Unsteady Lift and Radiated Sound from a Wake Cutting Airfoil

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An experimental study of the transient response of an airfoil to a passing wake, commonly known as "wake cutting," has been carried out in order to contribute to the basic understanding of interaction between successive blade rows in turbomachinery. An open jet was traversed periodically by moving circular rods in pinwheel fashion and a periodic row of oblique wakes was created. An instrumented airfoil was placed in the jet and microphones were used to obtain the radiated field. By using periodic sampling and averaging technique on all signals, the random, turbulent portion was suppressed, and only the periodic component was extracted. The periodic component of the instantaneous chordwise surface pressure distribution on the airfoil and the radiated sound field from the airfoil were measured and compared with the existing theories.

I. Introduction

SEVERAL important interaction problems between successive blade rows in axial flow compressors and in other rotating machinery depend on a basic phenomenon commonly referred to as "wake cutting." The term refers to the transient interaction between an airfoil (blade) and the approaching nonuniform, unsteady flow carrying vorticity in the form of viscous (or turbulent) wakes. This interaction phenomenon has importance not only because of the additional energy losses and induced vibrations, but especially for its role in the generation of sound.

Theoretical investigation of the problem related to this phenomenon has been carried out by a number of authors, such as Küssner,[1] von Kármán and Sears,[2] and Sears[3]; the work most directly relevant to this experimental study is by Meyer,[4] which develops earlier results[1-3] more explicitly. He has obtained the instantaneous chordwise distribution of the pressure gradient for the particular case when an airfoil cuts through a single isolated wake of arbitrary velocity defect distribution. Furthermore, he has obtained an even simpler general form for the limiting case when the wake is very narrow compared to the chord of the airfoil.

Wake cutting was studied experimentally in a water tunnel by Lefcort[5] who measured the instantaneous pressure on an airfoil cutting into the laminar wake of a circular cylinder, and the results obtained supported qualitatively Meyer's predictions.

In recent times another important problem that depends on "wake cutting" is the generation of sound. The fluctuation of the body force (lift) during the wake passage will generate a radiated pressure field and this phenomenon is one of the major sources of periodic sound produced by turbomachinery. Curle[6] has extended Lighthill's[7] elegant theory of aerodynamic noise to include the effect of solid boundaries. This theory was reasonably confirmed by the experimental study of Clark and Ribner[8] which correlated the fluctuating lift of a small airfoil in a turbulent flow with the radiated sound.

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Detailed experimental studies on "wake cutting," especially at high Reynolds numbers, when both the wake and the boundary layer on the airfoil are turbulent, were clearly needed. In recent years it became possible to detect and measure by periodic sampling and averaging the recurring or "deterministic" portion of a flowfield by suppressing the random fluctuating "background noise" caused by the nonrecurring turbulent contribution. The goal of the work reported here was to make definitive measurements on the response of an airfoil to wake cutting in order to clarify the validity and the limitations of the existing theoretical approaches especially those in Refs. 4 and 7.

II. Theory

A. Chordwise Pressure Distribution

As previously mentioned, Meyer[4] using thin airfoil theory, has obtained the instantaneous chordwise pressure gradient and also the instantaneous velocity distribution along an airfoil cutting into a wake of arbitrary velocity defect distribution. Furthermore, he has developed a much simpler asymptotic formula for the limiting case of a very narrow wake. Here the notation is somewhat simplified but the following discussion is based largely on Ref. 4.

As shown in Fig. 1, the two-dimensional thin airfoil is approximated by a flat plate of chord length c. The wing span is in the z direction, perpendicular to the x-y plane and the midchord point of the airfoil is chosen as the origin of the coordinate system, while x is positive in the direction of the undisturbed flow and y is perpendicular to it and to z. The oncoming wake is represented by an upwash pattern v_g carried downstream with a nominal transport velocity U_o. It is convenient here to define a wake coordinate ξ which is moving with the stream opposite to x, so that v_g is maximum, where $\xi = 0$, or

$$\xi = U_o t - x \tag{1}$$

Here it is assumed, without loss of generality, that the wake axis passes through the origin of the x, y coordinate system at

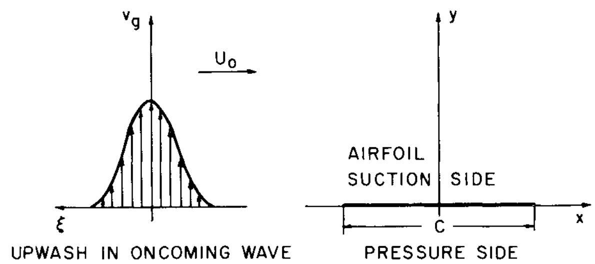

Fig. 1 Airfoil and approaching wake.

