Static and Dynamic Aeroelasticity

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1 PURPOSE AND SCOPE

The purpose of this chapter is to describe aeroelastic effects on fixed wing air vehicles. Rotating systems such as propellers, helicopter blades, or gas turbines are not addressed. Aeroelasticity is a design activity concerned with the consequences created and tradeoffs required by interactions among aerodynamic forces, structural deformation (elasticity), and motion (dynamics) of aerodynamic and hydrodynamic lifting surfaces. These highly interactive, interdisciplinary activities are summarized in Figure 1 with interactions among four major technical areas involved in aeroelastic phenomena (Friedmann, 1999). The term *aeroelasticity* was coined in the early 1930s by Alfred Pugsley and Harold Roxbee Cox, two distinguished engineers at the Royal Aircraft Establishment (RAE).

Aeroelastic interactions determine airplane loads and performance in four primary areas: (i) wing and tail surface lift redistribution that changes external loads; (ii) stability derivatives, including lift effectiveness, that changes flight

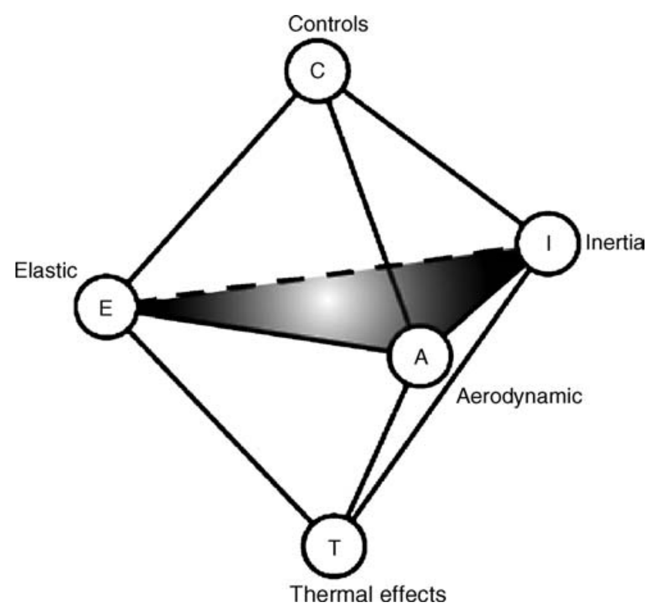

Figure 1. Aeroelastic interactions among major technical areas create new problems and opportunities.

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Figure 2. Boeing 767 static structural proof test to wing failure showing extreme strength with large deformation.

static and dynamic control features such as aircraft trim and dynamic response (Abzug and Larabee, 2002); (iii) control effectiveness, including aileron reversal, that limits maneuverability; and (iv) aircraft structural dynamic response and stability, in particular buffeting and flutter.

Highly efficient wing and tail surface structural design permits substantial structural bending and twisting deformation during flight, as indicated in the picture of the Boeing 767 ground test shown in Figure 2. Wing twist produces changes in local angle of attack on swept and unswept surfaces. Sweptback wing bending reduces local angle of attack, whereas sweptforward wing bending increases local angle of attack.

Figure 3 shows two different aerodynamic load distributions along a hypothetical, 35° sweptback wing, one for a rigid wing, the other for an identical but flexible wing. The total lift, shown as the areas under each of the two curves, is equal; the aircraft angle of attack for the flexible surface must be larger for the rigid surface.

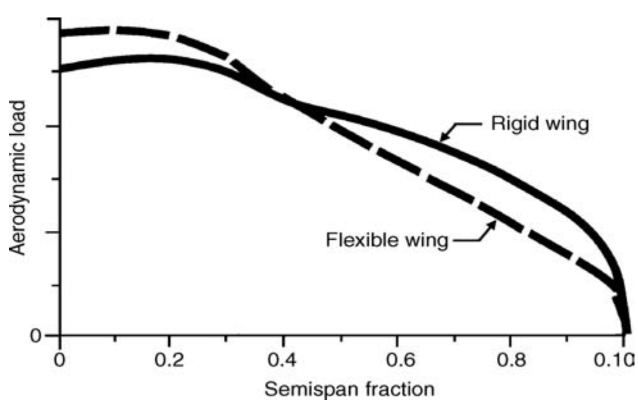

Figure 3. An example flexible 35° sweptback wing lift redistribution caused by aeroelastic effects at cruise.

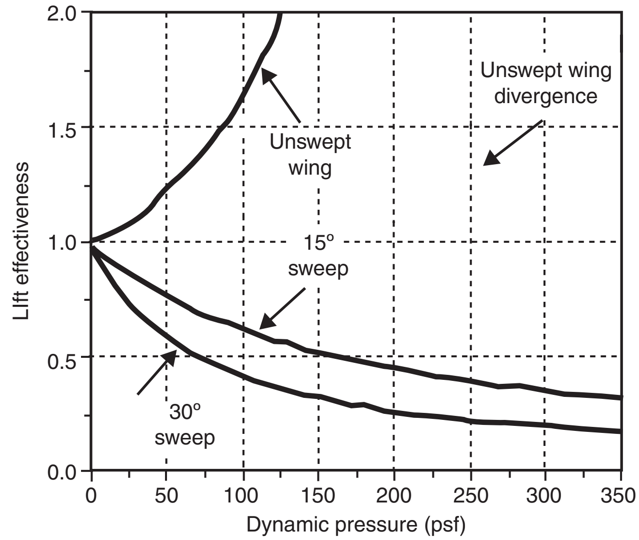

Figure 4. Lift effectiveness, showing increased lift per unit angle of attack associated with unswept wings.

Unswept wing torsional deformation causes the loading to shift outboard, but wing bending produces no change in local angle of attack. The aerodynamic load on an unswept wing resembles the curve marked “flexible wing” in Figure 3, whereas the unswept flexible wing lift distribution resembles the curve marked “rigid wing.”

Lift effectiveness is defined as the ratio of the total lift on a flexible surface divided by the lift on the identical, but rigid, surface at identical angles of attack. Lift effectiveness changes with airspeed. Figure 4 shows three plots of typical wing lift effectiveness as a function of airspeed. Sweptback wings are *lift ineffective* (lift effectiveness less than one) because of lifting surface bending. Conversely, flexible *unswept* lifting surfaces generate more lift than similar rigid surfaces.

Control effectiveness is the ability of a control surface such as an aileron or a rudder to produce aerodynamic forces and moments to control airplane orientation and maneuver along a flight path. Asymmetrical aileron rotation produces rolling acceleration and roll rate. Aircraft rolling motion creates damping in roll that opposes rolling motion. The ability to create a *terminal* or *steady-state roll rate* is the primary measure of aileron effectiveness.

Consider Figure 5. Without wing torsional flexibility, the terminal roll rate is a linear function of airspeed. Rotating the aileron downward produces lift, but also twists the wing surface nose-down, reducing the local wing angle of attack as well as the lift force and moment. The size of the nose-down twisting moment and nose-down twist depends upon the: (i) size of the control surface; (ii) amount of aileron deflection; (iii) structural stiffness; and, (iv) dynamic pressure, q.

