19
Introduction to Stability and Control Flight Testing

19.1 Introduction

In order to perform its intended mission an aircraft must have adequate stability and control. The quantity of each of these items will depend on the mission or purpose of the aircraft. For instance, a transport aircraft is interested in a high stability level in order to give the passengers a smooth ride through turbulence, while at the other end of the spectrum, the fighter aircraft needs a high level of controllability for air combat maneuvering. So, like aircraft performance, the stability and control is a function of the aircraft mission. As a result, the applicable regulations that govern the levels of stability and control are somewhat different depending on the aircraft's mission.

19.2 Regulations

For civilian airplanes, FAR Part 23 specifies the requirements for light aircraft under 12,500 lb gross weight.

For transport aircraft or aircraft over 12,500 lb gross weight, FAR Part 25 specifies the level of stability and control.

Stability and control requirements for military airplanes are covered in Military Specification MIL-F-8785. Although this specification covers all military airplanes, except some army airplanes that are certified under the FAA Regulations, it specifies certain classes and categories of airplanes. The requirements of the specification are different for each class of airplanes. These classes and categories are based on the airplane mission.

19.3 Reference Axes Systems

In order to quantify stability and control parameters and have a common reference system, several systems of axes have been established. Fig. 19.1 (Ref. 1) illustrates one of the axis systems in common use known as the body axis system. Other axes systems referenced to the Earth's surface or inertial space are also useful in evaluating stability and control but are not used in this text. For the body axis system the X axis, or longitudinal axis, runs fore and aft in the aircraft and is located on a plane of symmetry. The positive direction for this axis is in the direction of flight. Motion about the longitudinal axis is called roll and is positive when it is to the right as viewed from the cockpit. The notation for a rolling moment about the longitudinal axis is the capital letter L. The Z or vertical axis is also in the plane of symmetry and the

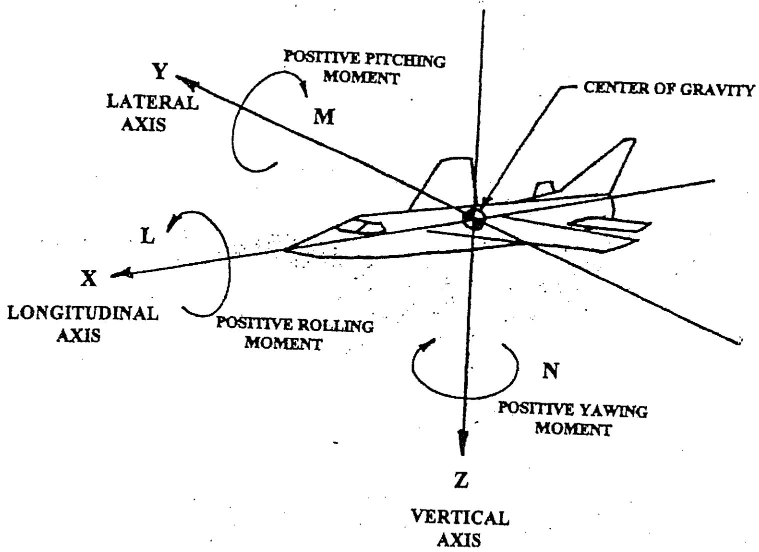

Fig. 19.1 Body axis reference system.[1]

positive direction is down. This convention was established for the aerodynamic theorist by the NACA; however, in the real world of flight test we generally consider up as the positive direction. This is because down is where the ground is and no one wants a collision with the ground; therefore, how could down be positive? A moment about the vertical axis is called a yawing moment. Its notation is the capital letter N and is also positive to the right when viewed from the cockpit. The Y or lateral axis is perpendicular to the plane of symmetry and is positive to the right side of the aircraft. A moment about this axis is a pitching moment and is positive in the nose up direction. It is noted by the capital letter M.

19.4 Definitions of Stability and Controllability

The stability of an airplane can be divided into two basic types: 1) static stability; and 2) dynamic stability.

19.4.1 Static Stability[1,2]

An airplane is said to exhibit positive static stability if, when displaced from a condition of equilibrium, it has a tendency to return. We call the condition of equilibrium the trim condition. If the airplane has a tendency to continue its movement when displaced from a condition of equilibrium we can say that it exhibits negative static stability. If the airplane exhibits neither a tendency to return nor a tendency to continue its movement when disturbed then we say it exhibits neutral static stability. Fig. 19.2 gives a visual presentation of the three types of static stability. The static stability may be further divided into stick-fixed or stick-free static stability, since the airplane will exhibit different levels of stability with the controls fixed or the controls free. In flight testing we cannot actually measure these types of stability so we measure things that are measurable and are indications of them.

