ACTIVE CONTROL OF SEPARATION ON A WING WITH CONFORMAL CAMBER

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ACTIVE CONTROL OF SEPARATION ON A WING WITH CONFORMAL CAMBER

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A preliminary investigation of a wing with a conformal camber is discussed. The wing uses an adaptive actuator mounted internally to alter the shape of the suction surface which results in a change in the effective camber by increasing the maximum thickness and moving the location of maximum thickness aft. Since the actuator motion can be altered continuously, this allows the wing shape to be either static or dynamic. For the static mode, effects of profile perturbations are tested in a wind tunnel and tow tank to determine the change in wing efficiency due to variations in the camber at low Reynolds number. Various actuator locations and frequencies are tested from \( Re = 2.5 \times 10^5 \) to \( Re = 2 \times 10^5 \) at varying angles of attack. Preliminary comparisons are made with numerical predictions to demonstrate the effect of adaptive wing shaping on flight performance and the difficulty of predicting separated flow at low \( Re \). For the dynamic mode, a wing with 5 modular span-wise sections each with a separately controlled internal actuator was constructed. The dynamic effects are currently being examined.

Nomenclature

\[ \begin{align*}
    c & \quad \text{Chord length} \\
    Re_c & \quad \text{Reynolds number based on length } c \\
    f & \quad \text{Actuation frequency, Hz} \\
    St & \quad \text{Strouhal number} \\
    U & \quad \text{Freestream velocity} \\
    \alpha & \quad \text{Angle of attack, deg} \\
    \nu & \quad \text{Viscosity}
\end{align*} \]

Introduction

Range of Interest

In this paper we are concerned with wings which must operate in a range of Reynolds numbers far below those encountered in the usual practice of aeronautical design.\(^1\) Design of wings for use at low Reynolds numbers is motivated largely by the increased interest in small-scale air vehicles such as Micro Aerial Vehicles, and Unmanned Aerial Vehicles (\( \muAVs \) and UAVs).\(^2\) Low Reynolds number flight also is of interest in the design of vehicles of ordinary scale for use in the earth’s upper atmosphere, and potentially for flight within other atmospheres where the kinematic viscosity is driven to high values by low ambient density.

Sea-level flight of \( \muAVs \) will require wings to operate at Reynolds numbers in the range from \( 10^6 \) to \( 10^7 \). Such devices must be small in order to avoid detection and destruction on the battlefield. Their size should be on the order of 10 cm. Their speed is also limited to one at which they can gather useful intelligence. On the other hand they may be expected to have a limited life, and may therefore exploit power technologies which provide high output for short periods.

For operation at relatively low altitudes, the requirements for \( \muAVs \) are less stringent. They are larger than \( \muAVs \), may fly somewhat farther from their targets, and may thus fly faster and still gather clear imagery. They operate in a \( Re \) range of \( 10^5 \) to \( 10^6 \). As they fly at higher altitudes, however they encounter the low \( Re \) effects from low density, and may indeed have to deal with high Mach numbers as well.

The difficulty with the operation of a wing at low Reynolds number is that the inevitable adverse pressure gradient is encountered at a point at which the boundary layer is quite likely to still be laminar. Since a laminar boundary layer is incapable of negotiating any but the slightest adverse pressure gradient, the flow will separate. The separated flow then transitions to turbulence, entrains fluid and re-attaches to form a turbulent boundary layer. The resulting structure is the laminar separation bubble (figure 1) and has been described nicely and briefly by Lissaman\(^3\) and at greater length by Chalmichael.\(^4\)

If one measures airfoil performance as the ratio of lift to drag (\( L/D \)) then these laminar bubbles make static airfoil performance fall off by more than an order of magnitude as Reynolds number drops from \( 10^5 \) to \( 10^4 \). The problem, then, for the wing designer who must work in this range of \( Re \) is to control and limit the formation of these bubbles. Such may be accomplished through flow control. The parameter space is shown in figure 2 adapted from Lissaman.\(^3\)

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interested in the detailed manipulation of the flow to control some flow parameter. In our application the parameter of greatest interest is separation. We desire to modify the flow in order to minimize separation, and thus maximize lifting efficiency of a wing.\textsuperscript{5} Our range of interest is just that range where separation causes the most difficulty. That is, the low-Reynolds-number regime where separation begins in a laminar boundary layer and spreads over a significant portion of the wing.

Flow control can be either passive or active. It is active flow control that interests us here. Active methods require an expenditure of energy, but they can respond rapidly to changes in the flow field. Such changes, of course, require monitoring of the flow field in order to respond to it, but benefits of aerodynamic tailoring can be exploited if the response of the active system is fast enough.

Active flow control is not a mature technology, but there are numerous signs, which show promise. Continuous blowing and sucking have long been shown to have pronounced effects. More recently intermittent blowing and sucking in the form of synthetic jets have shown their effectiveness.\textsuperscript{6,7} These latter methods also suggest “sweet-spots” in the range of frequency inputs, which may translate to other oscillatory inputs.

