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CALCULATION OF DYNAMIC STALL ON AN OSCILLATING AIRFOIL

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A mathematical model of an unsteady separated flow around an oscillating airfoil is considered. This model is based on a viscid-inviscid approach. The points of separation and the intensity of vorticity displaced into the external flow are determined using boundary-layer equations in an integral form. Dynamic stall on an oscillating airfoil is studied. The mechanism and nature of antidamping are discovered.

A number of critical regimes of the flow around the blades of helicopters, wind turbines, compressors, and also the wings of airplanes are caused to a large extent by flow separation. These regimes can be accompanied by worsening of the lifting properties, ambiguity of the characteristics on direct and reverse strokes, loss of damping, and the appearance of self-induced oscillations. Experimental and numerical investigation of the flow and its unsteady aerodynamic characteristics in these regimes, and also the attendant processes is a challenging problem that has to be solved to develop recommendations for designing lifting systems. A number of papers (see, for example, [1–6]) are devoted to the study of unsteady aerodynamic characteristics of airfoils in separated flow regimes. The rather simple numerical approach proposed in this paper allows one to obtain results that are important in practice in considering a separated flow around an oscillating airfoil whose study has not been completed yet.

1. An unsteady separated flow around an airfoil oscillating in accordance with a given law in a viscous incompressible fluid flow is considered. The algorithm of solving the problem of determining unsteady aerodynamic characteristics is constructed on the basis of the viscid-inviscid approach [1, 7]. The essence of this approach is a consecutive solution of two problems at each time step. At the first stage, based on the assumption of a potential external flow, the characteristics of an inviscid flow are determined, including the velocity distribution on the body surface. Then, using the known parameters of the external flow, separation points (for example, R_1 in Fig. 1) and the parameters of the separated portions of the boundary layer are found. Boundary-layer equations in an integral form are used for this purpose.

In considering the inviscid flow problem, it is assumed that one of the lines of the contact discontinuity L_2 emanates from the upper smooth surface of the airfoil and the other L_1 from its sharp trailing edge (Fig. 1). This flow scheme for the class of problems considered, i.e., for a separated flow around airfoils of relative thickness $c > 0.12$ (the thickness is normalized to the airfoil chord b) and Reynolds numbers $\mathrm{Re} > 10^6$, is justified by the known experimental data [8]. These data allow us to conclude that a simplified flow scheme with an assumption of prevailing turbulent separation can be used for numerical simulation of an unsteady separated flow at high, practically important Reynolds numbers ($\mathrm{Re} > 10^6$) for rather thick airfoils ($c > 0.12$). This assumption cannot be extended to thin airfoils and to low Reynolds numbers for which the effect of closed circulation regions (bubbles) can be determining.

2. In the course of numerical solution of the problem, the airfoil contour is approximated by a polygon, and each side of this polygon is an elementary panel (Fig. 1). On each panel, there is a continuous vortex layer with an unknown dimensionless overall running intensity varying in accordance with a linear law. Free

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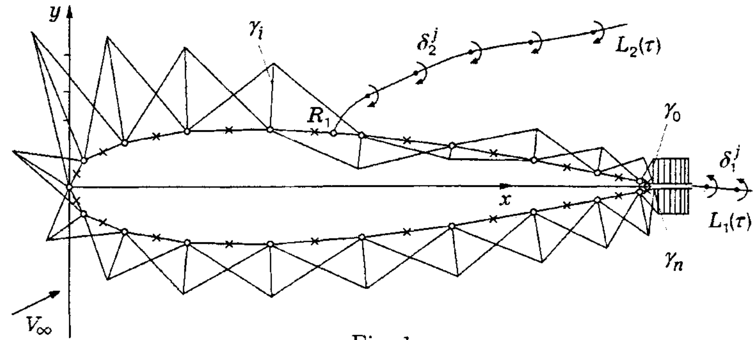

Fig. 1

vorticity shed from the upper and lower surfaces of the airfoil on its trailing edge is simulated by two panels with constant intensities γ_0 and γ_n (Fig. 1). At the next time steps, these panels are replaced by one discrete free vortex with circulation δ_1^j.

