i &= \Delta w_i^s + \Delta w_i^a \end{aligned} \tag{2.3.1}$$

or

$$\begin{aligned} \Delta u_{i+N} &= \Delta u_i^s - \Delta u_i^a \\ \Delta v_{i+N} &= -\Delta v_i^s + \Delta v_i^a, \qquad i = 1,(1),N \\ \Delta w_{i+N} &= \Delta w_i^s - \Delta w_i^a. \end{aligned} \tag{2.3.2}$$

In expressions (2.3.1) and (2.3.2.) $\Delta u_i^{s,a}$, $\Delta v_i^{s,a}$ and $\Delta w_i^{s,a}$ represent components of the perturbed velocity [1] at the i-th control point of the global coordinate system in the cases of symmetrical and antisymmentrical flows, respectively.

Projections of the total velocity vector over the axis of the global coordinate system at control point i can be obtained by relations:

$$\begin{aligned} U_i &= \Delta u_i + \cos\alpha\cos\beta, \\ V_i &= \Delta v_i - \sin\beta, \qquad i = 1,(1),2N \\ W_i &= \Delta w_i + \sin\alpha\cos\beta. \end{aligned} \tag{2.3.3}$$

In case of the planar boundary condition option of lifting surfaces due to wing thickness effect, the increase of velocity components in expressions (2.3.3) should also be considered.

Thus, the pressure coefficients can be calculated as in [1] on the basis of the exact formula of isentropic flow

$$c_{p_i} = \frac{2}{\kappa M^2}\left\{\left[1 + \frac{\kappa - 1}{2}(1 - q_i^2)\right]^{\kappa/(\kappa-1)} - 1\right\}, \tag{2.3.4}$$

where q_i is defined in the following manner:

$$q_i^2 = U_i^2 + V_i^2 + W_i^2, \tag{2.3.5}$$

while κ is air adiabatic coefficient ($\kappa = 1,4$).

2.4. Calculation of forces and moments on the basis of given pressure distribution. Aerodynamical forces and moments acting upon the aircraft configuration can be calculated by using numerical integration. Vertical force, tangential force and pitching moment arround the coordinate origin of the panel i local coordinate system are calculated in [1], i.e., as:

$$\begin{aligned} N_i &= -A_i c_{p_i} \cos\theta_i \cos\delta_i, \\ T_i &= A_i c_{p_i} \sin\delta_i, \\ M_i &= N_i x_i - T_i z_i, \end{aligned} \tag{2.4.1}$$

where A_i is the area of panel i, θ_i i-th panel inclination angle defined in [1] and x_i, z_i control point coordinates.

When the flow is asymmetric, integration will be done for each configuration side separately. In such a case the lateral force and moments of yawing and rolling are:

$$\begin{aligned} Y_i &= A_i c_{p_i} \cos\delta_i \sin\theta_i, \\ M_{z_i} &= -Y_i x_i - T_i y_i, \\ M_{x_i} &= -Y_i z_i + N_i y_i. \end{aligned} \tag{2.4.2}$$

The usual coefficient of aerodynamic forces and moments acting upon each configuration side are obtained by a summation of forces and moments on all panels of the given configuration:

$$c_N = \frac{1}{S}\sum_{i=1}^{N} N_i, \qquad c_T = \frac{1}{S}\sum_{i=1}^{N} T_i, \qquad c_M = \frac{1}{S\bar{c}}\sum_{i=1}^{N} M_i,$$

$$c_Y = \frac{1}{S}\sum_{i=1}^{N} Y_i, \qquad c_{M_z} = \frac{2}{Sb}\sum_{i=1}^{N} M_{z_i}, \qquad c_{M_x} = \frac{2}{Sb}\sum_{i=1}^{N} M_{x_i}. \qquad (2.4.3)$$

It should be emphasized that the total aerodynamic coefficients for the given configuration are obtaind by adding up those for both configuration sides calculated by the expression (2.4.3.)

Now it is possible to determine the lift and drag coefficients in the following way:

$$c_L \approx c_N \cos\alpha - c_T \sin\alpha \qquad (2.4.4.a)$$

$$c_D \approx c_N \sin\alpha + c_T \cos\alpha. \qquad (2.4.4.b)$$

The computer program "USSAERO", as already mentioned in [1], calculates forces and moments on the body, wing, and tail surfaces as well as entire configuration. It also computes local values of aerodynamic coefficients on body and wing segments.

3. Conclusions

The example of the ogive-cylindric body [1] was used for the analysis of sliding effects in subsonic and supersonic flows. Due to the double symmetry in respect to xOz and yOz planes, the results in the case of the attack angle α without sideslip β (symmetrical flow) and sideslip at zero angle of attack (antisymmetric flow included) are identical if the values of these angles are the same, whereby one should have in mind that a 90° rotated panel in the asymmetric flow corresponds to a panel in a symmetrical one.

Comparing the results of the modified "USSAERO" program in the case of mechanical deflections (ailerons) with the results of the program "LOAD" [2], the conclusion is drawn that the coincidence of global and local aerodynamic coefficient values obtained as well as pressure distributions are satisfactory in respect to the wide spectrum of subsonic flow of Mach numbers. It should be added that the results were compared for the case of the high aspect ratio rectangular wing of zero thickness in the "USSAERO" program. The introduction of thickness slightly increases the values of the aerodynamic coefficients obtained for zero thickness wings (shape of the flat plate).

The results for the case of rotation around Ox axis (rolling) for the rectangular wing from the previous example in the subsonic flow also coincided properly with the results obtained by "LOAD" program.

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МОДИФИКАЦИЯ ПРОГРАММЫ "USSAERO" С АСПЕКТА НЕССИМЕТРИЧЕСКОГО ТЕЧЕНИЯ

В работе предлагается алгоритм численного анализа течения около сложной конфигурации в трехмерном несимметрическом полете. С помощю модификации и дополнения программы "USSAERO" возможен расчеть аэродинамической нагрузки самолета в условиях скольжения, крена и отклонения командной механизации. Выведенный анализ относится только в случае плоскостных граничных условиях.

MODIFIKACIJA PROGRAMA "USSAERO" SA ASPEKTA NESIMETRIČNIH OPTEREĆENJA

U radu je razvijen postupak kompjuterske analize strujanja oko složene konfiguracije u trodimenzionalnom prostoru u nesimetričnim slučajevima leta. Modifikacije i dopune paketa „USSAERO“ omogućile su proračun aerodinamičkog opterećenja letelice u slučajevima klizanja, valjanja i otklona komandnih površina. Izvedena analiza odnosi se samo za slučaj planarnog (ravanskog) predstavljanja uzgonskih površina.

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