Dramatic effects can be produced through mechanical momentum transfer as has been shown in various experiments.\textsuperscript{8,9} Flow has been energized acoustically, and mechanically. Finally one can control the flow over an airfoil by changing the shape of the foil itself. Such an approach employs an adaptive wing.\textsuperscript{10}

Adaptive Wings

By an adaptive wing we mean one which can change its shape to adapt to flow conditions. Simple examples of wing adaptation include simple flaps, droops, and slats which allow a wing to adapt to the differing demands of differing flight regimes. Such simple adaptations can be thought of as a series of static airfoil shapes a given wing may take on.

The kind of adaptation which interests us in the present paper is that in which the rate of actuation is rapid, and may be able to respond quickly enough to arrest or limit the formation of laminar separation bubbles. Such speeds either require large forces, or light-weight actuators or both. Different approaches span the range of mechanical actuation techniques.\textsuperscript{11–25} Since we are interested in \( \mu \)AVs which are small and light, a natural choice for our wings is the use of piezoelectric actuation. There have been several other attempts to influence flow over surfaces using piezoelectric devices.\textsuperscript{10}

Experimental Approach

The guiding philosophy of adaptive wing design is to find the most cost-efficient system using cur-
Fig. 3 A portion of an adaptive actuator using piezoelectric material. This specific design offers a good balance between force and deflection.

rent or nearly available technology. Several considerations must be taken into account when designing this subsystem. These include cost, weight, frequency response, force/deflection, and ease of control. Piezoelectric actuators offer a good balance between these concerns. They also lend themselves well to novel applications. Such a design is shown in figure 3. A device such as this is currently under development for use in helicopter blades. A piezoelectric material is bonded to a substrate. When a voltage is applied to the smart material, the actuator deflects. This deflection is easily controlled by the input voltage. In addition, this system can also be used as a sensor by measuring the output voltage for a given force/deflection input. Thus, this technology can be typically serve dual functions.

Experiments with a circular-arc airfoil suggest that an airfoil with oscillating camber will produce a higher lift coefficient than the same airfoil at any fixed camber setting. The present experiment will attempt to extend this finding to airfoils of more ordinary and more efficient cross-section.

Adaptive Wing

The adaptive wing design is based upon a test article constructed by Pinkerton and Moses as a feasibility test for drag reduction. A molded airfoil section is constructed with a recess cut in the upper surface to receive a THUNDER actuator as shown in figures 4 and 5. The Thunder actuator is mounted in such a way that it is even with the un-recessed airfoil section when it is at its smallest effective radius (when it is most curved). A thin plastic sheet is then placed over the actuator to smooth the profile, and then the entire assembly is wrapped in a latex membrane to hold it together and provide a seamless outer surface. When the THUNDER is at its smallest effective radius, the assembly has the same cross-section as the base airfoil; and we expect the flows and the coefficients of lift and drag to be the same as the base airfoil. When the THUNDER is shifted to its greatest effective radius (closest to being flat) it protrudes through the upper cross-section. The plastic sheet and latex membrane smooths the upper surface and the effective camber

Fig. 4 Adaptive wing with actuator in stowed and extended configuration.

Fig. 5 Adaptive wing module. Covering is removed to show the actuator beneath.

of the upper surface is increased and the point of maximum thickness is moved aft.

Five identical modular wing sections were constructed as shown in figure 5. These can be placed end-to-end along with wing-tip caps to construct a single wing with up to 5 separately controlled actuators. Each modular wing section has a chord of 8 inches and a width of 3.25 inches, giving the completed wing an approximate maximum span of 17 inches with wing-tips. Shorter wings can be built from fewer modules to test the effects of aspect ratio. This is shown in figures 6 and 7.

The modules each contain a single THUNDER 7R piezoelectric actuator. These have an operating range of -300V to +600V and produce deflections at the tip of nearly a centimeter. The THUNDER actuators are driven by either an HP E3631A DC power supply or an HP 33120A waveform generator passed through an array of Physik Instrumente PI model E-107 high voltage amplifiers. These have a fixed gain of 100. The amplifiers have been modified to produce a range from -270V to +730V. They have a built-in DC bias setting so an offset can be applied to nearly match the asymmetric range of the actuators.

1Manufactured by Face Corp, Norfolk, VA.
Fig. 6 Wing plan-form with modular sections - each section has a separately controlled actuator.

Fig. 7 Wing plan-form with modular sections - each section has a separately controlled actuator.

Fig. 8 Wing perturbation profiles.

In addition to the actuated model, a series of five solid models matching the predicted profiles for the range of actuator settings were made for additional testing. These models were made of solid plastic and could be immersed in water. The models are identified by the effective radius of the actuator, which ranges from 4 to 8 inches. The profiles of foils 4, 6, and 8 are shown in figure 8. The base profile is a NACA 4415. The 4415 was chosen since it provided sufficient thickness to make room for the actuator and its attachment fittings, and because it is a relatively flat-bottomed foil. The profile was them modified slightly to make the bottom truly flat to ease surface finishing the models. The foils are eight inches in chord, and the modeled actuator has the leading edge of its active surface 1.1 inches behind, and .66 inches above the leading edge of the chord line. It is placed so that its tangent at the leading edge is 20 degrees above the chord angle. Solid models were then constructed for actuator radii of 4, 5, 6, 7, and 8 inches.