For the stick-fixed case we measure the control surface position that, by its direction of movement, is an indication of positive or negative stick fixed stability.

The magnitude and direction of control force is an indication of positive or negative stick-free static stability.

These two indications of stability are sometimes referred to as control position stability and control force stability. They are definitely things to which the pilot relates when flying the airplane and are easily measured in flight test.

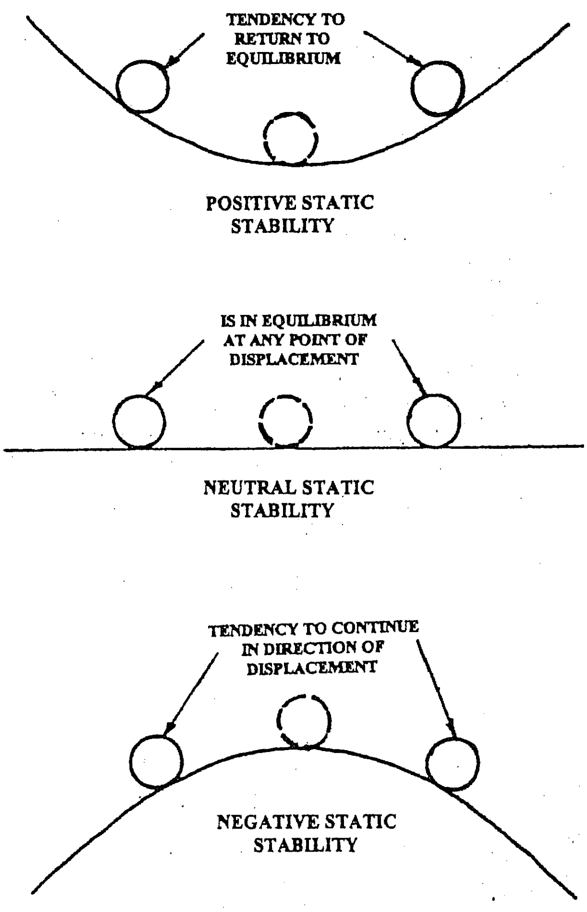

Fig. 19.2 Static stability.[1]

19.4.2 Dynamic Stability

While static stability is concerned with the tendency of the airplane to return to trim, dynamic stability can be defined as the resultant motion of the airplane with time, after being displaced from trim. An airplane is said to display positive dynamic stability if the amplitude of the resulting motion decreases with time. The various possibilities of dynamic behavior are shown in Fig. 19.3. If the airplane is disturbed from trim and the motion subsides without oscillation, the dynamic stability is said to be deadbeat and is positive. If the aircraft is displaced from trim and the amplitude continues to increase

