Numerical Simulation of Separated Flows around a Wing Section at Steady and Unsteady Motion by a Discrete Vortex Method

By

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Summary: Separated flows around a wing section at fixed attack angle of an oscillating airfoil are simulated numerically by a discrete vortex method combined with a panel method. The potettial flows around wing sections are expressed by vortex sheets and separated shear layers are expressed by discrete vortices.

In the calculation a separation point is determined by solving boundary layer equation. The strength of shed vortex is estimated using local velocity near separation point. Also modification for the estimation of pressure coefficients around wing section are proposed.

The estimated pressure distributions show good agreements with experimental results. Aerodynamic characteristics of a wing section of NACA4412 are predicted with excellent agreement with experiments. Also separated flows around pitching airfoils are simulated for various conditions. A hysteresis of lift of airfoil at dynamic stall is obtained. Further unsteady flow caused by abrupt change of the attack angle from 10 to 20 degrees is simulated numerically. The transient flow patterns show significant characteristics of unsteady flow. The hysteresis of the lift of piching airfoil is simulated properly.

1. Introduction

A combination of the discrete vortex method and a panel method, in which potential flow around a body is expressed by a set of singular points distributed on the body surface, was proposed[1]. The present authors have conducted the calculation of separated flows around wing sections. In the calculation a wing section is expressed by a set of linearly distributed vortex sheets and the separation points are determined by boundary layer calculations. Separated shear layers are expressed as rows of discrete vortices. A new procedure for the estimation of pressure coefficients is proposed. The calculated pressure distributions show good agreements with experimental results. The separated flow around a wing section at a specified angle of attack are simulated properly and the results showed good agreements with experiments[2].

Also separated flows around sinusoidally pitching wing sections are simulated by the same method. The applicability of the method for the calculation in those

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severe situations is examined. The results show good agreements with experimental results. Especially a hysteresis of lift coefficient at higher angle of attack is predicted.

Further unsteady flow field around a wing section under the abruptly increase of attack angle is simulated. The transient change of the flow fields is simulated properly. The maximum peak of lift during transient processes are well predicted.

2. Analytical Method

The complex potential f of the flow field is expressed by the following form:

$$f = Ve^{-i\alpha}z + i\int_B \frac{\gamma(\zeta)}{2\pi}\log(z-\zeta)\,d\zeta$$
$$+ i\sum_{K=1}^{M}\left\{\frac{\Gamma_{A_k}}{2\pi}\log(z-z_{A_k}) + \frac{\Gamma_{B_k}}{2\pi}\log(z-z_{B_k})\right\} \tag{1}$$

, where U is the uniform flow velocity and α is the angle of attack. For the calculation of the flow around the pitching wing section, incident angle of the freestream is sinusoidally changed. Then the angle of attack of the wing section is given by following form:

$$\alpha = \alpha_0 + \alpha_1 \sin\omega t \tag{2}$$

, where the angular frequency ω is related to the reduced frequency k by the following form:

$$k = \frac{\omega c}{2V} \qquad (c\text{: chord length}) \tag{3}$$

In the estimation of pressure coefficient, two modifications are proposed. One is for the evaluation of the potential of the flow field and the other is for the correction of loss of the kinetic energy in the separated region[2].

3. Results and Discussions

The wing section used for the present calculation is NACA 4412 as shown in Fig. 1. As the separation point on the wing section is not obvious a priori, the separation point is determined by the boundary layer calculations. The Thwaites'method[3] is used for the laminar boundary layer calculation and the Truckenbrodt's method[4] is used for the turbulent boundary layer calculation.