$t = 0$. At this point it is immaterial whether the wake is indeed perpendicular to the x axis or oblique to it since within the framework of the linearized theory only the component normal to the airfoil's plane matters. The streamwise velocity distribution u_g in the wake is arbitrary provided that u_g and v_g are uniform in the z direction, satisfy the two-dimensional incompressible continuity equation, and are "frozen" when carried downstream with U_o. It is also convenient to introduce nondimensional quantities by using $c/2$ the half-chord length and U_o the nominal transport velocity of the wake (in the experiment U_o is the nominal mean velocity outside the wake) as follows:

$$x' = \frac{2x}{c}; \qquad \xi' = \frac{2\xi}{c}; \qquad t' = \frac{2U_o t}{c} \tag{2}$$

The upwash pattern may be normalized as

$$g(\xi') = [v_g(\xi')/WU_o]$$

where

$$W = \int_{-\infty}^{\infty} v_g(\xi')\,d\xi' \Big/ U_o$$

so that

$$\int_{-\infty}^{\infty} g(\xi')\,d\xi' = 1$$

The Fourier transform of $g(\xi')$ is defined as $f(w')$

$$g(\xi') = \frac{1}{2\pi}\int_{-\infty}^{\infty} f(w')\,e^{iw'\xi'}\,dw' \tag{3}$$

where w' is the nondimensional circular frequency.

$$w' = \frac{wc}{2U_o}$$

and

$$w = 2\pi v$$

where v is the conventional frequency in hertz.

The chordwise pressure gradient of a two-dimensional thin airfoil cutting into a sinusoidal upwash pattern can be obtained easily; i.e., if

$$v_g = v_o\,e^{i(t'-x')w'}$$

then

$$\frac{\partial p}{\partial x'} = \pm \rho v_o U_o e^{iw't'} S(w) \frac{1}{(1+x')(1-x'^2)^{1/2}} \tag{4}$$

where $S(w')$ is the Sear's function defined as in Ref. 9

$$S(w') = \frac{1}{i'w'[K_o(iw') + K_1(iw')]}$$

For $w' = 0$ the problem is quasi-steady and $S(0) = 1$. The function $S(w')$ is complex in general and the degree to which it differs from unity both in amplitude and in phase shows the degree to which unsteadiness is important in the problem. Meyer[15] has shown that $S(w')$ has Hermitean symmetry and numerical values were tabulated extensively by Kemp.[9] The upper sign in Eq. (4) corresponds to the suction side of the airfoil and the lower sign to the pressure side. By integrating chordwise Eq. (4) one obtains the pressure distribution as

$$p(x', t') = \mp \rho v_o U_o e^{iw't'} S(w) \left(\frac{1-x'}{1+x'}\right)^{1/2} \tag{5}$$

This equation illustrates the interesting result that the chordwise pressure distribution is proportional to the factor $[(1-x')/(1+x')]^{1/2}$, the same way as for steady flow and there is no propagation of the pressure field along the airfoil chord, contrary to what might be expected intuitively. Since in most theoretical airfoil analyses the validity of the Kutta condition is assumed, one of the first experiments performed in the course of the present work was an experimental verification of the validity of the Kutta condition for unsteady flow. The results were reported by present authors in Ref. 10.

Now, by replacing v_o, the amplitude of the sinusoidal upwash, by the gust's Fourier component $WU_o f(w')(2\pi)^{-1}dw'$, one obtains the pressure distribution for an arbitrary upwash pattern by integrating over all Fourier components as follows:

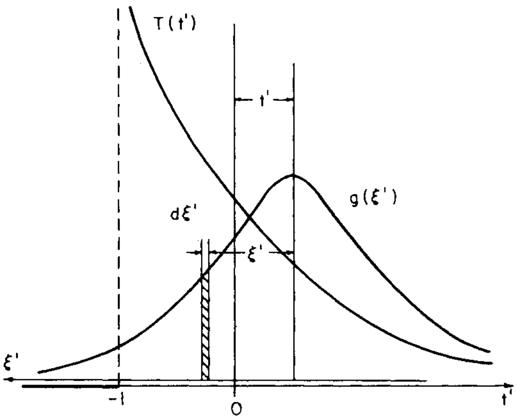

Fig. 2 Convolution integral of $g(\xi')$ and $T(t')$.

