

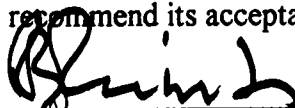
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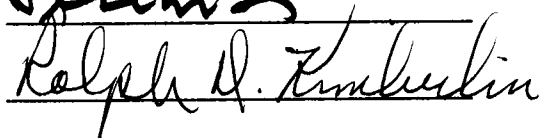
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**STABILITY AND CONTROL FOR THE LIGHT AIRPLANE PILOT**

**A Thesis**

**Presented for the**

**Master of Science**

**Degree**

**The University of Tennessee, Knoxville**

**Richard Noble Clark**

**August, 1994**

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## **DEDICATION**

This thesis is dedicated to my parents

Mr. William H. Clark

and

Mrs. Frances Lou Clark,

memories of whom still inspire me.

## **ACKNOWLEDGEMENTS**

I would like to thank the following people who have reviewed my manuscript or given me counsel: Dr. Vern Gordon, Mr. JJ McCue, Mr. Jim Lewis, Mr. Dan Canin, Professor Bob Richards, and my thesis advisor, Dr. Peter Solies. I would especially like to thank my wife, Kristen, and my sons Sam and Alex, for the free giving of their time to let me pursue this work.

## ABSTRACT

This thesis is intended to be a literary work to fill a void in the lightplane and homebuilder library. It is the author's view that, in this aviation segment, there is a general lack of fundamental knowledge in the areas of aerodynamics, stability and control, and flight test applications.

In this manuscript, the reader will be led down the path of a deeper understanding of these subjects. Aerodynamics is presented first, as the foundation for stability and control study. Most chapters are subdivided into "Fundamentals" and "Details" sections. In the first section, a basically text-based discussion and development of concepts will occur. This section is intended to be readable by nearly any pilot. In the following section, some of the mathematical relationships of classical airplanes will be presented to facilitate a deeper discussion of theory and practical application. Following aerodynamics as the foundation comes the major work of this manuscript, a focus on classical airplane stability and control. It is the fervent desire of this author that a smarter, safer general aviation pilot will result.

In the text, italicized entries introduce new terms, and are also listed in the Glossary. Numerous Bibliography entries provide the reader with further material should this thesis inspire more study or interest.

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# CHAPTER 1

## INTRODUCTION AND PURPOSE

If you are a lightplane or homebuilt airplane pilot, and want to know more about how airplanes work, this manuscript was written for you. It is my goal to lead the reader through a tangle of physics and flight test that, once read, will lead to a much clearer understanding about airplanes and flying.

The general approach will be to divide the major areas of interest into separate chapters, which may be further subdivided into sections, e.g. "Fundamentals" and "Details." In the first section, basic concepts will be introduced and developed. In the second section of each chapter some of the underlying mathematics will be presented for those readers interested in truly understanding. (It was once said that "application without theory is like blind-man's bluff.") Finally, Appendix A will acquaint the reader with the fundamentals of designing and executing a proper flight test program.

This purpose requires some elaboration. Most lightplane pilots know little else of aerodynamics but "lift equals weight" and "thrust equals drag." Though these statements may be generally true, they leave much unanswered. For example, if thrust really equaled drag, an airplane could never accelerate to takeoff speed! This shallow level of knowledge is worrisome in any pilot, but if the pilot is a home-builder of a lightplane, he is ill-equipped to

safely take on the challenge of testing or evaluating the basic flight characteristics of his new or modified airplane.

Used in this sense, "flight characteristics" are the collective description of what it is like to fly and perform useful maneuvers in an airplane. In a technical sense, these characteristics are defined by measurable or identifiable items of interest within a discipline generally referred to as "stability and control." The meaning of this term will be discussed shortly, but for now the terms "flight characteristics" and "stability and control" will be used interchangeably.

Stability and control is only one piece of the puzzle, albeit an important one. This subject addresses questions like "What are the control force characteristics?" and "How hard is it to flare the airplane for landing?" Airplane performance is a related but quite separate subject. Performance addresses questions like "How fast does it fly?" and "What is the speed to fly in order that the maximum distance might be covered?" Another subject of considerable importance is airplane strength and verifying flight air loads, that is, what kind of abuse (overloading) the structure of the airplane can stand before cracking, bending, or breaking. There are other subjects of undeniable interest as well, but the primary subject of this manuscript is stability and control. The other topics would be more properly discussed in a follow-on effort.

It is assumed that the reader is familiar and conversant with the aerodynamics and terms as presented in typical study materials for the Federal Aviation Administration (FAA) commercial certificate. Terminology and concepts will then be developed as necessary to arrive at a deeper understanding using terms and phrases well understood by engineers and

test pilots. Use of the proper terminology will aid in the understanding of other literature, and in communicating with engineers and test pilots.

Understanding of the words “stability and control” are required before proceeding. The American Heritage Dictionary [1], states in part that “stable” is defined as “1. resistant to sudden change, as of position or condition; and 2. maintaining equilibrium.” This concept might be best understood by reference to figure 1-1, while a more detailed discussion may be found in Hurt [3]. Part A of the figure represents a ball on the inside of a cup. If one moves it away from its equilibrium position to the left, its initial tendency will be to naturally return it to its starting point. For example, the typical airplane pitched up by a gust of wind will pitch down a moment later. More specifically, this situation is referred to as statically stable, a characterization of a system’s initial tendency to return to its original state when disturbed. The manner in which a statically stable system actually returns to equilibrium is covered in dynamics. Part B indicates a system that will tend to diverge if disturbed. A case in point is directional control of a conventional airplane, or “taildragger,” on takeoff or landing. If you

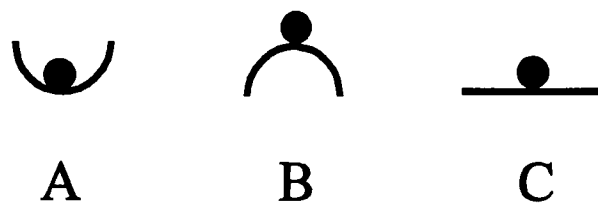


**Figure 1-1**  
Schematic of a A) Statically Stable System, B) Statically Unstable System, C) Statically Neutral System

aren't familiar with landing a taildragger, try balancing the end of a yardstick on the tip of a finger for a minute or two. It should be evident that unstable systems require considerable attention to maintain a desired condition. This system is in equilibrium at the depicted initial condition, and will not return to this condition without external assistance. Part C illustrates a system with neutral stability. Most airplanes display close to neutral stability when the roll attitude is left unattended, for example at 10° angle of bank (AOB)--the roll attitude tends to remain at the last AOB.

In reference 1, it is defined that "control" is "the power to regulate, direct, or dominate." Simply put, control is the facility the pilot has that allows him to point, roll, control attitude, or otherwise maneuver the airplane. There is a natural relationship between stability and control--the more stable a system is, the less control one has over it. Consider the obvious stability of the Washington Monument. No matter how much one might try, it would be very difficult or impossible to move. Conversely, the more unstable a system is the easier it is to displace it, and the more "control inputs" would be required to maintain its position (remember the yardstick). Stability and control, therefore, refers to the blending of enough stability and enough controllability in an airplane that a desired usefulness can be achieved. An airliner might be viewed as being more stable than a fighter jet, or a race car less stable than a sedan, yet each must display reasonable stability *and* control characteristics as determined by the vehicle's intended use.

**Before continuing, it is necessary to caution the reader. This text is intended to introduce a deeper understanding of a very complex and technical subject. It will not, by itself, prepare anyone to safely conduct test flights on production or homebuilt airplanes. These activities should be accomplished by a pilot experienced in other similar flights or test flying. Your local chapter of the Experimental Aircraft Association (EAA) should be able to refer you to competent personnel.**

## CHAPTER 2

### MECHANICAL SYSTEMS AND CHARACTERISTICS

In this chapter, a control system typical of most lightplanes will be discussed. This will be done by first presenting the control surfaces with their primary effects on the airplane. Secondly, a generic description of a flight control system will be presented. Lastly, miscellaneous characteristics of mechanical control systems will be introduced.

#### *Fundamentals*

Nearly all airplanes contain the same three basic controls as the Wright Flyer did in 1903--perhaps not in the same configuration, but certainly with the same intent. Refer to figure 2-1 for the following discussion. Forward and backward motion of the control stick or yoke controls the elevator (or canard, as in the Long Eze, or stabilator, as in the Cessna Cardinal) and produces a pitching motion; when one pushes forward, the nose of the airplane *itches* down. As this occurs, the airplane is actually rotating about its *lateral* axis. Think about the lateral axis as an imaginary line running the length of the wing of the airplane, and through the center of gravity (CG).

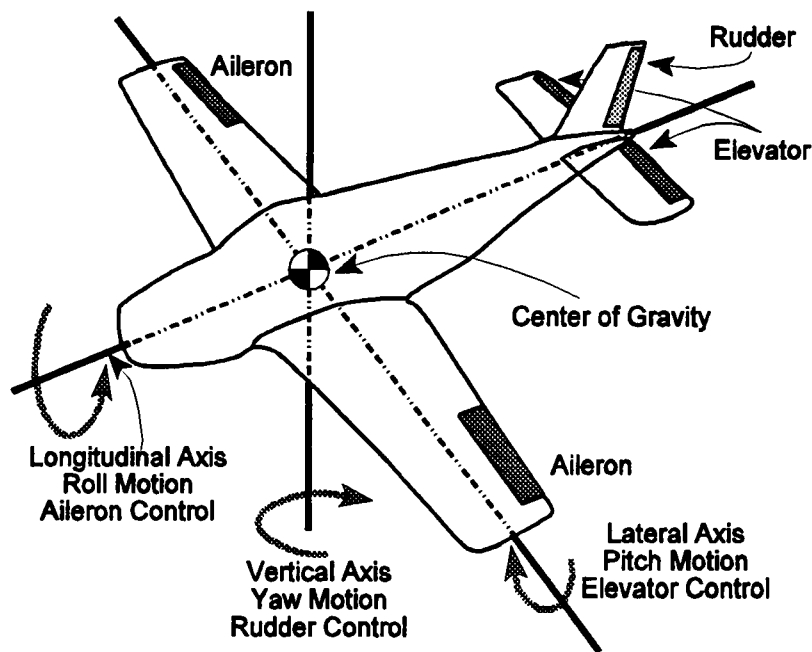


**Figure 2-1**

The Axes of the Airplane, Adjectives for Motion About Each Axis, and Primary Control Effector for Each Axis

Similarly, left or right motion of the stick or yoke controls the ailerons (and/or spoilers) and produces a banking movement, or *roll*. Rolling motion occurs about the *longitudinal* axis, which can be thought of as an imaginary line running the length of the fuselage, and through the CG.

Lastly, depression of the left or right rudder pedal controls the rudder(s) and produces a rotation about the *vertical* axis, and is called *yaw*. The vertical axis is perpendicular to the other two.

In a body fixed axis system, all of the axes move when the airplane moves. Don't be confused about this! As the attitude of the airplane changes, the axis system moves in precisely the same way--it is "attached" to the airplane.

## *Flight Control Systems*

Most general aviation airplane flight control systems (FCS's) consist of simple three-axis controls, that is pilot-operated controllers for each of the three axes. Refer to figure 2-2. These systems reflect aerodynamic forces back to the pilot through direct mechanical linkage, and are known as reversible control systems. What it means to "reflect forces" back to the pilot will be discussed in chapter 4. For now, realize that to move a flight control while flying requires force. It is true that increased stick deflection or deflection at increased speed requires greater force, which the pilot must supply to hold the control in that position.

A means of adjusting the elevator for zero longitudinal stick force (i.e. pitch trim) across a range of speeds is required. The means of adjusting these forces is referred to as elevator force trim, or just "trim." All reversible FCS airplanes today carry some sort of horizontal tail trimming facility, typically implemented through a movable tab on the trailing edge of the elevator (see figure 2-3). (In a canard or stabilator airplane, the technique is similar.) The basic concept may be described as follows. The pilot finds it necessary to hold

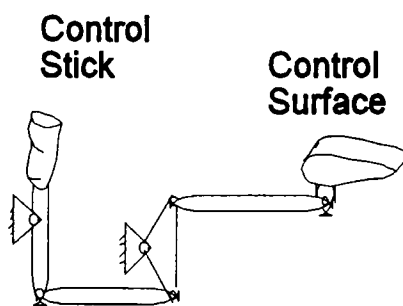


**Figure 2-2**  
Reversible Flight Control System

a force, for example 5 pounds, forward on the control stick after leveling off at cruising altitude. He adjusts the elevator trim in the nose down direction, which moves a tab on the trailing edge of the elevator in the up direction. This causes the elevator to move to a more trailing edge down position, pitching the nose down and reducing the forward pressure required of the pilot. This process is continued until the force is reduced to zero. Many airplanes can trim in two or all axes in exactly the same manner.

There are many variations of tabs and other “gimmicks” in use in reversible FCS airplanes, including servo tabs, geared tabs, bobweights, feel- and down-springs, and the use of aerodynamic balances. Askue [2], and Hurt [3], contain many excellent and amplifying remarks on these devices.

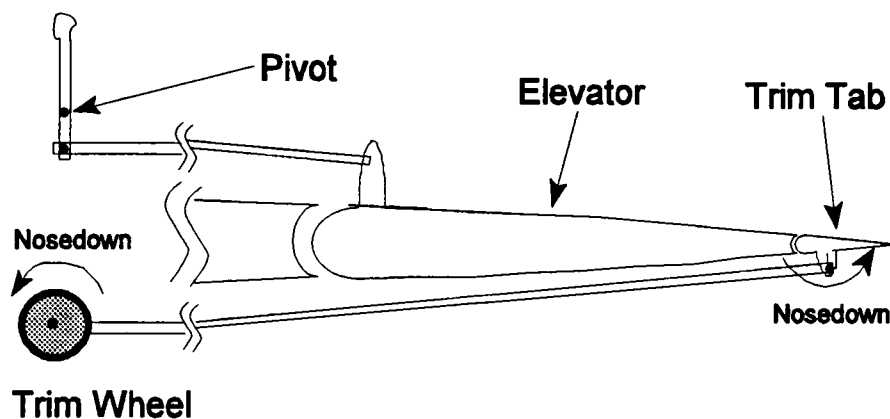


**Figure 2-3**  
Elevator Trim Tab Arrangement

## *Miscellaneous Mechanical Characteristics*

Any mechanical system can contain "impurities," or nonlinearities. Fortunately, most lightplanes have generally good mechanical characteristics. Nevertheless, an understanding of the causes and symptoms of the more common defects is warranted.

The force-versus-displacement characteristic is usually desired to be similar to that depicted in figure 2-4, part A. The most prevalent nonlinearity in mechanical FCS's is the presence of friction. Friction generates an uncertainty band in force about any stick position. For example, if a pilot was holding a few pounds of back stick force and slowly released pressure, the stick may hang up in a different position than if he had been holding forward force and slowly relaxed pressure. In this case, the friction has generated poor *centering*. In figure 2-4, part B illustrates such a system in flight (on the ground, reversible control systems are not experiencing air loads, and no force is usually required to move the controls). Observe the dashed line, which corresponds to a single stick position. Due only to the presence of friction, there exists a range of stick forces, between points 1 and 2, that can command this position. Since the stick is mechanically connected to the control surface, uncertainty in the stick position can cause uncertainty in the elevator position. Friction can therefore generate a band of speeds, for example 121-129 mph, where no longitudinal force is required. This is called a *trim speed band*. There is an analogous *roll rate band* caused by friction in the lateral control system.

A second nonlinearity found in mechanical FCS's is *freeplay* between the control position and the surface position. For example, if there was loose or worn linkage in the

control system, two stick positions could command the same surface position, as depicted in figure 2-4, part C. The stick positions between points 3 and 4 command the same elevator position. A pilot could observe this defect by noticing that to fly constant airspeed and altitude (throttle fixed), repetitive back-and-forth inputs, correction and counter-correction, were required. Freeplay can cause very undesirable flight characteristics, so the design of flight control systems should minimize it.

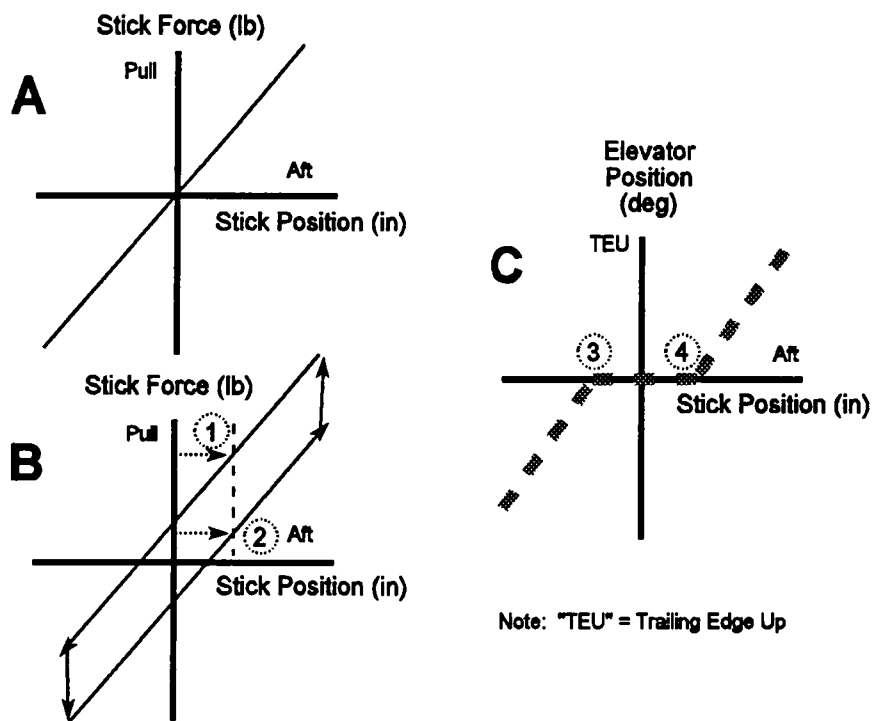


**Figure 2-4**

Representative Plots of Mechanical Characteristics: A) "pure" control system, B) control system with friction, and C) control system with freeplay

## CHAPTER 3

### AERODYNAMICS

This chapter presents principles of classical low-speed aerodynamics, a subject which receives cursory coverage in most lightplane-oriented texts. In the first section, the four forces are presented and discussed, while in the second section typical mathematical relationships used in the study of aerodynamics are introduced.

#### *Fundamentals*

**Balance of Forces.** Any object, including an airplane, must comply with Newton's Second Law, namely that the sum of all forces exerted on an object equal the product of its mass and its acceleration. (Mass is a measure of the property of an object that resists acceleration. On earth, an object's weight in pounds divided by the acceleration of gravity,  $32 \text{ ft/sec}^2$ , equals its mass measured in slugs.) In the case of steady flight (which means not changing its direction or speed), acceleration is zero, and the sum of all forces must equal precisely zero. This situation is referred to as *static*, and will be discussed in chapters 4 and 5. If the airplane is accelerating, e.g. increasing speed or maneuvering, the sum of all forces can not be zero; this subject will be addressed in the *dynamics* chapters. In aerodynamics,

forces are described in terms of the axis system presented earlier, in figure 2-1. Those forces contained in the geometric plane formed by the longitudinal and vertical axes (also referred to as the plane of symmetry) are normally the forces discussed in typical lightplane texts. Forces in this plane are usually broken down into four components:

- ▶ Lift, which is produced primarily by the wing.
- ▶ Drag, which is produced by the passage of the airplane through the air.
- ▶ Weight, generated by the gravitational attraction to the earth.
- ▶ Thrust, generated by a jet or a propeller. Of course, this force does not exist in a glider.

Lift. One does not have to understand Bernoulli's Theorem to know or learn how to fly airplanes, but to understand how a wing produces lift, it certainly helps! To convince yourself that lift is real, take a piece of paper about 2 inches by 8 inches long, grasp it on one end and hold it against your lower lip, and blow horizontally. The loose end of the strip of paper rises, in obedience to Bernoulli's Theorem. You can learn more by varying the strength of the wind you create--blow gently and the paper rises gently, blow forcefully and the paper rises to nearly the horizontal. Simply put, the more speed the wind has, the more lift it produces. Another variation in technique could be adjusting the angle at which the wind meets the paper.

Typical study materials present and describe how air, in meeting a wing, must accelerate as it passes over a wing, an action which produces lift. This subject will not be

repeated here. See Kershner [4]. A more subtle outcome to the production of lift is the generation of *downwash* behind the wing. As depicted in figure 3-1, the airstream is deflected downward as the wing passes through it. It is worth emphasizing that Newton's Third Law predicts this outcome: for every action (the wing is pushed up as lift is generated), there is an equal and opposite reaction (the airstream is pushed down, or downwash is generated). When driving in an automobile at moderate speeds, one can generate downwash by holding a hand out the window at an angle to the ground. At the same time, the hand is forced up by lift. In Langewiesche [5], the author dismisses Bernoulli's work altogether, and concentrates only on the application of Newton's Third Law. In fact, dismissing Bernoulli's analysis can lead to erroneous predictions, but has the advantage of being somewhat more intuitive.

Worth considering is that the generation of lift and other aerodynamic forces is really

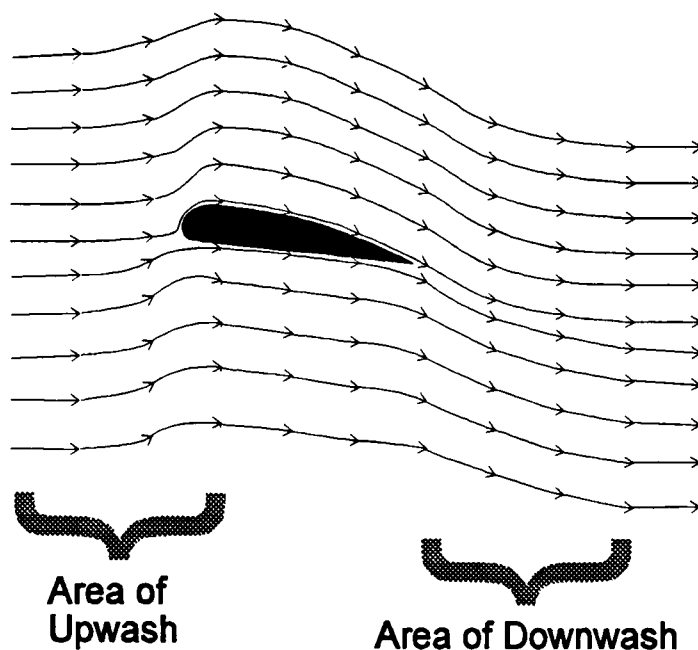


**Figure 3-1**  
Airflow Over Wing Section With Resultant Downwash

a summation of tiny incremental forces along the entire surface. Think of it in terms of “lift per square foot” of surface area. The point is that the lift distribution along a wing is not constant in either the lateral or longitudinal dimension. Indeed, the variation in lift distribution is in part responsible for the generation of wing tip vortices.

To ease and simplify the analysis of aerodynamics, “aerodynamic coefficients” have been defined. These coefficients are defined to be independent of the units of measure. There are many of these coefficients used in the study of airplanes, aerodynamics, and stability and control, and the one used commonly in the study of lift is the *coefficient of lift*, denoted  $C_L$ . As shown in Hurt [3],  $C_L$  is mathematically defined as:

$$C_L = \frac{L/S}{q} \quad (1)$$

where

$L$  = aerodynamic lift, lb

$S$  = wing area,  $\text{ft}^2$

$q$  = dynamic pressure (a measure of speed),  $\text{lb}/\text{ft}^2$ .

The “ $L/S$ ,” with units of  $\text{lb}/\text{ft}^2$ , illustrates the concept of “lift per square foot” introduced above. Accurate determination of  $C_L$  can be made with the help of a wind tunnel. Once known, the amount of lift generated for a specific flight condition for a specific airplane can be calculated.