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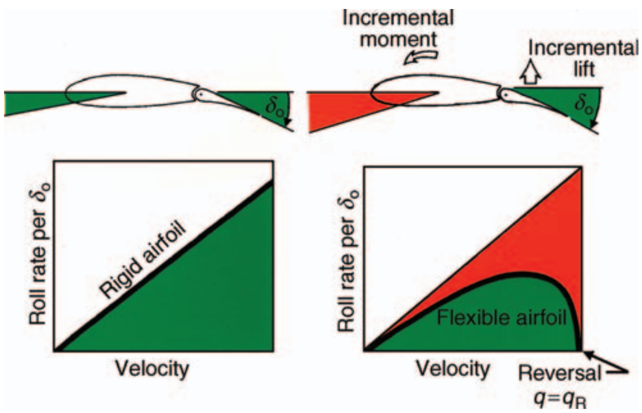

Figure 5. Control effectiveness declines with increasing airspeed.

Dynamic pressure is one half the product of air density times airspeed squared.

Because aileron induced twisting reduces the lift force, the rolling acceleration and the steady state roll rate will be less than the rigid wing roll rate. As indicated in Figure 5, the roll rate is maximum at a certain flight speed and then declines rapidly as airspeed increases. At a special airspeed, called *reversal speed*, the ailerons will not generate a rolling moment even though there is substantial wing surface distortion and substantial aileron rotation. At speeds above the reversal speed, the aileron produces a roll rate, but in the opposite direction to that intended.

2 AIRCRAFT STRUCTURAL DYNAMIC RESPONSE, STABILITY, AND FLUTTER

Unsteady dynamic loads such as those created by abrupt control movement, gusts, or buffet may produce moderate or even severe transient structural dynamic response. *Buffet* is an unsteady aerodynamic loading created by aerodynamic wakes shed from wings, nacelles, and fuselage pods during operation at high angles of attack. Buffet can create severe dynamic loads and stresses, particularly on vertical stabilizers.

Aircraft and missile resonant natural frequencies depend on stiffness and mass distribution. Lifting surface structural flexibility depends on the amount of displacement created by forces and moments. If large deflections are created by small forces then the surface is very flexible. Stiffness is the inverse of flexibility. Because the source of aeroelastic stiffness or flexibility results from the structure and the aerodynamic displacement-dependent forces, aircraft flexibility and stiffness, as well as resonant frequencies and mode shapes, change with airspeed or altitude.

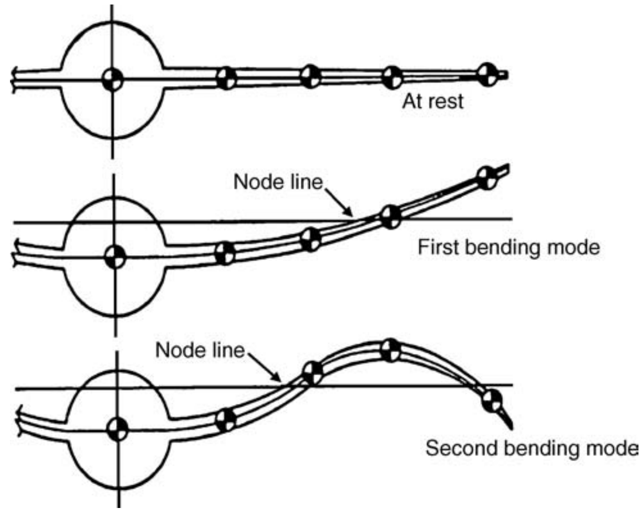

Figure 6. Wing vibration mode shapes.

Analysis and test are used to compute and then measure aircraft natural frequencies and the mode shapes associated with these dynamic pressure independent natural frequencies. Natural frequencies are identified during ground resonance tests. The vibration shapes at these natural frequencies are the mode shapes, unique to each natural frequency. A typical result is shown in Figure 6.

Flutter is a self-excited structural dynamic instability that occurs when an oscillating wing or tail surface vibrates in such a way as to extract energy from the airstream; the amplitude of vibratory motion increases exponentially. Flutter belongs to a special class of mechanics problems called *nonconservative* problems. The flutter mechanism depends on flying at the right speed and altitude to allow two or more aircraft vibration modes to interact or *couple* together. In rare instances flutter occurs as the result of the loss of aerodynamic damping. Flutter can be loosely categorized into at least five different areas: (i) classical – wing bending and torsion; (ii) control surface – surface rotation and wing bending; (iii) empennage – fuselage torsion and tail torsion; (iv) stall – wing torsion; and (v) body freedom – wing bending and fuselage pitch.

Figure 7 shows time histories of three possible types of dynamic responses at a point on a wing surface responding to a disturbance during flight at different airspeeds (Ricketts, 1983). Disturbances decay with time at an airspeed corresponding to point A where the resonant natural frequencies are well separated. At Point B a disturbance produces harmonic oscillatory motion with fixed amplitude. The two natural frequencies have changed and are very close to each other. At point C two natural frequencies are extremely close together. Any disturbance grows rapidly. This is flutter.

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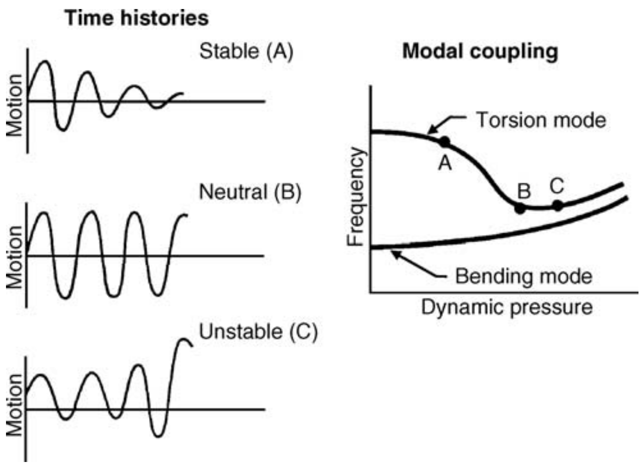

Figure 7. Aircraft vibration modes couple together to allow energy to be absorbed from the airstream.

The proximity of the two natural frequencies indicates that the two modes of motion are interacting in a way such that the lifting surface is naturally extracting energy from the airstream and feeding it into wing vibration.

Flutter *is not forced resonant response* requiring a fixed frequency harmonic force input at a system natural frequency. The airstream causing flutter is steady and nonoscillatory until the system is disturbed. Without internal structural damping, resonance response amplitude grows linearly with time, whereas the flutter dynamic response has an exponential increase until the structure is destroyed or some nonlinear mechanism comes into play.

The flutter instability mechanism is illustrated in Figure 8, which shows a wing section undergoing torsional and bending motion. The torsion mode creates harmonic aerodynamic forces, whereas the bending or "plunge" mode supplies the displacements for the aerodynamic forces to do work.