$$p(x', t') = \mp \frac{\rho U_o^2 W}{2\pi} \int_{-\infty}^{+\infty} f(w') S(w') e^{iw't'} \left(\frac{1-x'}{1+x'}\right)^{1/2} dw' \tag{6}$$

When the upwash pattern is very narrow, one may approximate $g(\xi')$ by the Dirac delta function $\delta(\xi')$ then

$$f(w') = \int_{-\infty}^{\infty} \delta(\xi')\,e^{-iw'\xi'}\,d\xi' = 1$$

and Eq. (6) simplifies to

$$p(x', t') = \mp \frac{\rho U_o^2}{2}\left(\frac{1-x'}{1+x'}\right)^{1/2} WT(t') \tag{7}$$

where $T(t')$ is a real function of t' and it is the Fourier transform of the Sear's function defined as

$$T(t') = \frac{1}{\pi}\int_{-\infty}^{\infty} S(w')\,e^{iw't'}\,dw'$$

The function $T(t')$ was first tabulated by Meyer[4] and later in more detail by Lefcort.[5] When the upwash pattern has a finite width, the Fourier integral in Eq. (6) may be avoided by taking the convolution integral of $g(\xi')$ and $T(t')$. Comparing Eqs. (3) and (6) it is easy to show that

$$\int_{-\infty}^{\infty} f(w') S(w') e^{iw't'}\,dw' = \pi \int_{-\infty}^{\infty} g(\xi') T(t'-\xi')\,d\xi' \tag{8}$$

The right-hand side of the Eq. (8) represents a superposition of contributions from thin slices of the wake as shown in Fig. 2. This is not surprising since the governing equation is linear. This way one obtains relatively simple form even for an arbitrary upwash pattern as follows:

$$p(x', t') = \mp \frac{\rho U_o^2}{2}\left[\frac{1-x'}{1+x'}\right]^{1/2} \overline{W}(t') \tag{9}$$

where

$$\overline{W}(t') = W\int_{-\infty}^{\infty} g(\xi') T(t'-\xi')\,d\xi' \tag{10}$$

By integrating Eq. (9) chordwise, the lift coefficient $c_L(t')$ is obtained as

$$c_L(t') = \pi \overline{W}(t') \tag{11}$$

B. Sound Generation

If the instantaneous lift distribution is known, the radiated pressure field can be calculated by acoustical theory. Curle[7] supplemented Lighthill's volume integral of quadrupoles by a surface integral of dipoles representing the radiation due to the presence of forces on a surface A. The instantaneous pressure $s(\mathbf{R}, t)$ is given as

$$s(\mathbf{R}, t) = -\frac{1}{4\pi}\frac{\partial}{\partial R_i}\int_A \frac{F_i[\mathbf{r}, t-(R_*/a_o)]}{R_*}\,dA(\mathbf{r}) \tag{12}$$

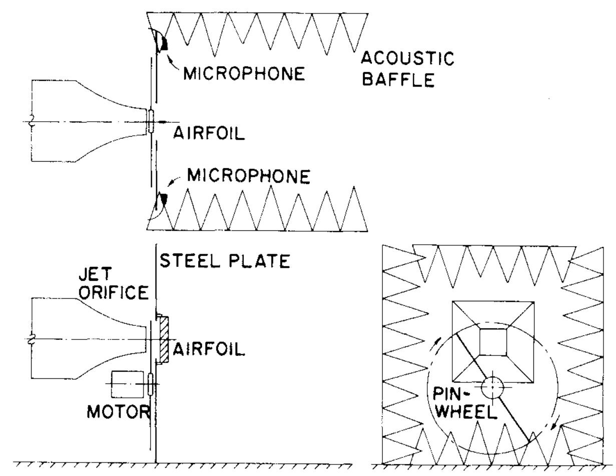

Fig. 3 Experimental configuration.

where a_o is the velocity of sound, F_i is the force per unit area exerted by the solid boundaries in the direction of R_i, and $R_* = |\mathbf{R}-\mathbf{r}|$ is the distance from the surface element $dA(\mathbf{r})$ to the field point $\mathbf{R}$. The integration is extended over the entire solid surface A.