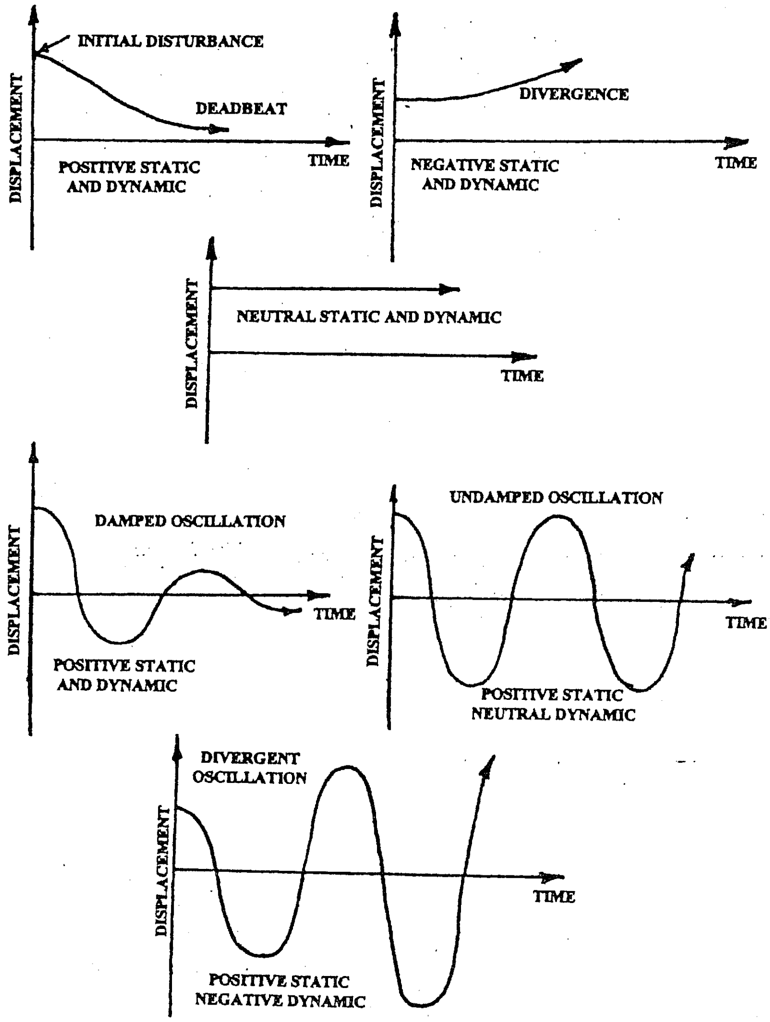

Fig. 19.3 Dynamic stability.[1]

without oscillation, the dynamic stability is said to be divergent without oscillation. In this case both the dynamic stability and static stability are negative. If we displace the aircraft from trim and it remains displaced without oscillation or change in amplitude it exhibits neutral dynamic and neutral static stability. The dynamic stability modes just described can be called nonoscillatory modes. We might then expect to have oscillatory modes. To have an oscillatory dynamic stability mode we must have positive static stability or a tendency to return to trim. A damped oscillatory mode is said to exhibit positive dynamic stability while a divergent oscillation shows negative dynamic stability. An undamped oscillation shows neutral dynamic stability. Dynamic stability may be described mathematically by equations similar to those for a spring-mass-damper system. The roots of such equations determine the modes of dynamic stability.

19.4.3 Trim[1]

Now let us turn our attention back to the trimmed condition. We mentioned earlier that the trimmed condition was a condition of equilibrium. If an aircraft is "in trim," we can say that the aircraft moments in pitch, roll, and yaw are all equal to zero. The ability to establish an "in trim" condition for the various flight conditions is a function of the aircraft controls and may be accomplished by pilot effort, trim tabs, or bias of a control surface actuator. However, when we speak of being trimmed it is generally taken to mean a movement of the trimming device to achieve a zero control force or hands-off condition.

19.4.4 Controllability[1]

Controllability can be defined as the ability of the aircraft to respond to control movements. There is a definite relation between the stability of an aircraft and its controllability. This relationship is sometimes misunderstood in the aviation community. It is sometimes thought that if an airplane has strong positive stability it is also controllable. This is not true. An airplane with strong positive stability is very difficult to control since it has a strong resistance to being disturbed from the trim condition. However, if the airplane has strong negative stability it also may be uncontrollable since it will not stay in any trim condition.

To get a little better understanding of this let us take another look at Fig. 19.2. If we try to maneuver the ball on the concave surface it always wants to return to the center, and to maintain it at some condition off-center would require that we constantly hold it there. This is the case of the airplane with strong positive stability. It requires considerable effort to maneuver.

The ball on the flat surface represents the airplane with neutral stability. If we move this ball it stays where we move it. It is also easy to move. We might say, then, that this case represents the peak in controllability. Control may be done precisely with little effort. It is for this reason that airplanes that need to be very controllable, like military fighters and aerobatic airplanes, have near-neutral stability.

The ball on the convex surface represents an airplane with strong negative stability. If we apply a force to displace this ball, we will rapidly need to apply a force in the opposite direction to stop its movement. If we remove that force, the ball will continue moving without us taking any action. As you can see, this ball can get out of control very rapidly.

We may sum up then by saying that to have an airplane that is highly controllable we do not want either strong positive or strong negative stability but one that has near neutral stability. However, for more mundane missions like cross-country flying the pilot likes to have positive stability, so a trade-off exists between stability and controllability that depends on the airplane's mission.

19.5 Relation of Stability and Control to the Aircraft's c.g. Envelope

The allowable movement or shift in the aircraft's c.g. due to different cargo, passenger, or fuel loads, called the center of gravity envelope (Fig. 19.4), is dependent on the level of stability and control exhibited by the aircraft.