3.1 The separated flows around a wing section at a fixed angle of attack

Fig. 2 shows the calculated stream lines at angles of attack of 14, 18, and 20 degrees. The shedding vortices from upper and lower surfaces of the wing section are described by circular symbol and triangle symbol respectively. Fig. 3 shows the stream lines at various instances at angle of attack of 20 degrees. The pressure distribution and the values of Cl and Cd are shown in Fig. 4. In the figure it is apparent that the calculated pressure distribution shows good agreement with that by experiment[5] including the separated region. Fig. 5 shows Cl and Cd with α. A

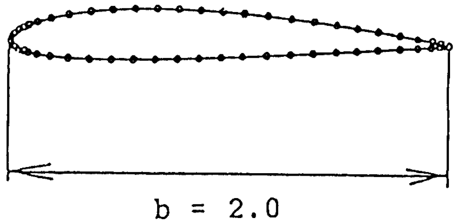

Fig. 1. Model used for the present calculation (NACA4412)

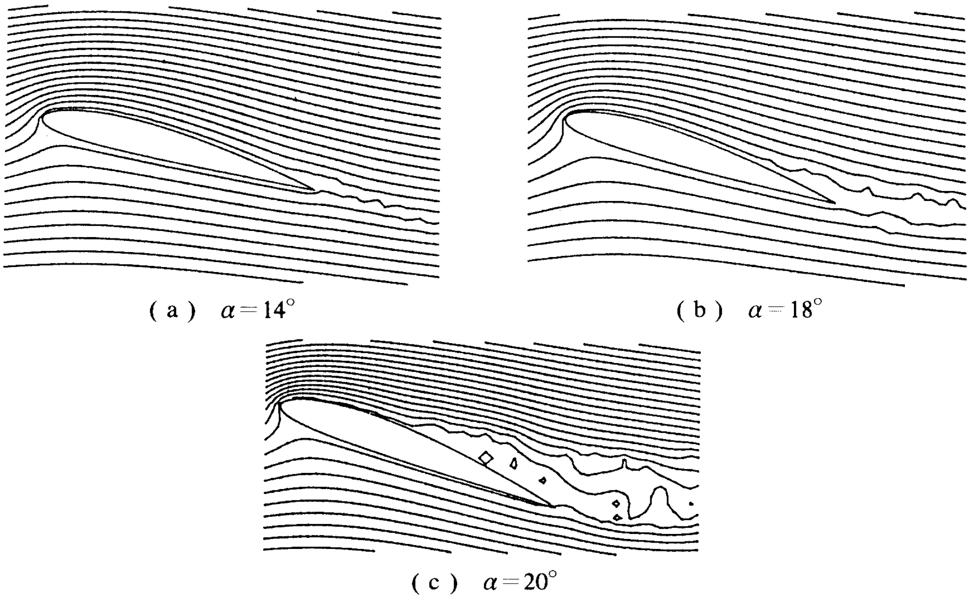

Fig. 2. Flow pattern (NACA4412)

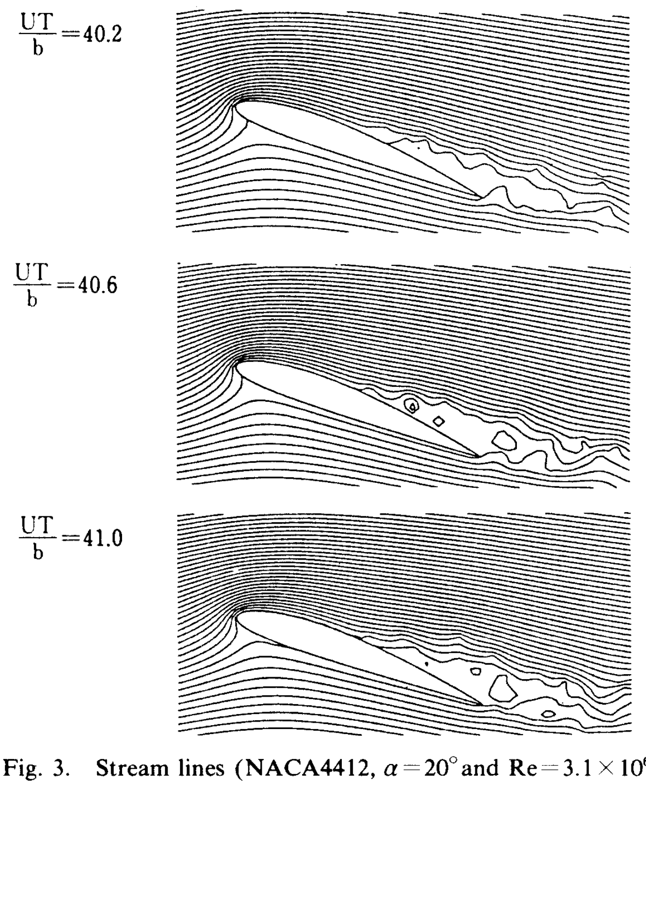

Fig. 3. Stream lines (NACA4412, $\alpha=20°$ and $Re=3.1\times10^6$)