For the far field when $|\mathbf{R}| \gg \lambda$, where λ is a typical wavelength of the sound, Curle simplified Eq. (12), in the same way as Lighthill did earlier in the case of the volume integral of quadrupoles, and Eq. (12) becomes

$$s(\mathbf{R}, t) = \frac{1}{4\pi a_o} \frac{R_i}{R^2} \frac{\partial}{\partial t} \int_A F_i\left(\mathbf{r}, t - \frac{R_*}{a_o}\right) dA(\mathbf{r}) \tag{13}$$

In the experiment reported here, however, the distance $\mathbf{R}$ from the center of the airfoil to the microphone is greater, but not much greater, than the characteristic acoustic wavelength of the radiated pressure field. Therefore the pressure field at the microphone location cannot be considered yet as "far field," and the approximation of Eq. (12) by Eq. (13) is not quite adequate. A less stringent condition, namely

$$R^2/l^2 \gg 1$$

is, however, satisfied reasonably well. Here l is the characteristic length of the airfoil, in this case the effective half span (15 cm) was used. Consequently, the term $1/R_*$ in the integrand to Eq. (12) can be taken outside, and after some calculation the following expression was obtained:

$$s(R', \theta, t') = \left[\frac{dc_L(t')}{dt'} + \frac{c_L(t')}{M_o R'}\right] \frac{qA'M_o}{4\pi R'} \cos\theta$$

or using nondimensional variables

$$s'(R', \theta, t') = \left[\frac{dc_L(t')}{dt'} + \frac{c_L(t')}{M_o R'}\right] \cos\theta \tag{14}$$

where

$$q = \frac{\rho U_o^2}{2}; \qquad R' = \frac{2R}{c}; \qquad A' = \frac{4A}{c^2}$$

$$M_o = \frac{U_o}{a_o}; \qquad s' = \frac{4\pi R's}{qA'M_o}$$

and θ is the angle between vector $\mathbf{R}$ and the y axis. The directivity factor $\cos\theta$ in Eq. (14) indicates that the radiated sound field is equivalent to that of an acoustic dipole.

III. Experimental Facility

A. Wind Tunnel

The steady uniform flow is produced by a simple open jet type wind tunnel, with a 30 cm square jet orifice. Nominal mean velocity of the jet is 38 m/sec and the nominal turbulence intensity is 0.3% of the nominal mean velocity, both measured at the center of the orifice. A steel plate (180 cm × 180 cm, about 1 cm thick) with a square opening cut out for the undisturbed exit of the jet is placed at 10 cm downstream from the jet orifice. It is used both for safety and also for the convenient mounting of models and traversing probes. In addition to these functions the plate also represents an acoustic mirror for the radiated sound measurement. Acoustic baffles made of triangular-shaped rubberized hair were placed on all four sides of the test section in order to avoid echoes from the laboratory walls. A schematic diagram of the flow facility is shown in Fig. 3.

B. Periodic Wake Generator

A "pinwheel" of 142 cm total diameter consists of two circular cross-section rods of 0.96 cm diam each (aluminum pipe). The wheel rotates typically at 300 rpm, resulting in a sweep velocity of the rods across the jet at the center section of 15.7 m/sec. The plane of the pinwheel is located at 6 cm downstream of the jet orifice. A periodic row of wakes is generated as the circular rods cut across the jet periodically. The ratio (rod velocity/nominal jet velocity) is about 0.41 and the wake makes an angle of 22° to the streamwise direction at the center of the span. Since the pinwheel is of a finite diameter, the wakes are not plane and there is a slight variation of the wake angle across the wingspan (about ±4°) but the wake arrives exactly the same time at the leading edge of the airfoil over the entire wingspan. Since the wakes are narrow there is a relatively long interval of undisturbed flow between successive wake passages so that the wakes do not interfere with each other; each can be observed as a single solitary event.