An important concept is that of *relative wind*. Schematically, relative wind is presented in figure 3-2. The relative wind is defined and described in Langewiesche [5] as:

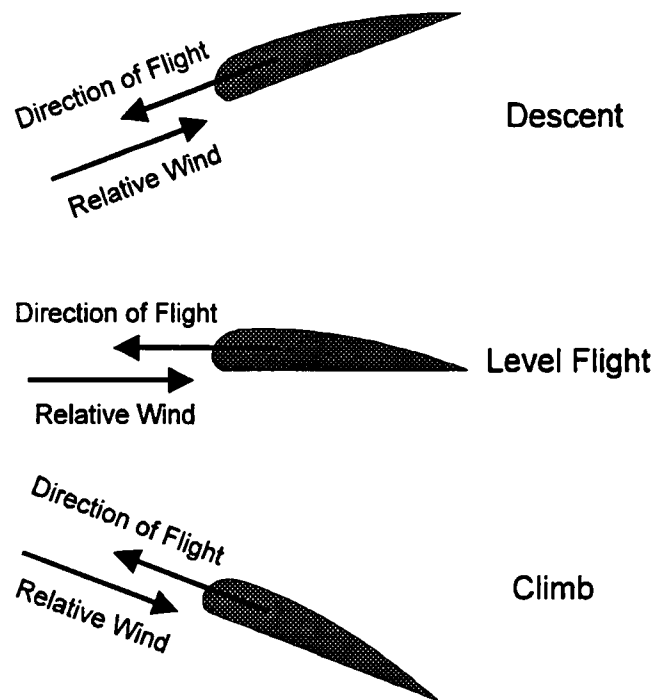


**Figure 3-2**

Diagram Showing the Effect of Flight Path Upon Relative Wind

"...the onrush of air [which] always comes at the airplane from the direction toward which the airplane is moving. If the airplane flies level, the Relative Wind blows back at it level. If the airplane slides sideways through the air in a sideslip to the left, this [relative wind] blows back at the airplane's left side. In a glide, when the airplane moves downward along a sloping flight path, the [relative wind] comes blowing up that slope. In a vertical dive, the [relative wind] blows at the airplane straight up! In a steep climb, when the airplane climbs up a sloping flight path, the [relative wind] blows down the same slope! In a spin, when the airplane is going [straight] down, the relative wind

blows [straight] up; but, since in a spin the airplane goes down twisting, the resultant [relative wind] blows up against it twisting!”

Implicit in this definition is that the relative wind has both a direction and a speed, in other words a velocity (velocity being a vector quantity containing information on both direction and speed). Other terms and concepts important to the understanding of aerodynamics and stability and control are (see figure 3-3, and Hurt [3]):

- ▶ **Angle of Incidence**--The fixed angle at which the wing is attached to the fuselage. This angle is determined by the designer, and incorporated (hopefully with great care) by the builder.
- ▶ **Chord**--The imaginary line connecting the trailing edge and the center of the circle that forms the leading edge of the wing. It is also referred to as the “chord line.”
- ▶ **Angle of Attack**--The angle measured between the chord and the relative wind. It is commonly denoted by “AOA” or the Greek symbol alpha,  $\alpha$ . Do not confuse this fact:  $\alpha$  is not determined by the attitude of the airplane, but by the angular difference

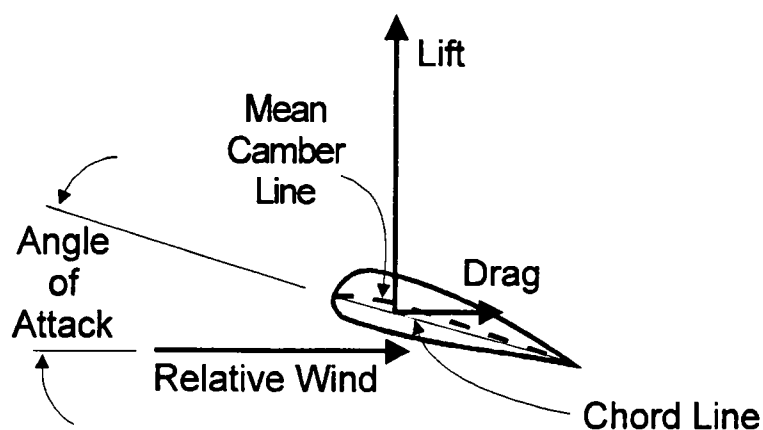


**Figure 3-3**  
Typical Airfoil Section Nomenclature

between the chord and direction the airplane is moving. In Langewiesche [5], the author does not exaggerate in stating "...[the] most important of all flying facts [or knowledge]--angle of attack."  $\alpha_0$  is closely related, and is defined as the angle of attack required for zero lift; this term is pertinent for asymmetric wing sections, such as those typical in most lightplanes. In the United Kingdom,  $\alpha$  is usually referred to as "incidence."

- ▶ **Camber**--A way to describe the curvature of the mean camber line or airfoil shape. The more camber a wing has, the more lift it can produce (all other parameters fixed). The use of ailerons and wing flaps change the camber of the wing, which in turn changes the chord-wise lift distribution. Similarly, the rudder and the elevator change the camber of the vertical and horizontal stabilizers, respectively.
- ▶ **Mean Camber Line**--A line drawn halfway between the upper and lower surfaces of the wing. In a symmetric wing (used in many aerobatic airplanes), the mean camber line and the chord are the same.
- ▶ **Dynamic Pressure**--A frequently used aeronautical quantity which contains information about the velocity of the airstream as well as its density. It is commonly denoted by the symbol  $q$ . Mathematically,  $q = \frac{1}{2} \rho V^2$ , where  $\rho$  is the air density, and  $V$  is the true airspeed.

If one should substitute the definition of  $q$  into equation (1) and solve for  $V$ ,

$$V = \sqrt{\frac{2L/S}{\rho C_L}} \quad (2)$$

where

$\rho$  = air density (at sea level, standard day conditions,  
 $\rho=0.002377$  slugs/ft<sup>3</sup>)

$V$  = true airspeed (ft/sec)

which yields the very important result that in steady level flight, for an airplane's flight speed to change, the  $C_L$  at which the airplane is operating must change, for in steady level flight, lift must equal weight, and the wing area or air density are assumed not to change. How a pilot controls the  $C_L$  will be discussed further in the stability and control sections, however, a typical airplane wing has a characteristic similar to figure 3-4, generally referred to as a "lift

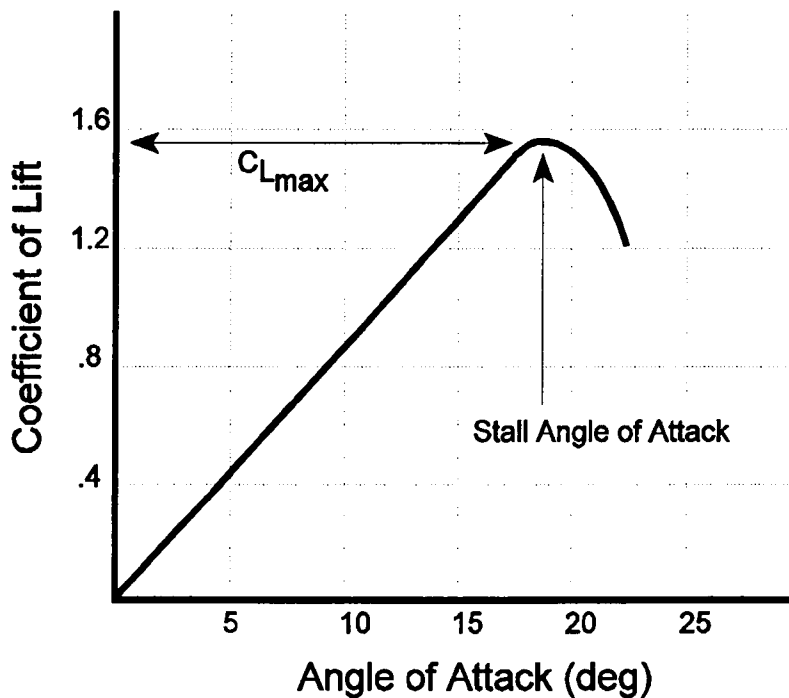


**Figure 3-4**  
Typical Lift Curve

curve.” Several features of this figure require elaboration. In this figure, no lift is produced at zero  $\alpha$ . However, lift may be produced at zero  $\alpha$  in some wings. Also, the curve is linear at the lower  $C_L$  values. This implies that if  $\alpha$  is doubled, so is the lift. Next, there exists an  $\alpha$  above which the curve is non-linear, and in fact  $C_L$  reaches a maximum value. This point is denoted as  $C_{L_{\max}}$ . At  $C_{L_{\max}}$  the wing is said to stall--it is incapable of producing more lift. Lastly, at angles of attack above stalling angle of attack, some lift is produced, albeit at reduced efficiency.

If one substitutes  $C_{L_{\max}}$  into equation (2), then  $V$  becomes  $V_S$ , the stall speed of the airplane:

$$V_S = \sqrt{\frac{2L/S}{\rho C_{L_{\max}}}} \quad (3)$$

where

$$\begin{aligned} V_S &= \text{stall true airspeed (ft/sec)} \\ C_{L_{\max}} &= \text{maximum coefficient of lift.} \end{aligned}$$

For example, if an airplane is in steady  $1g^1$  level flight, then lift equals weight. If the airplane weighs 2000 lb, has a wing area of 120 ft<sup>2</sup>, and a maximum coefficient of lift of 1.2, then its

---

<sup>1</sup> The use of “g” in the description of a turning or curved flight path is actually a misnomer, though commonly used. The ratio of the lift produced by the wing divided by the weight of the airplane is more properly called load factor. The symbol “g” actually represents the acceleration of gravity. Since a turn normally requires a pull force on the yoke or stick, “pulling g’s” has become the vernacular for the condition of operating the airplane at elevated load factors. Using “3g’s” implies operating at a flight condition with a load factor of 3.

stall speed is about 108 ft/sec, or 64 kt. If the airplane is in a steady 2g turn (60° angle of bank), then lift required increases to 4000 lb, and its stall speed increases to 90 kt.

**Drag.** When a body is moved through a fluid, drag is produced. This is true of boats and golf balls, as well as airplanes. Drag is generally divided into two parts (Kershner [4]):

- ▶ Induced drag, which is that drag generated as a direct result of producing lift. That is, the generation of lift “induces” drag.
- ▶ Parasite drag, which is the drag generated by the passage of the body through the air. It also may be referred to by one of its three major sub-components, form drag (e.g. caused by landing gear), interference drag (e.g. caused by airstream interference, such as at a wing root), and skin friction (all over the airplane--not just the wing).

As was the case in lift, it is generally advantageous to discuss drag in aerodynamic coefficient form, in terms of  $C_D$ , the coefficient of total drag:

$$\begin{aligned} C_D &= \frac{D/S}{q} \\ C_D &= C_{D_i} + C_{D_o} \end{aligned} \quad (4)$$

where

$$\begin{aligned} D &= \text{aerodynamic drag, lb} \\ C_{D_i} &= \text{coefficient of induced drag} \\ C_{D_o} &= \text{coefficient of parasite drag.} \end{aligned}$$

A typical total drag characteristic is presented in figure 3-5. Note that the drag increases rapidly with increases in  $\alpha$ . Mentally overlaying figure 3-4, one can surmise that

drag is rising rapidly with increases in  $C_L$ . Usually, the predominant drag at higher AOA's is of the induced variety. Note also that at  $2\frac{1}{2}^\circ \alpha$ , the total drag is only fractionally greater than that at zero  $\alpha$ . These are the AOA's that notionally correspond to level flight maximum cruise speeds. The drag that occurs at zero lift coefficient is the parasitic drag.

The drag characteristic is a strong function of angle of attack. Consider this example, which takes place at a near-constant speed. A 2000 lb airplane is at  $5^\circ \alpha$  (and at steady and level flight), so  $C_D$  is approximately .04. If  $\alpha$  was increased to  $10^\circ$  to perform a 2g turn, the drag coefficient would nearly double to .07, so the airplane would slow somewhat unless thrust was increased to compensate.

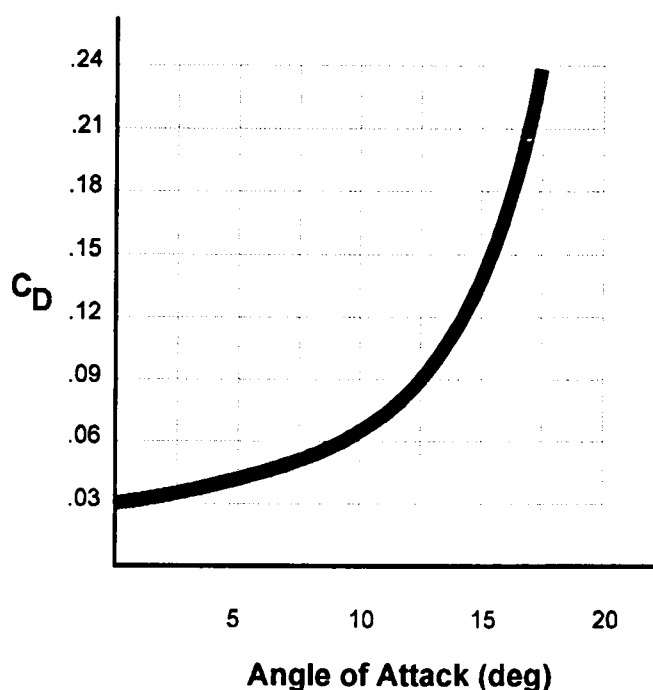


**Figure 3-5**  
Total Drag Coefficient as a Function of Angle of Attack

Weight. This is the easiest of the major forces to understand. It makes sense to rationalize that increased weights would require greater lift, and hence greater  $\alpha$  at a constant

speed, which would in turn generate more drag. It follows, then, that the fourth force, thrust, would need to increase to offset the increase in drag if the same speed was desired. The important point about weight is to remember that it always acts down, where “down” means in the direction of the center of the earth.

**Thrust.** An entire course on engines and propellers would be required to do this subject justice, but for these purposes a much simplified handling is appropriate.

Thrust is a force, measured in pounds. It is possible to define an aerodynamic coefficient of thrust in analogous fashion to  $C_L$ , but for the purposes of the discussion of stability and control in this thesis, this is not necessary. Thrust is generated by a jet engine or by a propeller.

The thrust force usually acts forward (ignoring the braking effect of a fast-turning, flat pitch propeller), e.g. the way the airplane is pointing. It is the only one of the four plane-of-symmetry forces that is “attached” to the airplane’s attitude--lift and drag are oriented with respect to the relative wind, and the direction of gravity varies with airplane attitude.

It is possible, even likely, that the line of action of thrust passes over or under the airplane’s CG. Consider the engine of a Lake amphibian, mounted in a pod well above the fuselage. When the location of the engine and its thrust line differs from the CG to a significant extent, the pilot can expect the airplane to pitch in response to power changes.

Lastly, the physical orientation of the engine with respect to the flight path can change the balance of forces. Consider an airplane in slow, level flight conditions. Suppose the attitude of the airplane is  $12^\circ$  nose-up and that the engine thrust line is parallel with the

longitudinal axis. If the airplane is in level flight, the relative wind and drag act parallel to the horizon, while lift acts perpendicular to the horizon. Note however that the thrust vector is not parallel to the relative wind (or perpendicular to lift), a condition which imparts some component of thrust that acts in the same direction of lift; nearly 21% of the thrust acts in the “lift” direction. Therefore, if the attitude of the airplane differs significantly from the flight path, or if the thrust line of the engine differs from the longitudinal axis, these differences may affect flight characteristics and should be taken into account. (Another thrust effect, “p-factor,” will be discussed in Chapter 8.)

### *Details*

Lift. In level, steady flight, a basic understanding of the force of lift requires no additional information than that provided in the previous section. If one knows the weight, wing area, and the flight condition of the airplane, then the coefficient of lift may be computed.

More advanced considerations in the subject of lift include compressibility effects, wing sweep angle, aspect ratio, variation in lift curve slope, and high lift devices.

*Compressibility effects* are a significant concern as the flight speed begins to approach the speed of sound, perhaps at 0.4 to 0.7 Mach number, depending on the application. (Mach number is defined as the ratio of true air speed to the speed of sound at the current atmospheric condition, Hurt [3].) Since these speeds are a reality for only a small fraction of

the general aviation market, this subject will be ignored. Hurt [3], provides additional information should the reader's interest warrant.

The *wing sweep angle* on almost all general aviation airplanes is essentially 0°. The main reason why this is so is that the structure is easier to build, and that compressibility is not a factor. However, there are several designs, particularly in the homebuilt market that use wing sweep angles of 10-20° in conjunction with canards and strongly tapered wings. These design features have been incorporated to place the vertical stabilizers and rudders at a good location from a lateral-directional stability and control viewpoint.

*Aspect ratio* is defined as:

$$\begin{aligned} AR &= b / \hat{c} \\ AR &= b^2 / S \end{aligned} \quad (5)$$

where

$$\begin{aligned} AR &= \text{aspect ratio} \\ b &= \text{tip to tip wingspan, ft} \\ \hat{c} &= \text{chord of the wing centroid, or mean geometric chord,} \\ &\quad \text{ft.} \end{aligned}$$

The first definition in equation 5 is useful for simply shaped wings, such as rectangular, while the second definition is useful for more complicated shapes.

Aspect ratio is a very obvious feature of an airplane. A sailplane can have an aspect ratio exceeding 30, while the F-104's was about 3.

According to simplified aerodynamic theory, the theoretical maximum lift curve slope (e.g. the slope of the linear part of figure 3-4) for a wing of infinite  $AR$  is  $2\pi$  per radian, or approximately 6.28 per  $57.3^\circ$ , or about  $0.11 C_L$  per degree  $\alpha$ . When considering real wings, the resultant lift curve slope is usually less, however, sailplanes can achieve perhaps 95% or this value. More conventional airplanes might achieve  $\frac{2}{3}$  of this value. Figure 3-6 presents the effect of aspect ratio on lift curve slope.

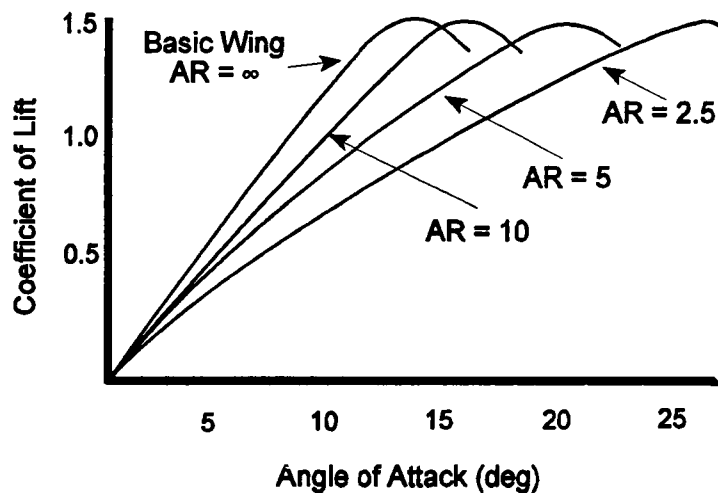


**Figure 3-6**  
Effect of Aspect Ratio for an Unswept Wing on Lift Curve Slope

High lift devices, such as wing flaps, are generally added to a wing to increase lift coefficient for landing, and in military fighters, for maneuvering as well. Drag is increased also, which can be beneficial to both flying qualities and engine acceleration performance. This higher lift coefficient is achieved by moving the lift curve shown in figure 3-4 up and somewhat to the left by increasing the wing camber. The same general lift curve shape is usually retained, just that the  $C_{L_{max}}$  and the zero-AOA lift coefficient is increased as the effective camber is increased; the  $\alpha$  required for  $C_{L_{max}}$  is reduced.

Four general types of wing flaps are common (see Hurt [3]). The plain flap is just a hinged section at the trailing edge area of the wing. The split flap is a hinged section below the trailing edge area of the wing. The slotted flap is a hinged section at the trailing edge, but as the flap is deflected a slot is developed between the flap and the wing. Lastly, a Fowler flap incorporates an extensible trailing edge section that moves aft as it rotates down, effectively increasing the wing area as well as camber. Table 3-1 presents flap information, and figure 3-7 depicts these flap types.

**TABLE 3-1**

Typical Flap Configuration Data for a 25%<sup>1</sup> Chord Flap Deflected 30°

| Flap Type  | $C_{L_{max}}$ | $C_D$ at $C_{L_{max}}$ | Example Application             |
|------------|---------------|------------------------|---------------------------------|
| Basic Wing | 1.5           | .05                    | --                              |
| Plain      | 2.0           | .10                    | Simple light airplane           |
| Split      | 2.2           | .10                    | Typical light twin              |
| Slotted    | 2.6           | .10                    | Single engine STOL <sup>2</sup> |
| Fowler     | 3.0           | .13                    | Full size airliner              |

Source: Hurt, H. H. Jr. *Aerodynamics for Naval Aviators*. U. S. Navy, 1965.

Notes: (1) If the chord of the flap surface alone is ¼ of the chord of the wing, it is termed a "25% flap."  
 (2) Short takeoff and landing

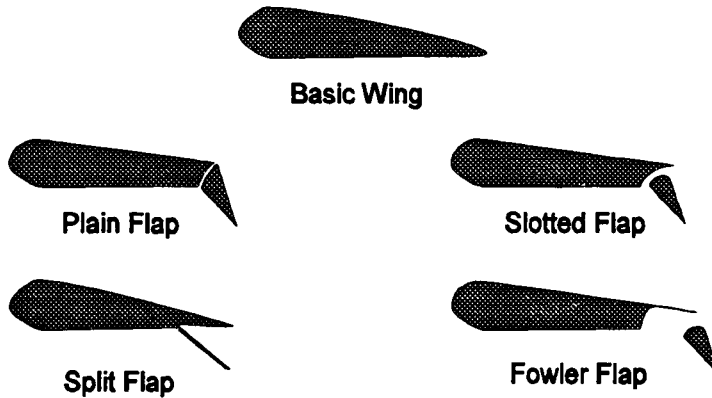


**Figure 3-7**  
Various Flap Configurations

**Drag.** As foreshadowed earlier, the total drag coefficient may be thought of as having both a constant and a variable component. The constant component is the intercept value where the drag curve intercepts the vertical axis, or about .03 in figure 3-5. This is parasitic drag,  $C_{D_0}$ . The remaining term is quadratic in nature, and has been shown to be the lift induced drag (defined as that drag that varies with  $\alpha$ ). Dommasch [6] derived

$$C_{D_i} = \frac{C_L^2}{\pi A R e} \quad (6)$$

where

$e =$  span efficiency, a variable factor which is used to compensate for non-ideal wings.

This is the so-called parabolic drag polar, derivation of which requires calculus and elliptical wing theory. Though this derivation is beyond the scope of this text, the origin of induced

drag can be visualized. Consider that as the atmosphere passes the wing, airstream momentum, the product of a unit of mass times its velocity (a vector quantity), can only be changed by the application of a force. Consider figure 3-8. A unit of air mass, in going by the wing, is effectively acted upon and deflected in the production of downwash. Part A of the figure shows the magnitude and direction of the acting force that must have caused the momentum to change from the initial condition to the downwash condition. Due to Newton's Laws, the reaction force is equal and opposite, as shown in Part B. If one resolves the reaction force into components parallel and perpendicular to the relative wind, the lift and drag components are produced. In fact, the drag component depicted is the induced drag.

The aerodynamics of sailplanes offer some interesting considerations. Since using thermals to gain altitude requires the use of relatively low speeds, usually just above stall (i.e. high  $C_L$ 's), equation (2) illustrates the need to minimize  $L/S$ . For steady and level flight conditions,  $L/S$  is the same as  $W/S$ , a ratio usually referred to as the wing loading of the airplane, measured typically in pounds per square foot. The most straightforward way to

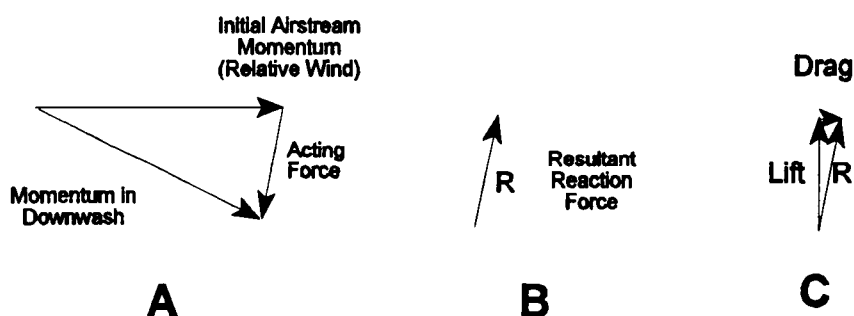


**Figure 3-8**

Vector Analysis of Momentum Change. A) Momentum Change Induced by Wing Presence; B) Resultant Reaction Force; and C) Resolution of Resultant Force into Lift and Drag Components

accomplish this is by using large wing areas. However, operating at these high angles of attack predicts high values of induced drag through the  $C_L^2$  in the numerator of equation (6). The sailplane has no means of thrust, and other performance considerations require minimization of drag. Therefore, the denominator of equation (6) must be maximized in order that induced drag be minimized, requiring the large aspect ratios typically seen in sailplane designs.

By way of example, a designer might wish to build a sailplane that weighs 1200 lb, stalls at 40 kt, and has 200 ft<sup>2</sup> of wing area. Substituting this information into equation (1), a  $C_L$  of approximately 1.1 is required. If the design included an  $AR$  of 30 (assuming  $e=1$ ), the induced drag coefficient at stall would be 0.013, which equates to only about 14 lb of drag.

## CHAPTER 4

### LONGITUDINAL STATIC STABILITY AND CONTROL

This chapter will present static longitudinal stability and control topics. In the *Fundamentals* section, using the aerodynamics presented in the previous chapter, the author will explore the meaning and some theory behind the longitudinally static condition, including moments and equilibrium, the effects of CG shift, desirable flight characteristics, indicators of static longitudinal stability, maneuvering stability, and the meaning of stick-fixed versus stick-free stability. In the *Details* section, pitching moment coefficient, wing, tail, and fuselage contributions to static stability, total airplane static stability, elevator surface float, and trim.