Following the panel method [9], we consider a triangular distribution of the vortex intensity in panel nodes γ_i. From the no-slip conditions satisfied at control points located in the middle of the panels and from the condition of constant circulation over a closed liquid contour covering the entire vortex system, we write a system of linear algebraic equations with respect to unknown intensities γ_i. This system is solved at each time step. The boundary layer separated from the upper surface of the airfoil is modeled by a system of discrete vortices with circulations δ_2^j. Unsteady processes in the external flow are assumed to be significantly more inertial than the boundary-layer processes for the class of problems considered [10]. Therefore, the boundary-layer parameters are calculated using the method of E. Truckenbrodt in a quasi-steady formulation: apart from good agreement with experimental results, this method is remarkable for its relative simplicity and compact form of calculations. The positions of free vortices with circulations δ_1^j and δ_2^j at an arbitrary time are determined by solving the equations of motion. The Cauchy–Lagrange integral is used to find unsteady aerodynamic characteristics of the airfoil.

The algorithm proposed is universal and allows one to pass easily to consideration of an unsteady nonseparated flow (for $\delta_2^j = 0$) or a steady nonseparated flow (for $\delta_1^j = 0$ and $\delta_2^j = 0$).

The studies performed allow us to evaluate the effect of the number of vortex panels on the airfoil and the time step on calculation accuracy. In particular, by comparison with exact solutions and known numerical and experimental data, the number of panels on the airfoil n was chosen equal to 40, and the time step was $\Delta\tau = 0.05$. Here $\tau = tV_\infty/b$ is the dimensionless time (V_∞ is the free-stream velocity and t is the time).

3. Dynamic stall refers to the most interesting and practically important critical regimes of the flow around airfoils. As an example, we consider an NACA 0012 airfoil oscillating relative to the angle of attack in accordance with a harmonic law, $\alpha = \alpha_0 + \theta \sin(\omega\tau)$. Numerical and experimental results for the parameters $\alpha_0 = 15°$, $\theta = 10°$, and $\omega = 0.3$ are presented below. The center of gravity (i.e., the dimensionless coordinate of the axis of revolution of the airfoil along the chord) was $x_{\rm pitch} = 0.25$ and $\mathrm{Re} = 2.5 \cdot 10^6$. This regime was previously studied experimentally [11], but some special features of the mechanism of formation of unsteady aerodynamic characteristics were not explained.

Along with investigation of the total unsteady aerodynamic characteristics, the algorithm developed allows one to study the flow structure around the airfoil. Figure 2 shows a sequence of physical processes and the corresponding unsteady aerodynamic characteristics obtained for an oscillating airfoil in numerical (Fig. 2, left) and physical [11] (Fig. 2, right) experiments. Note that the flow structure shown in scheme (e), which was obtained in the numerical experiment, could not be found in the physical experiment [11]. The dashed curves in Fig. 2 show the static values of the coefficients c_y and m_z versus the angle of attack, which were obtained in the direct stroke of the airfoil ($\dot\alpha > 0$) both in the experiment and calculation for $\tau \to \infty$.

The flow pattern shown by scheme (a) is gradual evolution of diffuser stall (shedding from the trailing edge). Because of flow inertia, the pre-separation flow is rather extended.