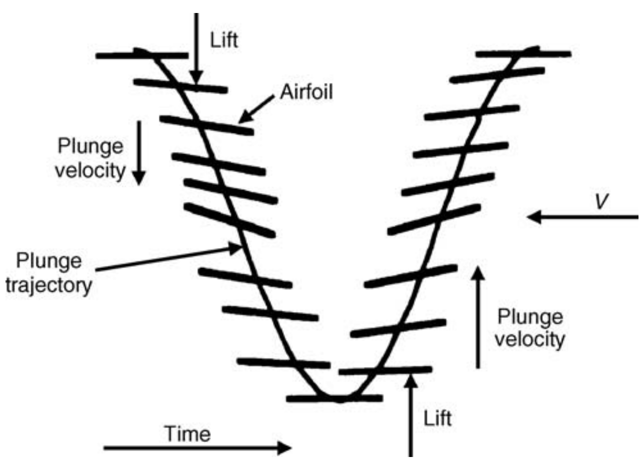

Figure 8. Airspeed dependent phasing between two distinct types of oscillatory motion at nearly the same frequency is required to create flutter dynamic instability.

As the wing plunges downward it also rotates downward to place the lift force in phase with the downward motion. At the beginning of the downward motion the lift is zero and reaches its maximum downward value when the plunge displacement is zero.

The aerodynamic forces behave like a sine wave, whereas the displacement is a cosine wave. These two functions are 90° out-of-phase. As a result, the integral of the product of the lift force times the plunge velocity is positive. Energy is extracted from the flow and fed into the system.

3 THE ORIGINS OF STATIC AND DYNAMIC AEROELASTIC PROBLEMS

Two factors drive aviation development: the quest for speed and the competition for new air vehicle military and commercial applications. These factors trigger the appearance of new aircraft shapes, new materials, and applications of other new technologies such as avionics. This in turn has created and continues to create new challenges for aeroelasticity.

Static aeroelastic phenomena appeared at the very beginning of powered flight. Samuel Langley's *Aerodrome* failed

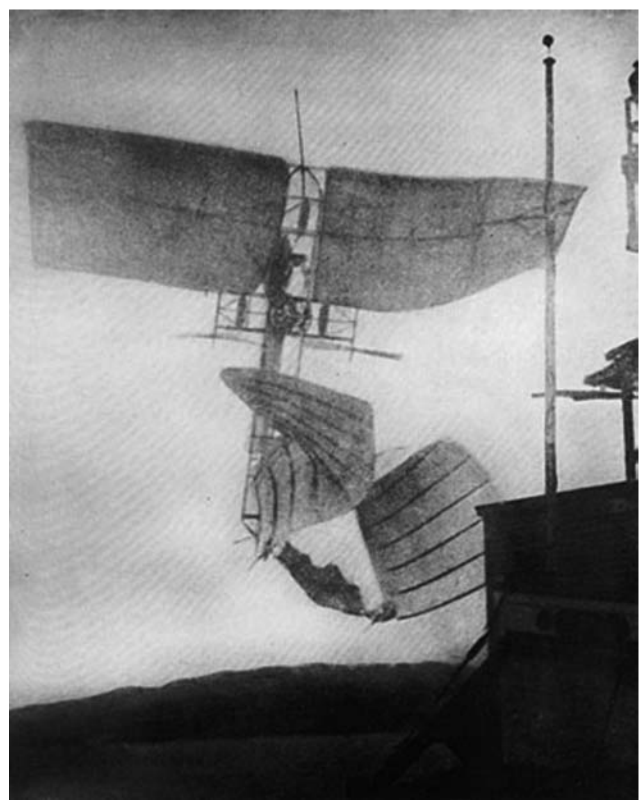

Figure 9. Langley Aerodrome aeroelastic launch failure December 1903 showing forward wing collapse. Reproduced with permission from Phil Callihan.

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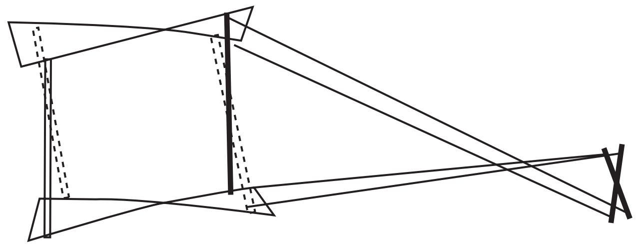

Figure 10. 1901 Wright Brothers wing warping kite.

twice, including once several days before the Wright Brother's first flight in 1903 (Figure 9). Collar (1978) attributes this failure to a static structural instability called *wing divergence*. Garrick and Reed (1981) disagree with this conclusion, citing recent evidence that rules out static instability, but identify excessive flexibility that led to an overload situation.

In 1901 the Wright Brothers used a tethered kite, sketched in Figure 10, to demonstrate *wing warping* – controlled, antisymmetrical biplane wing twisting – to create active roll control. Wing warping depends on building torsionally flexible wing surfaces easily distorted by the pilot. Wing warping surfaces must be very flexible; they are also easily distorted by the airstream to produce unintended airloads. This concept was used on the Wright Flyer in 1903. Most early monoplanes were controlled by wing warping. These included the Bleriot XI, shown in Figure 11, the British Bristol Prier Monoplane, and, as late as 1915, the Fokker Eindecker. The most often cited early aeroelastically induced wing failure occurred on the Fokker D-VIII monoplane (Fokker, 1931). Strengthening

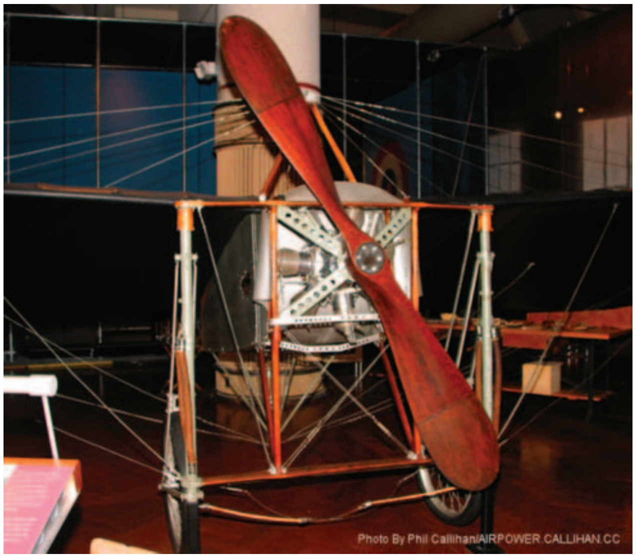

Figure 11. The wire-braced Bleriot XI with a wing warping control system first flew between England and France in 1909.