For aircraft with conventional tail locations (other than canard configured aircraft) positive stability decreases as the c.g. moves aft. We can then say that the aft c.g. limit is generally set by the minimum acceptable level of positive stability. There are cases, however, where other items like stall characteristics or spin recovery characteristics may establish this limit.

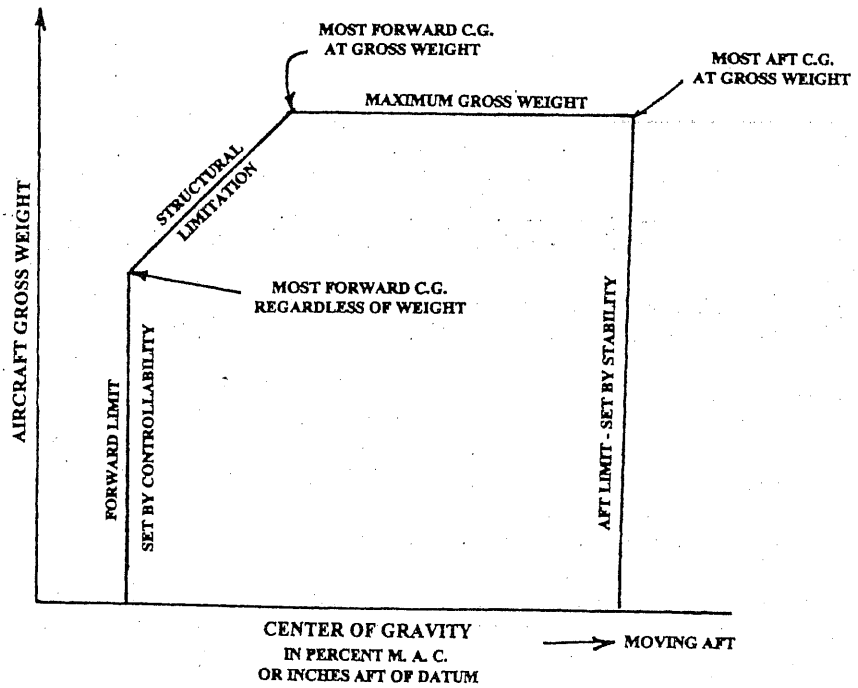

Fig. 19.4 Aircraft c.g. envelope.

As the c.g. moves forward the airplane becomes more stable and less controllable. Therefore, a controllability function such as nosewheel liftoff for takeoff or elevator required to land will establish this forward limit.

As you can see, in flight testing of stability and control it is very important to know the location of the aircraft's c.g. precisely.

19.6 Control System Characteristics

The control system characteristics play an important part in the stability and control of the aircraft as seen from the pilot's perspective. Two of the most important characteristics are control system friction and breakout force.

19.6.1 Control System Friction

It is desirable to keep control system friction to a minimum as it may mask low levels of static stability. In flight, high control system friction will make the airplane difficult to trim precisely. In pitch this will result in a large band of airspeed where the airplane appears to be in trim. During the design and fabrication of the control system considerable effort should be expended to reduce control system friction to a minimum and the friction level should be checked and reduced to a minimum prior to starting flight testing.

19.6.2 Control System Breakout Force

A low level of breakout force in the control system may be desirable so that the pilot has some feedback from the control system should there be a distractioin and an unwanted force then applied. However, these forces should not be excessive or they may lead to over controlling or pilot-induced oscillations. Breakout force levels are somewhat pilot subjective so several pilots should be used to determine if they are acceptable.

19.6.3 Control System Gearing

The gearing ratio of the control system may be used to improve or reduce apparent levels of static stability on airplanes with reversible control systems. However, by the time the airplane reaches the flight test stage it is unlikely that this gearing ratio can be changed.

References

[1]Hurt, H. H., Jr., "Aerodynamics for Naval Aviators," NAVAIR 00-80T-80, U.S. Navy, U.S. Government Printing Office, Washington, D.C., 1960, rev. Jan. 1965.

[2]Langdon, S. D., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-NO. 103, 1 Aug. 1969, rev. 1977.

20
Static Longitudinal Stability Theory

20.1 Introduction

When we speak of longitudinal motion we are speaking about motion in the plane of symmetry of the aircraft, or motion about the lateral Y axis. For small disturbances, longitudinal motion does not generally couple with motion about other axes and can therefore be handled as two-dimensional motion which greatly simplifies its analysis.