#### *Fundamentals*

Overview. If an airplane has “good” stability and control (S & C) characteristics, then that airplane will empower the pilot to select and command the direction of the flight path with ease and precision, and in a timely fashion (USNTPS [8]). For example, suppose a lightplane pilot found himself 3 miles outside the outer marker, preparing for an Instrument Landing System (ILS) approach. The airplane’s heading is perpendicular to the approach

course, the speed is 25 kt above approach speed, and the landing gear is up. The pilot rolls into a 30° angle of bank (AOB), adds a bit of power, adds a few pounds of aft stick force, selects the landing gear handle down once on final, and begins his descent while referencing localizer and glideslope indicators. This is a very typical task, and one that is performed thousands of times per day. The competent pilot might say it was an easy task to perform.

It *could be* an easy task: if adding the power doesn't cause the nose to pitch excessively, if stick force change and retrimming requirements aren't excessive in the deceleration to approach speed, if the stick force required to establish the required load factor for the 30° AOB turn is not excessive or too light, if the pitch change with landing gear extension is minor, etc. Most pilots have been exposed to airplanes with generally "good" S & C characteristics, these are, after all, a requirement for FAA certification. However, many homebuilts have less than adequate characteristics in some specific area, and even certificated airplanes may have some S & C problems. It is axiomatic that poor S & C characteristics will prevent the pilot from using the full performance capability of his airplane. As an extreme example, suppose a typical aerobatic airplane, capable of  $\pm 9g$ , also required 50 lb of longitudinal stick force to pull each increment of 1g--nobody could fly the airplane to its 9g limit, as it would require 400 lb of stick force (in increase of 8g times 50 lb/g).

Non-maneuvering static stability. Perhaps the reader now has an intuitive feel for S & C. In *static* S & C, equilibrium conditions of flight are examined, e.g. straight and level flight, and steady climbs and descents. These are referred to as "non-maneuvering static" conditions, for which the sum of all aerodynamic forces must be precisely zero. The reader

must not think that because an airplane is climbing that lift is greater than weight, for if lift was greater, the airplane would be curving in its climb, as in the beginning of a loop. In the static sense, an airplane climbs because its flight path is inclined above the horizon. This inclination causes a component of weight to act in the direction and sense of drag; thrust must be increased to compensate if the same equilibrium condition (constant speed) is to be maintained.

In longitudinal static S & C, equilibrium conditions in the plane of symmetry are examined. From the pilot's perspective, this means equilibrium in pitch, or about the lateral axis. It is a quite straightforward concept: if an airplane is to maintain its static condition, then the sum of all forces in the plane of symmetry must be precisely zero. An additional requirement, not previously presented, is that the *sum of all moments* in the plane of symmetry, about a desired point, must also equal zero. Both conditions are required for a static case, that is they are both necessarily true for a static case, and to specify that a static case exists, one must prove that both the sums of forces and moments are zero.

Any person who has taken the FAA Private Pilot Written exam understands much about moments, nevertheless, the concept will be reviewed. A force's moment about a point is defined as the product of that force and a distance, where the distance measured is the perpendicular distance between the force's line of action and the point about which the moment is computed. Referencing figure 4-1, where  $F$  is the force,  $d$  is the perpendicular distance between the line of action of the force and the point, then the moment  $M$  equals the product of  $F$  and  $d$ . Observe that, in this case,  $M$  has a counter-clockwise direction; in longitudinal descriptions, the direction may be referred to as "nose up" or "nose down."

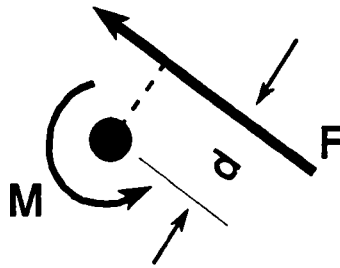


**Figure 4-1**  
Diagram Showing Moment Definition

It would be an omission not to point out that the pilot's perception of S & C characteristics are affected by the pilot himself (is he aggressive, or timid in his inputs?), displays (a properly designed Heads-Up Display can dramatically improve accurate control), the mechanical characteristics of the flight control system as discussed in chapter 2, as well as the basic underlying S & C characteristics.

Equilibrium. In order for an airplane to be in static equilibrium in the plane of symmetry, the following equalities will be true:

$$\begin{aligned} \Sigma Forces &= 0 \\ \Sigma Pitching Moments &= 0 \end{aligned} \quad (1)$$

where

$\Sigma$ , the Greek letter Sigma, means "the sum of."

The analysis of the level flight, static condition now requires a bit more detail than that described in chapter 3. For purposes of summing forces the location of the forces is irrelevant, but this is not true for the moments these forces generate (refer to figure 4-2). In particular, we have ignored the subject of where lift forces act. There is a great deal of

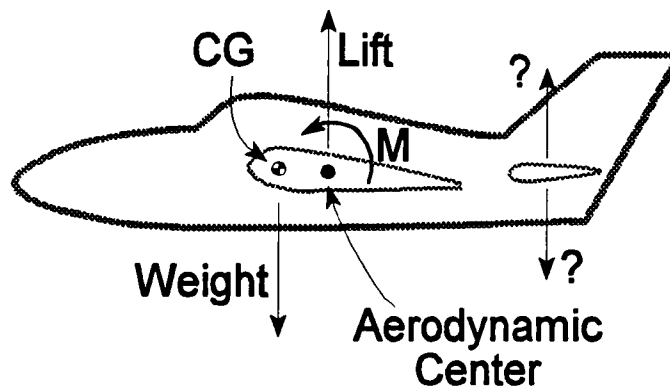


**Figure 4-2**  
Sum of Forces for Longitudinal Statics Case

science behind the answer to this question, but the result is simple. Lift forces act through the *center of pressure* (or c.p.). The c.p. may be considered to be the “center of gravity” of the lift distribution, and is a function of camber and lift coefficient (Hurt [3]). More useful is a point known as the *aerodynamic center* (or a.c.); for typical wings in subsonic flight, this point is 25% aft from the leading edge on the mean chord (see Hurt [3]). The a.c. is typically more useful because while the location of the c.p. moves around with camber and lift coefficient changes, the a.c. does not. (The laws of mechanics permit the replacement of the force acting through the c.p. by a force located at the a.c., and the moment of the original force about the a.c.) In a positively cambered wing (as in figure 3-3), the a.c. is forward of the c.p.; observe that there is no lift-generated moment about the c.p. (this is where the force acts, so  $d=0$ ), but there is a nose-down pitching moment,  $M$ , generated about the a.c.

Consider for a moment the case when the airplane’s CG is aft of the a.c., and assume a straight and level initial condition. Since the a.c. is forward of the CG, lift production generates a nose-up pitching moment about the CG (in equilibrium, this would be offset by

M and tail moments). An incremental increase in pitch attitude, caused by a gust perhaps, would generate added  $\alpha$  and lift, and add nose-up pitching moment. Said another way, at this CG location the wing's angle of attack characteristic is unstable.

When the condition is as depicted in figure 4-2, with the CG forward of the a.c., an incremental increase in  $\alpha$  would generate more wing lift, generating a force couple which would tend to pitch the nose down, in the direction of restoring the initial conditions, or in the direction required for positive angle of attack static stability. In this case, tail "lift" must act down (in the direction of weight for straight and level flight conditions) in order that pitching moments sum to zero. Consider figure 4-3. If the airplane weighs 1600 lb, and the CG is 5 in ahead of the a.c., equations (1) require

$$\begin{aligned}\Sigma Forces &= 0 \Rightarrow W + T = L \\ \Sigma Moments &= 0 \Rightarrow L \times d = T \times D\end{aligned}\quad (2)$$

where

$$\begin{aligned}W &= \text{weight (lb)} \\ T &= \text{tail lift (lb)} \\ d &= \text{distance between the CG and a.c. (in)} \\ D &= \text{distance between CG and the tail (in)}.\end{aligned}$$

These two equations may be solved, since only  $T$  and  $L$  are unknowns:

$$\begin{aligned}1600 + T &= L \\ L \times 5 &= T \times 144 \\ \Rightarrow T &\approx 58 \text{ lb} \\ \Rightarrow L &\approx 1658 \text{ lb}\end{aligned}\quad (3)$$

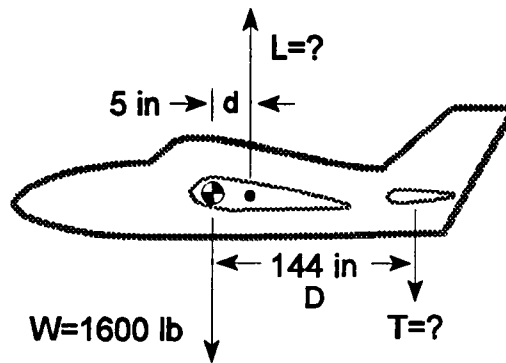


**Figure 4-3**  
Problem in Distribution of Wing and Tail Lift

Most lightplanes are strongly statically stable. What this means to a pilot is that, when disturbed from an equilibrium attitude or  $\alpha$ , the airplane wants to return to the initial equilibrium condition. If the nose is pulled up and released, it pitches down. If a gust pitches the nose down, the airplane naturally pitches back up toward the initial conditions. Some homebuilt airplanes do not display such trustworthy behavior. For example, as originally designed and flown for several years, the Lancair 320 displayed unstable longitudinal characteristics in the landing configuration, as noted in Lindorff [9], where the author, a test pilot, stated:

“...the [longitudinal static stability] in the Land configuration at an approximate trim speed of 75 kias [knots indicated airspeed] was negative...At airspeeds away from the trim speed of 75 kias, releasing the stick produced an aperiodic divergence. The longitudinal static stability characteristics were demonstrated to be poor under the best conditions and would not provide an adequate indication to the pilot of even a substantial deviation from trim airspeed.”

Effect of CG variation. The CG in most light airplanes is referenced to an arbitrary datum (instead of the aircraft CG itself, which moves as a function of the weight distribution), typically the leading tip of the propeller spinner, or the firewall of the engine bay. Aircraft designers utilize the concept of CG placement relative to the leading edge of the mean aerodynamic chord (MAC). However CG placement is computed, the effect is always that an allowable range of CG travel is defined. Since the longitudinal placement of the CG varies with weight distribution, including passengers, cargo, and fuel, forward and aft CG limits are always of interest. In many airplanes the change of weight and CG location due to fuel burn is also significant, and a special fuel transfer strategy may be required to maintain the CG within limits.

Airplanes fly differently at different CG locations. Consider figure 4-4, where only the major forces are shown for simplicity. In part A of the figure, the airplane is strongly stable as indicated by the small radius of curvature of the cup containing the ball. This condition is accompanied by a relatively far forward placement of the CG. If the CG is placed forward, additional wing lift is required to overcome the additional downward tail lift generated. Both wing and tail lift increases also increase induced drag. In the solution to figure 4-3 (equations 2 and 3), if the CG was 10 in forward of the a.c. instead of 5 in, the tail load would increase from 58 to 115 lb. This would increase the lift required for the static condition to 1715 lb. The forward limit may also be defined by ground operation concerns. The CG must be maintained between the forward and aft wheels to prevent tipping when standing motionless, and must be further constrained if tipping during braking is to be

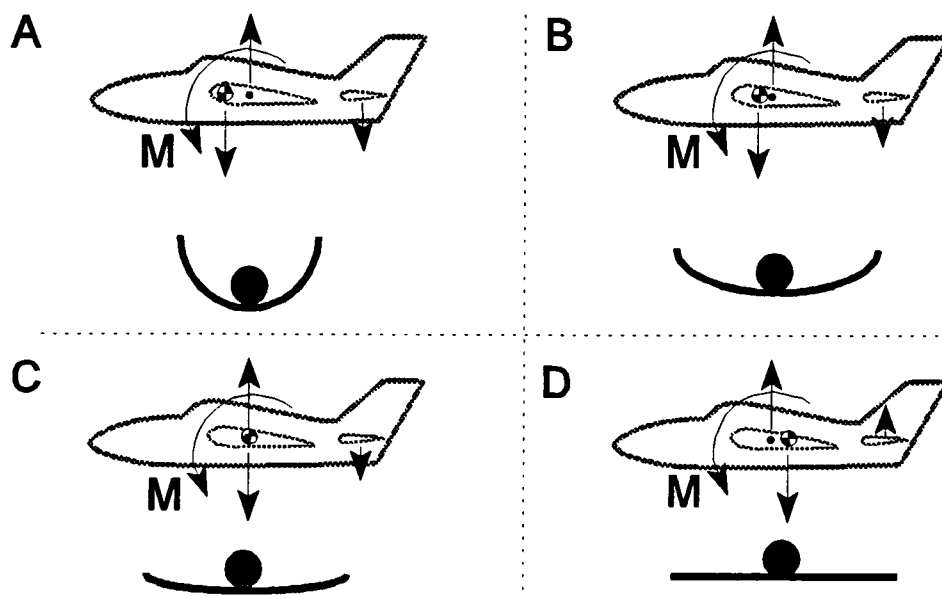


**Figure 4-4**

Diagram Showing the Effects of CG Shift on Static Stability (tail force exaggerated for clarity). A) Strongly stable, B) Weak stability, C) No wing contribution, weaker stability, and D) At static neutral point, wing contribution destabilizing, neutral static stability, typical tail force required for equilibrium.

controlled. Lastly, if the CG is too far forward, raising the nose for takeoff or flaring to land can be difficult or impossible.

In part B of figure 4-4, the CG has shifted aft somewhat. As a direct result, tail down load for equilibrium decreases, and the wing's contribution towards longitudinal static stability is reduced, but still positive. When a gust increases the angle of attack, the airplane still responds by pitching down, back toward equilibrium.

In part C, the CG overlies the wing's a.c. The only tail force required for equilibrium is that required to compensate for the wing's pitching moment,  $M$ . If  $\alpha$  is increased by a gust

(controls fixed), the airplane will tend to recover to the initial condition, but only due to  $M$  and a change in tail load--the wing is not helping to stabilize the airplane.

If the CG continued to move aft as in part D, a condition would have been reached where no pitching moment back towards the initial condition would occur in response to an  $\alpha$  increase. The wing's moment contribution in this case is nose-up (the CG is aft of the a.c.), and just equals the wing's (that is,  $M$ ) and the tail's nose-down pitching moment. At this point, the airplane displays neutral static stability, and the CG is said to be located at the *static neutral point*. This is sometimes referred to as the airplane's a.c. Further aft travel of the CG would result in an unstable angle of attack characteristic.

It can be seen that, with all other variables held constant, CG shift changes the tail lift requirements. Therefore, to achieve zero pitching moments over a range of CG's, the position of the elevator must change.

Indicators of static stability. Since aircraft pitching moments can not be measured in free flight conditions, it is necessary to rely on indirect methods to indicate stability. Typical evidence of non-maneuvering static stability is presented in figure 4-5. If a pilot trims carefully to 100 KIAS, and then (without retrimming or adjusting power) increases or decreases airspeed by about 10-15%, he could measure a force characteristic similar to the dark, solid line for a statically stable airplane. As the plane slows down, the pilot is required to hold ever-increasing pull forces to maintain a speed slower than trim. Conversely, if he accelerates (without retrimming!), he will be required to hold ever-increasing push forces. The dark grey line in figure 4-5 represents a decrease in, but still positive, longitudinal static

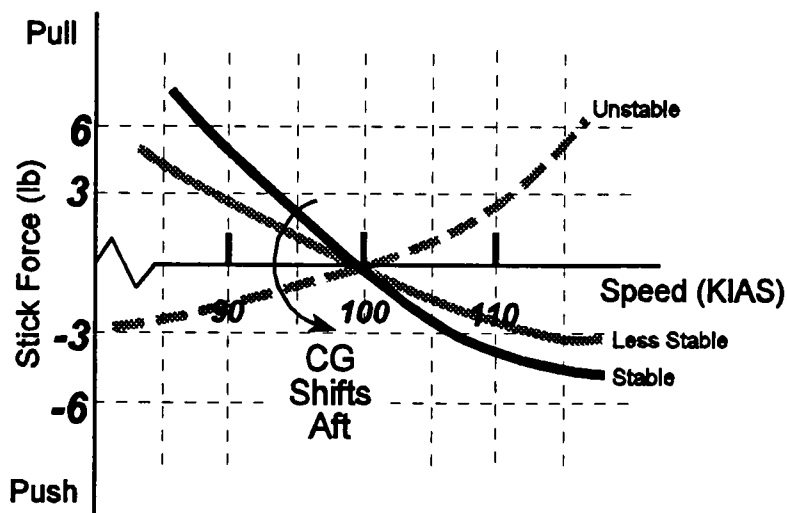


**Figure 4-5**

Typical Non-maneuvering Longitudinal Static Stability Characteristic, Including Effects as CG Shifts Aft

stability. This characteristic might be different from the solid line due exclusively to a CG shift aft. As the CG shifts farther aft still, the plot might overlay the airspeed axis of figure 4-5. In this case, the CG would be coincident with the static neutral point. (As will be discussed later, since this plot is of stick forces versus airspeed, this CG position would more precisely be defined as the stick-free static neutral point.) The absence of force change with speed change is usually considered to define a neutrally stable angle of attack response. Lastly, as the CG shifts behind the neutral point, the slope of the dashed grey curve becomes positive in figure 4-5, which is associated with negative static stability (or is an unstable characteristic). The pilot of this airplane might pull back a bit to begin a deceleration, but as the speed decayed, he would have to add stick forward of trim to stabilize. It might be controllable, as long as the pilot could pay much greater attention to the control task, as compared with a stable airplane. The Wright Flyer in 1907 was unstable, and the X-29 today

is dramatically unstable. In 1907, flights were short, and often ended in a premature landing or crash, and in the X-29, triply redundant digital computers keep the angle of attack under control. The reason for the great emphasis placed on CG control in light airplanes should now be apparent. If the CG should travel too far aft, unstable angle of attack characteristics may be generated, significantly complicating control of the airplane.

Maneuvering stability. Thus far the analysis and description has been about non-maneuvering longitudinal static stability. One may also consider a different kind of steady flight condition. If one placed an airplane in  $45^\circ$  angle of bank, and applied the required load factor to maintain level flight, another kind of stability may be discussed, maneuvering stability. The issue in maneuvering stability is observing the airplane's initial response when displaced from a steady maneuvering path.

In maneuvering stability, it is customary to discuss stick force vs load factor. An airplane with positive maneuvering stability typically requires higher stick forces for higher

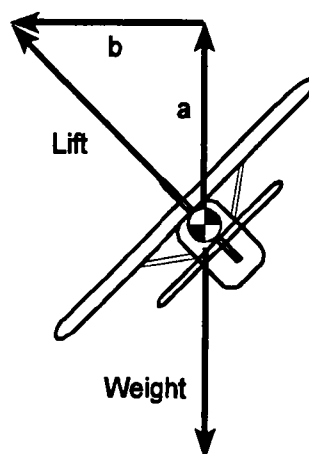


**Figure 4-6**  
Applied Forces in a Steady Turn

load factor. If an airplane required the pilot to pull 5 lb for a steady 2g turn, required 10 lb for 3g, and required only 11 lb for a load factor of 4, then that airplane would be very susceptible to being overstressed by the pilot. Referring to figure 4-6, the vertical component of lift,  $a$ , must be equal to weight if the airplane is in level flight. The unbalanced force,  $b$ , is what turns the airplane. The ratio of Lift/ $a$  is actually the load factor (or  $g$ ).

The effect of CG shift is quite similar to the previous discussion, as well. As the CG travels aft, stick force requirement for increasing  $g$  would decrease. There exists a position where the CG may lie aft of the static neutral point, where no increment in stick position would be required to stabilize at an increased load factor. This point is referred to as the *maneuvering neutral point*.

Stick-fixed and stick-free stability. A new concept is that of *stick-fixed* vice *stick-free* static stability. All stability characteristics discussed thus far have been the stick-fixed variety, that is when the control position is held fixed against any forces developed. Stick-fixed stability is also referred to as control position stability. If the elevator is subjected to an angle of attack change, and allowed to float freely (disregarding any airplane motion), the elevator would tend to float in the direction required to align with the new relative wind. This is called the stick-free characteristic, and any time the the elevator is released, the stability resulting can be somewhat less and the slope of any stability curves could be slightly less negative for stable airplanes, or more positive for unstable airplanes. (In stabilator-equipped airplanes, the horizontal tail might actually float in a way that it can increase stability.) Refer to figure 4-7. Stick-free stability is also referred to as control force stability.

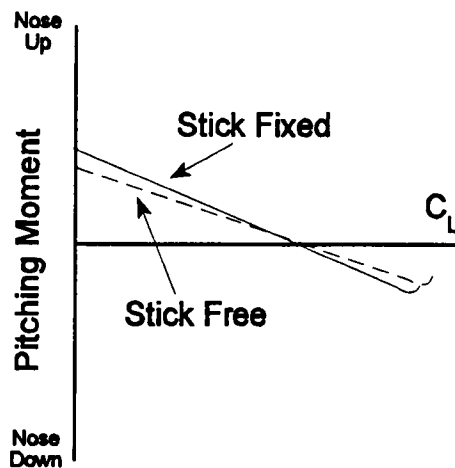


**Figure 4-7**  
Diagram Depicting Effect of Allowing the Elevator to Float for a Non-Maneuvering Statically Stable Airplane

In the case of maneuvering static stability, stick-fixed and stick-free characteristics are manifested through the control position change per load factor increment (stick-fixed) and the control force change per load factor increment (stick-free).

Other Airframe Effects. Two other pitching moment effects should be mentioned. The fuselage has a generally small destabilizing effect. Also, the location of the wing affects whether drag forces act above or below the CG, which would change pitching moment characteristics somewhat. A high wing airplane is slightly more statically stable than a low wing airplane, especially at high AOA's (Hurt [3], and Perkins and Hage [17]). See *Details* in this chapter for more information.

Thrust Effects. As mentioned in chapter 3, there can be thrust effects upon the pitching moment characteristic of an airplane. The most common effect is that of propwash. If an approach to a landing is being made at slow speed with some power on, and then the throttle is closed, the propwash over the tail decreases. If the tail force had been exerted downward, the reduction of airstream velocity (propwash) over the tail would decrease the tail down force, causing the nose to drop. Similarly, if the airplane was trimmed for a power off condition, adding power can cause the nose to rise.

The nose would also tend to rise at constant trim for two other reasons after a power addition, but only one of these is really caused by the power increase, that being the tendency of the propwash to accelerate with increased throttle which results in still greater airstream velocity over the tail. The other effect is really caused by the speed change after the power increase, as discussed in connection with figure 4-5.

The second thrust effect has been previously introduced in chapter 3, the effect of pitching moment change caused by the thrust line of the engine not overlying the CG. Consider again the Lake amphibian, where the thrust line is several feet above the CG. For the sake of discussion, assume the distance is 3 ft, and a modest thrust change of 100 lb was made. (It can be shown that  $P_{TH} = T \times V$ , where  $P_{TH}$  is the thrust power,  $T$  is the thrust, and  $V$  is the TAS in ft/sec. Converting to knots, a change of 100 lb of thrust at 100 kt would be a horsepower change of approximately 30 hp. See Kershner [7] for an excellent discussion.) A moment change of 300 ft-lb or 3600 in-lb would result. This is quite a significant pitching moment. For example, a CG change in a statically stable 3600 lb airplane of 1 in forward would generate approximately the same moment.

## Details

**Pitching Moment Coefficient.** A pitching moment coefficient,  $C_M$ , is developed and defined in a similar fashion to previous coefficients (USNTPS [8]):

$$C_M = \frac{M}{q S \bar{c}} \quad (4)$$

where

- $M$  = moment about a desired point, ft-lb
- $q$  = dynamic pressure, lb/ft<sup>2</sup>
- $S$  = wing area, ft<sup>2</sup>
- $\bar{c}$  = wing mean aerodynamic chord, ft.

In contrast to earlier-presented coefficients, this one uses an additional  $\bar{c}$  in the denominator, in order that the dimensionless nature of the coefficient be maintained. For the typical, statically stable lightplane,  $M$  is opposite in sense to an increase in  $\alpha$ . Also, since  $\alpha$  and  $C_L$  are directly related,  $M$  would also be opposite to an increase in  $C_L$ . These relationships, between  $C_M$  (or  $M$ ),  $\alpha$ , and  $C_L$ , form the mathematical foundation for longitudinal stability and control.

The most important contributions of individual components to the total airplane pitching moment can be expressed:

$$C_{M_{CG}}(Total) = C_{M_{CG}}(Wing) + C_{M_{CG}}(Fuselage) + C_{M_{CG}}(Tail) + C_{M_{CG}}(Power) \quad (5)$$

where the subscript “CG” indicates moments computed about the CG. The major power effects have been adequately discussed, and will not be covered further.