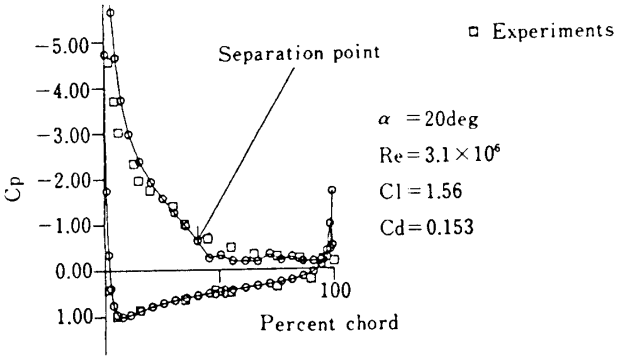

Fig. 4. Pressure distribution (NACA4412, $\alpha = 20^\circ$)

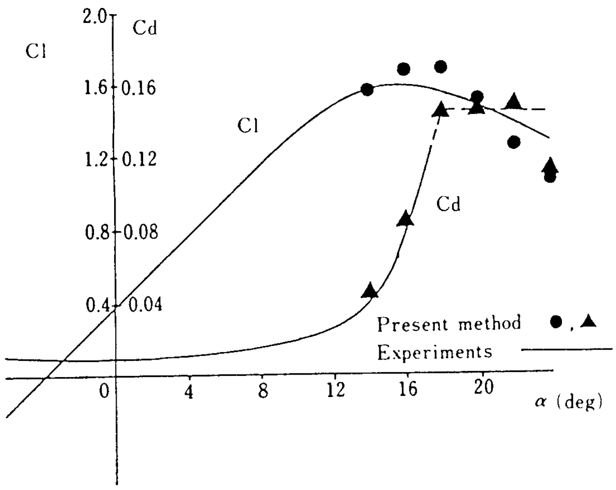

Fig. 5. Comparison of Cl and Cd curves with experiments (NACA4412)

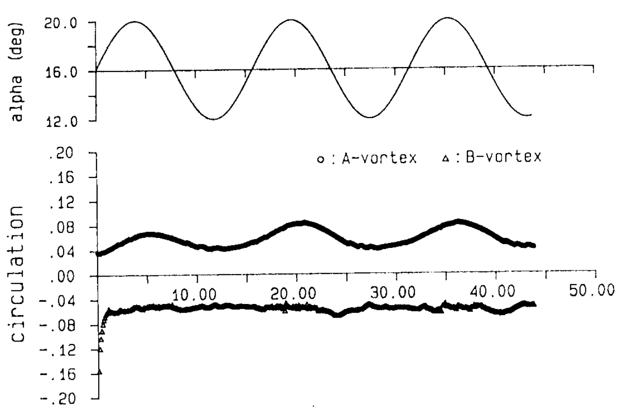

Fig. 6. Temporal changes of attack angle and strength of shed vortex of pitching airfoil (NACA4412)
($\alpha = 16^\circ + 4^\circ \sin \omega t,\ k = 0.2$)

good agreement of Cl between the calculations and the experiments is obtained. Also the results of Cd show good agreement with the experiments.