The in-wing actuator has a maximum range of displacement at a frequency \( f \) up to approximately 25 Hz depending on the applied force. After this frequency is exceeded, the displacement drops significantly. The reduced frequency of the camber oscillations is given by the Strouhal number

\[
St = \frac{fe}{U_\infty}
\]

Since tests are conducted at different velocities corresponding to identical \( Re \) in two different mediums, it is necessary to relate this to the chord-based \( Re \) given by

\[
Re = \frac{U_\infty c}{\nu}
\]

Combining the two relations, we see that

\[
St = \frac{1}{Re\left(\frac{f\nu^2}{c}\right)}
\]

This allows a range of \( St \) up to 0.11 for the tests in air at \( Re = 5 \cdot 10^5 \) and up to order 1 if \( Re \) is reduced by an order of magnitude. Corresponding tests in water at a similar \( Re \) would give \( St \) up to 1.6 for \( Re = 5 \cdot 10^5 \) and 16 for \( Re = 5 \cdot 10^4 \).

Wind Tunnel

The Low-Speed Wind Tunnel (LSWT) is a low-turbulence open-circuit blow-down wind tunnel. A 7.5
A 5W Argon-Ion laser is used to create a laser sheet across the length of the tank. The laser sheet is fixed relative to the laboratory reference frame, as shown in figure 9. Silver-coated hollow glass spheres 10 μm in diameter are used as tracer particles. A Pulnix TM-6701 with a square-pixel CCD array (684 x 484 pixels) and a frame rate of 60/120 Hz is used to image the flow. Images are captured real-time to a PC using a Matrox Pulsar PCI frame grabber. LPT is used to extract the two-dimensional velocity and vorticity fields from the images as described below.

The Lagrangian parcel tracking (LPT) algorithm developed by Sholl and Savas29 is used to determine instantaneous flow fields from the tow tank experiments. The LPT algorithm regards seeding particles as fluid parcel markers and tracks both their translations and deformations. During this tracking, fluid parcels registered by individual CCD pixels are advected with individually estimated velocities and total accelerations. The velocity field needed to initialize the LPT process is obtained from a standard DPIV algorithm which uses multiple passes, integer window shifting, and adjustable windows. Both the LPT and DPIV algorithms employ a rigorous peak-detection scheme to determine velocity vectors and use the local velocity gradient tensor to identify spurious velocity vectors. We have found that the LPT algorithm works well in the flow field of a vortex which is characterized by high deformation rates where DPIV algorithms are plagued by biasing and limited dynamic range. No smoothing algorithms or other post-processing techniques are employed on the data. In the tow tank data, vorticity, being a component of the velocity gradient tensor, is calculated spectrally at each grid point as an intrinsic part of the LPT algorithm.

**Preliminary Results**

Both flow visualization and force measurements were taken in the wind tunnel. The actuated positions stall earlier than the un-actuated position, but for both actuated positions there is an increase in the lift coefficient after the initial stall angle as α is increased. Wind tunnel measurements show a dramatic increase in both lift and drag. For modest displacements, profile perturbations have a negative impact on airfoil efficiency. This trend is reversed as the maximum actuator displacement is reached, however. Force measurements are not presented in the present paper.

PIV measurements in the tow tank reveal that separation over the airfoil is highly dependent upon both Re and geometry. Results are shown for α = 0° and α = 12° for Re = 2.5 × 10^5 in figures 10 and 11, respectively. In figure 11, separation first increases as the actuator is displaced and then decreases once maximum displacement is achieved. Separation is clearly seen at the lowest Re for all 3 wings.

The plots illustrate two points. (1) Slight varia-
Fig. 10  Results of tow tank tests for three static wing configurations at \( Re = 2.5 \cdot 10^5 \) and \( \alpha = 0^\circ \).

Fig. 11  Results of tow tank tests for three static wing configurations at \( Re = 2.5 \cdot 10^5 \) and \( \alpha = 12^\circ \).
region was dramatically reduced over a range from 20 Hz to 60 Hz with the maximum reduction at 45 Hz. (See figure 13.)

Summary
As this is an exploratory study, there are plenty of avenues to explore in the near future. In particular, we plan on measuring the flow field above the wing with oscillating actuation using PIV.

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References


a) No actuation.

b) Actuation at $f = 15$ Hz.

**Fig. 12** $Re = 2.5 \cdot 10^4$, $\alpha = 0^\circ$. 
a) No actuation.

b) Actuation at \( f = 45 \text{ Hz} \).

Fig. 13  \( Re = 2.5 \cdot 10^4, \alpha = 9^\circ \).