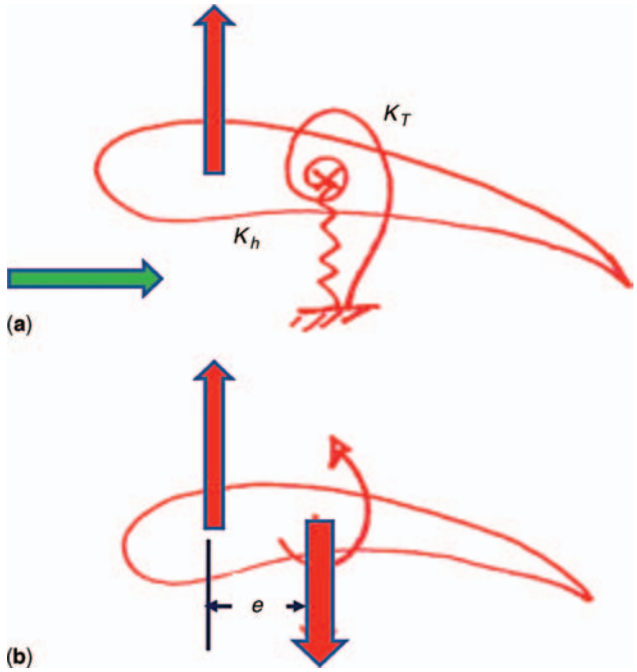

Figure 12. (a) (b) The typical section aeroelastic idealization.

its rear spar created a tendency of the leading edge to twist, producing excessive lift forces during dive pullout. Designers and operators struggled to understand the source of these problems.

The Bleriot XI and the Bristol Prier had fatal accidents around 1912. Bleriot strengthened the Bleriot XI guy wires and increased the main wing spar size, but wing failures still occurred. On 12 September 1912 the British War Office placed a ban on Royal Flying Corps pilots flying monoplanes after a series of accidents involving the Bristol Prier monoplane (Baker, 1994). Some witnesses to these crashes reported in-flight explosions. The French issued their own monoplane flight ban soon after. These temporary bans created an anti-monoplane bias lasting through World War I. Designers did not understand that aeroelastic problems are stiffness related, not stress related.

Figure 12 shows the *typical section* aeroelastic idealization used to illustrate static aeroelastic phenomena. The model has two degrees of freedom called pitch and plunge and is supported by two linear elastic springs located at the *shear center*. The two springs, with stiffnesses K_h and K_T, simulate the stiffness of wing spars and skin. The displacement h, called "plunge" represents wing surface bending displacement, measured at the wing section shear center, whereas θ represents wing structural torsion. The shear center or *center of twist* is a point on the wing section where the application of a concentrated force creates plunge/bending deflection, but no twist.

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The typical section lift vector and pitching moment act at the *aerodynamic center*. Thin airfoil theory for incompressible flow predicts the aerodynamic center to be at the 1/4 chord position. At the aerodynamic center the aerodynamic pitching moment depends on airfoil camber but does not change with angle of attack. The lift and moment are expressed in coefficient form as:

$$L = qSC_{L_\alpha}(\alpha_0 + \theta) \tag{1}$$

$$M_{AC} = qScC_{MAC} \tag{2}$$

with wing chord dimension, c, planform area S, airspeed V, dynamic pressure $q = \frac{1}{2}\rho V^2$ and initial angle of attack, α_o. The C_{MAC} term is an aerodynamic coefficient that depends on wing camber. For positive camber (curvature downward) C_{MAC} is negative.

Aerodynamic lift and moment twist the airfoil an amount θ, creating additional lift because the aerodynamic center is offset from the shear center by the distance e. The structural twist is:

$$\begin{aligned}\theta &= \frac{\frac{qSeC_{L_\alpha}\alpha_o}{K_T}}{1-\frac{qSeC_{L_\alpha}}{K_T}} + \frac{qScC_{MAC}}{K_T}\left(\frac{1}{1-\frac{qSeC_{L_\alpha}}{K_T}}\right) \\ &= \left(\frac{\bar{q}}{1-\bar{q}}\right)\alpha_o + \left(\frac{cC_{MAC}}{eC_{L_\alpha}}\right)\left(\frac{\bar{q}}{1-\bar{q}}\right)\end{aligned} \tag{3}$$

The typical section lift is:

$$\begin{aligned}L &= \frac{qSC_{L_\alpha}}{1-\frac{qSeC_{L_\alpha}}{K_T}}\alpha_\mathrm{o} + \frac{qSC_{L_\alpha}}{1-\frac{qSeC_{L_\alpha}}{K_T}}\left(\frac{qScC_{MAC}}{K_T}\right) \\ &= qSC_{L_\alpha}^{\mathrm{flex}}\alpha_\mathrm{o} + qScC_{MAC}^{\mathrm{flex}}\end{aligned} \tag{4}$$

The nondimensional aeroelastic parameter $\frac{\bar{q}=qSeC_{L_\alpha}}{K_T}$ represents the ratio of aerodynamic overturning moment divided by structural restoring moment. When $\frac{qSeC_{L_\alpha}}{K_T}=1$ we have *wing divergence*. An initial angle of attack, no matter how small, creates an infinitely large lift and twist. Wing divergence is a static structural instability.

The typical section model results in equations (3) and (4) show that flexible surfaces with large camber, such as the Langley *Aerodrome* wing and tail, develop large nose-down twist at low speeds because the second terms in equations (3) and (4) are negative. With large camber the second term in equation (4) dominates even if the aerodynamic center/shear center offset e is close to zero. Structural tests (Garrick and Reed, 1981) performed on the original Langley *Aerodrome* in 1980 found the front wing shear center to be very close to the wing aerodynamic center. This lack of a measurable offset between the shear center and wing aerodynamic center excludes wing divergence as a mode of failure but not aeroelastic overloading caused by high camber and low torsional stiffness.

The typical section effective lift curve slope C_{L_α} is modified by aeroelasticity to be:

$$C_{L_\alpha}^{\mathrm{flex}} = \frac{C_{L_\alpha}}{1-\frac{qSeC_{L_\alpha}}{K_T}} = \frac{C_{L_\alpha}}{1-\bar{q}} \tag{5}$$

Because e is usually positive, the typical section develops more lift than a similar rigid surface at a given angle of attack. For large values of $\bar{q}$ wing lift is sensitive to small changes in angle of attack.

Aircraft load factor, n, is defined as $n = L/W =$ lift/weight. For the typical section model the change in the load factor with respect to the aircraft angle of attack is

$$\frac{\partial n}{\partial \alpha_o} = \frac{\partial \frac{L}{W}}{\partial \alpha_o} = \frac{qC_{L_\alpha}}{\left(\frac{W}{S}\right)(1-\bar{q})} \tag{6}$$

Braced-wing monoplanes, such as the Bleriot XI, had small wing torsional stiffness and operated at large values of $\bar{q}$. Small changes in the angle of attack, either by pilot input or by atmospheric gusts, created large increases in wing external force and led to the documented failures.

When two typical section wings, each with area S, provide total lift, L, to support an airplane with weight, W, the required wing angle of attack is:

$$\alpha_\mathrm{o} = \frac{1}{2}\frac{W}{S}\left(\frac{1-\bar{q}}{qC_{L_\alpha}}\right) - \alpha_M \tag{7}$$

The wing section twist angles, θ, are both:

$$\theta = \frac{1}{K_T}\left(\frac{eW}{2} + qScC_{MAC}\right) \tag{8}$$

No aeroelastic parameter appears in this fixed lift equation. If camber is zero, then the twist angle is constant at all flight speeds; the airplane angle of attack decreases to zero as the flight q approaches the divergence condition $\bar{q}=1$. However, sensitivity to changes in angle of attack remains. The divergence instability still exists.