3.2 The separated flows around a pitching wing section

At first the calculated results under the condition of $\alpha = 16^0 + 4^0 \sin \omega t$ and the reduced frequency k of 0.20 is shown in Figs. 6–9. Fig. 6 shows the time histories of the circulations of shed vortices and the angle of attack. It shows that the circulations vary periodically with the angle of attack. Fig. 7 and Fig. 8 show the flow pattern and the stream line respectively. The figures show that the separation point, the flow pattern and the extent of the separated region are different both in α-increasing process ($\omega t = 3.56\pi - 4.33\pi$) and the α-decreasing process ($\omega t = 4.71\pi -$

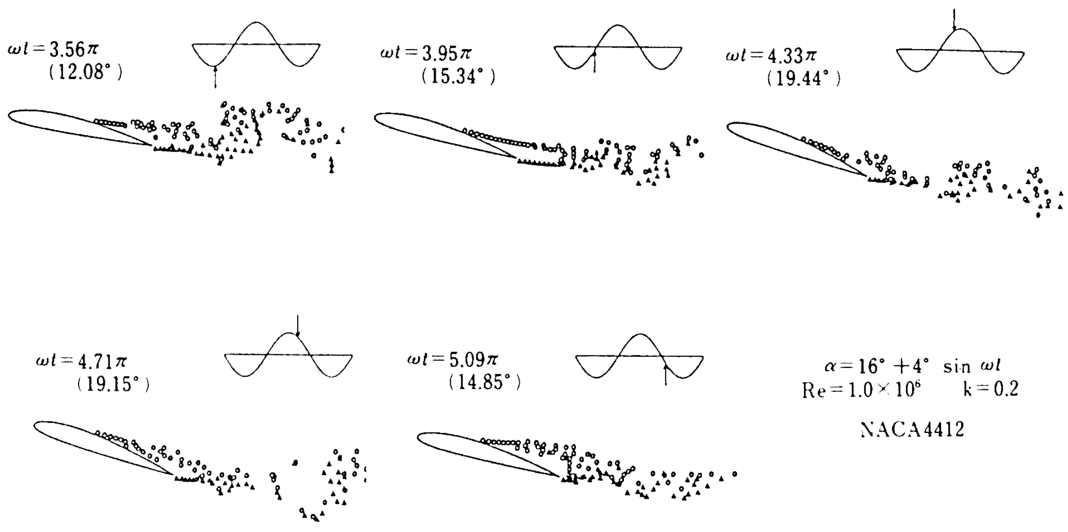

Fig. 7. Flow pattern around pitching airfoil (NACA4412)
($\alpha = 16^\circ + 4^\circ \sin \omega t,\ k = 0.2$)

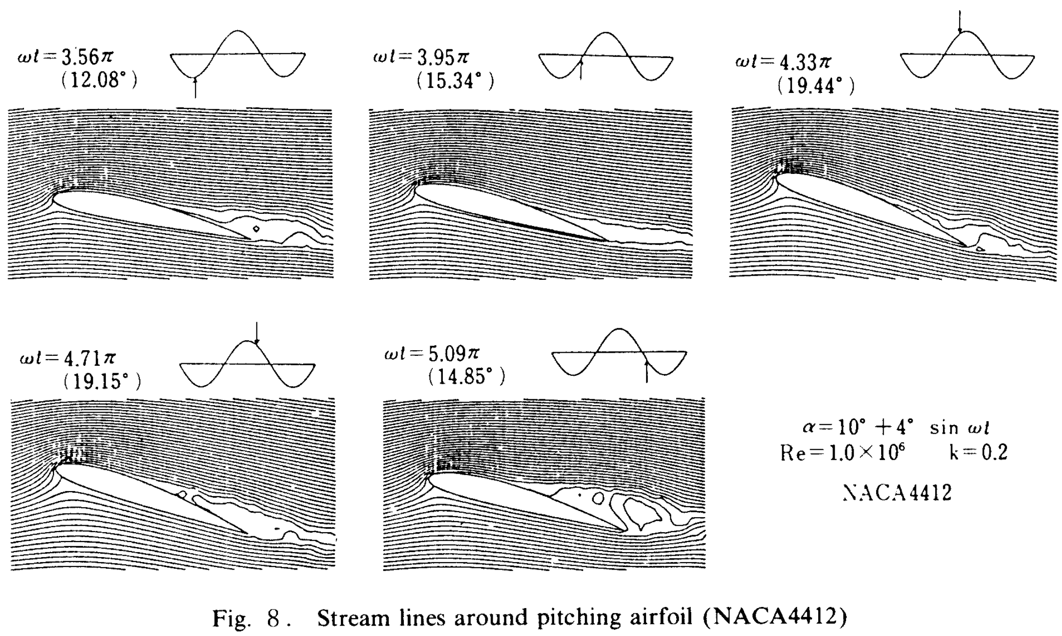

Fig. 8. Stream lines around pitching airfoil (NACA4412)
($\alpha = 16^\circ + 4^\circ \sin \omega t,\ k = 0.2$)