Wing Contribution. Examining the first term of equation (5), it can be shown that for a subsonic airplane operating at relatively small AOA’s (USNTPS [8]),

$$C_{M_{CG}}(Wing) = C_{M_{a.c.}} + \frac{\Delta C_M}{\Delta C_L} \times C_{L_W} \quad (6)$$

where

$C_{M_{a.c.}}$  = coefficient of pitching moment about the a.c.

$\frac{\Delta C_M}{\Delta C_L}$  = rate of change of wing pitching moment coefficient  
with changes in wing lift coefficient

$C_{L_W}$  = coefficient of wing lift.

Examining these wing contribution terms,  $C_{M_{a.c.}}$  is relatively constant, except at high AOA’s. As presented earlier, for a positively cambered airfoil,  $C_{M_{a.c.}}$  is in the nosedown direction. The main reason why referencing pitching moments to the a.c. (instead of the c.p.) is so useful is that the pitching moment about the a.c. is constant over a large range of AOAs or  $C_L$ s. Plots of this characteristic are presented in figure 4-8, including the effects of the three possible CG positions.

For the wing,  $\frac{\Delta C_M}{\Delta C_L}$  can be seen in figure 4-8 to be a function of CG placement relative to the a.c. Once a CG location is established, this characteristic is well established, and in fact, linear (ignoring  $C_L$  approaching stall).

$C_{L_{\alpha}}$  is also typically linear with respect to  $\alpha$  at low and moderate angles of attack. In figure 3-4, this linear relationship is maintained until quite close to stall  $C_L$ .

In summary, the wing contribution to the airplane total pitching moment is described by linear relations between  $\alpha$ ,  $C_M$ , and  $C_L$ . The location of the CG with respect to the a.c. is the determining factor whether the wing contribution is destabilizing, neutral, or stabilizing.

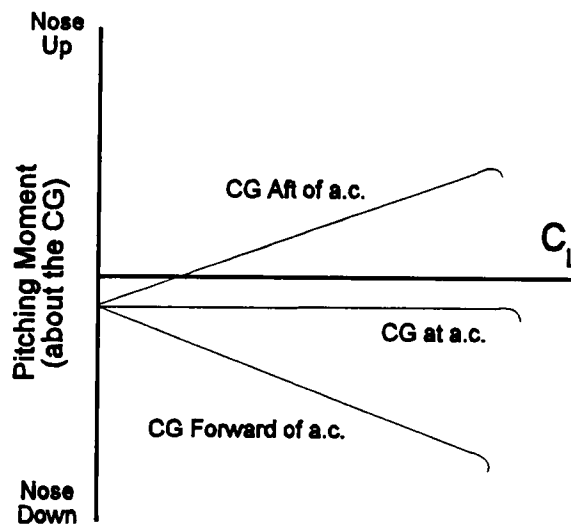


**Figure 4-8**  
Characteristic Wing Contribution to Pitching Moment,  
Including Variation Due to CG

Tail Contribution to Longitudinal Stability. The analysis of the tail contribution toward longitudinal static stability is similar, but somewhat more complicated. The longitudinal stability term is (there are other terms):

$$C_{M_{CG}}(Tail) = [-\frac{a_t}{a_w}(1-\frac{\Delta \epsilon}{\Delta \alpha})\eta_t \bar{V}]C_{L_w} \quad (7)$$

where

$$\begin{aligned} a_{t,w} &= \text{lift curve slope of the tail and wing, respectively} \\ \frac{\Delta \epsilon}{\Delta \alpha} &= \text{rate of change of downwash angle, } \epsilon, \text{ with variation of } \alpha \\ \eta_t &= \text{tail efficiency factor, equal to the ratio of } q \text{ at the tail over } q \text{ of the free airstream ahead of the aircraft} \\ \bar{V} &= \text{tail volume coefficient, equal to the ratio of tail area to wing area times the ratio of the distance between the CG and the a.c. of the tail to the mean aerodynamic chord: } \frac{S_t l_t}{S \bar{c}} \end{aligned}$$

This is a relatively simple part of the total tail contribution equation, but contains some important information. First, the greater the lift curve slope of the tail, the more pitching moment can be generated per increment of AOA change. This seems sensible, in that the more lift the tail can produce, the more moment can be generated. Breaking apart the tail volume coefficient, it makes sense that the greater the ratio of tail area to wing area, the greater the stabilizing influence. It is also intuitively correct that the greater the distance from the CG to the tail, the greater the influence of the tail is.

Not contained in equation (7) is the term that describes tail control. It also contains two factors discussed in the previous paragraph:  $a_t$ ,  $\bar{V}$ . Just as increases in these factors

introduce greater forces and therefore greater stability, they also prescribe greater control of pitching moment. Figure 4-9 presents a representative plot of the tail contribution to stability for a statically stable airplane, and depicts the dependence of the tail pitching moment contribution on elevator position.

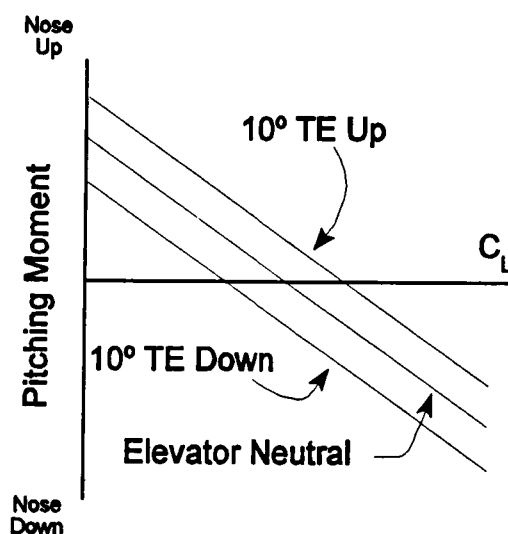


**Figure 4-9**  
Tail Contribution to Pitching Moment, with Variations in  
Elevator Deflection

Fuselage Contribution. The contribution of the fuselage towards total airplane pitching moment is generally somewhat destabilizing. No net lifting forces are usually generated in light planes. See Hurt [3] and USNTPS [8] for further information.

The Total Airplane. The total airplane, for the typical wing a.c. and CG placement, with representative components added to form the total characteristic for a single elevator position, is presented in figure 4-10. The total pitching moment characteristic for several

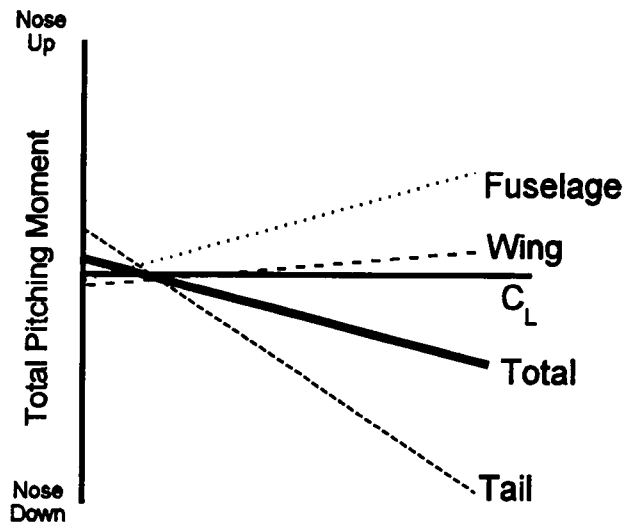


**Figure 4-10**

Total Airplane Pitching Moment, Composed of Wing, Tail and Fuselage Contributions

elevator positions is presented in figure 4-11. This latter figure requires elaboration. If the airplane was operated at point A, the elevator position is  $10^\circ$  and the airplane is at equilibrium at some  $C_L$ . If the elevator was adjusted  $10^\circ$  in the nose up (TEU) direction, equilibrium could then be reestablished only at point B, commanding a new coefficient of lift (or alternatively, a new  $\alpha$ ). This figure illustrates the fundamental truth that elevator position commands angle of attack. In the longitudinal sense, a pilot flies angle of attack.

Maneuvering Stability. The control positions required for steady turning flight may be obtained in flight; typical relationships are presented in figure 4-12. In part A, the typical relationship between load factor and the elevator change required to achieve that load factor is plotted. At  $1g$ , of course, no change is necessary, but as the loading increases, so does aft stick and TEU elevator position. Part B of the figure shows the effect of a CG shift. As the CG shifts aft, less and less control displacement per increment in load factor is required, and

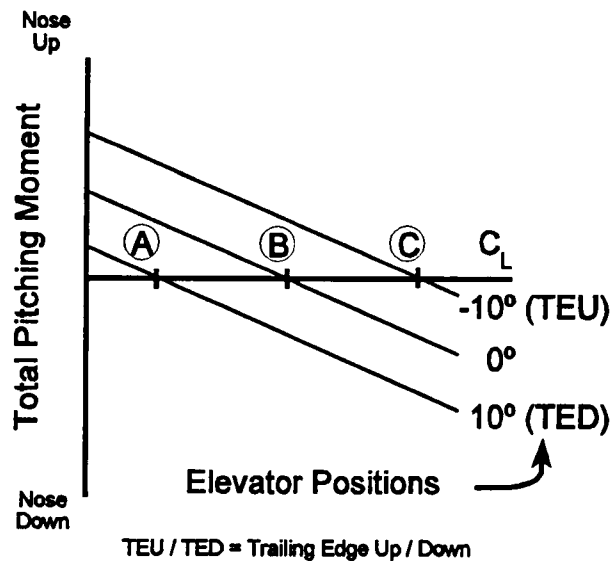


**Figure 4-11**  
Total Airplane Pitching Moment at Several Elevator Positions

when the CG is at the maneuver neutral point no displacement is required to sustain elevated load factors. In part C of the figure, the effect of increased speed is apparent, as is the effect of increased altitude in part D.

Part E shows the effect of the two basic types of turning flight. The steady pull is best described as the maneuver performed upon entry to a loop, i.e. a constant heading is maintained, and all motion resides in the plane of symmetry. The steady turn is a turn at constant altitude, while the heading varies so that some motion is occurring out of the plane of symmetry. The airframe's pitch response is slightly different in these two cases, in that *pitch rate* (the rate of motion, deg/sec, about the lateral or pitch axis) for a given load factor is greater in the level turn. (This fact may not be obvious, however it can be shown that pitch rate,  $q = g(n_z - 1)/V$  for a steady pull, and  $q = g(n_z - 1/n_z)/V$  for a steady turn. In these equations,  $g$  is the gravitational constant, and  $n_z$  is the load factor.) Lastly, part F presents the difference in elevator position required when engine gyroscopic effects or torque are a factor, as is

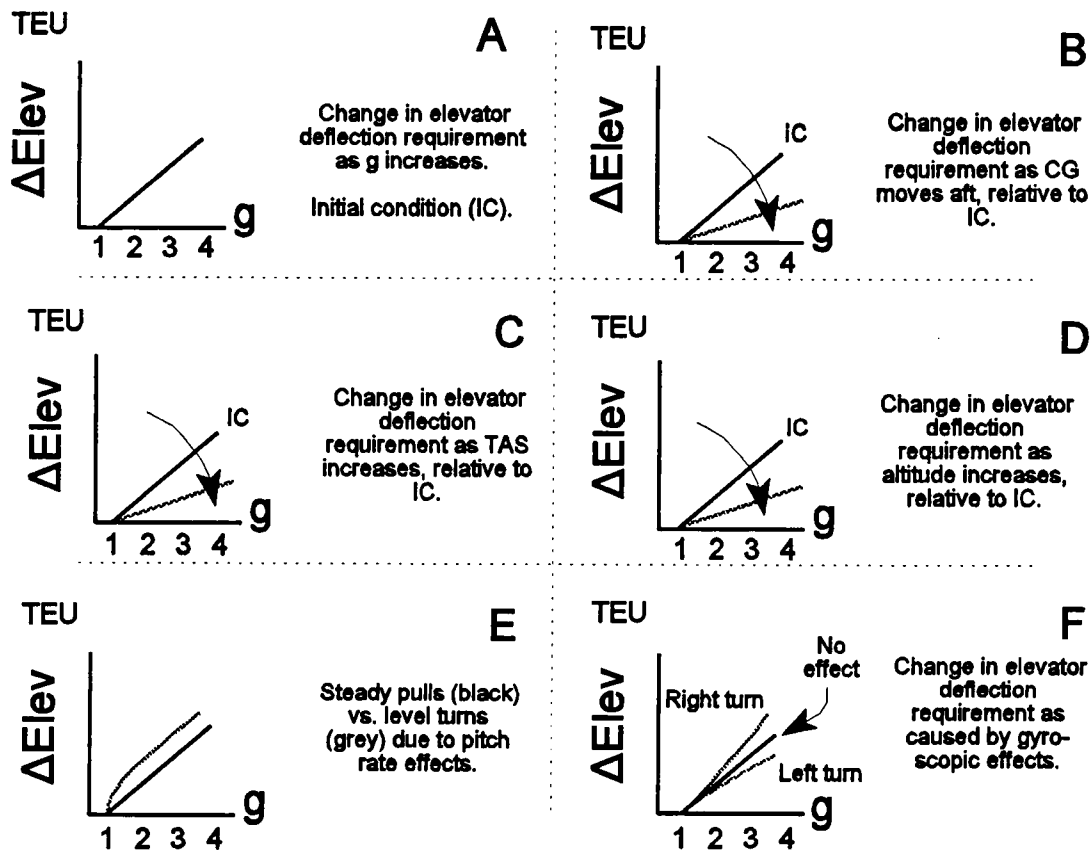


**Figure 4-12**

Maneuvering Stability Characteristics. A) Initial condition, B) CG shift effect, C) Airspeed effect, D) Altitude effect, E) Steady vs. level turns, and F) Gyroscopic effects

common in most lightplanes. This effect is greatest at low speeds and high *yaw rate* (the rate of motion, deg/sec, about the vertical or yaw axis). The depicted gyroscopic effect is based on a typical US-engine direction of rotation, clockwise when viewed from the rear.

Pitch Damping. The total pitching moment of an airplane with a pitch rate is effectively reduced by a term known as pitch damping. If one took a model airplane, drilled a hole big enough to insert a pencil laterally through the fuselage at the CG, and spun the airplane on the pencil, it would stop rotating even if the pencil interface was without friction.

Air flow impinging on the horizontal tail provides a damping force on it, which decreases any pitching moment applied. In the case of the model airplane, once the model is rotating, no pitching moment is applied, so the pitch rate slows to a stop. The equation describing pitch rate damping is

$$C_{M_q} = -2a_t \eta_t \bar{V} \frac{\dot{\theta}}{c} \quad (8)$$

where

$$C_{M_q} = \text{coefficient of pitch rate damping.}$$

The minus sign indicates the damping nature of this coefficient, while the factors of tail lift curve slope, tail efficiency factor, tail volume coefficient, and aft fuselage length (distance between the CG and the horizontal tail) all make sense as being contributory to the damping of pitch rate. Reconsidering figure 4-12 part A, the single plot shown may be thought of as being the sum of two lines with lesser slope, one for the basic maneuvering stability of the airplane, and one required to overcome pitch rate damping.

Surface Float and Trim. Consider the force on the elevator surface due to the trim tab deflection as presented in figure 4-13. Assume the flight controls are released. It is intuitive that the deflection of the tab generates a moment on the elevator surface, attempting to deflect it counterclockwise, or opposite to the tab's deflection. In fact, each degree of tab deflection generates a certain amount of hinge moment on the elevator. As the elevator deflects, however, the tab's moment decreases until the elevator comes to a rest; in a sense,

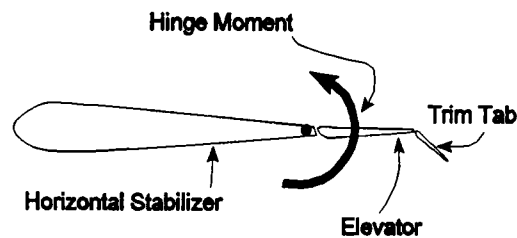


**Figure 4-13**  
Generation of Hinge Moments Via Deflection of a Trim Tab

the elevator has *floated* to this new position, and the angle that it moved is called the float angle. If the airplane is static in this condition with zero stick force, it is said to be *trimmed*. As discussed in connection with figure 4-11, it should be apparent that one actually trims to an angle of attack (or  $C_L$ ), vice a speed.

For example, suppose an airplane is to be trimmed to level flight at the conclusion of a climb. The pilot establishes the angle of attack and attitude for 1g level flight at climb speed. As the airplane accelerates to cruise speed, the angle of attack required for 1g decreases. While holding the airplane at this new AOA by pushing forward, forces are trimmed out by moving the trim tab in the direction required to cause the elevator to float to the position where the controls can actually be released. If the forces are trimmed out while the airplane is still accelerating, however, more trimming will be required as the AOA continues to change. Only when the speed is constant can the AOA (or  $C_L$ , or elevator position) required for level flight be constant. In the non-maneuvering case, if power and airspeed are constant, then forces should easily be reduced to zero; the airplane may still climb or descend if the attitude is not adjusted for level flight. Similarly, a pilot can also trim to the AOA required for a 2g turn, and perform a “hands-off” 60° AOB turn.

The total force felt through the control system is made up of two parts. The stability component is a function of the stability requirement to balance pitching moments using the elevator. The tab component of force is a function of the tab position, and is itself a function of the dynamic pressure.

Effects of CG shift on stick force are presented in figure 4-14; for example, if the same speed is trimmed for all CG's, subsequent airspeed variation would result in force characteristics similar to those depicted.

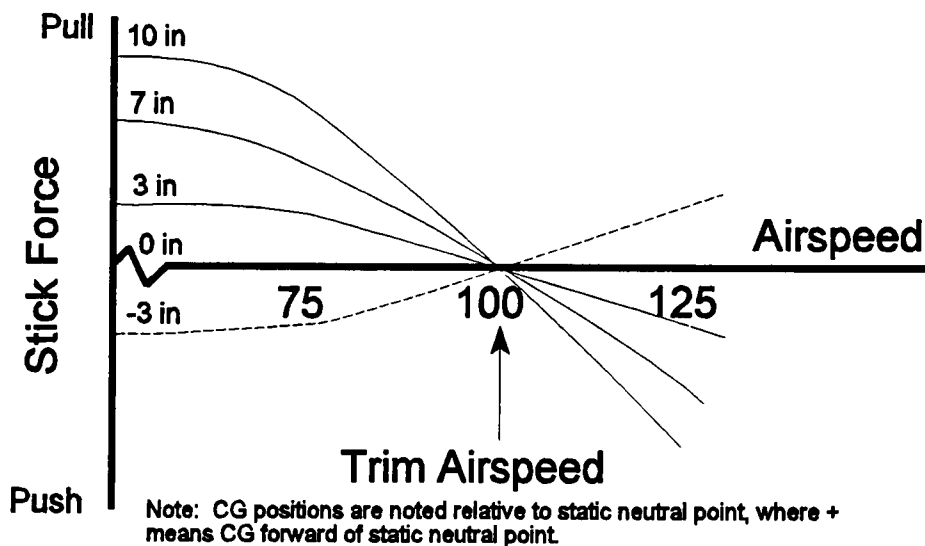


**Figure 4-14**

Diagram Depicting Effect of CG Shift on Stick Force at a Single Trim Airspeed

## CHAPTER 5

### LATERAL-DIRECTIONAL STATIC STABILITY AND CONTROL

In this chapter, the reader is introduced to the concepts of lateral-directional static stability and control. In the *Fundamentals* section, pertinent aerodynamic factors are presented and discussed, as well as steady heading sideslips and crosswind landings. In the *Details* section the balance of forces and moments, additional discussion about a few of the aerodynamic factors, roll control, and conventional landing gear (“taildragger”) lateral-directional flying qualities during landings are presented.

#### *Fundamentals*

Overview. The subject of lateral-directional stability and control, and associated handling qualities of the airplane, is usually a new subject area for most lightplane pilots. The subject of the “four forces” discussed in chapters 3 and 4 was an expansion of a previously learned subject, and was concerned with forces and moments in the plane of symmetry. In lateral-directional stability and control, or just “lat-dir” for short, statics and dynamics in the remaining two planes are presented, the planes containing the lateral and directional axes (see figure 5-1). Hurt [3], and USNTPS [8] contain amplifying information.

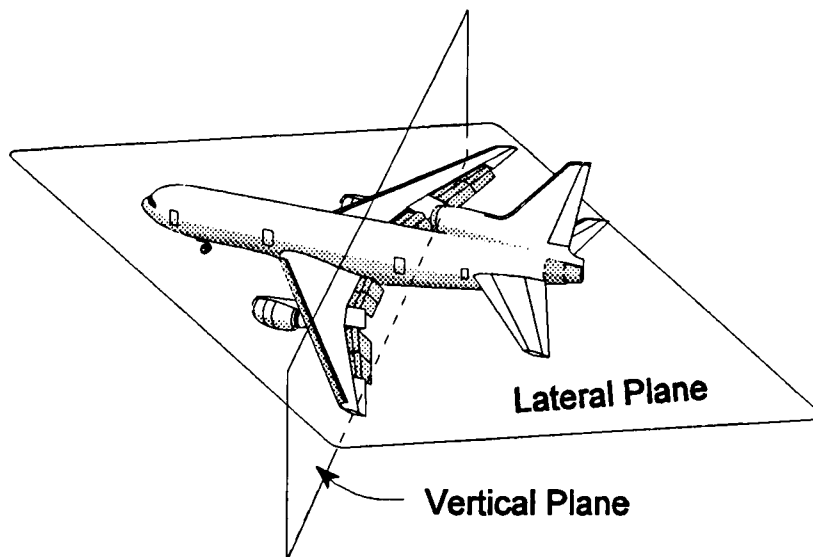


**Figure 5-1**  
Depiction of Lateral and Directional Planes

The reason for the differentiation between these two cases may be of interest. The mathematics that fully describe forces, moments, and motion are contained in a set of many simultaneous equations of motion. In their most general form, these equations are difficult to solve. It turns out, however, that if a number of simplifications are introduced, the resulting mathematics is much more workable. One outcome to these simplifications is that three major sets of equations are developed. One set describes activity only in the plane of symmetry, and is typically presented as a stand alone subject, longitudinal stability and control. The remaining two sets describe activity in the remaining two planes, both of which contain "cross coupling" terms, where activity in one plane is tied directly to an activity occurring simultaneously in the second plane. Therefore, it is not possible to deal with lateral and directional S&C separately, and these are typically presented together. As a result of

containing the subject matter regarding two planes of motion, lat-dir can be somewhat lengthy and cumbersome.

Helping to nullify the complexity is the consideration that longitudinal S&C is of primary importance from a pilot's point of view. For example, an airplane with excellent longitudinal flying qualities and weak lateral-directional flying qualities is typically useful, though possibly challenging to fly. An airplane with excellent lat-dir but poor longitudinal flying qualities may have little utility for any purpose.

At this point, the reader must be thinking, "But what are lat-dir handling qualities?" Lateral-directional static S&C is centered around the airplane's response to *sideslip*, or  $\beta$ , where  $\beta$  can be thought of as the directional "angle of attack" of the fuselage and vertical tail (refer to figure 5-2). Any pilot who has performed wing-down, top-rudder approaches for a crosswind landing already knows something about this! Lat-dir dynamics include the nature of the airplane's response when returning to trim, which includes *dutch roll*. Any person who has flown a V-tail Beechcraft Bonanza knows something about dutch roll. Another component of dynamics is termed *spiral stability*, which is the airplane's response when left

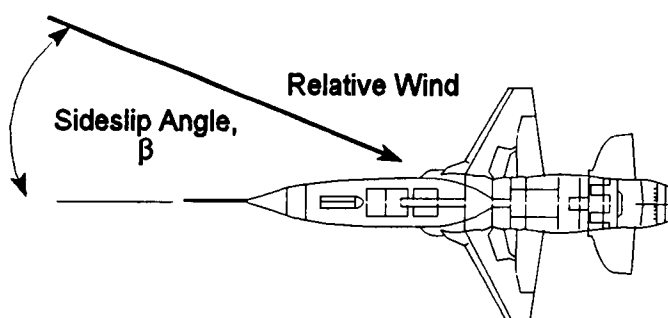


**Figure 5-2**  
Definition of Sideslip Angle, or  $\beta$

unattended at a small bank angle. These two dynamic modes will be presented in chapter 7. Another part of lat-dir is the response of the airplane to the lateral control, typically ailerons. Though rolling is really a dynamic mode of motion (it involves non-steady motion), the author will include an introduction to this characteristic in this chapter.

Lateral-directional static stability. As in the longitudinal case, the lat-dir static situation requires that

$$\begin{array}{lcl} \Sigma \text{ Forces} & = & 0 \\ \Sigma \text{ Moments} & = & 0 \end{array} \quad (1)$$

and due to this analysis covering moments and forces in two planes, there are numerous individual elements. Of utmost importance in the understanding of lat-dir is that all these factors may produce forces in the lateral or vertical planes, or moments about the longitudinal or vertical axes; the reader is encouraged to visualize the forces and the placement of these forces which result in moments. The requirement for the static situation will dictate that the sum of the side forces (forces in the lateral plane), the sum of the yawing moments (i.e. about the vertical axis), and the sum of the rolling moments (about the longitudinal axis) must be precisely zero. The most important of the lateral-directional terms are now presented.