In 1926, Hans Reissner published a landmark paper "Neuer Probleme aus der Flugzeugstatik" (New Static Structural Problems of Wings) detailing static aeroelastic phenomenon such as lift effectiveness and wing divergence. Similar papers soon appeared in other countries; static aeroelastic problems began to be understood and controlled.

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4 COMPRESSIBILITY: MACH NUMBER EFFECTS

Wing divergence dynamic pressure depends on the Mach number. Mach number, M, is the ratio of airspeed divided by the local speed of sound and measures the importance of flow compressibility. Dynamic pressure is written in terms of Mach number as:

$$q = \frac{1}{2}\rho V^2 = \frac{1}{2}\rho a M^2 \tag{9}$$

The symbol a represents the speed of sound and depends on altitude; this relationship is called the *atmosphere line*.

The *Prandtl–Glauert transformation* provides an estimate of the lift curve slope for compressible flow written as:

$$C_{L_\alpha} = \frac{C_{L_{\alpha_0}}}{\sqrt{1-M^2}} \tag{10}$$

As a result,

$$q_{\mathrm{D}} = \frac{K_T\sqrt{1-M_{\mathrm{D}}^2}}{SeC_{L_{\alpha_0}}} = \frac{1}{2}\rho a M_{\mathrm{D}}^2 \tag{11}$$

The solution to this equation is called the *match point*. The Mach number used to compute the aerodynamic features of the typical section matches the local Mach number used to compute the dynamic pressure.

Figure 13 plots divergence speed as a function of altitude for a typical section that will diverge at 250 lb ft^{-2} in incompressible flow. This is compared to the divergence speed computed when the Mach number effects are ignored. Compressibility effects are particularly important at high altitude.

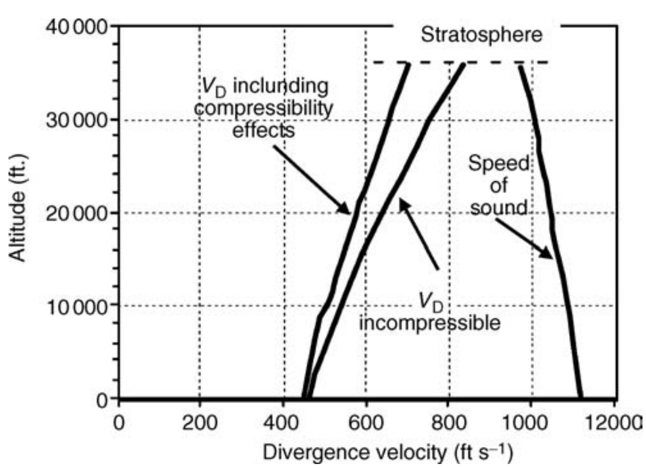

Figure 13. Altitude vs. divergence speed showing compressibility effects ($q_{D_0} = 250$ lb ft^{-2}).

5 CONTROL EFFECTIVENESS

Aeroelastic flight control problems began in the 1920s. In 1927 the British Bristol Bagshot fighter prototype encountered aileron reversal. Figure 14 shows the typical section with a control surface added. An angle of attack α_{o} and a control surface rotation δ_{o} produce lift, L, and a pitching moment at the aerodynamic center, M_{AC}, expressed as:

$$L = qSC_{L_\alpha}(\alpha_{\mathrm{o}} + \theta) + qSC_{L_\delta}\delta_{\mathrm{o}} \tag{12}$$

$$M_{AC} = qScC_{MAC} + qScC_{M_\delta}\delta_{\mathrm{o}} \tag{13}$$

where

$$\frac{\partial C_L}{\partial \delta} = C_{L_\delta} \tag{14}$$

and

$$C_{M_\delta} = \frac{\partial C_{MAC}}{\partial \delta_{\mathrm{o}}} \tag{15}$$

a negative number. The twist, θ, is:

$$\theta = \frac{qSeC_{L_\alpha}\left(\alpha_{\mathrm{o}} + \frac{c}{e}\frac{C_{MAC}}{C_{L\alpha}}\right) + qSeC_{L_\delta} + \frac{c}{e}C_{M_\delta}\delta_{\mathrm{o}}}{K_T - qSeC_{L_\alpha}} \tag{16}$$

The lift only resulting from δ_{o} (with $\alpha_{\mathrm{o}} = 0$) is

$$L = qsC_{L_\delta}\delta_{\mathrm{o}}\frac{1 + q\frac{Sc}{K_T}\frac{C_{L_\alpha}}{C_{L_\delta}}\left(C_{M_\delta}\right)}{1-\overline{q}} \tag{17}$$

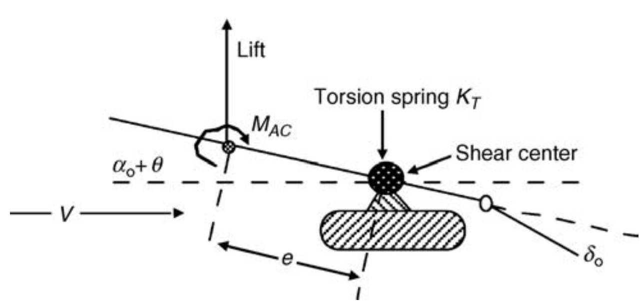

Figure 14. Typical section with control surface.

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with

$$\frac{C_{L_\delta}}{C_{L_\alpha}} = \frac{\partial C_L}{\partial \delta} \times \frac{\partial \alpha}{\partial C_L} = \frac{\partial \alpha}{\partial \delta} \tag{18}$$

defined as the apparent change in section angle of attack resulting from a unit control deflection, δ_o. This term is a function of the ratio of the control surface chord to wing chord ratio. The lift expression is also written as:

$$L = L_r \left[\frac{1-(q/q_R)}{1-(q/q_D)}\right] \tag{19}$$

where

$$L_r = qSC_{L_\delta}\delta_o \quad q_R = \frac{-K_T}{ScC_{M_\delta}}\frac{C_{L_\delta}}{C_{L_\alpha}} = \frac{-K_T}{ScC_{M_\delta}}\frac{\partial \alpha}{\partial \delta_o}$$
$$q_D = \frac{K_T}{SeC_{L_\alpha}} \tag{20}$$

The term q_R is called the *aileron reversal dynamic pressure* and is a positive number since $C_{M\delta}$ is a negative number. *Control effectiveness* for the typical section is defined as:

$$\frac{L}{L_r} = \left[\frac{1-(q/q_R)}{1-(q/q_D)}\right] \tag{21}$$

If the flight dynamic pressure, q, is less than the divergence dynamic pressure, then, as dynamic pressure approaches q_R, L/L_r decreases; it becomes zero when $q = q_R$. At this speed the control surface deflection produces no lift because lift created by nose-down twist cancels control deflection lift.

Control reversal is not a structural instability. Near the divergence speed a very small input α_o produces a very large structural response θ. At reversal a large input δ_o produces no output (L/L_r). At reversal the twist angle θ may be large or small. Above the reversal speed, a control input δ_o produces lift force opposite to that intended, but the structure is still stable.