C. Airfoil

There were three airfoils fabricated. In most of the experiments a symmetrical circular arc cross-section airfoil was used (instrumented airfoil), of 10.16 cm chord length, 1.46 cm thickness, and equipped with an internal pressure transducer connected to five pressure holes through a pressure switch. The holes were of 0.065 cm diam each, distributed over one-quarter of the circumference, at 0%, 5%, 10%, 25%, and 50% of the chord length from the leading edge. They were staggered along a 29° line spanwise in order to avoid the effect of the wake of an upstream hole on those farther downstream. Since the airfoil was a symmetrical one it was reversible in two ways or, in other words, it was installed in 4 independent positions; thus a total of 16 pressure points were obtained (leading and trailing edges, two midpoints plus 3 × 4 = 12). Each of the five pressure holes was switched to the pressure transducer by a five-position pressure switch operated from outside of the airfoil, and a single condenser microphone served as an a.c. pressure transducer. Two additional airfoils, not equipped with the pressure transducer, were used in order to study the thickness and the chord length effects but only the radiated sound was measured. These were also symmetrical circular arc cross-section airfoils of 0.63 cm thickness one with a 10.16 cm chord (thin airfoil) and the other with a 5.08 cm chord (short airfoil). The flow over the airfoils appeared to be satisfactory with laminar boundary layer on the front part and a turbulent one on the rear portion.

IV. Instrumentation and Signal Processing

A. Probes

Constant temperature hot-wire anemometers[10] were used with temperature compensated linearizers[11] with an "X probe" to measure simultaneously the streamwise and the normal velocity components in the oncoming wake.

The pressure probe installed in the airfoil is a frequency modulating condenser microphone originally developed at the University of Hiroshima.[13] Since the carrier frequency (about 90 MHz) is within the range of the commercial FM broadcasting bands, a commercially available FM tuner was used for the signal detection.

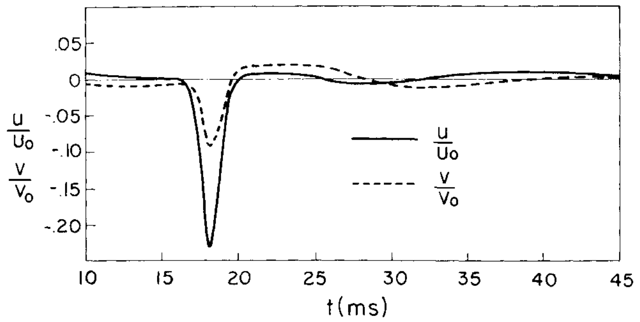

Fig. 4 Periodically averaged u and v velocity components of wake without airfoil.

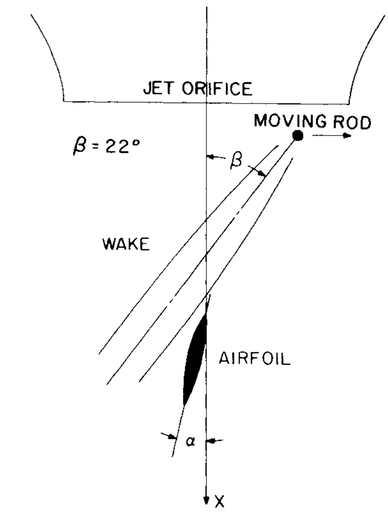

Fig. 6 Definition of α and β.

The over-all frequency response of the pressure probe was limited because of the cavity resonance of the tubing (about 7 cm long) leading from the pressure holes on the airfoil surface to the pressure probe, and electronic compensation was made in order to obtain wide enough frequency response for the measurement of the periodic component of the signal (d.c.-1500 Hz). Condenser microphones manufactured by the B & K Corp. were used for the measurement of the radiated sound.

B. Periodic Sampling and Averaging

Since the object of the research is to study the response of an airfoil cutting through a periodic row of viscous wakes generated by moving circular rods, the high turbulence in the wake flow as well as in the boundary layer along the airfoil surface superimposes a great deal of random fluctuation upon the "deterministic" periodic signal. In order to extract the deterministic (periodic) component and suppress the random signals, a particular technique defined here as "periodic sampling and averaging" was used. The periodic average is defined as

$$\widetilde{f(t)} \equiv \underset{K \to \infty}{Lt} \frac{1}{K} \sum_{n=1}^{K} f(t+nT)$$

where T is the period and K is a positive integer.

In actual experiments however, K, the number of samples, cannot be increased indefinitely, but for large K the relative rms value of the random component will decrease as $K^{-1/2}$. Typically $600 < K < 1200$ samples were taken and the random component was essentially removed. For periodic averaging Princeton Applied Research Waveform Eductor Model TDH-9 was used and the synchronizing pulses were obtained from a magnetic pickup coil sensing a small permanent magnet attached to the pinwheel. The periodic wakes had a repetition rate of about 10/sec and the passage of a wake width corresponded to about 5–10 msec so regarding each wake passage as a single event was a realistic approximation.

