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High Lift Characteristics of an Airfoil Placed in a Narrow Channel*

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Aerodynamic characteristics of an airfoil moving in a narrow channel bounded by two parallel walls, i.e. ground and ceiling, are studied theoretically and experimentally. Lift acting on an airfoil at a positive attack angle increases as the airfoil approaches either ground or ceiling. This study deals with the combination of ground and ceiling effects to realize a high lift wing. A theory is presented for the calculation of two-dimensional inviscid flow around an arbitrary airfoil placed in a narrow channel, using singularity method in which proper circulation is distributed directly on the airfoil contours. Sample calculation shows the influences of ground and ceiling on the lift and moment coefficients of the airfoil. Two-dimensional experiment is conducted to measure the pressure distribution of airfoil, from which lift and pressure drag coefficients are evaluated. Theoretical and experimental results are compared and the characteristics of this kind of high lift wings are discussed.

1. Introduction

The lift of an airfoil in a uniform stream at a positive attack angle increases as the airfoil nears the ground. This aerodynamic interference between airfoil and ground is well known as ground effect and plays an important role in the landing and take-off characteristics of airplane wings [cf. Fig. 1(a)].

In the similar way the existence of an upper plane (ceiling) on the suction side of the airfoil augments the lift. This phenomenon can be understood as a negative ground effect, i.e. ground effect at a negative attack angle [cf. Fig. 1(b)].

When an airfoil is placed in a stream bounded by two parallel planes, its aerodynamic characteristics are different, of course, from those of the airfoil in a free stream due to the combined ground and ceiling effects [cf. Fig. 1(c)]. This problem emerges frequently when the aerodynamic characteristics of an airfoil are measured in the test section of a wind tunnel. Numerous studies have been made on this wall interference effect in order to estimate the correction factor which converts the measured characteristics to those in an unbounded free stream^{(1)~(6)}. The interference effect of two parallel walls, called hereafter wall effect, includes ground or ceiling effect as its extreme case.

The present paper deals with the possibility of utilizing wall effect positively to realize a high lift wing. For this purpose the aerodynamic characteristics of an airfoil placed in a relatively narrow channel have been studied analytically and experimentally, where the distance between two walls is comparable to the chord length of the airfoil.

This kind of high lift wing operating in a bounded channel could be applied, for instance, to the lifting element of high speed train, if its aerodynamic characteristics, especially the maximum attainable lift coefficient and the corresponding drag to lift ratio, are more advantageous than those of a conventional wing.

Many analytical methods have been proposed for two-dimensional incompressible potential flow around an airfoil placed between two parallel walls. Among them the case of flat plate has been studied most extensively, since this imposes the simplest condition for the analysis and, nevertheless, can reveal the general tendency of the wall interference⁽⁷⁾. An exact general solution for this problem was given, including ground and ceiling effects as its special case, by S. Tomochika and his collaborators⁽⁸⁾ based on the principle of con-

Fig. 1 Ground and ceiling effects

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formal mapping. This solution was derived in a very lengthy mathematical process and the characteristics values such as lift and moment coefficients were expressed in the form of infinite power series. Due to the poor convergence of this power series for small values of wall distance to chord ratio and/or airfoil gap to chord ratio, this solution can be applied practically only to the wall interference in a wind tunnel and therefore can not to the present case.

In this paper a numerical method is adopted instead of reducing an analytical solution. The theory is based on the singularity method introduced by H.J. Oellers⁽⁹⁾ for the calculation of a cascade of airfoils with large camber, thickness and solidity. In contrast to the linearized singularity method introduced, for example, by H. Schlichting⁽¹⁰⁾, in which a given airfoil is generated by properly selected circulation and source/sink distribution on the airfoil chord line, the airfoil contours are created in this method by circulation distributed directly on the airfoil contours. This circulation must be so selected that streamlines of the induced flow field coincide with the airfoil contours. For the present case in which the distance between two adjacent blades can be very small, linearized singularity method cannot assure a satisfactory accuracy and the introduction of Oellers's method becomes inevitable. For the flow around an airfoil close to ground, Widnall et al. presented a suitable analytical method⁽¹¹⁾, but the effect of the ceiling cannot be taken into account by this method.