6 FLUTTER AND BUFFET APPEAR

The term *flutter* first appeared in 1924 in a published document by R.T. Glazebrook in the *Yearbook of the British Aeronautical Committee*. Flutter is a dynamic, oscillatory structural instability enabled by interactions between unsteady aerodynamic forces and moments created by vibratory motion of lifting surfaces and the vehicles to which these surfaces are attached. Empennage/tail flutter appeared first in 1915 when the British Handley bi-plane bomber experienced severe tail flutter.

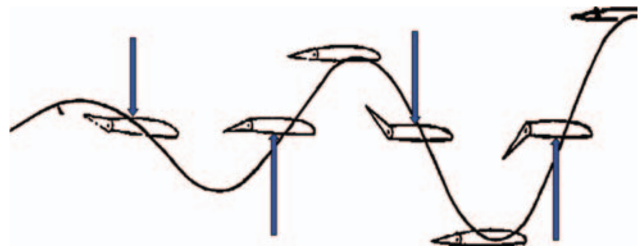

Figure 15. Interactions between ailerons and wing bending motions at high speed produced the first catastrophic flutter events in the 1920s (Cook, 1991).

After World War I airspeeds increased; low drag, semi-monocoque, monoplane designs appeared and with them *wing-aileron flutter*. As indicated in Figure 15, at the onset of wing-aileron flutter, phasing between the aileron motion and the wing motion is such that lift from the aileron motion inputs energy into the wing (Cook, 1991). The result is undamped, exponentially divergent oscillations. The flutter speed depends on the aileron/wing connection stiffness, wing bending frequency and the location of the aileron center on gravity with respect to the aileron/wing hinge.

Wing/aileron flutter and its counterpart, rudder/vertical tail flutter, were common in the 1920s. The British Gloster-Grebe biplane fighter encountered wing/aileron flutter in 1923. In the Netherlands, the Van Berkel W.B. seaplane experienced violent wing oscillations in flight.

In Germany, Von Bambauer and Koning (1923) identified the cause of and solutions to wing/aileron flutter by conducting experimental and theoretical investigations of wing/aileron dynamic interaction. They were the first to understand that flutter can be prevented by decoupling modes of motion. Their decoupling was the result of attaching small weights ahead of the aileron hinge line; this "mass balancing" solution is still used. In 1925 the last wooden British fighter, the Gloster Gamecock, encountered rudder/vertical tail flutter. This was also eliminated by mass balancing the rudder.

Accurately predicting flutter speed depends strongly on unsteady airload calculations. German researchers took the lead developing unsteady aerodynamic theories that addressed flutter problems (Herschel, Prem and Madelung, 2004). In 1922 Walter Birnbaum provided expressions for the oscillatory aerodynamic loading on a two-dimensional flat plate (much like the typical section) undergoing low-frequency oscillatory motion in an airstream. Two years later Herbert Wagner developed the first transient solution, the Wagner function, describing lift development on a

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two-dimensional flat plate airfoil when it was given an instantaneous increase in its angle of attack. By the end of the 1920s the fundamentals of flutter were clearly understood (Frazer and Duncan, 1920).

In the United States, Theodorsen's work on unsteady aerodynamics added further fidelity to aeroelastic analysis of the *typical section* (Theodorsen, 1935). A key parameter in these calculations is the reduced frequency of oscillation, defined as $k = \frac{\omega b}{V}$. The parameter b is the wing section semichord, whereas V is the airspeed, and ω is the frequency of oscillation. Flow unsteadiness is very important for calculations for systems with reduced frequencies beginning about $k = \frac{\omega b}{V} \approx 0.04$. Smilg and Wasserman (1942) outlined a flutter analysis procedure called the *V-g method*. This method was a standard computational technique for several decades.

7 SWEPT WING AEROELASTICITY

In 1935 Dr Adolph Busemann first proposed sweeping wings to delay the onset of wave drag resulting from high speed flow compressibility. High-speed swept wing aircraft designs appeared in Germany late in World War II. There are three reasons to sweep a wing forward or backward: (i) to improve longitudinal stability by reducing the distance between the aircraft center of gravity and the wing aerodynamic center; (ii) to provide longitudinal and directional stability for tailless (flying) wings; and (iii) to delay transonic drag rise (compressibility).

Although not the first swept wing aircraft, the American B-47 jet bomber shown in Figure 16, was the first to encounter and address high-speed swept wing aircraft aeroelastic issues ranging from aerodynamic load redistribution to flutter. The B-47s high aspect ratio swept wings and speed of 607 mph

Figure 16. The Boeing B-47, first flown 17 December 1947, was the first aircraft to address swept wing aeroelasticity.

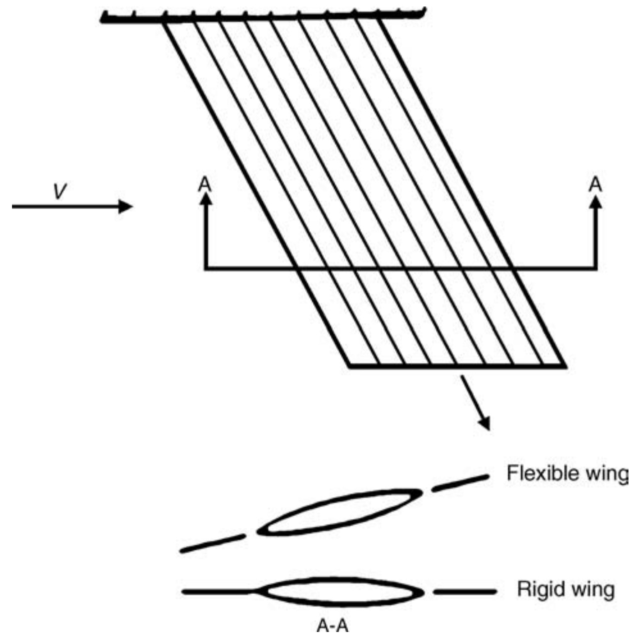

Figure 17. Swept wing bending flexibility produces washout that reduces local angle of attack at wing sections such as A-A.

led to radically new ways of analyzing aeroelastic behavior (Bhatia, 2003).

As indicated in Figure 17, sweptback wing bending displacements reduce wing section angles of attack, leading to three static aeroelastic problems: (i) flexible sweptback wings are lift ineffective because wing bending reduces the total wing lift for a given wing angle of attack; (ii) bending deformation moves the wing center of pressure inboard and forward; and, (iii) bending displacement reduces sweptback wing aileron effectiveness as well as sweptback tail/rudder effectiveness so much that ailerons are often replaced by spoilers. On the positive side, local angle of attack reduction created by swept wing bending counters the increased angle of attack created by torsion. This cancellation makes wing divergence unlikely if wings are swept back >10–15°, but more likely if the wings are swept forward.

The increased complexity of high-speed swept wing jet aircraft construction challenged engineers. Swept wing static aeroelasticity studies appeared in the late 1940s and early 1950s (Diederich and Budiansky, 1948; Pai and Sears, 1949; Diederich and Foss, 1952). Static aeroelastic methods for the complete airplane were developed at the Boeing Company about 1950 and published in an NACA report (Gray and Schenk, 1953). Design, testing and certification created new demands for analytical fidelity. These demands led Boeing engineers to develop a new aero/structural analysis method called the *finite element method*. The paper by Turner *et al.* (1956) coupled with the emergence of high-speed computers,

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laid the groundwork for a new powerful method of structural analysis that has since been embraced by the entire structural engineering community.