5.09π) even at the same angle of attack in a cycle. The flow pattern in the α-decreasing process is a little different from that in the α-increasing process as separation point is located more forward in α-decreasing process. The thickness of separated region is of the order of the wing section thickness and that is the feature of the light stall regime of the dynomic stall[7]. Fig. 9 shows the behavior of Cl versus α. At higher angle of attack lift in the α-increasing process is larger than that of α-decreasing process. However at lower angle of attack no difference between in α-increasing process and in α-decreasing process can be seen.

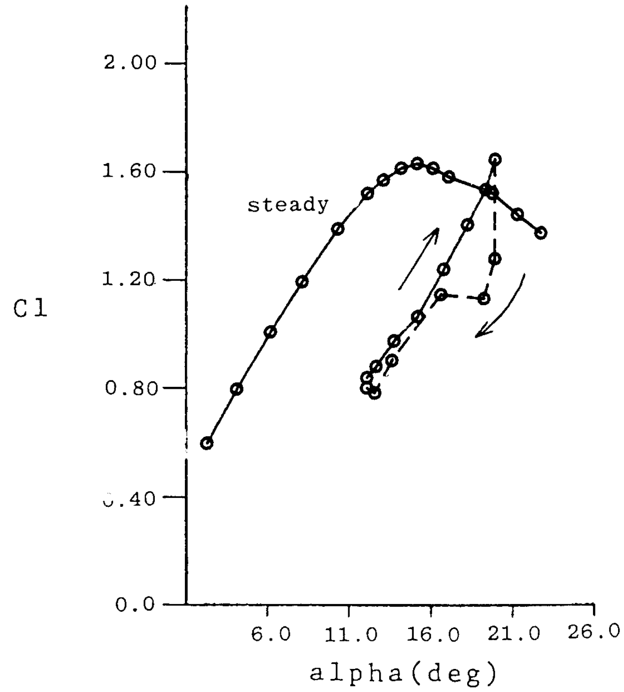

Fig. 9. Hysteresis of Cl of pitching airfoil (NACA4412) ($\alpha = 16° + 4° \sin \omega t$, $k = 0.2$)

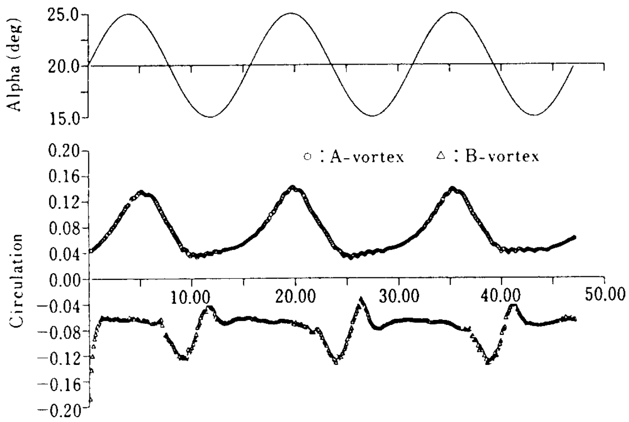

Fig. 10. Temporal change of attack angle and strength of shed vortex of pitching airfoil (NACA4412) ($\alpha = 20° + 5° \sin \omega t$, $k = 0.2$)

Figs. 10–13 show the calculated results under the condition of $\alpha = 20^0 + 5^0 \sin \omega t$ and the reduced frequency k of 0.20. When α is decreased, the separation point is located more forward and the separation region over the wing section is more extensive and complex compared with α-increasing process. In Fig. 13 extensive region of reversed flow and vortices over the wing section can be seen. Also the thickness of separated region is of the order of the wing section chord during the α-decreasing process. Those are the features of the deep stall regime. Fig. 12 shows the behavior of Cl versus α. Lift in the α-increasing process is larger than that in the α-decreasing process. And the maximum lift is larger than that for steady case.