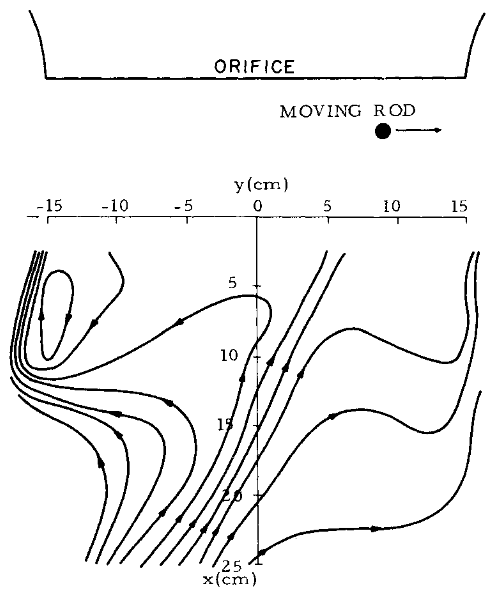

Fig. 5 An example of the instantaneous flow pattern as seen from the coordinate system moving with the nominal velocity U_o.

V. Experimental Results

A. Periodic Wakes without the Airfoil

The basic flow, the uniform jet with the periodic row of wakes, was fully investigated by a hot-wire anemometer before the airfoil was placed in it. Both streamwise (u) and normal (v) components of the velocity were measured, and by integrating these velocities, values of the stream function were obtained and from these the instantaneous streamline patterns were plotted.[6,14] Figure 4 shows traces of $u(t)$ and $v(t)$ at the location where later the leading edge of the airfoil was placed and Fig. 5 shows an example of the flow pattern plotted as seen from a coordinate system moving downstream with nominal mean velocity U_o. The uniformity of the flow without the wakes and the reproducibility of the wakes were both deemed satisfactory.

B. Unsteady Pressure Distribution and Instantaneous Lift on the Airfoil

In the experiment the angle of attack of the airfoil, was defined as shown in Fig. 6. Airfoil surface pressure was measured for four values of the geometrical angle of attack: $\alpha = 0°$, $-10°$, $-20°$, and $+10°$. Because of the finite span and an open jet configuration, the effective angle of attack α_{eff} was inferred to be about one-half of α the geometrical angle of attack. This inference was made from the measurement of $dc_L/d\alpha$ in steady flow and comparison was made with the calculated value for finite aspect ratio.[14]

The instantaneous chordwise lift distribution was obtained as the difference of the instantaneous pressures at the two sides, and the results are shown in Fig. 7. Broken lines show the theoretical lift distributions given by Eq. (9) but normalized in such a way that the total instantaneous lift of theoretical and experimental curves would be equal. For $\alpha = 0$, the theory agrees reasonably well with the experiment. The small negative portion of the experimental data near the trailing edge is probably due to the local separation of the boundary layer. By chordwise integration the lift coefficient is obtained and it is shown as a function of time in Fig. 8. For the case $\alpha = 0$, the experimental result (solid line) is compared with the prediction (broken line) based on the measured upwash profile of the oncoming wake