By means of the presented method sample calculations have been made to evaluate lift and moment coefficients and pressure distribution on the airfoil surface of flat plate, NACA 0010 and NACA 4410 placed at various positions in a channel.

Two-dimensional experiment has been conducted to measure the pressure distribution under wall effect, and the comparison between theory and experiment has been made. In the application of this high lift wing to a high speed train, for example, the airfoil moves between fixed ground and ceiling. In order to obtain the same relative flow around a fixed airfoil, both walls must move with the same velocity as that of

inlet flow. This can be realized by adopting a test section with moving walls, but the test rig becomes necessarily very complex.

In the present experiment fixed walls are adopted instead of moving ones and most efforts are made for reducing boundary layer thickness on the walls at the test section. It is known that the test results differ in both cases even when the boundary layer removal is made as perfect as possible⁽¹²⁾. Generally speaking, separation occurs earlier in a test section with fixed walls than in one with moving walls. Therefore, the performance limit obtained in this experiment is rather underestimated than the one in the real application.

2. Theory

Integral equation As illustrated in Fig. 2, an arbitrary airfoil with chord length c is placed between two parallel walls at an attack angle α . The distance between two walls, d , and between trailing edge and lower wall, g , are also given. Instead of moving the airfoil with velocity U_∞ in the channel, the flow enters the channel with uniform inlet velocity U_∞ relative to the fixed airfoil, so that the flow is analysed by coordinates system fixed to the airfoil. Leading edge of the airfoil is chosen as the origin, while x - and y -axis are set parallel and normal to the chord line. The location of a point on the physical plane is indicated by a complex number $z = x + iy$. Further a point on the contour is identified by variable s , which is equal to the distance measured along the airfoil contour from the trailing edge.

The existence of a wall near an airfoil is equivalent to the condition that the streamline is straight along the wall. This condition can be realized by putting a mirror image of the airfoil with respect to the wall plane. In the case of two parallel walls, a direct image M_{u1} with respect to the upper wall, for example, requires its own image M_{u2} with respect to the lower wall, then again with respect to the upper wall and so on. As the result of this infinite extension of image with respect to either walls, an infinite lattice system consisting of alternating upright and inverse images of the real airfoil is obtained as shown in Fig. 2. The real flow domain corresponds to a slice of this periodical flow field.

Indicating the complex coordinates of a point P on the real airfoil contour by $z(s)$, the complex coordinates of point P' , which is the direct image of point P with respect to the lower wall, can be expressed as

$$\zeta(s) = e^{2i\alpha} \bar{z}(s) - i k e^{i\alpha} \quad (1)$$

where $\bar{z}(s) = x(s) - iy(s)$ and $k = 2(g + c \sin \alpha)$.

Now we distribute circulation $\gamma(s)$ directly on the contour of each airfoil, real as well as imaginary one. Then the velocity at a point is the vectorial sum of given

Fig. 2 Flow configuration

uniform flow U_∞ and velocities induced by the distributed circulation on the contours. The complex potential f of this resultant flow is written in the form;

$$f = U_\infty e^{-i\alpha} z + \frac{i}{2\pi} \oint \gamma(s) G(z, s) ds \quad (2)$$

where

$$G(z, s) = \ln \left\{ \sin \pi \frac{ie^{-i\alpha} [z-z(s)]}{2d} \div \sin \pi \frac{ie^{-i\alpha} [z-\zeta(s)]}{2d} \right\}$$

The condition which the distributed circulation should satisfy, is that the airfoil contour coincides with one of streamlines of induced flow field, in other words, that the stream function Ψ of the induced flow holds a constant value on a contour.