### Sideslip Terms (refer to figure 5-3)

- The side force that is generated by sideslip. It can be seen in the figure that when the airplane is subjected to  $\beta$ , some aerodynamic force is generated laterally, after all, the fuselage and the vertical tail are aerodynamic lifting surfaces which would generate lateral "lift" as indicated in the figure by  $Y_o$  and  $Y_t$ . For reasons which will be obvious soon, the lateral force is considered to be made of two parts, that of the airframe and that of the vertical tail. The sum of these forces would cause the airplane to be "lifted" or shifted to its left. It is sensible that if the sideslip angle was zero, that side forces would be zero (neglecting any airframe asymmetries).
- The yawing moment that is generated by sideslip. It is straightforward to consider that the sideforce generated by the vertical tail alone acts through a moment arm of  $l_t$ , causing a yawing moment that attempts to bring the nose of the airplane into the relative wind. Called *directional stability* by the engineer, this is the familiar "weathercocking" tendency of most airplanes. As in the case with angle of attack,

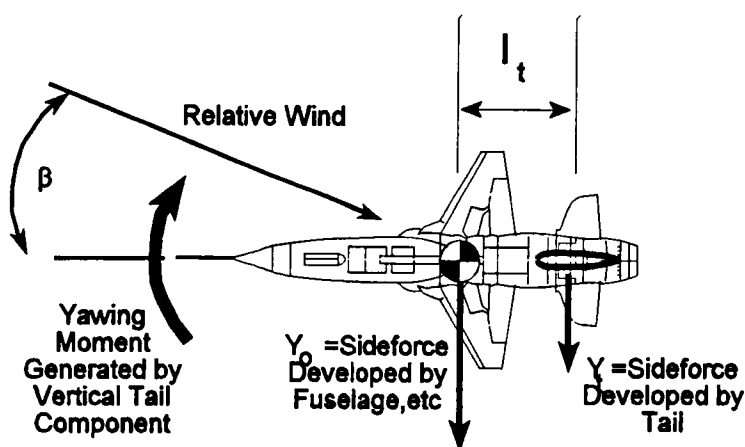


**Figure 5-3**  
Effect of Sideslip

there is a sideslip at which no greater  $Y_t$  can be achieved. This could be caused by vertical fin stall. Other causes are possible, however, if no greater yawing moment could be achieved for an increase in  $\beta$ , then directional stability would be lost, and the airplane could depart controlled flight. This should obviously not occur in ordinary flight conditions. Good directional stability is fundamental to a pilot's opinion about the flying qualities of an airplane; if directional stability is poor, invariably the airplane will require considerable or continuous compensation.

- The rolling moment that is due to sideslip, also known as *dihedral effect* and *lateral static stability*. One demonstrates this characteristic most easily by depressing a rudder pedal to generate  $\beta$ , and observing that the typical airplane rolls, usually in the same direction as the rudder is depressed. This characteristic is fairly important to the pilot, but the required levels of dihedral effect vary with the purpose and configuration of the airplane. If excessive dihedral effect is present, the task of crosswind takeoff and landing could be more difficult. As will be seen, excessive dihedral effect can generate undesirable rolling characteristics as well. High wing airplanes typically have fairly strong dihedral effect naturally, however, low wing airplanes achieve positive dihedral effect by using *geometric dihedral*, or just *dihedral*. These terms describe the attachment of the wings to the fuselage at an angle, such that the wing tip is above the wing root. To counter excessive dihedral in high wing tactical airplanes, such as the AV-8B Harrier, negative dihedral (also called *anhedral*), is used. In this case, the wing tips are below the wing root. Any airplane will likely display different lateral static stability effects at different places in the flight envelope.

### Effects of rudder deflection (refer to figure 5-4)

- The side force that is developed by a rudder deflection. Figure 5-4 illustrates the effect on total airplane sideforce by increasing the camber of the vertical tail through the deflection of the rudder. If one steps on the right rudder, then the turn-and-slip ball would shift to the left, an effect caused by the curving of the flight path to the right. More sideforce can be developed by increasing the rudder deflection, or by increasing the size of the rudder.

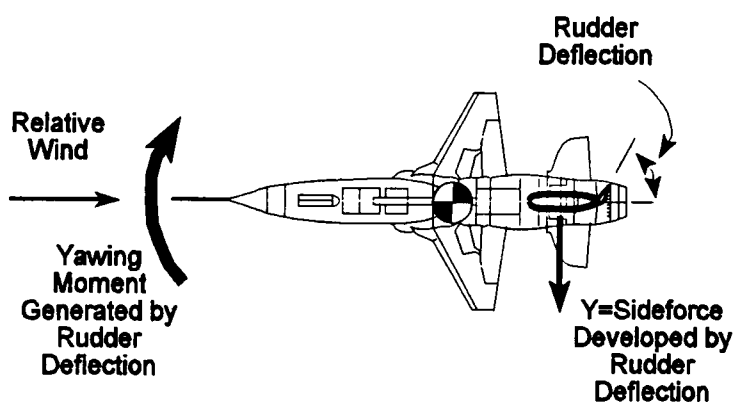


**Figure 5-4**  
Effect of Rudder Deflection

- The yawing moment that is due to rudder deflection. Due to the generation of side force described above, and through the moment arm distance to the airplane CG, a yawing moment is generated. The greater the angle that the rudder is deflected, the more yawing moment can be achieved. Similarly, a bigger rudder would also generate more yawing moment. Lastly, a greater distance to the airplane CG would generate

greater yawing moment, so forward CG positions may show slightly greater yawing moment response to rudder deflection than aft CG positions.

- The rolling moment that is due to rudder deflection. Refer to figure 5-5. Due to the relative vertical distance between the airplane CG and the centroid of the lateral “lift” of the vertical tail, a rolling moment opposite to rudder deflection may be generated, i.e. right rudder would cause left roll if the rudder centroid was above the CG. Clearly, the height of the vertical tail in relation to the rest of the airplane is the determining factor. Note that this rolling moment is opposite in direction to that generated from the sideslip, so it might be easy to confuse this characteristic with negative dihedral effect. The difference is that the “lift” and subsequent rudder-induced roll is generated almost instantly, and that dihedral effect rolling takes place as sideslip develops, over the span of a few seconds, perhaps.

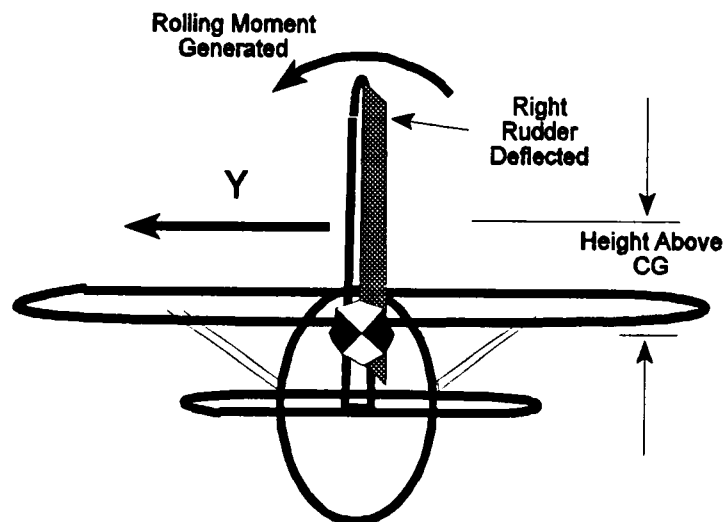


**Figure 5-5**  
Diagram Depicting Rolling Moment Generation Due to Rudder Deflection

Effects of lateral control deflection (refer to figure 5-6)

- The yawing moment that results from a lateral control deflection. This is normally referred to as *adverse yaw*, though more properly it should be referred to generically as *aileron yaw*. As can be seen in the figure, the down-going aileron causes increased camber, increasing lift and increased induced drag on that portion of the wing. Similarly, the up-going aileron effectively decreases camber, lift, and induced drag. The changes in induced drag at both ailerons generate a force couple that tends to yaw the airplane opposite to the applied lateral control. This is typical for aileron-equipped airplanes, and when the yaw moment is opposite the lateral input, then this characteristic may be properly referred to as adverse yaw. (In spoiler equipped airplanes, including some sailplanes and the Mitsubishi MU-2, it can be rationalized that the up-going spoiler may cause additional drag as well as loss of lift, causing yaw into the lateral deflection, or *proverse yaw*.) Since the ailerons are typically as far as

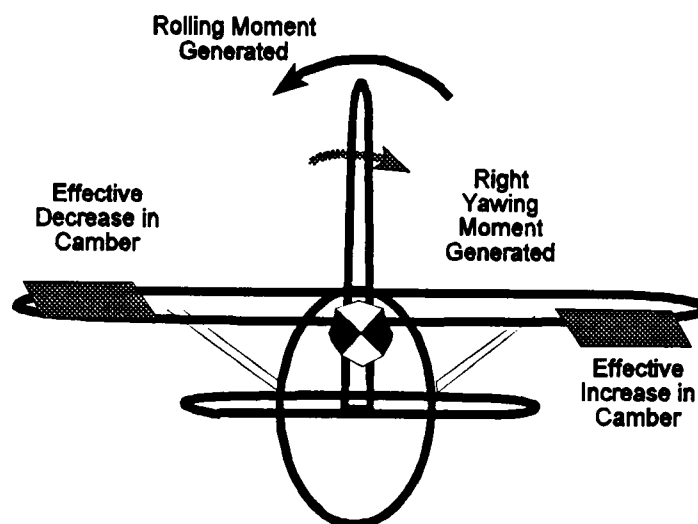


**Figure 5-6**  
Diagram Depicting Typical Effect of Deflected Ailerons

possible from the centerline of the airplane, enough moment is applied to yaw the airplane. It forces the pilot to add rudder when making a lateral input, such as rolling into or out of turns. Adverse yaw seems generally more powerful when the wing is operated at high  $C_L$ , such as approaching stall or with wing flaps extended. Most airplane designers help to decrease the effects of adverse yaw by using an aileron gearing scheme that deflects the down-going aileron less than the simultaneously deployed up-going aileron. The easiest way to observe the effects of adverse yaw is to roll briskly into a turn, and to aggressively capture (i.e. using large lateral control deflections) a desired heading while keeping the feet off the rudders. The typical airplane will overshoot the heading, and begin an oscillation that makes it very difficult to settle on the desired heading.

- The rolling moment that is generated by a unit of lateral control deflection. Since excessive aileron deflections could cause stall in a localized area of the wing, aileron deflections are limited, usually to about  $25^\circ$ . Therefore, ailerons must be sized both in the span and chordwise directions to achieve the desired maximum rolling moments and (therefore) maximum roll rate (in degrees per second). The SU-29, the Russian-built champion aerobatic airplane, has ailerons that are very nearly the same as the span of each wing. One might consider increasing the span of the wing to increase the moment that the aileron might provide, but this would likely prove ineffective (see the section Roll Control in this chapter).

The steady heading sideslip. There is a steady maneuver which can provide significant information about the most important of these characteristics. This maneuver, termed a *steady heading sideslip (SHSS)*, achieves a static condition in bank angle, side force, and sideslip. To perform this maneuver (in this case, to the right), right sideslip is generated by depressing the left rudder pedal. In order that the rolling moments be brought to zero, right aileron is applied to stop the roll rate. In this condition, the side forces generated due to  $\beta$ , rudder deflection, and bank angle cancel out (when the angle of bank is not zero, there is a component of the airplane's weight that acts laterally--right angle of bank, right side force). The rolling moments from dihedral effect, rudder deflection, and lateral control deflection cancel out. Lastly, the yawing moments generated due to  $\beta$ , aileron yaw, and rudder deflection cancel out. Though the maneuver does not actually illustrate the level of directional or lateral stability, it does confirm that positive stabilities exist. An example of a typical set of SHSS data for a lateral-directionally stable airplane is presented in figure 5-7.

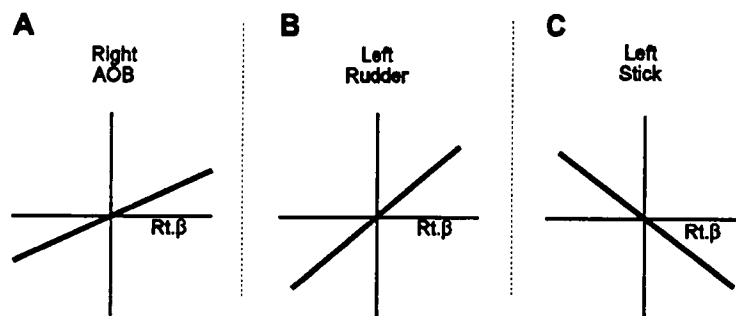


**Figure 5-7**

Plot of Results of Performing a Steady Heading Sideslip to the Right. Part A illustrates the side force balance, part B the apparent directional stability, and part C the apparent dihedral effect (lateral stability).

The crosswind landing. An interesting case of static lateral-directional stability and control is presented when performing a crosswind landing. Airplanes generally fall into one

of two categories in crosswind landings, those where a wing-down top-rudder technique is recommended, and those where a crabbed landing approach is recommended.

Analysis of the crabbed approach follows. The pilot adjusts the heading of the airplane to achieve a desired ground track, while superimposing the ground track on the runway centerline. The only issue in controllability is that, upon touchdown, side forces are generated on the wheels which may damage the landing gear or sharply yaw the airplane. Some operating manuals prescribe a "kick-out" of the crab with rudder just prior to landing. If the crab angle is small, these techniques work well. Most sailplanes use a crabbed approach, since with the large wing span, even small bank angles could drag the wing tip.

The wing-down, top rudder approach is somewhat more interesting. In practice the pilot uses bank angle to control lateral position on the runway, and rudder position to align the nose of the airplane with the runway (and the desired direction of motion). For an analysis, assume a steady state condition with a crosswind component from the right. Some angle of bank to the right is required to stop lateral drift, and some right sideslip,  $\beta$ , is generated with left rudder to align the nose of the airplane with the runway. Since the airplane is in a right sideslip, the flight path through the airmass is  $\beta$  degrees to the right of the airplanes nose, and if this is where the relative wind is coming from, a steady landing can be performed. Often more interesting and challenging is what happens immediately after the landing. Once all three wheels are on the ground, the crosswind is exerting a side force to the left, and weathercocking will cause many airplanes to tend to yaw to the right. Left rudder deflection helps to counteract both of these effects, by generating a side force to the right, and a left yawing moment.

## Details

In longitudinal S&C, the sum of forces along the vertical and longitudinal axes are analyzed, as well as the moments about the lateral axis (pitching moments). The remaining forces and moments are analyzed in lat-dir S&C: forces in the lateral plane, and moments about the longitudinal and vertical axes (rolling and pitching moments, respectively). Re-stated in mathematical terms, and referencing figure 5-8:

$$\begin{aligned}\Sigma Y_{aero} + W \sin \phi &= 0 \\ \Sigma L &= 0 \\ \Sigma N &= 0\end{aligned}\tag{2}$$

where

$$\begin{aligned}Y_{aero} &= \text{aerodynamic forces in the lateral plane} \\ W &= \text{weight of the airplane} \\ \phi &= \text{bank angle} \\ L &= \text{rolling moments} \\ N &= \text{yawing moments.}\end{aligned}$$

The  $Y_{aero}$  forces are those described by side force generated by sideslip, side force generated by rudder deflection, and the side force caused by a component of the weight acting in the lateral plane due to a bank angle,  $\phi$ .

A basic description of pertinent terms was provided in the *Fundamentals* section, and additional discussion about some of these is now provided to enhance understanding.

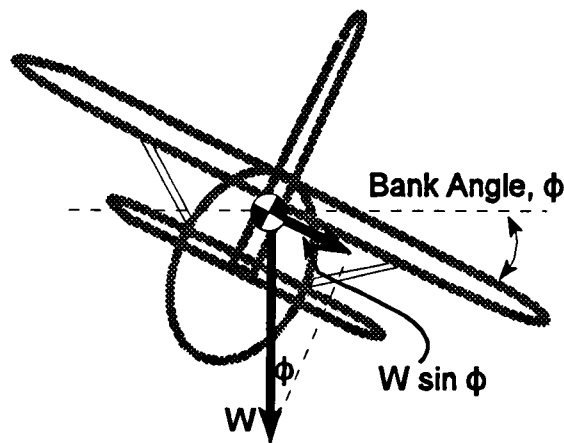


**Figure 5-8**  
Diagram Showing Weight Component Acting in the Lateral Plane

Yawing moment due to sideslip. This parameter describes the yawing moment of the airplane with an incremental change in sideslip. Also referred to as directional stability, this is one of the most important characteristics of an airplane. As a heuristic proof to this statement, consider the flight of an arrow without tailfeathers; the arrow tumbles end over end through the air, the same thing that would happen to an airplane without a vertical tail. Similar to angle of attack stability, one can consider the total airplane directional stability to be composed of two components, the vertical tail component, and everything else (the tail-off characteristic). The tail-off contribution is generally quite unstable, as the arrow analogy illustrated. Obviously then, the tail contribution must be quite powerfully directionally stabilizing.

Focusing on the tail-off characteristic, two generalities may be stated. First, the fuselage aft of the CG should be stabilizing, while the fuselage forward is not. Some World

War II fighters have quite large noses with huge radial engines, a configuration which is quite destabilizing. Also, a straight wing has little effect on directional stability.

The basic definition of directional stability should be considered as well. If an airplane is at equilibrium at zero sideslip, and an incremental increase in sideslip should occur, is the tendency of the airplane: (a) to return towards the previous sideslip (positive directional stability)?; or (b) to remain at the new sideslip (neutral directional stability)?, or (c) to diverge away from the original sideslip (unstable directionally)? Note the similarity to angle of attack stability, in that replacing “sideslip” with “angle of attack” in the previous sentence defines AOA stability. Sideslip is nothing else but angle of attack in the lateral plane. Consider an aerobatic airplane, perhaps a Pitts Special, doing a knife edge pass at the beginning of an airshow. The airplane is flown at a steady  $90^\circ$  AOB with about  $15\text{-}20^\circ$  sideslip, and for an airplane in this condition, the side force generated (by both the tail-off and the vertical tail contributions) opposes weight and can truly be considered lift.

High angles of attack usually adversely affect directional stability. If, as angle of attack increases, the vertical tail is increasingly blanked out or subjected to turbulent airflow, the restoring moment it normally provides is reduced, as illustrated in figure 5-9. This happens in many airplanes, especially those with lower aspect ratios (see Hurt [3]). Airplanes are subjected to reduced dynamic pressures approaching the 1g stall, which also has a deleterious effect on directional stability.

In another similarity to the longitudinal case, the directional stability of an airplane is greater with the rudder held fixed rather than allowed to float. With the rudder fixed, the rudder becomes “part of” the vertical stabilizer, effectively increasing its size and the area aft

Airflow at Low  
Angle of Attack



Turbulent  
Airflow at  
Higher Angle of  
Attack

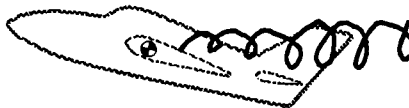


**Figure 5-9**

Depiction of Disturbed Airflow Over the Vertical Stabilizer at Low and High Angles of Attack

of the CG. If the rudder is allowed to float, the pressure distribution around the vertical tail changes in a way that reduces the ability of the vertical tail to generate lateral forces, which contribute directly to directional stability; in other words, the effective size of the vertical stabilizer is reduced.

The effect of propwash on directional stability is the same as that previously described for the elevator. If the power is set high, high velocity propwash tends to increase the effectiveness of the vertical tail (and rudder), while reduced local velocities will reduce it.

Yawing moment due to rudder deflection . This is the directional control parameter, describing how much yawing moment is produced by an incremental change in rudder deflection. In normal flight conditions, increasing deflection produces increasing yawing moment in a generally predictable fashion. However, some airplanes exhibit an undesirable characteristic called *rudder lock* if the rudder is free to float at high sideslips. If the float

angle becomes as large as the required deflection of the rudder to achieve that sideslip, no force will be required to hold the airplane at that sideslip (see figure 5-10). Moreover, increasing sideslip further will result in float angles up to the limit of rudder deflection. In order to return the airplane to normal flight conditions, opposite rudder force will be required. In a mild case, this characteristic can be little more than a nuisance, but in a more severe case, aircraft departure or overstress can occur. At a minimum, vertical tail size is effectively decreased, degrading directional stability. The apparent directional stability should be strong enough to require reasonable force to displace the airplane from balanced, zero sideslip flight. If the force is inadequate, the pilot will have a difficult time attaining balanced trimmed flight conditions. For example, some float planes, with the addition of significant surface area (floats) ahead of the CG, display weak directional stability, and are difficult to trim.

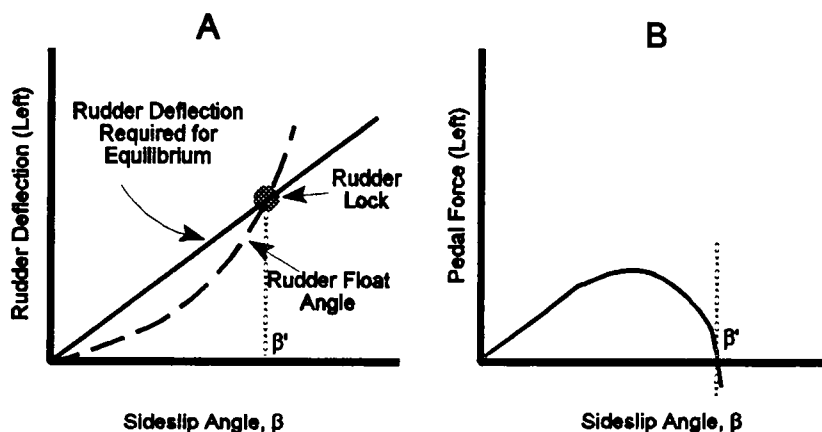


**Figure 5-10**  
Effect of Rudder Lock. A) Deflection characteristic, and B) Force Characteristic

An airplane must have sufficient rudder control for all anticipated flight conditions. Typically, the most demanding of the following five conditions will dictate the minimum rudder control requirements (Hurt [3]):

- ▶ Adverse yaw conditions achieved in high  $C_L$  flight during aggressive lateral control inputs may require large rudder deflections.
- ▶ Spin recovery typically requires strong rudder control, since sideslip is typically a driving force in out-of-control flight conditions.
- ▶ A propeller-powered airplane may require large rudder applications at high power settings in low speed flight due to the rotational nature of the propwash.
- ▶ Available rudder deflection may prescribe the maximum crosswind capability of the airplane, since as previously explained, large sideslips may be required to maintain steady flight.
- ▶ A multi-engine airplane (assuming wing-mounted engines) requires sufficient yawing moment generation capability to counteract the effects of losing power on an engine. The most demanding conditions are typically immediately after takeoff with full power applied, with the failed engine not feathered. Chapter 8 will discuss asymmetric power considerations in greater detail.

*CAUTION: Whenever steady heading or intentional sideslips are performed in an airplane whose flying qualities are not well known, a slow, progressive increase in sideslip is important. Discontinue sideslip evaluation if rudder forces diminish with increasing rudder deflection or sideslip.*

Roll Control. The asymmetric deflection of the lateral control surfaces creates a differential lift situation, where one wing has increased lift (on the down-going aileron side)

and one has decreased lift (on the up-going aileron side). If the stick or yoke is moved left, the right aileron goes down, the left aileron goes up, and the airplane rolls left. The differential lift generates a rolling moment, which causes a roll acceleration, where acceleration is the measure of the rate of change of roll rate, or  $\text{deg/sec}^2$ . The acceleration will continue until an opposing moment is generated, and once this moment balances the aileron-induced rolling moment, roll acceleration decreases to zero and a steady roll rate results (i.e. 90  $\text{deg/sec}$ ). The balancing moment is called *roll damping*, and its chief cause is the wing itself. Consider trying to “sweep” the wing through a pool of water in a motion similar to rolling into an angle of bank. Clearly, enormous forces would be required to perform this feat; certainly the viscosity of air is much less than that of water, but the effect is identical. Moving the wing through air in this manner tends to damp its motion, hence the expression roll damping. The simplified roll equation is

$$\dot{p} - L_p p = L_{\delta a} \delta a \quad (3)$$

where

- $\dot{p}$  = roll acceleration,  $\text{deg/sec}^2$
- $p$  = roll rate,  $\text{deg/sec}$
- $L_p$  = roll damping,  $1/\text{sec}$
- $L_{\delta a}$  = roll acceleration/deg of aileron deflection, also known  
as Roll Control Sensitivity,  $\text{deg/sec}^2/\text{deg}$
- $\delta a$  = aileron deflection,  $\text{deg}$ .