In discussing the longitudinal stability and control, we subdivide the discussion into maneuvering and nonmaneuvering tasks.[1] The maneuvering tasks involve both static and maneuvering longitudinal stability and will be left for a later discussion. The nonmaneuvering tasks are tasks such as:

1) takeoff
2) climb
3) cruise
4) holding
5) gliding
6) descents
7) approach
8) go-arounds

These are items that do not involve much maneuvering and are primarily affected by the static longitudinal stability of the airplane. These tasks are also more affected by the long period dynamic stability than are the maneuvering tasks.

During this discussion we will also be speaking of reversible and irreversible control systems. A reversible control system can be defined as one in which a movement of the pilot's controls will move the aerodynamic control surfaces, and a movement of the aerodynamic controls will correspondingly cause a movement of the pilot's controls. The control system then is rigidly connected together. In the irreversible control system the connection between the controls is not rigid but through either hydraulic or electric actuators. In this system the pilot's controls will move the aerodynamic control surfaces, but an external movement of the aerodynamic control surfaces will not move the pilot's controls.

Also in our discussion of static longitudinal stability we will talk about the stability of the airplane in two cases: 1) with the controls fixed; and 2) with the controls free.

20.2 Stick-Fixed Static Longitudinal Stability

In discussing stick-fixed stability, we are saying that the elevator is fixed in position and not free to float with the relative wind. In this condition the variation of pitching moments C_m about the aircraft's c.g. with changing lift coefficient C_L is a function of the pitching moments of the individual components of the airplane for the given C_L. In other words, we may sum the individual components of pitching moment due to the wing, fuselage, nacelles, horizontal tail, powerplant, and so forth, and come up with the way the airplane pitching moment varies with C_L. Before discussing the total airplane, let us examine the way the pitching moment of the individual components vary with C_L.

First, let us look at the wing.[3,4] In evaluating the wing we must first determine the wing aerodynamic center (a.c.) or the point on the wing chord where the pitching moment due to the wing remains constant with changing lift coefficient. This is also the point through which we can consider the lift to act. Once we have determined this point we can see from Fig. 20.1 (Ref. 3) that as long as the c.g. stays ahead of the a.c., the wing pitching moment will be nose down, or a restoring (stable) moment. If the c.g. is aft of the a.c. the moment is nose up or unstable. Since the normal aircraft center of gravity range may vary from 10–40% MAC and the a.c. is usually located in the vicinity of 25% MAC, the wing's contribution could be either stable or unstable. However, if we consider the worst case of aft c.g. it will generally be unstable. This is shown in Fig. 20.2 (Ref. 3).

The fuselage contribution is also generally unstable.[3,4] This is due to its shape and the upwash and downwash of the wing acting on it, causing nose up pitching moments with increasing C_L. It is the fuselage ahead of the wing that is the largest problem, and long noses should be avoided where possible. Fig. 20.2 shows the additional unstable contribution of the fuselage to the overall pitching moment.

Nacelles act similar to a fuselage and may add considerable unstable pitching moment when located ahead of the c.g.

The effects of engine thrust depend upon the location of the thrust line with respect to the vertical location of the c.g. If the thrust line is located above the c.g. then the thrust effect is stabilizing. If below the c.g. it is destabilizing. In either case, however, the thrust line is usually so close to the c.g. that the thrust effect is not very significant.

The element of producing thrust that is significant is the force generated by the turning of the air when it comes through the propeller discs or when it enters a jet intake. This effect and its resultant forces are shown in Fig. 20.3.[3] As we can see from this figure the force created by turning the air generates a nose up or unstable moment about the c.g. This force is usually greater than any stabilizing force created by the thrust, so we can say that for jets and tractor-configuration, propeller-driven aircraft the overall power effects are generally destabilizing. A pusher configuration of a propeller-driven aircraft or jet engine nacelles mounted on the aft fuselage can reverse these effects and create stabilizing moments as shown in Fig. 20.4.

In our discussion so far, nearly everything has created instability in our airplane, and the only component we have yet to discuss is the horizontal tail.