Stream function at $s = \sigma$, where σ denotes a specific point on the contour, can be written as

$$\Psi = U_\infty [y(\sigma) \cos \alpha - x(\sigma) \sin \alpha] + \frac{1}{2\pi} \oint \gamma(s) K[x(\sigma), y(\sigma); s] ds$$

where $K(x, y; s) = K(\sigma, s) = \text{real part of } G[z(\sigma), s]$. Putting $\Psi(\sigma) = \text{constant} = \Psi_c$, the above equation can be written as

$$\frac{1}{2\pi} \oint \gamma(s) K(\sigma, s) ds - \Psi_c = U_\infty [x(\sigma) \sin \alpha - y(\sigma) \cos \alpha] \quad (3)$$

where

$$K(\sigma, s) = \frac{1}{2} \ln [(A-B)/(A-C)]$$

and

$$A = \cosh [(\pi/d)\{[x(\sigma)-x(s)] \cos \alpha + [y(\sigma)-y(s)] \sin \alpha\}],$$

$$B = \cos [(\pi/d)\{-[x(\sigma)-x(s)] \sin \alpha + [y(\sigma)-y(s)] \cos \alpha\}],$$

$$C = \cos [(\pi/d)\{-[x(\sigma)+x(s)] \sin \alpha + [y(\sigma)+y(s)] \cos \alpha + k\}].$$

Equation (3) represents a basic integral equation for determining circulation distribution on the airfoil contour.

Aerodynamic parameters When the circulation distribution on the contour satisfies Eq.(3), the resultant velocity of uniform and induced flow becomes zero anywhere inside of the airfoil contour. The velocity $U(s)$ at the outer surface of the airfoil is, therefore, equal to the intensity of the distributed circulation itself:

$$U(s) = \gamma(s)$$

Aerodynamic parameters can be then calculated as follows:

Pressure coefficient;

$$C_p = 1 - [\gamma(s)/U_\infty]^2 \quad (4)$$

Lift coefficient;

$$C_L = \sec \alpha \oint C_p(s) d[x(s)/c] \quad (5)$$

Moment coefficient around leading edge;

$$C_M = \oint C_p(s) \{ [y(s)/c] d[y(s)/c] + [x(s)/c] d[x(s)/c] \} \quad (6)$$

Circulation around an airfoil;

$$\Gamma/U_\infty c = \oint [\gamma(s)/U_\infty] d(s/c) \quad (7)$$

Center of pressure;

$$a/c = C_M / (C_L \cos \alpha) \quad (8)$$

Method of numerical calculation Hereafter all variables are replaced by normalized ones, for example, γ by γ/U_∞ , x , y , s , σ by x/c , y/c , s/c , σ/c and Ψ_c by $\Psi_c/U_\infty c$ unless otherwise specified.

For the convenience of numerical calculation a new angular coordinate θ is introduced, which has geometrical correlation with x - y coordinates as illustrated in Fig. 3.

Hence,

$$x(s) = \frac{1}{2} (1 + \cos \theta) = x(\theta),$$

$$y = y(x) = y(\theta). \quad (9)$$

Defining a function

$$\rho(\theta) = ds/d\theta = \sqrt{1 + (dy/dx)^2} \times \left| -\frac{1}{2} \sin \theta \right| \geq 0 \quad (10)$$

Eq.(3) can be reduced to the following non-dimensional form;

$$\frac{1}{2\pi} \int_0^{2\pi} \gamma(\theta) K(\phi, \theta) \rho(\theta) d\theta - \Psi_c = x(\phi) \sin \alpha - y(\phi) \cos \alpha \quad (11)$$

where ϕ indicates angular coordinate of a specific point $s = \sigma$.