8 HIGH-SPEED FLUTTER PROBLEMS

In 1947 the Bell X-1 broke the sound barrier; soon many fighters and a few bombers were flying at supersonic speeds. In 1958 the X-15 flew at hypersonic speeds at the edge of the Earth's atmosphere. By 1982 the first Space Shuttle showed that it could operate at speeds ranging from subsonic to hypersonic.

During the post-World War II era, airplane size also grew, as did the variety of different configurations, including flying wings. A new phenomenon called *Body Freedom Flutter* appeared. Body freedom flutter occurs as the aircraft short period mode, a basic flight mechanics parameter, increases with airspeed and comes in close proximity to a wing vibration mode. The modal coupling between aircraft pitching motion and wing structural vibration strongly affects the dynamics of the aircraft. This problem appeared on the RB-57 reconnaissance aircraft, a version of the B-57 with extended, high aspect ratio wings and a short fuselage. It is also a possible mode of instability on flying wings and forward swept wing designs.

High speed flight created new aeroelastic challenges for launch vehicle structures. *Supersonic panel flutter,* depicted in Figure 18, occurs because of modal coupling between the first two bending modes of a platelike surface. Early models of the German V-2 rocket were rumored to have been lost because of flutter of panels near the rocket nose. Panel flutter plagued X-15 tail surface panels, creating loads and acoustic problems. The Saturn V launch vehicle also suffered from this problem.

Panel flutter differs from other wing flutter because; (i) only one side of the platelike structure is exposed to high speed flow whereas the other is in contact with dead air; and (ii) its onset usually leads to limit cycle oscillations that create a severe, sustained, acoustic environment and lead to high cycle fatigue.

High-speed flutter instabilities include single degree of freedom flutter (Lambourne, 1968). Unlike classical flutter requiring modal interaction, single degree of freedom flutter involves instability mechanisms such as loss of damping, flow separation, pure pitching oscillations, or pure bending oscillations. Single degree of freedom pitch oscillations require oscillation at special reduced frequencies and pitch axis locations.

A common sustained, limit cycle flutter with one degree of freedom is *control surface buzz.* This instability requires shock wave–boundary layer interaction to trigger separated flow with periodic shock wave reattachment. This instability is highly nonlinear with the onset of a limit cycle oscillation dependent upon unsteady aerodynamic phenomena.

9 FLUTTER DESIGN SOLUTIONS

The key to preventing classical flutter events is to eliminate or delay the modal interaction shown in Figure 19. Flutter problems are ameliorated by combinations of: (i) mass distribution changes (adding or removing mass at specific lifting surface locations); (ii) structural stiffness changes (thickening wing skin or other structural members); or (iii) aerodynamic surface changes (changing wing span, chord dimensions or reshaping planiform areas).

Figure 20 shows that the location of the wing section center of gravity changes flutter speed. Placing the center of gravity near the 1/4 chord significantly increases flutter speed. Center of gravity position also depends on fuel

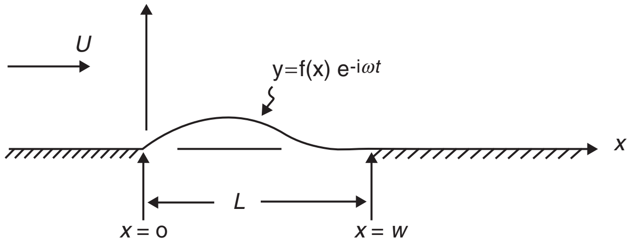

Figure 18. Panel flutter occurs at supersonic speeds and is characterized by bending vibration modal coupling.

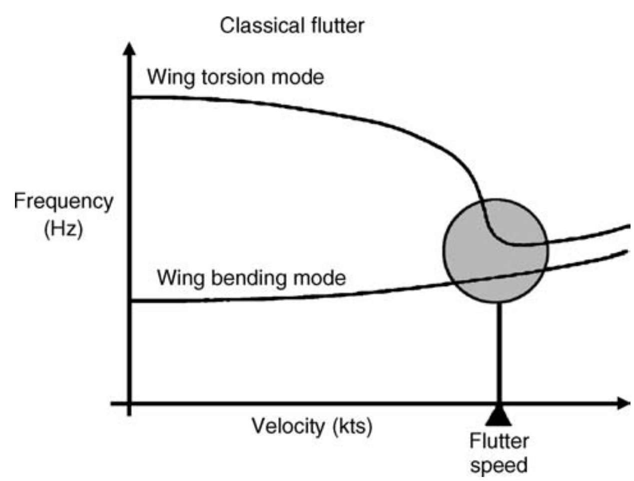

Figure 19. Delaying or eliminating modal coupling as airspeed increases is the key to preventing classical flutter.

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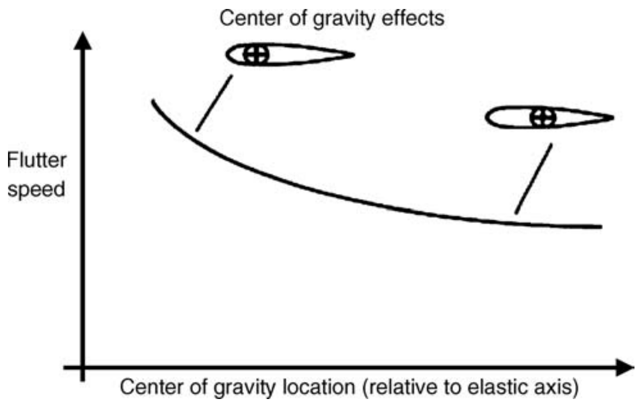

Figure 20. Wing cross-sectional center of gravity location is an important factor in determining flutter speed. Forward c.g. locations are better.

Figure 21. The DARPA X-29 research aircraft promoted aeroelastically tailored advanced composite materials to control undesirable static aeroelastic effects.

tank placement, under-wing missile or bomb placement, and engine placement. Re-positioning engines, including lengthening or shortening attachment pylons, affects flutter speed. Flutter speed increase may require *mass-balancing*, adding nonstructural mass, such as lead weights, to critical places on a wing, an engine, an aileron or an elevator.

Many fighter aircraft carry a wide variety of externally mounted, under-wing armaments or “stores.” Operational requirements may create a large number of possible store combinations. This, in turn, may result in missions where the flutter speed is less than possible operational airspeeds.

10 CONTROLLING AND EXPLOITING AEROELASTICITY: AEROELASTIC TAILORING AND AEROSERVOELASTICITY

With the advent of modern advanced composite materials, passive aeroelastic control, known as *aeroelastic tailoring*, has come to include structural stiffness design to control deformation mode coupling. Strong directional stiffness exhibited by advanced composite materials such as graphite/epoxy provides the ability to decouple or couple aeroelastic deformations. Aeroelastic tailoring through use of composite laminate design is used to increase lift effectiveness, control effectiveness, and flutter speed.