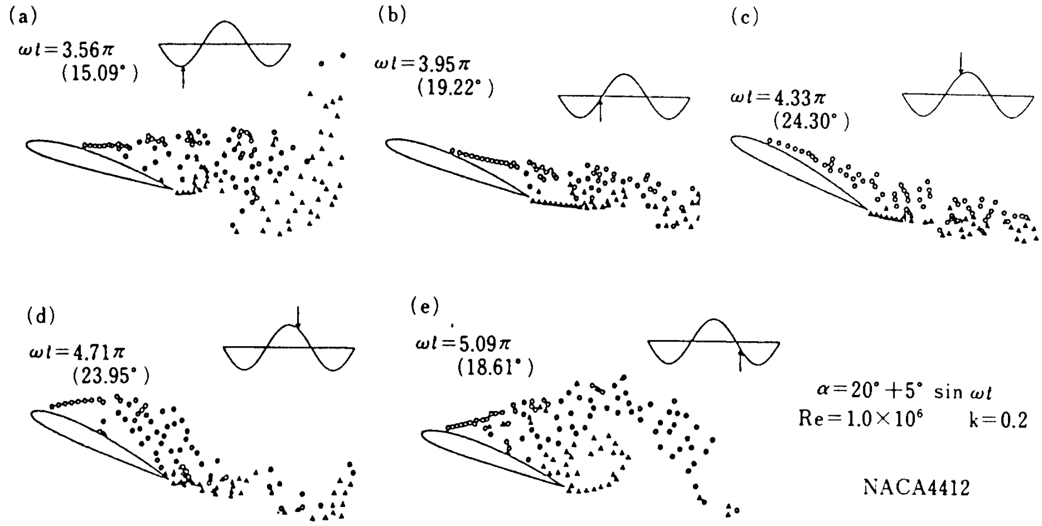

Fig. 11. Flow pattern around pitching airfoil (NACA4412)
($\alpha = 20^\circ + 5^\circ \sin \omega t$, $k = 0.2$)

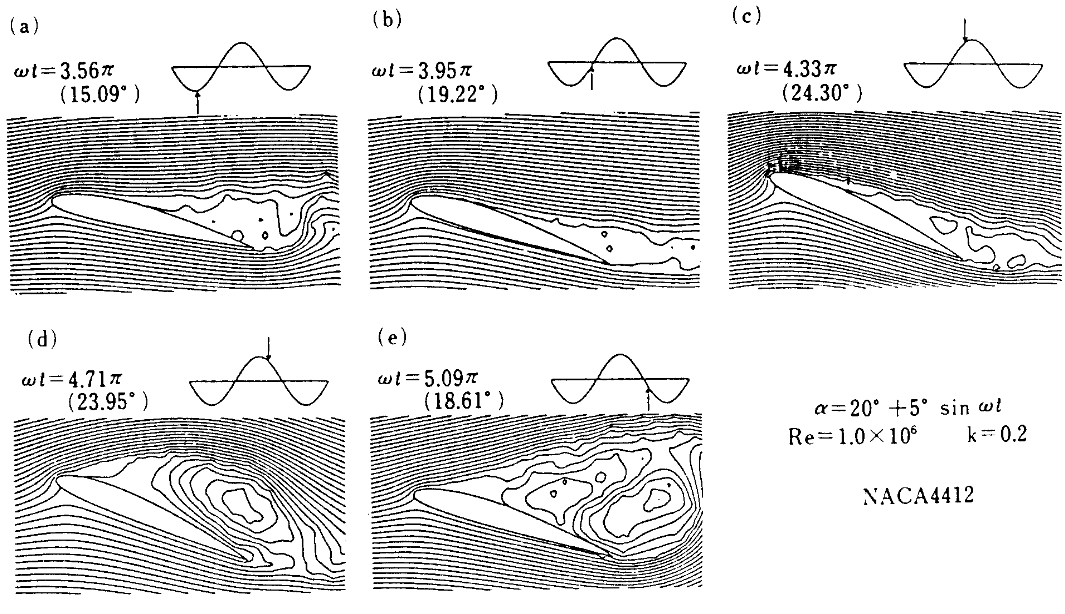

Fig. 12. Stream lines around pitching airfoil (NACA4412)
($\alpha = 20^\circ + 5^\circ \sin \omega t$, $k = 0.2$)