The solution to this equation, assuming a zero roll rate initial condition is

$$p(t) = p_{ss} \left[ 1 - e^{-\frac{t}{\tau}} \right] \quad (4)$$

where

$p(t)$  = roll rate as a function of time, deg/sec

$p_{ss}$  = steady state roll rate, deg/sec

$t$  = time elapsed from roll initiation, sec

$\tau$  = roll mode time constant, sec.

$p_{ss}$  may be interpreted as the maximum steady state roll rate, and is the product of  $L_{\delta a}$  and  $\delta a$  divided by the roll damping.  $\tau$  is equal to  $-1$  over the roll damping. Graphically, this solution is as appears in figure 5-11. For a given flight speed and maximum aileron deflection, the maximum steady state roll rate is that rate achieved after roll damping decays roll acceleration to zero (the slope of the curve in figure 5-11 is the roll acceleration). Also,

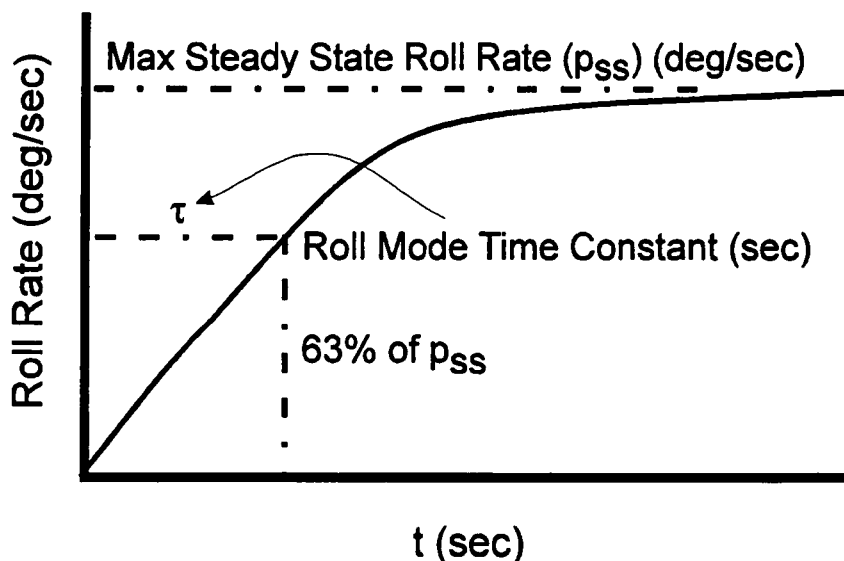


**Figure 5-11**

Solution to the Roll Equation Assuming an Aileron Deflection at  $t=0$

$\tau$  is defined to be the length of time it takes to reach approximately 63% of the maximum roll rate. The slope of the curve at  $t = 0$ , or the initial roll acceleration, is a function only of the product of  $L_{\delta a}$  and  $\delta a$ , which is the maximum rolling acceleration achievable. Of course, the final roll rate is a function of the maximum acceleration, aileron deflection, and roll damping.

Most airplanes today are certificated after having demonstrated a desired roll rate between specified angles of bank, i.e. from 45° left AOB to 45° right AOB. However, the achieved roll rate is a function of other parameters that were held constant in the previous discussion. In particular, within the range of airspeeds typical of lightplanes, the greater the flight speed, the greater the maximum steady state roll rate. Also, the previous discussion assumed that sideslip was unchanged during the maneuver (this type of analysis is referred to as the “single degree of freedom” roll response). If one considers the effect of adverse yaw in generating sideslip, then this induced sideslip, through dihedral effect, could cause a much less ideal response, such as that indicated in figure 5-12. This is referred to as the multi-

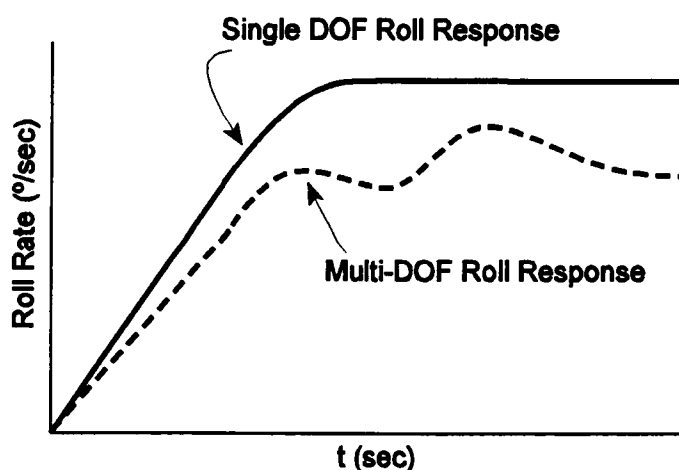


**Figure 5-12**

Variation in Single Degree of Freedom Roll Response when Positive Dihedral Effect Reacts to Adverse Yaw

degree of freedom roll response. The steady state roll rate achieved in the multi-degree of freedom roll response will be less than in the single-degree of freedom case. The periodicity of this curve is related to the dutch roll characteristic of the airplane, which will be discussed in chapter 7. The greater the dihedral effect or adverse yaw, the greater the amplitude of the roll oscillations. Reduced directional stability can also cause greater oscillations, while high levels of directional stability would tend to limit  $\beta$  and therefore the roll oscillations. It becomes clear why these characteristics must be kept to manageable levels, and that the task of the designer is a formidable one.

Directional control of a conventional airplane during takeoff and landing. As mentioned in chapter 1, directional control of a "taildragger" may present some interesting challenges during landing. The pilot is trying to maintain a static lateral-directional situation, but a perturbation occurs which may require a flurry of rudder pedal activity to maintain (or regain!) control. Consider figure 5-13, where a diagram of the situation is presented. (The CG must be aft of the main wheels, or the airplane would not rest on the tail wheel when at rest.) The major control problem is that if either the nose of the airplane is not aligned perfectly with the runway, or if any lateral motion exists relative to the runway, the nose of the airplane will tend to diverge from the desired heading upon touchdown. In the figure, assume that the airplane is in a slight right drift, which is the same as saying it has slight right sideslip, as depicted. At the moment of touchdown, the CG of the airplane has inertia in the direction opposite to the relative wind; this is the direction towards which the airplane will continue to move, if undisturbed. As the main wheels touch the surface, however, friction

forces impart a side force to the airplane opposite the direction of lateral drift. Considering these forces as moments about the CG, the airplane would yaw left, and require rapid footwork if a groundloop is to be avoided. Rudder-generated yawing moments must be sufficient not only to arrest the yaw divergence, but to bring the nose of the airplane back to runway heading.

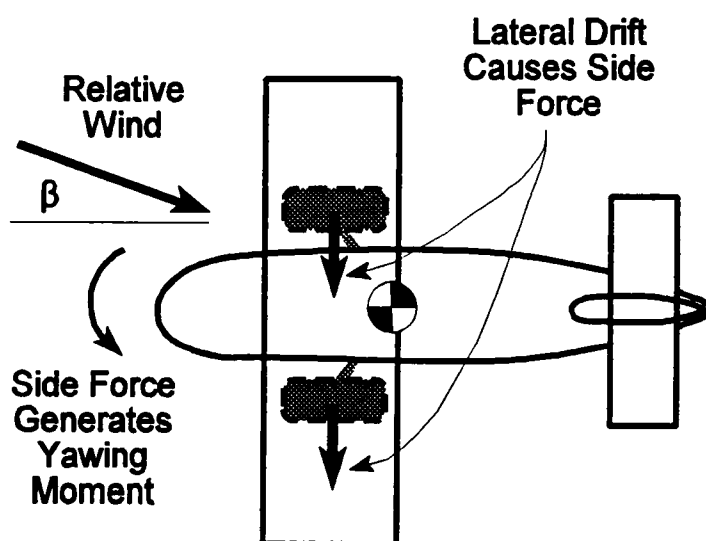


**Figure 5-13**  
Diagram Showing the Effects of Lateral Drift or Sideslip on Conventional Airplane Landing

## CHAPTER 6

### LONGITUDINAL DYNAMIC STABILITY AND CONTROL

In this chapter, an introduction to dynamics is presented, where typical conditions and types of motion are described. Following this tutorial, presentation and discussion of the longitudinal dynamics modes are presented.

#### *Introduction to Dynamics*

Static stability referred to the initial tendency of a system when displaced from an equilibrium condition. For example, if a longitudinally statically stable airplane was in equilibrium at some angle of attack, and was subjected to a perturbation in angle of attack and left to its own devices, the airplane would seek its original condition. Likewise, if a directionally statically stable airplane was in equilibrium at zero sideslip and was perturbed, the airplane would also seek its original condition. In dynamics, the nature of the response after the perturbation, with respect to time, is studied. The perturbation can be of natural origins, for example from turbulence; of course, the perturbation could also be caused by deflection of the elevator or rudder. No matter what the source, after the angle of attack or sideslip is changed, observing the airplane's response over time is the subject of dynamics.

Longitudinal dynamics is the response of the airplane in pitch (or in the plane of symmetry), due to angle of attack perturbations. Lateral-directional dynamics are the responses in the lateral and vertical planes, and are typically generated through sideslip perturbations. In general, a system will demonstrate positive dynamic stability if the size of the deviations, or amplitude of motion, decreases with time. A parameter's deviation plotted on the y-axis versus time on the x-axis is referred to as a *time history*. Hurt [3] contains an excellent discussion on dynamic stability.

This may sound rather esoteric, but an airplane with good static stability does not necessarily have good dynamic stability, though most lightplanes have good characteristics as a requirement of their certification. Homebuilt airplanes are not "guaranteed" to have as good stability characteristics, as is the case for most production airplanes.

Of general importance in dynamic stability, once again, is the effect of float (see Kershner [7]). Consider figure 6-1. If the airplane and horizontal tail have been subjected to an angle of attack disturbance, the airplane whose control surface is allowed to float is usually less dynamically stable than one whose surface is held fixed. As the figure shows in a heuristic way, the fixed elevator surface contributes to the overall surface area of the tail, which increases the moment generation capability. The same can be said of the rudder and vertical tail regarding directional stability.

In dynamics, there are six possible responses to a disturbance. Three are non-oscillatory, or aperiodic, and three are oscillatory (Hurt [3]). Refer to figure 6-2. In part A of the figure, the disturbance quickly decays to the equilibrium value. This response is characteristic of a statically and dynamically stable system, and is termed *deadbeat* or

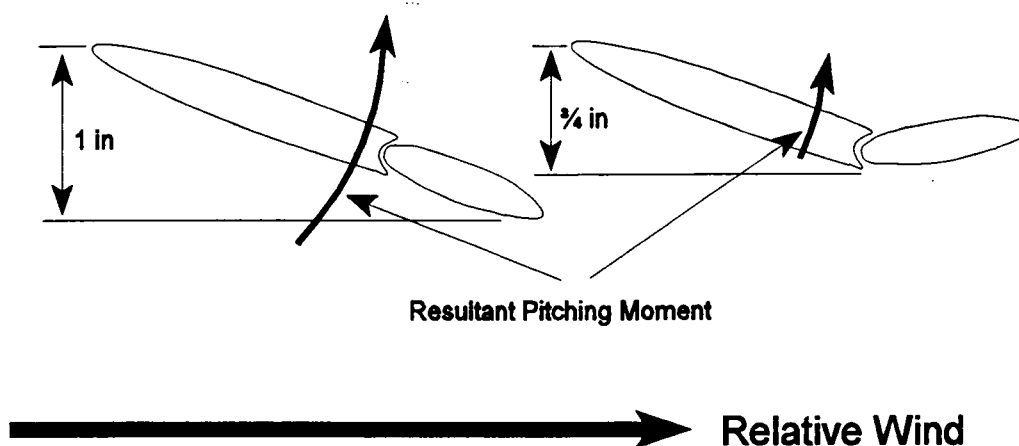


**Figure 6-1**

Effect of Elevator Float on Dynamic Stability. The relative wind "sees" less stabilizer if float occurs.

*subsidence*. In part B of the figure, the disturbance diverges away from the equilibrium condition, a characteristic that implies negative static and dynamic stability. This characteristic is called *aperiodic divergence*. Part C illustrates a response after a disturbance for a neutrally stable, static and dynamic case. There is no tendency to return to the starting condition, and the amplitude of the deviation does not change with time. In part D, a damped oscillation occurs. Since the system is constantly tending to the equilibrium position, it is statically stable, and since the response decays over time, it is dynamically stable. In part E, an undamped oscillation occurs. Since the system tends to equilibrium, it is statically stable, however, since the amplitude of the time history is not decreasing, it has neutral dynamic stability. Lastly, in Part F a divergent oscillation is illustrated. The system tends to equilibrium, so it is statically stable, however since the magnitude of the deviations is increasing, it is dynamically unstable. An example of this can occur in flight if the pilot unknowingly makes inputs in phase with the airplane's natural frequency. *Pilot induced oscillations (PIO)* may result. Damping of oscillatory motion is highly desirable (Hurt [3]):

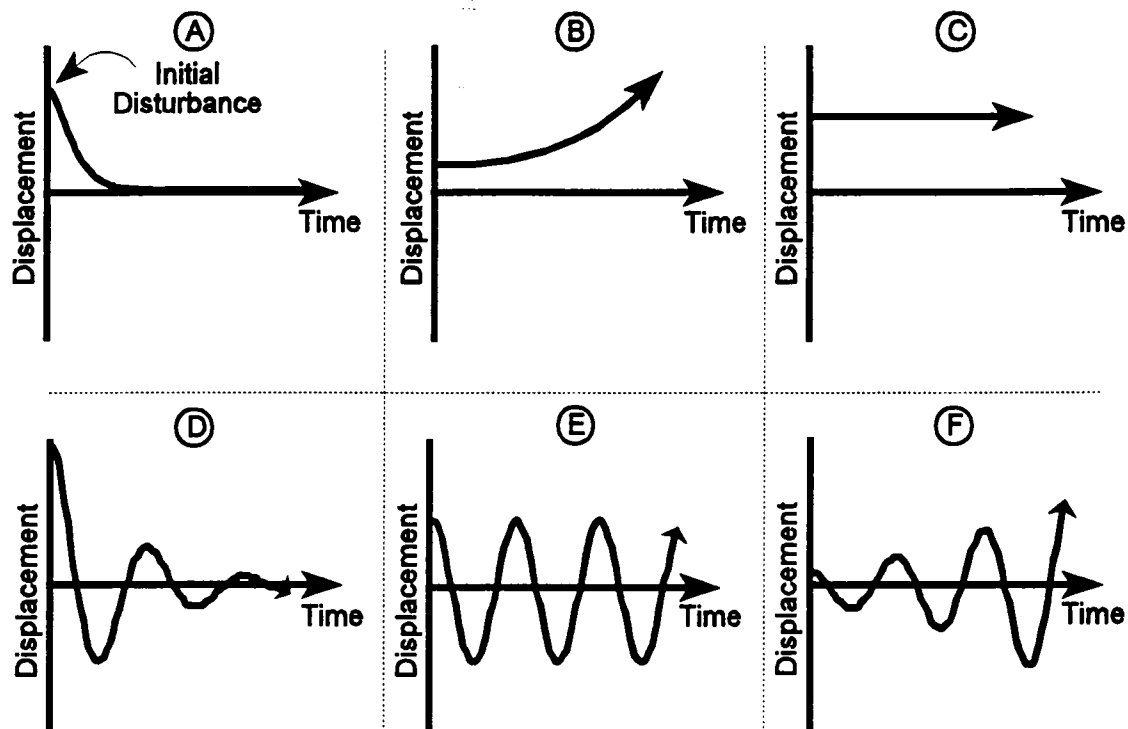


**Figure 6-2**

The Six Possible Dynamic Modes of Motion. A) Statically and dynamically stable, B) Aperiodic divergence, C) Statically and dynamically neutral, D) Damped oscillation (statically and dynamically stable), E) Undamped oscillation (statically stable, dynamically neutral), and F) Divergent oscillation (statically stable, dynamically unstable)

“As an example, an automobile with worn shock absorbers (or “dampers”) lacks sufficient dynamic stability and the continued oscillatory motion is neither pleasant nor conducive to safe operation. In the same sense, the aircraft must have sufficient damping to rapidly dissipate any oscillatory motion which would affect the operation of the aircraft.”

The existence of positive static stability does not predict the nature of the dynamic response, though if a system is dynamically stable, it must be statically stable. As stated earlier, any aircraft must have the required degree of both static and dynamic stability, where the

requirement is defined by the mission of the airplane. Converging or diverging time histories may be described by expressing the time it takes to reach half the initial, or twice the initial amplitude, respectively. These times are referred to as *time to half amplitude* and *time to double amplitude*.

Discussing the difference between parts A and D of figure 6-2 is of interest. Both cases are statically and dynamically stable. Imagine the cup and ball analogy presented in chapter 1. The dynamic response of the ball depends on the curvature of the cup (level of static stability), among others. If one released the ball at the edge of the cup, the ball would accelerate down towards the bottom of the cup, overshoot the bottom, and continue up the other side. One can rationalize that the ball does not go up the other side as high as it was when it was released. Again the ball would accelerate towards the bottom, and overshoot, and again, it would not travel quite as far up as it did after the first overshoot. The motion of the ball is oscillatory, and damped, as the deviations are reducing in size. Friction between the ball and the cup expends system energy, and provides the damping medium. Now consider what would happen if the cup was filled with water. The ball certainly would not overshoot much. Switching to motor oil, it is easy to believe that the ball would not overshoot at all. It is apparent from this thought experiment that the viscosity of the fluid is a factor in the damping character of the system. If an oscillatory system is well damped, it can be deadbeat (as the vertical motion of an automobile with effective shock absorbers should be).

## *Longitudinal Dynamic Modes of Motion*

Since longitudinal dynamics deals with motion in the plane of symmetry, the typical variables to consider in this chapter are the pitch attitude, angle of attack, flight path velocity, and elevator displacement if float is permitted. Statics requires that the sums of forces and moments are zero; now no such restriction holds. In particular, the airplane may have unbalanced forces and moments in the plane of symmetry. The mathematics required to solve the equations of motion are complex, and yield an equation which predicts that two natural longitudinal oscillatory modes of motion exist for the conventional airplane. These are not just “theoretical” results or an engineer’s abstraction--these two modes are the primary ways an airplane moves in pitch when left unattended.

The long period, or phugoid mode. The first mode of motion is generally called “the long period,” or the *phugoid* mode of motion. The use of “long” in “long period” foreshadows the relative frequency of this motion. Most phugoid responses for light airplanes are similar to figure 6-3. Important concepts regarding characteristics of the phugoid mode include the continuous energy exchange between airspeed and altitude, that the entire maneuver is performed at essentially constant angle of attack, and that the phugoid is lightly damped. It must be emphasized that the nature of the response might be greatly affected by float. A specific airplane might have a well-damped phugoid in the absence of elevator float. Float can be eliminated by the pilot (or even better, by an autopilot) restraining the longitudinal control. Any float would typically decrease the damping, and increase the time

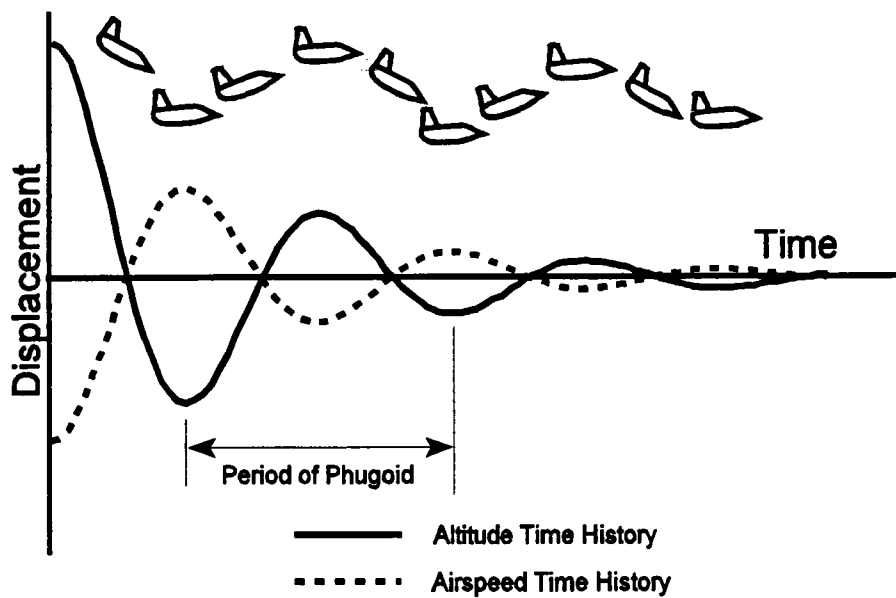


**Figure 6-3**

Time Histories of Phugoid (Long Period) Longitudinal Dynamic Mode

to settle to the initial condition. Large amounts of float could actually cause the system to go dynamically unstable, or increase the amplitude of the deviations over time. Appropriate mathematical simplifications produce the approximation

$$T_{Phugoid} = 0.233 \times V_T$$

where

$T_{Phugoid}$  = the period, that is the amount of time that passes from one peak to the next peak of the time history (actually, any two points may be chosen as long as they are separated by one complete cycle of the oscillation), sec.

$$V_T = \text{True airspeed, kt.}$$

For example, suppose a light airplane had a cruise speed of 100 kt. Simplified theory would predict that the phugoid would have a period of about 23 sec. An airplane at a final approach speed of 60 kt would have a phugoid period of about 14 sec.

A test pilot might refer to the “ease of exciting” the phugoid, by which is meant “how often” does the phugoid become a factor in flight path control, and “how large” is the resulting oscillation. For example, does light turbulence excite a phugoid response? For this to occur implies that turbulence upsets the static equilibrium, and begins the phugoid response. If an airplane could be flown at precisely the correct attitude to maintain a constant speed while holding power constant, then the phugoid will be continuously damped to zero amplitude. It follows that either power, airspeed, or attitude errors from the trimmed condition will tend to excite the phugoid, and only a pilot’s action can dampen it more quickly than the natural response. Of course, an autopilot with attitude- or altitude-hold dampens the phugoid quite effectively. Mechanical characteristics, for example excessive freeplay, can exacerbate the phugoid, as can the installation of a downspring.

The origin of phugoid damping is pitch rate damping; important considerations are the size of the horizontal tail and the length of the fuselage. By far the most important single characteristic, however, is that the airplane is statically stable, and the most important variable in this type of stability is the CG position.

Many airplanes have very lightly damped or even neutral-to-divergent phugoids. Clearly, such an airplane might require more attention to pilot than another airplane with better damping, however, Bingelis [11] overstated this:

“...These [phugoid] oscillations should die out within 3 or 4 cycles--if the airplane is stable in its pitch axis. If not, the airplane is unstable...”

Bingelis referred to the phugoid as if it was the only type of longitudinal motion. Also, if it took 5 cycles to damp, the airplane must be statically stable (it tends toward the initial condition) and dynamically stable (it is a damped oscillation). Much closer to the truth is Kershner's [7] understanding:

“An airplane with neutral or negative dynamic stability in [the phugoid mode] can be flown quite safely with little or no effort, as the periods may be many seconds or even minutes in length.”

Some airplanes might have good phugoid damping if the speed deviations are less than 5-10%, while the phugoid might diverge if the speed change is greater.

The short period mode. Believing that the phugoid exists is relatively easy, after all, one can excite the mode, and watch it over a period of many seconds. The existence of the short period mode is less obvious for several reasons, but its existence is no less real. The difficulty in observing the short period begins with the fact that most light airplanes have very well damped short period responses, and essentially no oscillatory behavior can be noticed. Next is that the period of the motion is much shorter, on the order of a second or so. Nevertheless, all airplanes exhibit the short period mode. The short period is the facility by which an airplane returns to trimmed angle of attack when displaced by a perturbation such as turbulence or releasing the elevator while in a pitch rate. Consider figure 6-4.