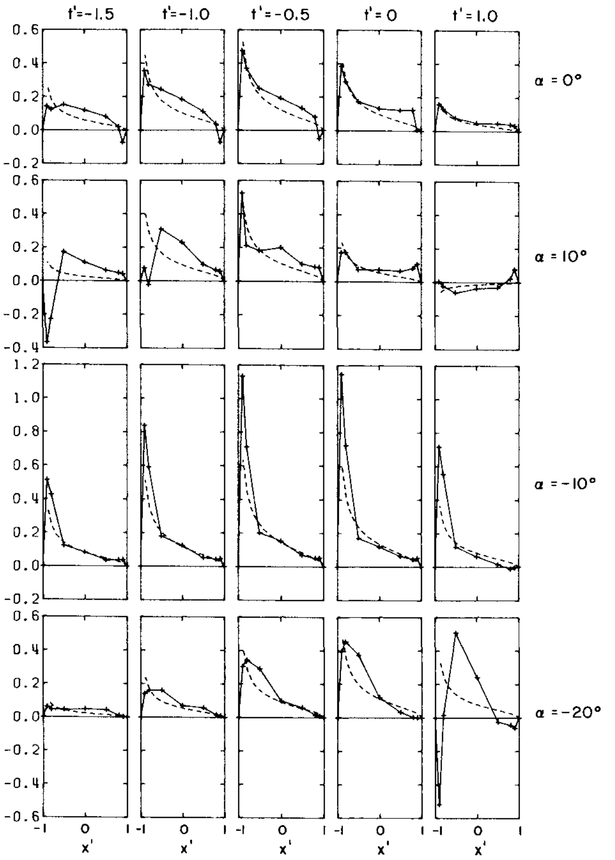

Fig. 7 Instantaneous chordwise lift distribution at various times for four values of the angle of attack α. Dotted line is theory.

given by Eq. (9). In order to facilitate calculation of the radiated sound from Eq. (14) both c_L and dc_L/dt' were plotted in Fig. 9 for all four values of α. It is interesting to notice the difference between $\alpha = 10°$ and $\alpha = -10°$. Since the instantaneous flow deflection due to the passing wake increases the absolute angle of attack of the airfoil for $\alpha = 10°$, a temporary leading edge separation seems to occur when the wake arrives at the leading edge ($-1.5 < t' < -1.0$) but later quickly recovers around $t' \approx -0.5$ as shown in Fig. 7. This recovery causes a sharp rise of the lift coefficient around $t' = -1$ yielding a large value of the time derivative of c_L as shown in Fig. 9 and this in turn radiates a higher sound level than in the other cases. For the case $\alpha = -10°$, the flow deflection reduces the instantaneous angle of attack and the theoretical lift distribution agrees well with the experiment. For the case $\alpha = -20°$, the flow seems to be fully separated most of the time. The large negative lobe of c_L for $4 < t' < 9$ must be a slow hysteresis phenomenon and has not been fully explained.

C. Sound Generation

In order to determine the instantaneous radiated pressure field, two microphones were placed symmetrically in both sides of the airfoil at a distance of 75 cm from the airfoil center perpendicular to both the span and the flow direction as shown in Fig. 3. Both the sum (corresponding to source radiation) and the difference (dipole radiation) of the two microphone output signals were periodically averaged and recorded. The steel plate was

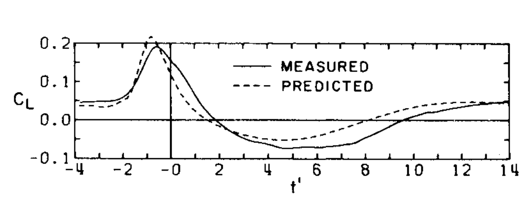

Fig. 8 Measured and predicted unsteady lift coefficient for $\alpha = 0$.

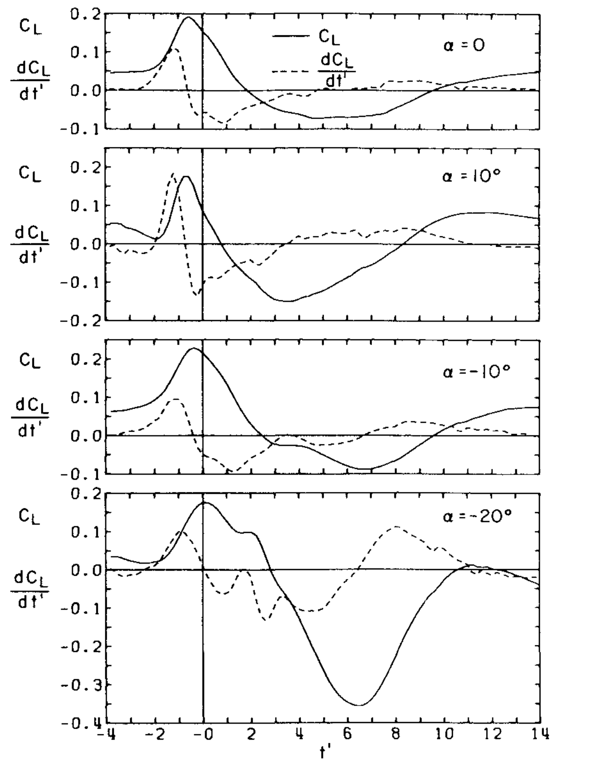

Fig. 9 Lift coefficient c_L and time derivative dc_L/dt' for four values of α.