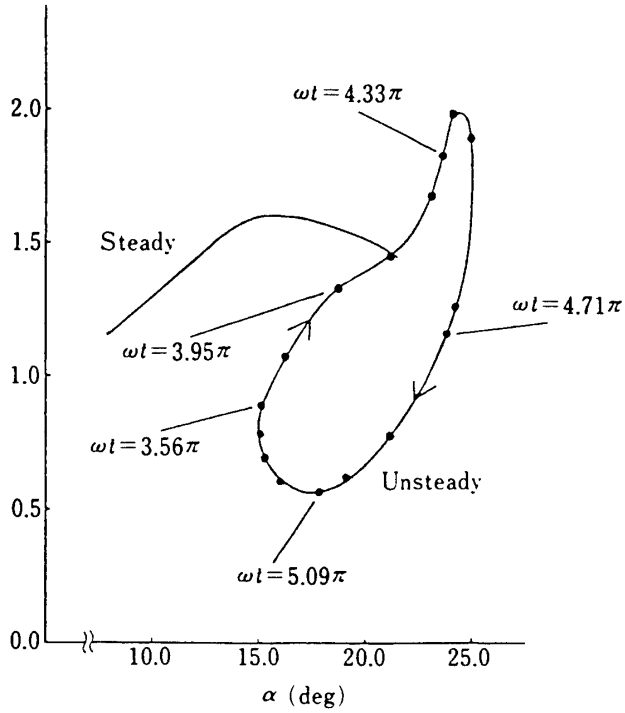

Fig. 13. Hysteresis of Cl of pitching airfoil (NACA4412) ($\alpha = 20° + 5° \sin \omega t$, $k = 0.2$

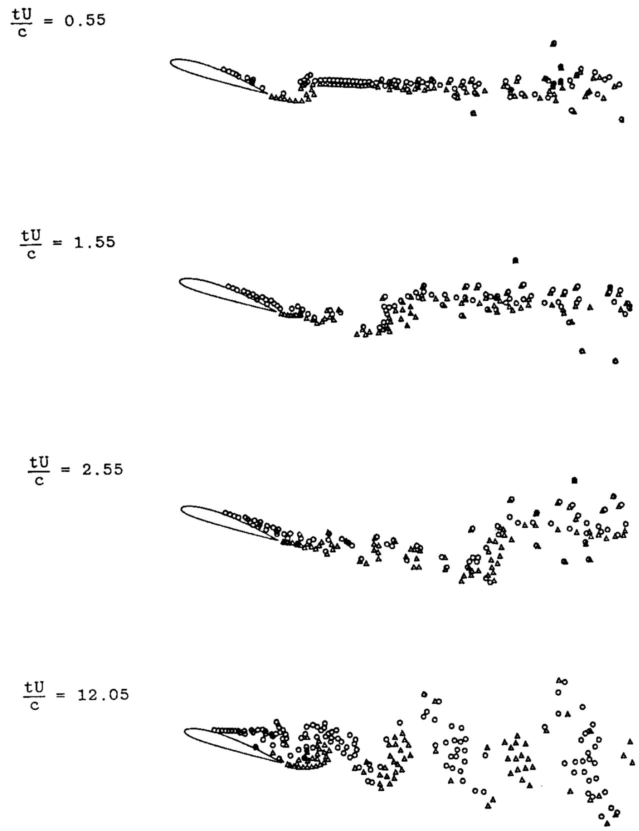

Fig. 14. Transient flow patterns around wing section under abrupt increase of attack angle (NACA4412)