In order to avoid the singularity of integral kernel $K(\phi, \theta)$ at $\theta = \phi$ on the path of integration, an additional kernel

$$L(\phi, \theta) = \frac{1}{2} \ln (4 \sin^2 \frac{\phi - \theta}{2}) \quad (12)$$

is subtracted from the original kernel $K(\phi, \theta)$, and a new kernel is defined as

$$R(\phi, \theta) = K(\phi, \theta) - L(\phi, \theta) \quad (13)$$

where $R(\phi, \theta)$ has the following definite value;

Fig. 3 Definition of angular coordinate

$$R(\phi, \phi) = \lim_{\theta \rightarrow \phi} R(\phi, \theta) = \ln \left[\frac{\pi \rho(\phi)}{2d} \div \left| \sin \pi \frac{-x(\phi) \sin \alpha + y(\phi) \cos \alpha + k/2}{d} \right| \right]$$

The product of $\gamma(\theta)$ and $\rho(\theta)$, hereafter denoted by $\hat{\gamma}(\theta)$, can be expressed in the form of Fourier series;

$$\hat{\gamma}(\theta) = \gamma(\theta) \rho(\theta) = A_0 + \sum_{k=1}^{\infty} A_k \cos k\theta + \sum_{k=1}^{\infty} B_k \sin k\theta \quad (14)$$

Since the additional kernel of Eq. (12) can be expressed also by Fourier series

$$L(\phi, \theta) = - \sum_{m=1}^{\infty} \frac{1}{m} \cos m(\phi - \theta) \quad (12')$$

the following relation holds;

$$\frac{1}{2\pi} \int_0^{2\pi} \hat{\gamma}(\theta) L(\phi, \theta) d\theta = - \frac{1}{2} \sum_{k=1}^{\infty} \left(\frac{A_k}{k} \cos k\phi + \frac{B_k}{k} \sin k\phi \right) \quad (15)$$

In the numerical calculation of Eqs. (14) and (15), only a finite number of terms from $k=1$ to $k=N'$ can be taken into account, where $N'=(N-1)/2$ represents an integer.

Application of Kutta-Joukowski's condition at the trailing edge requires that

$$\hat{\gamma}(0) = \hat{\gamma}(2\pi) = 0,$$

and Eq. (14) can be further transformed to

$$\hat{\gamma}(\theta) = \sum_{k=1}^{N'} A_k (\cos k\theta - 1) + \sum_{k=1}^{N'} B_k \sin k\theta \quad (14')$$

Substitution of Eqs. (13), (14') and (15) into Eq. (11) yields a linear equation with respect to N unknown parameters, that is, A_k ($k=1$ to N'), B_k ($k=1$ to N') and Ψ_C .

For simplicity all unknown parameters are hereafter reassigned to a part of A_k ($k=1$ to N) by converting A_k to A_k , B_k to $A_{k+N'}$ and Ψ_C to A_N .

Now, N arbitrary points are selected on the airfoil contour, where the integral equation (11) should be satisfied strictly. This means that a streamline coincides with the given airfoil contour at least at these N points. Putting ϕ_i ($i=1$ to N) of each calculating points into Eq. (11), N simultaneous linear equations for N unknowns are obtained. This can be expressed in a matrix form;

$$[C_{ik}] [A_k] = [D_i] \quad \begin{matrix} i=1 \text{ to } N, \\ k=1 \text{ to } N \end{matrix} \quad (16)$$

where

$$C_{ik} = - \frac{1}{2k} \cos k\phi_i + \frac{1}{2\pi} \int_0^{2\pi} (\cos k\theta - 1) R(\phi_i, \theta) d\theta \quad \text{.....for } k=1 \text{ to } N',$$

$$= - \frac{1}{2(k-N')} \sin (k-N')\phi_i$$

$$+ \frac{1}{2\pi} \int_0^{2\pi} \sin (k-N')\theta R(\phi_i, \theta) d\theta \quad \text{..... for } k=N'+1 \text{ to } N-1,$$

$$= -1 \quad \text{.....for } k=N,$$

$$D_i = x(\phi_i) \sin \alpha - y(\phi_i) \cos \alpha \quad \text{..... for } i=1 \text{ to } N.$$

All unknown parameters A_k can be determined by the determinant calculation of Eq. (16), and aerodynamic parameters are then calculated by Eqs. (4) through (8).