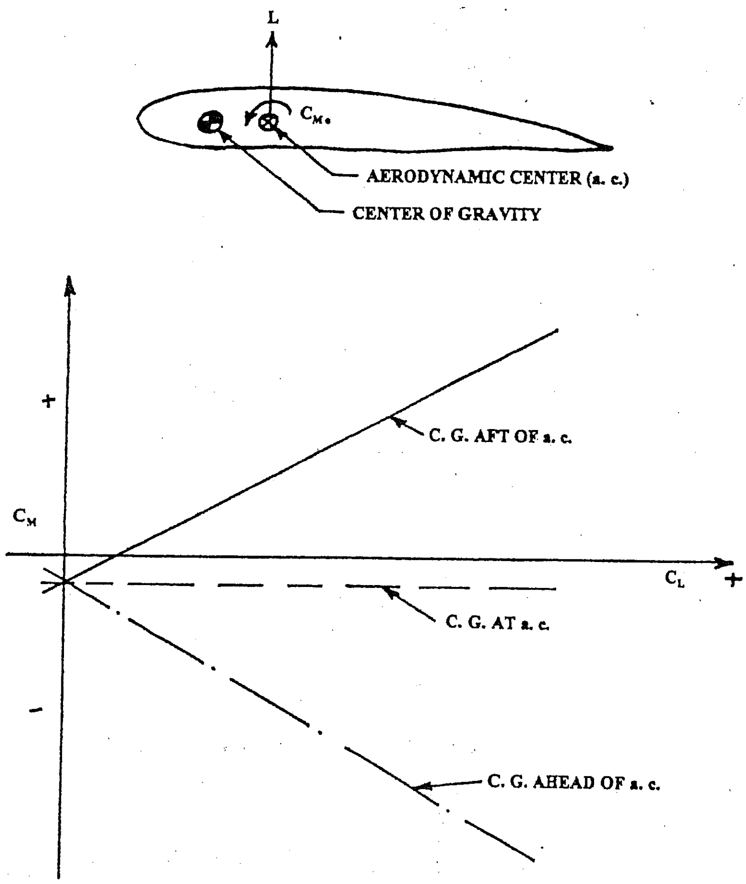

Fig. 20.1 Wing contribution to stability.[3]

Therefore, we can deduce that it is the horizontal tail that provides the stabilizing moments to provide positive stability for the entire aircraft. Fig. 20.2 shows the pitching moment coefficient for the tail as a function of C_L and its contribution to the overall aircraft pitching moment. As can be seen from this figure, the contribution of the horizontal tail to static longitudinal stability is powerful and stabilizing.

A term that is used frequently by airplane designers to give an indication of the power of the horizontal tail stabilizing the airplane is the tail volume coefficient,[1,2,4] which is determined by the following terms:

$$\overline{V_H} = \left(\frac{S_t}{S_W}\right)\left(\frac{l_t}{C}\right) \tag{20.1}$$

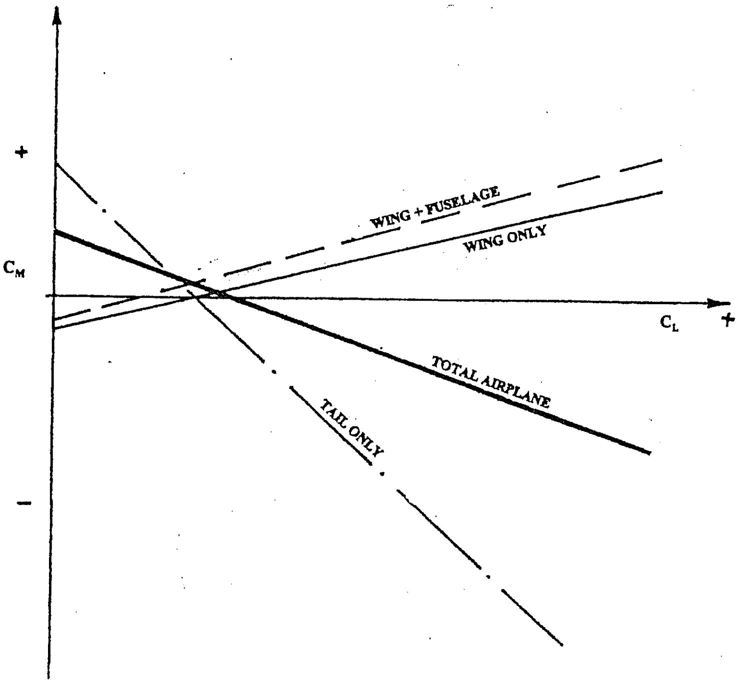

Fig. 20.2 Contribution of airplane components to stability.[3]