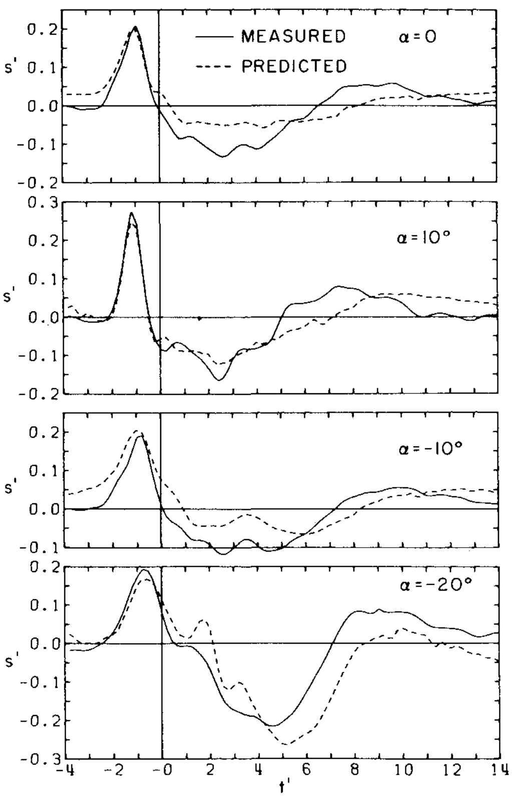

Fig. 10 Measured and predicted radiated sound for four values of α.

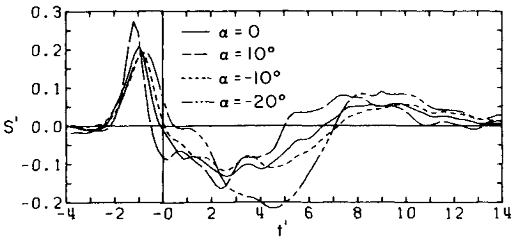

Fig. 11 Comparison of radiated sound at different angles of attack.

considered to be a perfect acoustic mirror and half of the measured sound was considered to be radiated. The background sound caused by the rods cutting the jet boundaries was first measured without an airfoil in place and was later subtracted from all the measured signals in order to obtain only the sound originating in the wake airfoil interaction. The sum signal for all cases was small and the difference signal was a sharp pulse. The dominance of the difference over the sum confirmed that the radiated sound was actually dipole radiation.

Figure 10 shows the measured sound radiation from the instrumented airfoil for all four angles of attack with the prediction based on Eq. (14). The prediction agrees well with the experiment during the wake passage, surprisingly enough even for $\alpha = 20°$ when the flow was separated. Disagreement at later times ($t' > 5$) is probably due to poor two dimensionality of the very large vortices created near the jet boundary by the entering and leaving of the rod. The measured sound signals for all four cases are shown together in Fig. 11. It is clear that in the case of $\alpha = +10°$ with the temporary leading edge separation and reattachment the airfoil radiates higher peak sound levels than all other cases.

Figure 12 compares the measured sound radiation from the three different kinds of airfoils at $\alpha = 0$. The fact that all three agreed indicated that the thickness-to-chord-length ratio has no significant effect in this range.

VI. Conclusions

Unsteady lift of an airfoil due to "wake cutting" and the sound radiation due to the unsteady lift were studied experimentally, and the measurements compared with the existing theories. Meyer's prediction[4] for airfoil surface pressure based on the measured flow pattern of the oncoming wake agreed reasonably well as long as no flow separation on the airfoil was involved. Curle's prediction[7] for radiated sound based on the measured lift on the airfoil agreed quite well with the acoustic measurement, even in the case when the flow was fully separated. The only surprising phenomenon observed was the increased sound pulse generated in the case of a local and temporary leading edge separation when during reattachment dc_L/dt' exhibited higher values than for the nonseparating cases.

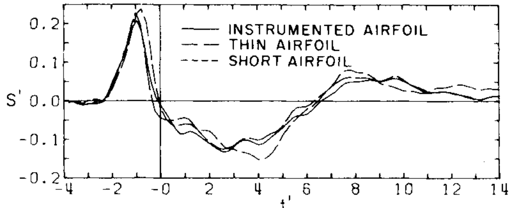

Fig. 12 Comparison of radiated sound at $\alpha = 0$ for three different airfoils.

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