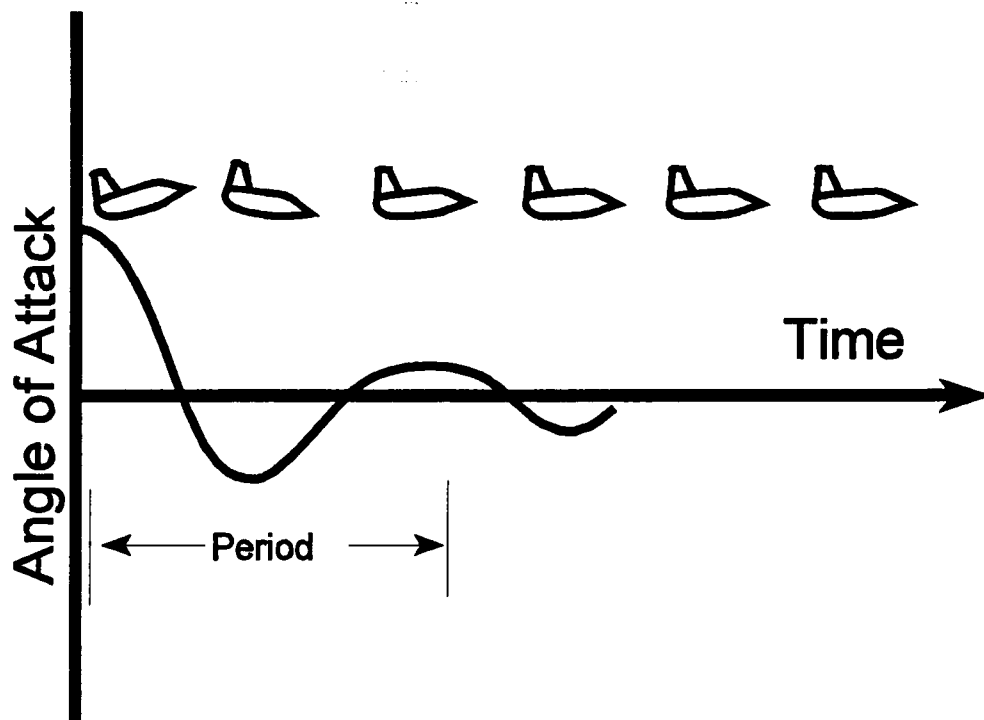


**Figure 6-4**  
Time History of Short Period Longitudinal Dynamic Mode

Important concepts regarding short period include that the motion is conducted at constant airspeed (as it happens quite quickly), that the angle of attack varies (the flight path does not), and that at least moderate damping is generally desired in order that objectionable flying qualities can be avoided. In the same manner as the phugoid, the existence of float generally decreases the stability and damping. During short period motion, the angle of attack moves or oscillates in the manner of the y axis displacement shown in figure 6-2, hopefully as in part A or D. Since the period is on the order of a second, if divergence was to occur, the pilot would have to react very quickly to prevent the airplane from destroying itself. (In an unstable design such as the X-29, whose stability is augmented by a computer, the natural time to double amplitude of the short period is a fraction of a second.) Because of pilot

response time, if a pilot intended to damp this motion, it is possible that the short period oscillation would actually be reinforced in a PIO which could quickly prove disastrous. In cases where PIO is occurring, a generally recommended strategy would be to hold the controls fixed, or release them entirely.

The short period is excited during angle of attack changes, that is to say frequently, and during longitudinal maneuvering. Therefore, the short period must be sufficiently damped to prevent or limit uncomfortable uncommanded load factor excursions caused by the angle of attack variation during the return to trimmed angle of attack.

As was discussed in chapter 4, the elevator commands angle of attack, so it follows that use of the elevator excites the short period motion. The origin of damping is once more affected by the size of the horizontal tail and fuselage length. The more forward the CG is, the greater the angle of attack stability, resulting in a higher frequency (or shorter period) of the short period. Higher speed also increases the frequency of the short period. As the CG shifts aft, the frequency decreases, until when the CG is at the maneuvering neutral point there is no tendency of the angle of attack to converge to the initial conditions.

The typical airplane in cruise flight has a short period on the order of one-tenth the phugoid period. However, since the phugoid period gets shorter and the short period gets longer as speed is reduced, one can expect that the control strategy in landing configuration may change. If the periods of the two modes get close enough, a phugoid response may involve angle of attack change, and a short period may involve flight path changes. In these conditions, the two modes couple with each other.

## CHAPTER 7

### LATERAL-DIRECTIONAL DYNAMIC STABILITY AND CONTROL

In this chapter, the lateral-directional dynamic modes of motion are presented.

#### *Lateral-Directional Dynamic Modes of Motion*

Lat-dir dynamics are concerned with the modes of motion that occur in the vertical and lateral planes, or alternatively, that do not occur in the plane of symmetry. Excepting the roll mode, these modes of motion are generally less important to the pilot than the longitudinal modes. Nevertheless, if the S&C characteristics in lat-dir are unsatisfactory, the pilot will certainly have his hands full.

In chapter 5, all pertinent aerodynamic factors were presented, but in this chapter the sums of the forces and moments result in motion. As was the case in longitudinal dynamics, the mathematics is complicated, and yields an equation that predicts two linear acceleration, non-oscillatory responses and one oscillatory mode. One linear solution is that of the roll mode, which was presented in chapter 5. The remaining two solutions are presented here.

The dutch roll mode. The dutch roll mode has some analogy to the short period in the longitudinal dynamics case. The lat-dir equivalent of angle of attack is sideslip, so dutch roll is the response to a sideslip change. In the longitudinal case, appropriate simplifications led to the assumption that the short period did not couple out of the plane of symmetry, Unfortunately, in the lat-dir case, the typical airplane's response to sideslip excursions include both sideslip and roll attitude changes, in a maneuver called the dutch roll.

Presenting a good drawing of what a dutch roll looks like on paper is difficult, so the following "word picture" is provided. In response to sideslip excursions (which can be commanded by the pilot through rudder deflection, or by turbulent or gusty winds), the directional stability of the airplane will tend to bring the nose back to the trimmed position. If the airplane is dynamically stable directionally, the airplane's sideslip behavior should be similar to figure 6-2, part A or D, implying that the excursions diminish over time. Depending on the damping, the nose may overshoot the trimmed value (as in part D), or may slowly decrease towards the trimmed value (as in part A). So far, so good.

While the sideslip excursions take place, dihedral effect (lateral stability) causes roll moments and subsequent roll rates. For a typical airplane, as the nose yaws left, right sideslip is induced which in turn induces a left roll rate. At some point, the yaw rate and sideslip reverse (assuming the dutch roll is not heavily damped), which causes roll rate in the opposite direction. The final picture is a damped sideslip response which in turn causes a damped roll rate and bank angle response. If the dutch roll was excited from level flight, the roll rate oscillates the airplane about an approximate wings level attitude, say  $\pm 15^\circ$  angle of bank. It is rather like waving your wings at someone while simultaneously waving your tail!

Depending on the strength of directional and lateral stabilities, the resulting ride can be mildly annoying to extremely uncomfortable. Whatever the quality of the ride, dutch roll is a nuisance. Figure 7-1 presents a dual time history of sideslip and roll attitude (angle of bank).

As in longitudinal dynamics, float generally degrades the airplane's natural stability, and must be controlled by the use of aerodynamic balance and hinge placement (Hurt [3] and Askue [2] have excellent discussion on these items). Important other concepts regarding dutch roll include that the maneuver takes place at essentially constant speed and altitude, that it is generally somewhat damped, that the sideslip and angle of bank tend to vary in an oscillatory fashion, and that the period of the dutch roll is generally less than that of the phugoid, but longer than that of the short period. A period of 1 second to a period of

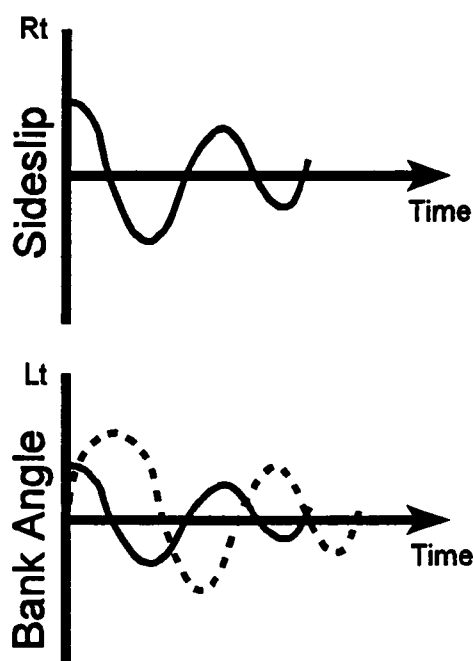


**Figure 7-1**

Time Histories of Sideslip and Angle of Bank in the Dutch Roll Dynamic Mode. Solid bank angle trace depicts in-phase response, while dotted bank angle trace is nearly 180° out-of-phase.

5-10 seconds (for an airliner) is typical. Lastly, the dutch roll frequency gets higher (period gets shorter) with increases in flight speed.

Due to the many variables that affect dutch roll period and damping, such as levels of directional stability and dihedral effect, weight, and inertia about the roll and directional axes, different airplanes have widely varying dutch roll characteristics. These differences affect the amount of roll rate and attitude change developed during sideslip oscillations. For example, in the V-tail Bonanza, turbulence excites a relatively undamped dutch roll response with only a small angle of bank, perhaps only 2-3° angle of bank for several degrees of sideslip. A dutch roll that exhibits a small roll response might be referred to as a "snaky dutch roll." Other airplanes, typically those with large values of dihedral effect, 5° of sideslip can generate angles of bank upwards of 15-20° angle of bank. This dutch roll might be referred to as "rolly." The bank angle response is often characterized by a test pilot or engineer as the *phi-to-beta ratio*,  $\phi/\beta$ , where  $\phi$  is the shorthand for bank angle, and  $\beta$  remains the sideslip angle. Airplanes with high  $\phi/\beta$  are roly, and those with low  $\phi/\beta$  are snaky. (One can gain a good general impression of  $\phi/\beta$  by stepping on and releasing a rudder while observing the wing tip. The wing tip of an airplane with a  $\phi/\beta = 1$  will scribe a circular motion, while a  $\phi/\beta = 5$  would imply an elliptical motion with the major axis oriented in the vertical and 5 times longer than the minor axis.)

As figure 7-1 shows, not only can the magnitude of the roll response change from airplane to airplane, but the phasing of the roll response can change as well. For example, an airplane with low inertia in the roll axis will tend to roll quickly when a rolling moment exists (like a fighter jet), however an airplane with large values of inertia tends to accelerate quite

slowly in roll (like a Boeing 747) These differences can change how quickly the roll response is observed for a specific sideslip excursion. In some airplanes it can seem like the roll response leads the sideslip response, but it is probably lagging sideslip by nearly an entire period.

Since the dutch roll is a nuisance mode (like the phugoid) and not generally used as a control mode (like the roll mode), an important handling quality of an airplane is the ease of exciting the dutch roll and the damping of the dutch roll. The greater the  $\phi/\beta$ , the more objectionable the dutch roll seems, although low  $\phi/\beta$  ratios combined with very low damping (as in the V-tail Bonanza) can get quite trying and tiring. In general, pilots prefer airplanes which do not have easily excited dutch roll modes, and should the dutch roll get excited, the damping should be high enough to limit the number of overshoots.

The airplane designer will attempt to minimize dutch roll by controlling float and providing high directional stability. Unfortunately, increasing directional stability degrades the other dynamic mode, the spiral mode.

The spiral mode. The spiral mode is another nuisance mode, the existence of which may hamper the pilot's control of the airplane. Most pilots would like to be able to establish an angle of bank, and release lateral control and have the airplane stay in that angle of bank. Of course, the desired angle of bank might be zero, as well. However, some sideslip is typically generated over time, which induces some rolling moment. Consider an airplane in level flight. Suppose a small control input or some turbulence excites a bit of right sideslip (the relative wind is right of the nose). Dihedral effect causes a bit of a left roll. With

controls free, the airplane starts an easy left turn. The right wing is advancing a tiny bit faster than the left, thereby producing more lift, and rolling further left. The bank angle increases slowly into an ever-tightening spiral. This aperiodic divergence in bank angle typically progresses slowly, with a time to double amplitude on the order of 10-30 sec for a typical lightplane with negative spiral stability. Some airplanes, a relative few, actually have positive spiral stability, with times to half amplitude of about the same number of seconds.

As already mentioned, the designer's goal of minimizing dutch roll competes with his desire to have long times to double amplitude of the spiral mode. Two flight control devices can assist him if the airplane's characteristics warrant. If the dutch roll damping is poor, he can add a *yaw damper*, which is typically an electrically powered device designed to detect uncommanded yaw rates, make opposing rudder inputs, and stop the resulting yawing motion. This device is designed to quickly stop dutch roll activity. Other airplanes can incorporate one, two, or three axis autopilots. A single axis autopilot is usually referred to as a "wing leveler." When activated, the angle of bank is held at zero by an electrically powered aileron actuator. Many Mooney single engine airplanes contain PC™ (for "Positive Control"), which is a wing leveler. A two axis autopilot includes some pitch control, which can hold pitch attitude, or by reference to an altitude source, can maintain a desired altitude. This type system effectively controls both spiral and phugoid nuisance modes. A three axis autopilot adds a heading control feature, where heading control or capture of a desired heading is incorporated.

It is worth mentioning that descriptions regarding dutch roll in many texts vary substantially. Some texts offer only a few words, others are inaccurate. Kershner [7]

provides a good summary of the lat-dir dynamics modes, while in Askue [2], the author leads the reader astray discussing dutch roll:

“Dutch roll occurs when inadequate yaw stability couples with excessive dihedral effect. [S]ideslip causes the airplane to roll away from the relative wind and slow down. The nose will drop, and the roll will cause a sideslip in the opposite direction.”

First, directional stability and dihedral effects must be managed within limits to control both the dutch roll and spiral modes. Improving “inadequate” yaw stability may well result in dramatic reduction in the time to double amplitude of the spiral mode. Also, for nominal dutch roll maneuvers excited by 5-10° of sideslip, speed change is insignificant.

## CHAPTER 8

### ASYMMETRIC POWER CONSIDERATIONS

In this chapter, a special case in static and dynamic stability and control is presented. When a non-centerline thrust multi-engined airplane loses power on an engine, the lateral-directional handling qualities may be challenging, especially at low speed. Asymmetric power characteristics provide a good opportunity to apply previous material to a new subject. For these purposes, assume the airplane under discussion is a light twin-engined airplane. The balancing of the yawing moments, balancing of the rolling moments, and miscellaneous characteristics will be discussed. Consult Hurt [3] and USNTPS [8] for further information. (In this chapter, instead of using non-dimensional aerodynamic coefficients, the author uses the dimensional quantities. For example,  $C_L$  is the dimensionless coefficient, while  $L$  (lift) is the dimensional quantity, in this case the amount of lift in pounds.)

#### *Balancing the Yawing Moments*

An obvious feature about operation with an engine failed is the generation of an asymmetric yawing moment by the remaining engine,  $N_t$ . See figure 8-1, where it is assumed the failed engine is feathered. The actual value of this moment is a function of thrust

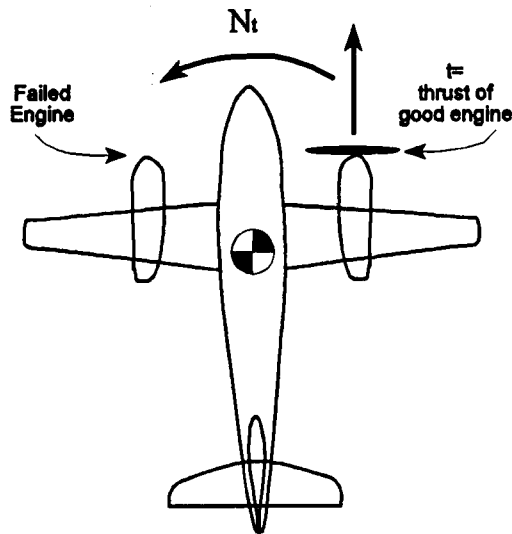


**Figure 8-1**  
Asymmetric Power Yawing Moment

produced by the good engine, and distance from the airplane center line. The logical way for the pilot to oppose the asymmetric yawing moment is to use the rudder to generate a countering moment  $N_{\delta r}$  as shown in figure 8-2. However, although the yawing moments are now balanced, there is an unbalanced sideforce due to rudder deflection,  $Y_{\delta r}$ , which will generate a lateral acceleration on the aircraft and curve the flight path. In order to attain steady, straight flight in this asymmetric condition, therefore, the airplane can be sideslipped into the failed engine by adding an additional increment of rudder to generate a balancing sideforce (through sideforce due to sideslip,  $Y_{\beta}$ ) and stop the airplane from turning. This situation is depicted in figure 8-3. As an aside, this sideslip also generates a yawing moment,  $N_{\beta}$ , which is countered by the increment in  $N_{\delta r}$ .

There is a drag penalty associated with flight in a steady sideslip, and it is possible to generate the balancing sideforce by banking the airplane towards the operative engine and allowing a component of its weight,  $W \sin \phi$  (where  $\phi$  continues to be the bank angle), to act

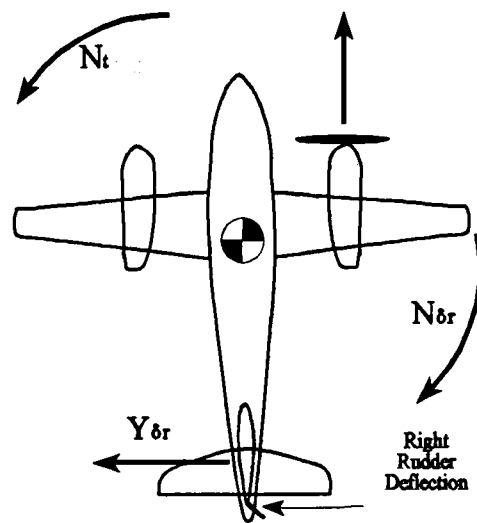


**Figure 8-2**  
 Moment Equilibrium at Zero Bank Angle and  
 Unbalanced Side Force

to oppose the sideforce due to rudder,  $Y_{\delta r}$ . This technique has the advantage that sideslip is eliminated and drag is reduced; it follows that this is the most efficient configuration for long distance cruise. The rudder requirement will be reduced compared to the wings level sideslip method because  $N_{\beta}$  is eliminated. The bank angles involved will usually be small and generally do not cause discomfort. On the other hand, most airplanes do not have any display for sideslip, so it becomes difficult to properly establish the zero sideslip condition. As a result, most pilots will likely fly in some compromise condition between these two.

There is a third equilibrium condition which can be achieved with no rudder deflection at all. By increasing the bank angle towards the good engine, the airplane will sideslip in that direction which will generate both an opposing yawing moment,  $N_{\beta}$ , and a sideforce  $Y_{\beta}$  to balance the lateral component of weight,  $W \sin \phi$ . In other words, the airplane's directional stability offsets the yawing moment. This technique is inefficient due to the large sideslips

required and uncomfortable for the occupants of the airplane due to the large angles of bank generated. This technique is also potentially dangerous because of the risk of vertical tail stall at large sideslips, as well, and is not typical or to be recommended.

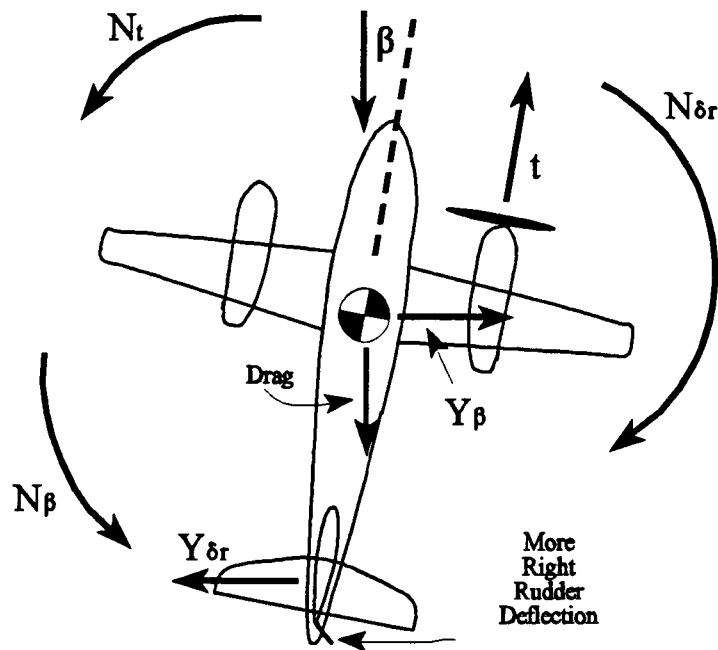


**Figure 8-3**  
Wings Level Equilibrium With Sideslip

### *Balancing the Rolling Moments*

Lateral control problems may dominate the asymmetric handling qualities (especially in the dynamic case). Consider figure 8-4. In wings-level equilibrium, as in figure 8-3, roll due to sideslip is in the direction of the operating engine (assuming normal lateral stability).

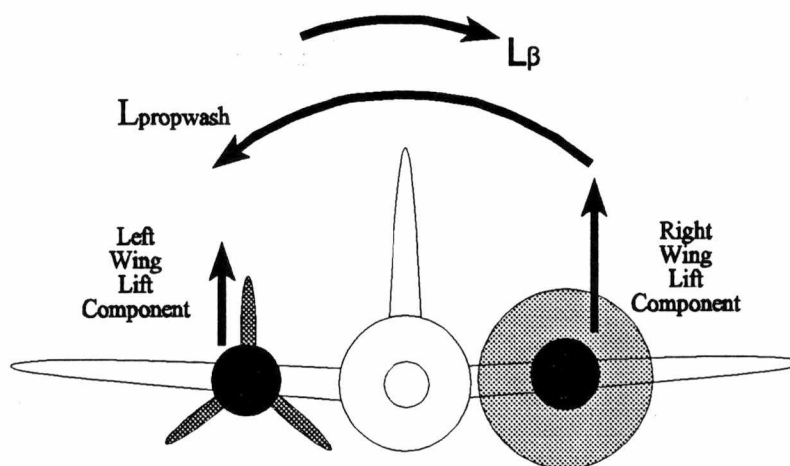


**Figure 8-4**

Depiction of Adverse Rolling Moments In Asymmetric Powered Flight

However, a propeller airplane will also suffer a loss of high-energy propwash behind the failed engine, decreasing lift on that wing. Maintaining lateral control may well be the limiting factor in some airplanes.

### *Static Minimum Control Speed*

At constant power setting, the thrust of the good engine increases as speed is decreased in the normal speed range, so the yawing moment the good engine generates will increase as speed is reduced. The total aerodynamic yawing moment,  $N_{\delta r} + N_{\beta}$ , is dependent on dynamic pressure (i.e. on TAS squared) so as speed is reduced at constant power at some point all available aerodynamic control in one or another axis (using maximum rudder or maximum aileron deflection) will be required for equilibrium. When the good engine is set at full power and the deceleration is conducted smoothly and slowly, this speed is called the static minimum control speed, or  $V_{mc}$ . It is normally first investigated with wings level, to

provide the minimum speed for the wings level case. If the pilot runs out of rudder at the wings level  $V_{mc}$ , the test may be repeated with up to 5° angle of bank toward the good engine. These speeds represent the absolute minimum speeds at which control of the aircraft can be retained. Any loss of control near  $V_{mc}$  may cause pitch, yaw, and roll rates which can not be arrested with the remaining aerodynamic control. If one considers the case of an engine failure immediately after takeoff, the effects of a loss of an engine can be much more calamitous than a slow deceleration towards  $V_{mc}$ . Additional aerodynamic control must be available for these cases, so it appears that an artificially low  $V_{mc}$  may have been determined in the above discussion.

#### *Dynamic Minimum Control Airspeed*

When an engine fails in flight there will always be a delay before the pilot recognizes the situation and applies the correct controls. During this interval, yaw, roll, and pitch rates can build up which will require more than the final equilibrium control deflections to stop. Thus the dynamic minimum control speed,  $V_{mc(dyn)}$ , will never be lower and will quite possibly be higher than the static  $V_{mc}$ . The magnitude of the rates achieved before control is regained will depend on how long the pilot delays before taking corrective action. The worst condition for  $V_{mc(dyn)}$  in most airplanes will usually be the takeoff. Some airplanes (airliners, for example) may not develop very noticeable rates in the first few seconds, and may mask an engine failure until more significant rates have built up. Limited aerodynamic control and large values of roll and yaw inertia may then delay recovery.  $V_{mc(dyn)}$  will not be associated

with a unique speed at which something unmistakable happens, as was the case in determining static  $V_{mc}$  when the airplane can no longer be kept straight. It is essentially a judgement call on the part of the pilot that this speed represents the minimum speed at which an average pilot could recover to straight flight following an engine failure.

The lateral handling problem may be exacerbated both by roll rates towards the failed engine and by large aileron deflections applied to correct them. A roll rate will increase the angle of attack of the down-going wing and any downward aileron deflection will increase the effective angle of attack still further. If the airplane was near the stall before the engine failure, this increase in angle of attack may be enough to stall the down-going wing and cause loss of control. Large aileron deflections also increase any unfavorable yawing moment due to adverse yaw and make recovery more difficult.

It is worth mentioning that  $V_{mc}$  varies with the amount of thrust produced by the operating engine. If throttle is reduced, the resulting yawing moments are reduced and the resultant  $V_{mc}$  would be lower. This leads one to an intimate understanding of how to recover from a  $V_{mc}$  situation: increase speed, decrease asymmetric thrust, or both.

### *Miscellaneous*

Engine failure during takeoff. There are several important differences between loss of an engine during the takeoff run and an airborne engine failure. Ignoring performance considerations (i.e. rate of climb, etc.), it can be seen first that much smaller deviations from the planned flight path can be tolerated. Secondly, the airplane is constrained to zero bank

angle because the wheels are on the ground. Finally, the airplane's response to control inputs consists of motion not about the CG as in flight, but about the wheels. With tricycle gear this means that the rudder will operate at a reduced moment arm. Crosswinds can also have a definite effect on the results. Minimum control speed for an engine failure on the takeoff roll is termed  $V_{mcp}$  for "ground." For typical light twin-engine airplanes, it seems likely that closing both throttles will permit recovery of heading control in these situations, and permit stopping the airplane on the runway.