It is advisable to select N calculating points on the airfoil contour according to three-quarter method, so that the maximum accuracy can be expected within a given number of calculating points.

Choice of number N is made from the consideration of the expecting accuracy and the required computing time. Sample calculations were made for several representative cases with $N = 5, 7, 9$ and 11 . The result revealed that overall aerodynamic parameters such as lift and moment coefficients agreed without any significant difference for all cases, though pressure distribution showed a slight change in the vicinity of the trailing edge depending on N . For the systematic numerical calculations $N = 7$ was adopted in this paper.

Result of numerical calculation is presented and discussed in the later section in connection with the corresponding test result.

3. Experiment

Test equipment and method When wall effect is applied to a high lift device in reality, an airfoil travels in a channel bounded by two walls, and the fluid filling the channel stands still until it is disturbed by the traveling airfoil. In order to realize the same relative condition to a fixed airfoil, the inlet velocity must be constant across whole channel section upstream of the airfoil. When a fixed airfoil is installed between two fixed walls, this condition can not be satisfied due to the boundary layer formation on the walls. This problem inherent in a fixed airfoil can be avoided by introducing, for example, moving walls or mirror airfoils, although such test equipment becomes usually very elaborate one. In the present experiment, a fixed airfoil was placed between two fixed walls, but efforts were made to reduce the boundary layer thickness on the walls as far as possible.

The test setup is illustrated in Fig. 4 schematically. Tested airfoil has chord and span lengths of 100 and 400 mm, respectively, and the section profile is flat plate, NACA 0010 and NACA 4410. Flat plate is approximated in experiment by a plate of 5 % thickness with elliptically rounded (between $x/c = 0$ and 0.15) leading edge and parabolically tapered off (between $x/c = 0.7$ to 1) trailing edge.

Each airfoil is equipped with 21 static pressure taps on its upper and lower surfaces at the center of the span.

Two bounding parallel walls, though they should ideally be infinitely long, have a total length of 600 mm. Test airfoil is installed in the middle at a given angle and distance to the wall. At 150 mm downstream of wall leading edge, walls are sliced with sharp angle and shifted slightly to form slits for blowing off boundary layers. Expected boundary layer thickness on the walls at the position of test airfoil is around 4 mm at $U_\infty = 20$ m/s. Nineteen static pressure taps are installed at the spanwise center of each wall.

Because of boundary layer growth on the walls, the inlet velocity which is uniform across a section outside boundary layer, increases gradually in downstream direction and, therefore, it becomes difficult to define and measure the inlet velocity which should be referred to uniform velocity U_∞ in the processing of the test result. A preliminary test was conducted to correlate the total pressure at the wind tunnel exit to the velocity measured by a standard Pitot tube installed in place of test airfoil. The referring velocity U_∞ with test airfoil installed in its position is evaluated from the measured total pressure at the tunnel exit by using the same correlation curve. Two-dimensionality

Fig. 4 Arrangement of test setup

of the flow at the spanwise center was satisfactory according to preliminary test.

Wind tunnel used for this experiment has a nozzle dimension of 400 times 200 mm, and the exit velocity is maintained at 20 m/s. Reynolds number based on the uniform velocity U_∞ and the chord length c is about

$$Re = U_\infty c / \nu = 1.4 \times 10^5.$$

Evaluation of result Lift, drag and moment around leading edge were calculated from measured pressure distributions on the airfoil by integrating them numerically along the airfoil contour.

Although the influence of shear stress acting on the airfoil is comparatively insignificant on lift and moment coefficient, the influence on drag coefficient can be large, especially when no separation occurs on the airfoil. Therefore, lift and moment coefficient evaluated from the measured pressure distribution can be adopted with sufficient accuracy, while the evaluated drag coefficient represents only a fraction of real one, that is, pressure drag coefficient.