In the early 1970s aeroelastic tailoring was applied to sweptforward wing designs with inherently low wing divergence speeds. The DARPA X-29 research aircraft shown in Figure 21 was the result of these efforts.

Aeroelastic tailoring introduces structural bending/torsion elastic coupling by offsetting the direction of laminate fibers, as indicated in Figure 22. A sweptforward wing with the fibers aligned along the wing swept axis leads to deformation called *wash-in* and increased airloads. This reduces the wing divergence airspeed compared to an unswept wing. Substantial added structural stiffness (and weight) is required to provide aeroelastic stability.

Re-orienting laminate fibers changes bend/twist coupling. As the wing bends upward it twists in the nose-down direction, creating *wash-out*. This reduces the local airloads and increases wing divergence speed without extra weight. Figure 23 summarizes the effects of laminate fiber orientation on sweptback wing aeroelastic phenomena (Shirk, Hertz and Weisshaar, 1986).

Aeroelastic phenomena such as aileron reversal have been exploited to create actively controlled, light-weight structures. The X-53 Active Aeroelastic Wing (AAW) test aircraft

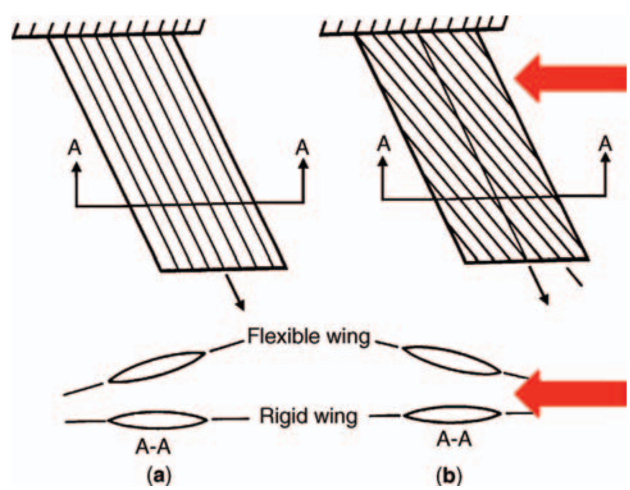

Figure 22. Sweptforward wing aeroelastic tailoring increases wing divergence speed.

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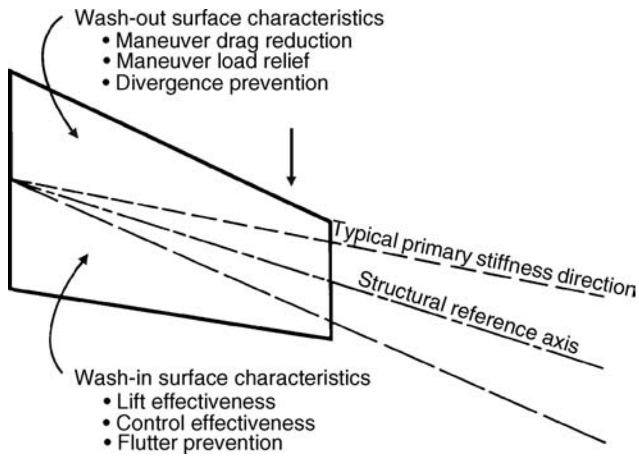

Figure 23. Aeroelastic tailoring applied to sweptback wing aeroelastic performance, showing preferred directions for improving aeroelastic performance.

shown in Figure 24 uses a combination of active and passive aeroelastic control to produce a highly flexible, lightweight control system (Pendleton *et al*., 2007). The X-53 wing has a relatively low reversal speed; it uses ailerons to create wing distortion much like wing warping to roll the aircraft.

Aeroservoelasticity uses interactive, active flight control to modify aeroelastic dynamic response and stability. In the past few decades, aircraft active flight control has brought flight mechanics much closer to aeroelasticity than it has been in the past. Until a few decades ago, except in unusual cases, aeroelasticians isolated lifting surface aeroelastic response from vehicle response. Aeroservoelasticity began by improving XB-70 supersonic bomber ride quality. Later active flutter suppression using actively controlled ailerons was demonstrated on a B-52 test aircraft (Roger, Hodges and Felt, 1975).

Figure 24. X-53 Active flexible wing aircraft.

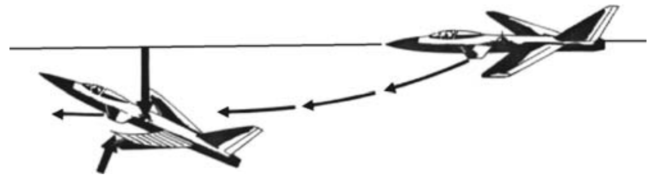

Figure 25. Forward swept wing body-freedom flutter replaces static divergence as an instability mode.

11 STATIC STABILITY OR FLIGHT DYNAMIC STABILITY: DIVERGENCE OR FLUTTER?

Wing static divergence is a theoretical phenomenon; it has been called "flutter at zero frequency." Divergence of lifting surfaces is unlikely to appear in flight because, unlike a wing rigidly mounted in a wind tunnel, freely flying aircraft respond to large forces developed when an aerodynamic surface deforms. This is particularly true of flying wings, sweptforward wings, joined wings, and oblique wings.

When the airplane is treated as a dynamic system, the wing divergence mode appears as a low frequency wing-body vibration mode. This mode displays a classical modal interaction with other flight mechanics modes, in particular the short period mode, to create body freedom flutter as shown in Figure 25.

As a result, most aeroelastic analyses are conducted with models that include *rigid-body degrees of freedom*. These mathematical models are far more complex than those first imagined decades ago when the first flutter analysts began their quest for understanding and control. In addition, this approach has moved aeroelasticity from a specialty area into the mainstream of aircraft structures that recognizes the need to treat the system as a truly integrated flying system.

12 SUMMARY

The strong interactions among various aeroelastic constituencies, such as structures, flight mechanics and control, unsteady aerodynamics, and vehicle performance has created unusual problems for aircraft development during the past century. Creative, skilled aeroelasticians have risen to the occasion and solved these problems. Today, the wealth of comprehensive analytical methods available to engineering analysts allows developers to examine aeroelastic phenomena before they produce unpleasant surprises. However, aeroelasticity continues to be a vibrant discipline required when new and unusual aircraft configurations are designed. Future vehicles such as those with rapidly changing mass properties, trans-atmospheric aircraft or vehicles with rapidly

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changing shape, such as morphing aircraft may hold interactive surprises. This statement is particularly applicable to new generations of unmanned vehicles, micro air vehicles and hypersonic vehicles. Each new flight era brings with it the prejudices of the past and the challenges for the future. So too it will be with aeroelasticity.

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TEXTBOOKS AND FURTHER READING

Classic aeroelasticity textbooks describe aeroelastic phenomena in all flight speed regimes and provide methods to analyze them. These methods have been largely supplanted by modern computational developments, but textbooks remain an excellent source of fundamental explanations, classic examples, and valuable, basic computational techniques that can be used for first estimates of aeroelastic effects. The following references are recommended for further study.

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