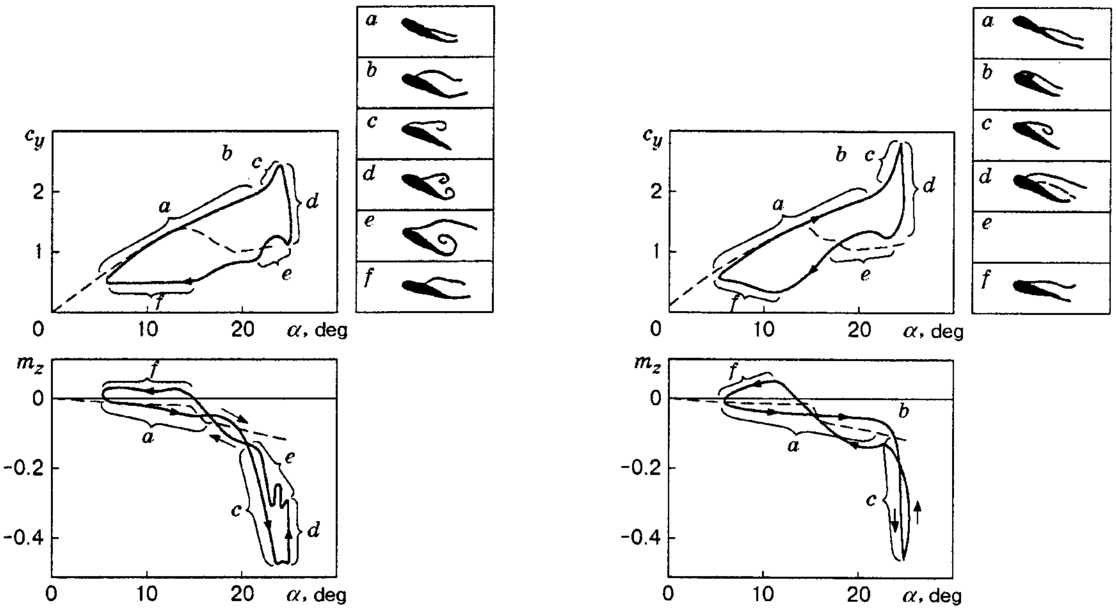

Fig. 2

Both in physical and numerical experiments, scheme (b) corresponds to the moment of vortex incipience in the vicinity of the leading edge of the airfoil.

The subsequent dramatic increase in the lifting force and pitching moment is associated in both experiments with the action of a developing vortex of dynamic stall [scheme (c)]. For brevity, we will call it the dynamic vortex.

Scheme (d) corresponds to detachment of the dynamic vortex, which results in dramatic worsening of the lifting properties of the airfoil. In the numerical experiment, it was found that a trailing vortex originates in the vicinity of the trailing edge, whose powerful induction is responsible for the drastic increase in the coefficient c_y at the reverse stroke [scheme (e)]. A similar phenomenon is also observed in the experiment, but this effect has not found a physically grounded explanation in [11].

The interaction of the dynamic (leading) and trailing vortices is manifested most clearly in the behavior of the pitching moment. It is known that, under oscillations of the airfoil with one degree of freedom relative to the pitching moment, the dynamic stall is accompanied by the loss of stability of motion [11], which is manifested in changing the direction of motion along the loop $m_z(\alpha)$, i.e., an antidamping regime arises, which corresponds to the clockwise motion along the loop.

We describe the formation mechanism of antidamping. Because of the delay in displacement of the separation point, a nonseparated flow is retained in the vicinity of the tip of the airfoil, and the moment does not "fall" into the negative region [the loop on the curve $m_z(\alpha)$ corresponding to scheme (a)]. Then a dynamic vortex arises [scheme (b)], which generates a large negative moment [scheme (c)]. Then there arises a trailing vortex, which displaces the dynamic vortex and, having the opposite direction, compensates the rarefaction at the rear end of the airfoil. Therefore, the moment increases dramatically [scheme (d)] but does not exceed the moment on the direct stroke, since the flow is already shed at the tip of the airfoil [scheme (e)]. A consequence of this process is the formation of an antidamping region in a certain vicinity of the mean angle of attack. It should be noted that displacement of the dynamic vortex by the trailing vortex hinders the development of a vast region of antidamping in the vicinity of the maximum angle of attack.

Scheme (f) corresponds to the flow around the airfoil in the regime of deep stall. As the angle of attack decreases in a certain vicinity of $\alpha_{\min}$, the pre-separation characteristics are rather rapidly recovered.

The good agreement of steady aerodynamic characteristics obtained in the calculation and experiment

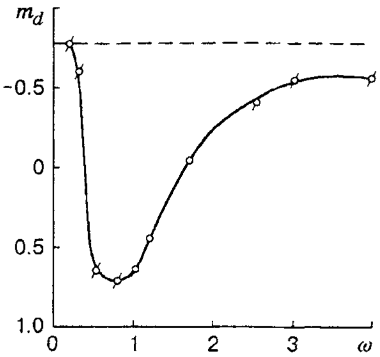

Fig. 3

(dashed curves in Fig. 1) indicates that the algorithm can also provide credible results for a steady flow around airfoils of moderate thickness ($c > 0.12$) at high Reynolds numbers ($\mathrm{Re} > 10^6$).

By means of numerical simulation, we considered physical features of formation of unsteady aerodynamic loads on an NACA 0012 airfoil in a wide range of variation of the reduced oscillation frequency ω. Figure 3 shows the dependence of the coefficient of harmonic linearization of the pitching moment $m_d = m_z^{\dot{\alpha}} + m_z^{\omega_z}$, which determines aerodynamic damping, on ω. The dashed line in this figure shows the dependence of this coefficient in an attached flow. It follows from the data in Fig. 3 that three characteristic ranges of oscillation frequencies ω can be distinguished: $\omega < 0.4$, $0.4 < \omega < 1.7$ (region of antidamping), and $\omega > 1.7$.

A comparison of the considered dependence of the coefficient of harmonic linearization of the pitching moment on ω with the corresponding dependence obtained for a nonseparated flow confirms that the antidamping region found is determined by dynamic stall of the flow.

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