In most test condition airfoil produces a high lift coefficient under wall effect, and this means usually that the airfoil operates in partly or wholly separated condition. In this case the portion of pressure drag to the total drag becomes large and the tendency of total drag can be understood even quantitatively from that of pressure drag.

4. Results and discussion

To understand general tendency of wall effect, aerodynamic characteristics of flat plate were calculated theoretically for various combinations of wall distance d/c , attack angle α and trailing edge gap g/c . The results are shown in Fig. 5. The change of lift ratio L/L_0 with trailing edge gap is illustrated in Fig. 5(a) for $\alpha = 5^\circ$ and 10° , where L_0 is a lift without wall effect at the same attack angle. There appears a minimum of lift ratio where the total influence of ground and ceiling effects becomes least. Since the ceiling effect is obviously stronger than the ground effect, there is

Fig. 5 Theoretical performance of flat plate

a case in which lift ratio increases monotonously with trailing edge gap as in the case of $d/c = 0.4$ and $\alpha = 10^\circ$. This is also the case for $d/c < 1.5$ and $\alpha = 15^\circ$ though not shown in the Figure.

Wall effect on moment coefficient is shown in Fig. 5(b) for $\alpha = 10^\circ$ and the tendency is very similar to that of lift. Fig. 5(c) indicates the change of aerodynamic center at $\alpha = 10^\circ$. It is seen that the aerodynamic center moves backward from the ordinary 1/4 chord point as the airfoil nears the ground, while it moves forward and approaches the leading edge as the airfoil nears the ceiling.

Fig. 6 Performance of flat plate, theory and experiment

Fig. 7 Performance of NACA 0010, theory and experiment

In Fig. 6(a) and (b) aerodynamic characteristics such as lift, pressure drag and moment coefficients of flat plate are shown for $d/c = 0.6$ and 1.0 , respectively, where both measured and calculated curves are drawn for comparison. Fig. 7(a) and (b) as well as Fig. 8(a) and (b) show the corresponding results of NACA 0010 and NACA 4410 respectively.

As seen generally from these Figures the lift increases and reaches once its maximum before it falls off sharply as airfoil approaches from ground to ceiling. This sharp drop of lift is caused by a fully developed separation on the suction side of airfoil. Since the passage bounded by ceiling and suction surface of airfoil forms a diffuser with large divergence angle as airfoil nears ceiling, the flow in this passage is exposed to a strong positive pressure gradient and is liable to separate. Even outside of this fall-off region of lift the measured lift agrees with the theoretical one only qualitatively and is substantially smaller especially at higher attack angle. This suggests that the flow around the airfoil is partly separated even though not fully when it operates under high lift condition.

The coincidence between theory and experiment is only fair when the airfoil operates at small attack angle ($\alpha = 5^\circ$) and not too close to ceiling. In this case the pressure distributions between theory and experiment agree naturally well as shown in Fig. 9(a) for $d/c = 1.0$, $\alpha = 5^\circ$ and $g/c = 0.6$. To the contrary Fig. 9(b) shows a typical pressure distribution when the coincidence is quite poor. In this case the airfoil locates so close to ceiling [$g/c = 0.813$, i.e. (gap between leading edge and

Fig. 8 Performance of NACA 4410, theory and experiment

ceiling)/ $c = 0.1$], that the flow on the suction side separates wholly. In Fig. 9 (a) and (b) the pressure distributions on upper and lower walls are also shown.

5. Applicability

To judge the applicability of wall effect to a lifting element, experimental data are summarized in Table 1. In the upper half of the table, trailing edge gap g/c , lift coefficient $C_{L \max}$, pressure drag coefficient C_{DP} and drag to lift ratio $C_{DP}/C_{L \max}$ at the maximum lift point are tabulated for every combination of $d/c = 0.6$ and 1.0 and $\alpha = 5^\circ, 10^\circ$ and 15° . In the lower half, aerodynamic characteristics for $d/c = \infty$ at $\alpha = 5^\circ, 10^\circ$ and 15° are shown for comparison. Since the case of $d/c = \infty$ is equivalent to the condition of isolated airfoil, these data are borrowed from the experimental results of Reference (13) at $Re = 6 \times 10^6$.