Critical Engine. In discussion and determination of minimum control speeds and handling characteristics, pilots are concerned with the worst case, and thus far no distinction between failures of left and right engines has been made. The engine whose failure gives rise to the most adverse outcome is called the critical engine. Some airplanes may lose electrical or hydraulic power as a result of losing an engine, which could cause a controllability problem, but any propeller airplane will have unequal asymmetric power effects (unless it has contra-rotating propellers). This is due both to the swirl imparted to the slipstream by propeller rotation (which will have asymmetric effects on the rear fuselage, vertical tail, and rudder) and to the fact that flight at a positive angle of attack results in the downgoing propeller blade experiencing a slightly higher blade angle of attack than the upgoing one (often called "p-factor"). The downgoing propeller therefore generates more thrust, which has the effect of moving the effective thrust center for each propeller towards the downgoing blade. In US-manufactured engines, with clockwise-rotating propellers, the engine on the right is effectively operated at a slightly larger moment arm than the engine on the left, and

the most adverse condition from a control point of view is when the left engine is failed while the right engine (with greater moment arm) is operating.

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## **APPENDICES**

**APPENDIX A**

**FLIGHT TEST FUNDAMENTALS, EVENTS, AND**

**CHRONOLOGY**

## **FLIGHT TEST FUNDAMENTALS, EVENTS, AND CHRONOLOGY**

In this appendix, a discussion of the fundamentals of flight test are presented. Flight test evolutions are simply not the same as other flights. An understanding of the difference in outlook is vital. Next, a discussion of typical flight test events and their chronology is presented.

### *Flight Test Fundamentals*

Philosophical Basis. The basis of properly conducted flight test, and the outlook of the test pilot, is rightly to be compared with the scientific method. Reference [12] states in part that:

“Scientific Method” is a term denoting the principles that guide scientific research and experimentation. Science applies itself in a rigorous manner. Definitions of scientific method include such concepts as objectivity of an approach to, and acceptability of the results. Objectivity indicates the attempt to observe things as they are, without falsifying observations to accord with some preconceived notion. Acceptability is judged in terms of the degree to which observations and experimentations can be reproduced. Scientific

method also involves the interplay of inductive reasoning (reasoning from specific observations and experiments to more general hypotheses and theories) and deductive reasoning (reasoning from theories to account for specific experimental results).”

The underlines are the author’s, and are provided because they represent the concepts that transfer directly to the discussion of flight test. It is sufficient to conclude that a flight test program is not to be entered into lightly, yet if the airplane to be tested is the product of thousands of hours of the reader’s labor, it should be an immensely satisfying activity to do so.

Preparation and principles of safe flight test. If the test pilot is experienced with thousands of hours and hundreds of airplane types in his logbook, and is test-experienced, the process of preparation will probably be somewhat reduced when compared with that of the case of the pilot with hundreds of hours. It is imperative that the evolution be a planned one, with emphasis on exercising every risk reduction measure reasonably available. Make no mistake, flight testing of a newly-built airplane is a potentially hazardous activity.

The pilot must know the airplane and its systems. In flight test lore, the stories of how the knowledgeable pilot saved the airplane and lived to fly another day are numerous. Of course, there are plenty of stories describing inadequate knowledge, too! If, from reading this manuscript, the reader desires to sample the S&C characteristics of a production airplane, the preparation might be reduced when compared to a first-flight, however even this activity must not be considered lightly. Production airplanes generally fly well, but the condition of the

structure might be unknown, and flying qualities at the edges of the permissible flight envelope may not be quite so good as those in the center, where the pilot has spent most of his time.

The agenda of what is to be tested, and how it is to be tested, should be determined in the sobering reality of a conscious thought process on the ground, not in the air. If the reader spends his effort in preparing well for a test the days and weeks before a proposed flight, there is little reason to suspect that a better decision can be made on the “what’s and how’s” once airborne. It is a rare pilot whose IQ increases after takeoff! Do the thinking and planning on the ground, including a plan for knock-it-off criteria, if required. Once airborne, carry out the plan.

Most people have heard the term “Drive Defensively.” Conscious planning for flying defensively is a must. The plan must include as many “outs” as possible, and should certainly ask the “what if” questions in order that the unanticipated can be planned for. Do not assume that the test will go perfectly, as thousands of test pilots will surely attest.

The concept of *build up* is an important one. Always plan to execute a test from a known condition. Typically, start in the middle of the envelope and progress to the edges. This is especially important on a first-flight, but a sound policy in all cases. For example, if flying qualities are under examination, sample them in the center of the flight envelope before looking at  $V_{max}$  or  $V_{stall}$ . If one is looking at stall characteristics, look at them at moderate altitude before low or high, and at moderate-CG positions prior to stalls at the forward or aft limit. If steady heading sideslips are the order of the day, perform them at  $\frac{1}{4}$ ,  $\frac{1}{2}$ , and  $\frac{3}{4}$  rudder deflection before using full rudder deflection. It is difficult to imagine when this piece

of advice could get a pilot in trouble. It is also a frequently overlooked piece of advice. Plan for build up, then execute the plan.

In preparation for a first-flight, take advantage of others' experience. Speak with pilots who have flown the airplane, and have flown first-flights in the airplane. If it is a lesser-built variety of homebuilt, speak to the designer if possible. The EAA is a great organization for developing a network of sources of information for any homebuilder.

### *Flight Test Events and Chronology*

Bibliography entries [2], [13], [18], [19], and [20] contain a wealth of experience and information on this subject. In particular, Askue's [2] volume is nearly required reading for anybody considering building or test flying a homebuilt airplane. Not only is the reader provided a strong foundation in the flight testing of light airplanes in an incremental approach, but correction strategies to numerous airplane problems are discussed for the homebuilder. In this section, an overview of first-flight preparations and activities is presented. The pilot interested in sampling the flying qualities of a previously flown and tested airplane may concentrate on those tests.

Systems tests. Long before the first flight is attempted, full systems check-outs must be conducted. Flight control system, landing gear, flaps, brakes, engine, electrical, and pitot-static systems must all be without defect, and have had enough operation to assure the pilot of the required reliability. Also, an accurate determination of the airplane's CG must be

made. Tests should consider a build up approach to endurance conditions. For example, a first engine run might be only a couple of minutes long, with only a minimum of fuel in the airplane. After shutdown, a complete inspection of the engine and installation should be performed before graduating to ever-increasing operating times. Clearly, engine cooling must be monitored. Complete testing of the fuel system and fuel feed capability to the engine must be demonstrated at the most nose-up attitude likely to be encountered.

Taxi tests. A build up approach to taxi and actual flight conditions should be performed, as well as light, moderate, and heavy braking. Very slow taxiing and easy turns, should be followed by higher speeds and quicker turns. At any point in this process, if anything unusual occurs, discontinue the testing until satisfactorily resolved. As confidence and experience is gained, and in the absence of wheel shimmy, unusual vibration, or PIO, speed should be increased to the point where a runway is the safest place to be. Know the anticipated takeoff speed (usually about 1.3 times  $V_s$ ) and an abort point down the runway which will assure you of a safe stop from liftoff speed. At speeds approaching  $V_s$ , attempt to raise the nose or tail wheel, immediately, but slowly, replacing it on the runway. If no problems need to be corrected, it's time for the first-flight.

First-flight. Establish a maximum operating speed for the first flight, one-third to one-half of way between stall and maximum level flight speed. Design a test card, a sort of sequential checklist with each desired maneuver, test condition, and test to be performed. The first flight should consist of a lift off at a speed above stall, when the airplane leaves the

ground naturally in a takeoff attitude. Reconfigure as required and conduct a gentle climb and immediate turn towards downwind or other available runway, and continued gentle orbiting climb over the airport. Watch all engine limitations carefully during the climb to and level off at about 5,000 ft AGL. Using no more than 45° AOB for the first flight, perform gentle turns in both directions using coordinated control inputs (aileron supplemented by rudder). Increase the bank of the turns slowly to 45° AOB.

Next perform part-power descents and gliding descents, being especially sensitive to pitching moments generated by power changes. Perform slow decelerations at low power and power off all the way to a wings level stall, recording each speed. Compute 1.4 times  $V_s$  for use during landing approach. Simulate the approach, flare, and touchdown in terms of attitude, airspeed and power. Simulate a go-around. If the test airplane has movable landing gear or flaps, repeat these tests with gear and flaps down.

Perform an approach to a landing, but be prepared to go-around on the first pass if necessary for any reason. Otherwise, land, close the throttle, and stop--the first flight is over. Conduct a thorough post flight inspection after removing all access covers.

Envelope expansion. In the next few flights increase the load factor and airspeed slowly, and conduct flutter tests. Flutter testing is a serious business, and should be accomplished only after thorough preparation. Again, Askue [2] contains excellent recommendations. Once the basic structural integrity of the airplane has been demonstrated, it is time to document the handling qualities.

Handling qualities. Some handling qualities tests can tax the airplane's structure, so approach all tests in a build up fashion, being sensitive to changes in handling qualities, and always conduct thorough post flight inspections. Define at least three test conditions for all tests: low speed (approximately  $1.3 V_s$ ), medium speed (halfway between  $V_s$  and maximum level flight speed), and high speed cruise with about 75% power. Refer to entries [18] through [20] for extensive discussion regarding appropriate test conditions. Starting with the low speed, and on successive flights for the increased speeds, conduct the following tests at the same trim, power, and CG position:

- Longitudinal static non-maneuvering stability. (As in other tests, estimates or measured control forces are required. Some fish scales may be suitable for use in flight, but in any event, use a scale before flight to "calibrate" your arm to improve your estimating capability.) Stabilize at the desired test airspeed, and set power and trim. Leaving power and trim alone, change speeds in no greater than 5 kt increments to about 10-20% above and below the trim condition. Record force data, and upon return, plot the data in a similar fashion to figure 4-5.
- Longitudinal maneuvering stability. Leaving trim and power alone, in level turns record and plot the control forces required to stabilize at elevated g, in a similar fashion to figure 4-11, part A.
- Steady heading sideslips. Using first  $\frac{1}{4}$ , then  $\frac{1}{2}$ , then  $\frac{3}{4}$ , and finally full rudder deflection, perform SHSS using both right and left rudders. For example, as left rudder is added from wings level, some right angle of bank will be required to stabilize a heading. At each stabilized condition, record angle of bank, relative rudder position,

lateral stick or yoke deflection, and sideslip. Consciously look for signs of rudder float and tail stall. See the caution in chapter 5 regarding sideslip evaluations.  $\beta$  may be estimated by observing the heading change the airplane experiences when the rudder is released from the stabilized condition. This also excites the dutch roll, so damping can be observed by counting overshoots, and  $\phi/\beta$  ratio can be estimated by looking at the wing tip's motion (see chapter 7). Recall that the level of apparent directional stability can be inferred by the slope of the rudder versus sideslip plot, and the existence of the dutch roll implies positive directional stability. Plot the SHSS in a fashion similar to figure 5-7.

- Phugoid mode. Smoothly decrease airspeed by 10-15%, establish the airplane in a level flight attitude, and release the controls. Measure the time from one maximum airspeed to the next (the period of the oscillation), while recording each maximum and minimum airspeed observed. If each successive maximum is less than the previous one, the phugoid is damped.
- Short period evaluation. As previously stated, most light airplanes have highly damped short period responses, so it may be difficult to observe. Any longitudinal control motion excites the short period, but a longitudinal *doublet* is a commonly used one. A longitudinal doublet is a forward stick motion followed by an aft stick motion of the same amplitude, followed by release of the controls. After the controls are free, observe the airplane's short period response. It will be damped (or the pilot will be very unhappy!), but attempt to observe any overshoots. Another way to excite the

short period is to apply a 2 or 3g turn, then release the controls and observe the airplane's response.

- **Spiral mode evaluation.** Most flight control systems have some friction, as described in chapter 2. For an evaluation of the spiral mode, it is important that friction be eliminated as a possible cause of AOB change. The easiest way to do this is to get the airplane in equilibrium at wings level. If the airplane will not trim laterally or directionally and displays non-equilibrium at wings level, there is no point in conducting the test. Using dihedral effect (rolling with rudder), establish a 5-10° AOB in either direction while holding the lateral control fixed. Time the bank angle change (time to double- or time to half-amplitude) from 10-20° for a divergent spiral, or from 10-5° for a convergent spiral.
- **Roll mode.** Using first  $\frac{1}{4}$ , then  $\frac{1}{2}$ , then  $\frac{3}{4}$ , and finally full lateral control observe the roll response using two techniques, lateral control only, and coordinated lateral supplemented by rudder. This is also a convenient time to observe and document the airplane's dihedral effect by using rudder only to roll the airplane. If right rudder is applied, the airplane should roll right if dihedral effect is positive. With reasonable levels of lateral stability, any small to moderate bank angle can be captured and maintained.
- For a twin-engined airplane, determine the half power stall speed, then perform a slow deceleration from  $1.4 V_s$  to static  $V_{mc}$ , once for each engine at full power. Do not decelerate below plus 5 kt, and discontinue evaluations if any indications of stall are detected. Mixing up a stall within a  $V_{mc}$  evaluation is a recipe for departure from

controlled flight. Then, starting at  $1.5 V_{mc}$ , establish takeoff power in a climb for the  $V_{mc(dyn)}$  evaluation. With the desired airspeed established, perform a throttle chop on the non-critical engine (left engine for engines that rotate clockwise when viewed from the cockpit). Using a 1 to 2 second reaction time, recover the airplane using any or all controls, but primarily the rudder. The test airspeed may be reduced by increments until a limiting condition is reached; the last airspeed tested is the  $V_{mc(dyn)}$  for this set of test conditions and reaction time.

These are the basic tools in the arsenal of the test pilot wishing to document S&C characteristics. Perhaps the most valuable tool has not been discussed, however, that of observing the performance of mission tasks. Any test pilot, before performing the above tests, will perform mission task elements that the airplane is likely to need. For example, a lightplane must taxi, takeoff, cruise and land. While performing these normal tasks, a pilot should observe the airplane's response and characteristics, paying particular attention to the relative difficulty in performing these tasks. If an airplane is difficult to fly at steady cruise conditions, then the phugoid, mechanical characteristics, or static non-maneuvering longitudinal stability may be suspect.

By performing the above qualitative and quantitative tests, a good understanding of the flying qualities of the airplane can be obtained.

Performance. Documenting the performance of the airplane is another subject. It is sufficient to state that all the structural integrity and all the fantastic handling qualities mean

little if the airplane's performance is not adequate for its mission. Bibliography entries [2], [3], [4], [7], and [18] contain much valuable information.

**APPENDIX B**  
**GLOSSARY**

## GLOSSARY

|                           |                                                                                                                                          |                           |                                                                                                                                             |
|---------------------------|------------------------------------------------------------------------------------------------------------------------------------------|---------------------------|---------------------------------------------------------------------------------------------------------------------------------------------|
| adverse yaw               | yawing action away from desired roll direction, as in left aileron causing the nose to yaw right. Also known, generally, as aileron yaw. | center of pressure (c.p.) | the actual location of lift forces; it is a function of camber and coefficient of lift                                                      |
| aerodynamic center (a.c.) | a point 25% aft on the MAC, where lift forces can be considered to act when accompanied by the wing's moment about the a.c.              | centering                 | the tendency of a control to return to neutral or trim                                                                                      |
| anhedral                  | description of geometric dihedral where the wing tip is positioned below the wing root                                                   | coefficient of lift       | $C_L$ , example of an aerodynamic coefficient                                                                                               |
| aspect ratio              | numerical term describing the relative shape of the wing, i.e. long and skinny ( $10 < AR < 60$ ), or short and stubby ( $AR < 5$ )      | compressibility           | term used to describe whether airflow is acting like a liquid (i.e. non-compressibly, such as at speeds less than 0.3 Mach) or compressibly |
| build up                  | flight test practice proceed from known conditions to unknown, or from the center of the envelope to the edges of the envelope           | couple                    | a pair of parallel forces with opposite direction, each generating a moment about a common reference point                                  |
|                           |                                                                                                                                          | deadbeat                  | the action of a displacement returning promptly to initial conditions with no overshoots; also known as subsidence                          |

dihedral effect also known as lateral stability, the tendency of the airplane to roll in response to sideslip (positive dihedral results in rolling away from sideslip)

#### geometric dihedral

also "dihedral," the fixed angle at which the wings are attached to the fuselage, describing the wing tip's position above or below the wing's root

directional airplane motion about the vertical axis

#### directional stability

also known as weathercock stability, the tendency of the airplane to return to an initial sideslip when displaced

downwash a term describing the descending airflow caused by the lifting wing passing through the air

dutch roll lateral-directional dynamic mode, typically a coupled roll and yaw maneuver

dynamic the condition when the sums of forces and moments are non-zero

float the deflection of a control surface in response to hinge moments as opposed to pilot inputs; usually reduces stability

freeplay the condition described when a cockpit controller can be moved without moving a control surface

lat-dir shortened form of lateral-directional

lateral airplane motion about the longitudinal (i.e. roll) axis

lateral axis the airplane axis oriented along the span of the wing (see also "longitudinal" and "vertical")

longitudinal airplane motion about the lateral (i.e. pitch) axis

#### longitudinal axis

the airplane axis oriented along the fuselage (see also "lateral" and "vertical")

#### mean aerodynamic chord (MAC)

the chord drawn through the geometric center of the wing area

p-factor the yawing moment generated by a propeller due to the down-going blade experiencing greater angle of attack and thrust

phugoid the long period longitudinal dynamic mode

#### pilot induced oscillations

or PIO, the inadvertent reinforcement of a natural airplane response by in-phase pilot inputs

|                 |                                                                                                                 |
|-----------------|-----------------------------------------------------------------------------------------------------------------|
| pitch           | airplane motion about the lateral axis                                                                          |
| pitch rate      | pitching motion measured in deg/sec                                                                             |
| proverse yaw    | yawing action into desired roll direction, as in right lateral control deflection causing the nose to yaw right |
| relative wind   | the airflow coming at the airplane from the direction in which the airplane is moving                           |
| roll            | airplane motion about the longitudinal axis                                                                     |
| roll rate band  | analogous to trim speed band, a band of roll rates where no stick force is required to stabilize                |
| roll damping    | the aerodynamic action and quantity that stops roll rate once started                                           |
| rudder lock     | rudder float that causes reduced or no force requirement to increase sideslip                                   |
| sideslip        | the angle measured in the lateral plane between the relative wind and the longitudinal axis                     |
| span efficiency | factor used to account for less than ideal lift distribution                                                    |

|                         |                                                                                                                                |
|-------------------------|--------------------------------------------------------------------------------------------------------------------------------|
| spiral                  | lat-dir dynamic mode, typically an aperiodic divergence in angle of bank                                                       |
| static                  | the condition when the sums of forces and moments are zero                                                                     |
| steady heading sideslip | or SHSS, a static maneuver performed in crosswind landings and for engineering analysis of directional and lateral stabilities |
| stick-fixed             | term describing characteristics with no surface float                                                                          |
| stick-free              | term describing characteristics with surface float                                                                             |
| sweep angle             | the angle that the wing leading edge or quarter chord line makes with the plane of symmetry                                    |
| thrust power            | or $P_{TH}$ , thrust times velocity                                                                                            |
| time history            | plotted data showing the variation of a parameter with time, such as angle of attack versus time                               |

|                               |                                                                                                                   |
|-------------------------------|-------------------------------------------------------------------------------------------------------------------|
| time to half/double amplitude | the number of seconds required for a displacement to decrease/increase to one-half/twice the initial displacement |
| trim speed band               | a range of speeds where no stick force is required to stabilize                                                   |
| trimmed                       | the state of having relieved control forces through the use of trim devices                                       |
| vertical axis                 | the airplane axis oriented up and down (when in level flight)adjective describing                                 |
| yaw rate                      | yawing motion measured in deg/sec                                                                                 |
| yaw damper                    | a device used to detect yaw rates and make countering rudder inputs, to damp dutch roll                           |
| yaw                           | airplane motion about the vertical axis                                                                           |

**APPENDIX C**  
**LIST OF ACRONYMS**

## LIST OF ACRONYMS

|      |                                   |
|------|-----------------------------------|
| a.c. | aerodynamic center                |
| AOA  | angle of attack, $\alpha$         |
| AOB  | angle of bank                     |
| CG   | center of gravity                 |
| c.p. | center of pressure                |
| EAA  | Experimental Aircraft Association |
| FAA  | Federal Aviation Administration   |
| FCS  | flight control system             |
| ILS  | Instrument Landing System         |
| MAC  | mean aerodynamic chord            |
| PIO  | pilot induced oscillation         |
| S&C  | stability and control             |
| SHSS | steady heading sideslip           |
| TAS  | true air speed                    |

**APPENDIX D**

**LIST OF SYMBOLS**

## LIST OF SYMBOLS

|                |                                                                                                  |
|----------------|--------------------------------------------------------------------------------------------------|
| $\alpha$       | angle of attack (usually in deg)                                                                 |
| $\beta$        | sideslip angle (usually in deg)                                                                  |
| $\rho$         | density of air (usually in slugs/ft <sup>3</sup> )                                               |
| $a_{t,w}$      | lift curve slope of tail, wing respectively (units of lift coefficient per degree or per radian) |
| $AR$           | aspect ratio (dimensionless)                                                                     |
| $b$            | wing span (ft)                                                                                   |
| $\bar{c}$      | mean aerodynamic chord (MAC) (ft)                                                                |
| $\hat{c}$      | mean geometric chord (ft)                                                                        |
| $C_L$          | coefficient of lift                                                                              |
| $C_{L_{\max}}$ | maximum coefficient of lift                                                                      |
| $C_D$          | coefficient of total drag                                                                        |
| $C_{D_i}$      | coefficient of induced drag                                                                      |
| $C_{D_o}$      | coefficient of parasitic drag                                                                    |
| $C_M$          | pitching moment coefficient                                                                      |
| $C_Y$          | coefficient of side force                                                                        |
| $C_N$          | coefficient of yawing moment                                                                     |
| $C_l$          | coefficient of rolling moment                                                                    |

|                |                                                                             |
|----------------|-----------------------------------------------------------------------------|
| $\delta a$     | lateral control deflection                                                  |
| $\delta r$     | rudder deflection                                                           |
| $\Delta$       | Greek letter delta, when prefacing a variable means "change in"             |
| $\epsilon$     | downwash angle (deg)                                                        |
| $e$            | span efficiency                                                             |
| $g$            | acceleration of gravity (32.17 ft/sec <sup>2</sup> )                        |
| $l_t$          | distance from the vertical tail to the CG (ft or in)                        |
| $L$            | lift (in lb), or rolling moments (ft-lb) depending on context               |
| $L_p$          | roll damping (1/sec)                                                        |
| $L_{\delta a}$ | rolling moment per degree of aileron deflection (deg/sec <sup>2</sup> /deg) |
| $M$            | pitching moment (ft-lb)                                                     |
| $N$            | yawing moments (ft-lb)                                                      |
| $\phi$         | common symbol for bank angle (deg)                                          |
| $p$            | roll rate (deg/sec)                                                         |
| $p_{ss}$       | maximum steady state roll rate                                              |
| $\dot{p}$      | roll acceleration (deg/sec <sup>2</sup> )                                   |
| $q$            | dynamic pressure (lb/ft <sup>2</sup> )                                      |
| $\phi/\beta$   | roll-to-yaw ratio                                                           |
| $S, S_t$       | wing area, tail area (ft <sup>2</sup> )                                     |
| $\tau$         | roll mode time constant (sec)                                               |
| $T$            | period of a dynamic mode (sec)                                              |
| $V, V_T$       | true airspeed (kt or ft/sec)                                                |

|               |                                                                 |
|---------------|-----------------------------------------------------------------|
| $V_{mc}$      | static minimum control speed (engine failed, kt)                |
| $V_{mc(dyn)}$ | dynamic minimum control speed (engine failed, kt)               |
| $V_{mcg}$     | dynamic minimum control speed on the ground (engine failed, kt) |
| $V_S$         | stall true airspeed                                             |
| $\bar{V}$     | tail volume coefficient                                         |

## **VITA**

Richard Noble Clark was born in Milwaukee, Wisconsin on October 5, 1951. He attended public schools in the Wauwatosa, Wisconsin school system through high school graduation in June, 1970. In September, 1970, he enrolled in the University of Wisconsin-Milwaukee. He graduated in December, 1973 with a Bachelor's Degree in Applied Mathematics and Physics. He joined the U.S. Navy in November, 1974, where he became an F-4 and F-14 carrier pilot, accumulating nearly 500 carrier landings. In July, 1985, he entered the 11-month course at the United States Naval Test Pilot School (USNTPS), Patuxent River, Maryland. Following the completion of that course, he has participated in numerous flight test programs.

He has returned to USNTPS staff, and is presently serving as the Chief Flight Instructor.