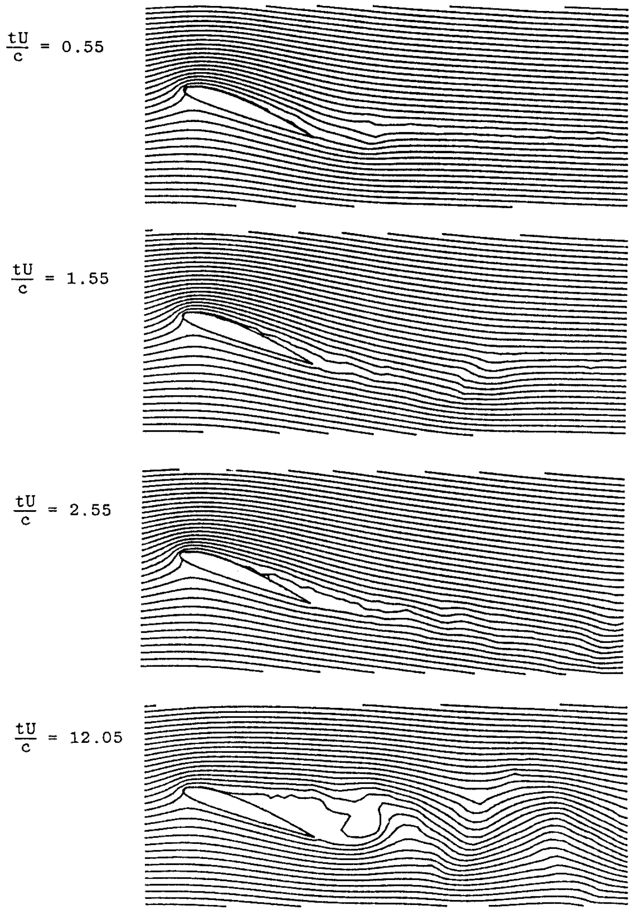

Fig. 15. Transient stream lines around wing section under abrupt increase of attack angle (NACA4412)

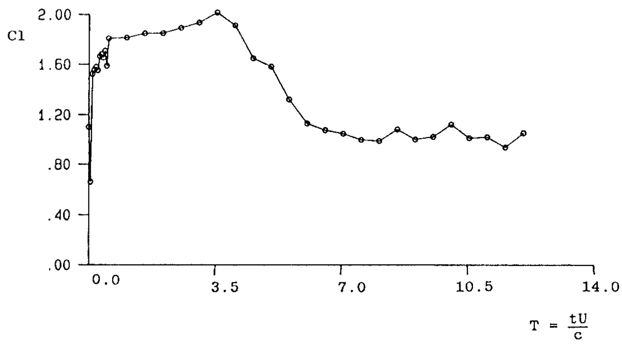

Fig. 16. Time histories of Cl of wing section under abrupt increase of attack angle (NACA4412)

Also those features express the characteristics of the deep stall regime qualitatively.

3.3 Separated flow around wing section under abrupt increase of attack angle

The calculated results under the abrupt increase of angle of attack from 10 degrees to 20 degrees are shown in Figs. 14–16. Fig. 14 and Fig. 15 show the calculated flow pattern and the stream lines respectively. As shown in the figure the vortex generated at the trailing edge leaves downstream and interacts with a row of vortices which have been shed before the attack angle is increased. Just after the increase of angle of attack, the separation region over the wing section is small and simple. However, the separation point moves forward gradually and the separation region over the wing section comes more extensive and complex. Fig. 16 shows the time histories of the Cl. After the increase of angle of attack, Cl reaches maximum value, and then decreases and becomes almost constant. The overshoot of lift coefficient in transient process is quite similar to the case which is observed by Aihara in the experiments with a wing section of NACA0012 during stepwise increase of attack angle[8).

4. Conclusions

The separated flows around wing sections have been calculated by the method that the body is expressed by a set of linearly distributed vortex sheets and the separated shear layer is expressed by a row of the discrete vortices. And new procedures for the estimation of the pressure coefficient are proposed. The calculated Cp, Cl and Cd show good agreements with experimental results. The results prove that the method is quite useful for the simulation of separated flows around wing sections. Also separated flows around the pitching airfoil and wing section under abrupt increase of attack angle are simulated by the same method. The results show that the method is quite useful even for those complicated flow fields.

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