Fig. 9 Pressure distribution on NACA 0010, theory and experiment

Although this experiment does not cover a very wide range of parameters, it is seen that the maximum lift coefficient of around 2.0 with $C_{DP}/C_L = 0.09$ can be achieved by proper combination of airfoil profile, d/c , α and g/c , and the performance limit of this kind of high lift device may lie at about this level. Properly cambered airfoil shows better performance than uncambered one. Effective application of boundary layer control on the suction surface would improve the high lift characteristics to some extent.

From the interest in the practical application, the stability of lifting element is very important. When an airfoil supports a constant weight, the loading of the airfoil is indicated by a straight line ① of Fig. 10. Between two crossing points A and B of lift and loading curves, the operation at point A is obviously statically unstable, and therefore, B must be the operating point for the given loading. So, in order to utilize the wall effect positively, the airfoil must operate under the wholly separated condition where drag to lift ratio is inevitably higher.

As to the dynamic stability of point B a serious problem must be also expected. A series of experiments⁽¹⁴⁾ was made on the dynamic stability when an airfoil was suspended elastically to give loading curve as indicated by line ② of Fig. 10. Under the condition that the crossing point C is

Fig. 10 Operating point and its stability

Table 1 Summary of experimental results

d/c	α	NACA 0010				NACA 4410			
		g/c	$C_{L \max}$	C_{DP}	C_{DP}/C_L	g/c	$C_{L \max}$	C_{DP}	C_{DP}/C_L
0.6	5	0.42	1.36	0.06	0.04	0.27	1.60	0.05	0.03
	10	0.24	1.66	0.07	0.04	0.18	2.14	0.19	0.09
	15	0.18	1.37	0.32	0.23	0.13	1.88	0.33	0.18
1.0	5	0.80	1.18	0.02	0.02	0.70	1.51	0.05	0.03
	10	0.57	1.36	0.05	0.04	0.55	1.81	0.14	0.08
	15	0.58	0.96	0.24	0.25	0.52	1.69	0.25	0.15
d/c	α	-	C_L	C_D	C_D/C_L	-	C_L	C_D	C_D/C_L
∞	5	-	0.50	0.008	0.015	-	0.95	0.007	0.007
	10	-	1.08	0.010	0.010	-	1.43	0.014	0.010
	15	-	1.23	-	-	-	1.63	0.020	0.012

between points P and Q of the lift curve and the inlet velocity U_∞ exceeds a certain limit, a self-excited oscillation of airfoil begins to develop. The energy to sustain the oscillation comes from the main stream due to the hysteresis or time lag of separated flow on the suction side. Therefore, much attention must be paid to the dynamic behavior of the airfoil and its suspension system. Boundary layer control seems very promising to suppress the dynamic instability, since it is closely connected with the dynamics of boundary layer.

6. Conclusions

Aerodynamic characteristics of an airfoil placed between two parallel walls with relatively narrow distance have been studied theoretically and experimentally. From theoretical result the general tendency of wall effect has been recognized, where overwhelmingly stronger effect of ceiling than that of ground is noteworthy.

High lift characteristics of flat plate, NACA 0010 and NACA 4410 have been measured and compared with the corresponding theoretical result. The lift increases up to its maximum before it falls off sharply, as the airfoil approaches from ground to ceiling. This fall-off is the result of a strong separation on the suction side of the airfoil.

From test result the applicability and performance limit of high lift wing utilizing wall effect have been estimated. Attention has been given to static and dynamic stability of airfoil and its suspension system, which might impose a serious problem in its application.

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