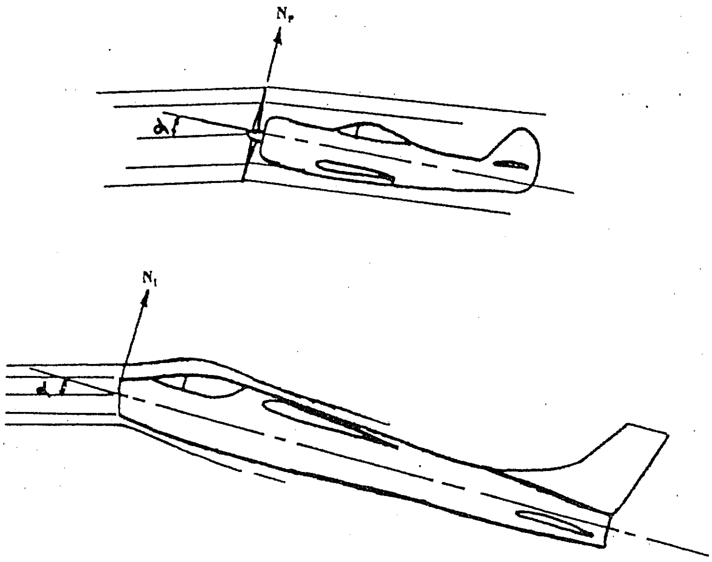

Fig. 20.3 Normal force created by propeller or jet intake.[3]

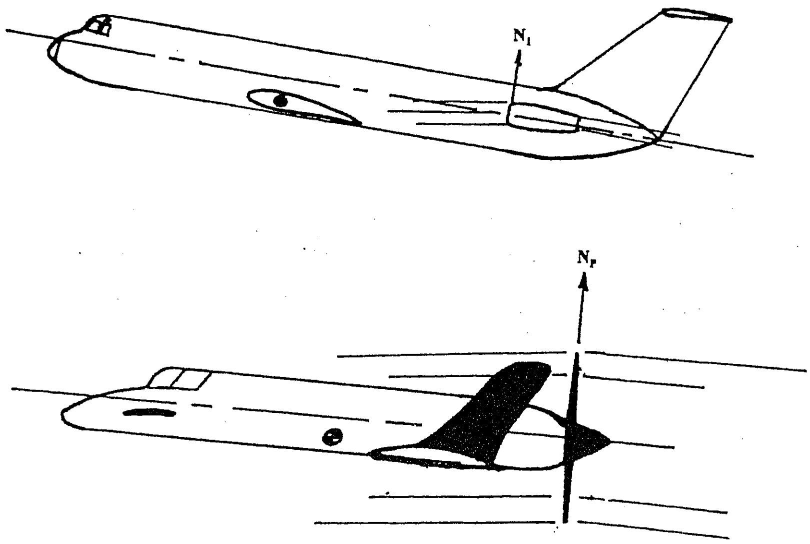

Fig. 20.4 Effects of aft-mounted jet engines or propellers.

where
$\overline{V_H}$ = the tail volume coefficient
S_t = the horizontal tail area
S_W = the wing area
l_t = the horizontal distance from the wing a.c. to the tail a.c.
C = the wing mean aerodynamic chord

This coefficient is used in comparing a new aircraft design to an existing design. One must be careful in using this coefficient because the location of the tail in the vertical direction also has considerable effect on longitudinal stability. For a conventional configuration, a T-tail that is up out of the strongest part of the wing downwash is more stable than a low tail design. Therefore, in using the tail volume coefficient for comparison purposes one must be careful to compare tails of similar vertical location.

The tail efficiency factor η_t is another measure of the tail effectiveness. This factor is obtained by comparing the dynamic pressure at the tail q_t with the free-stream dynamic pressure.[1,2]

$$\eta_t = \frac{q_t}{q_\infty} \tag{20.2}$$

For configurations in which the horizontal tail is located in the wing wake, or area of high downwash, the dynamic pressure at the tail is reduced. For these configurations the tail efficiency factor is generally taken to be about 0.9. For T-tail configurations it more nearly approaches 1.0.

We must also keep in mind when discussing the horizontal tail that the angle of attack of the horizontal tail is not the same as the wing. This is because of the differences in wing and tail incidence and the downwash created by the wing.

Again, by examining Fig. 20.2, we can see that the moment coefficient of the horizontal tail provides a strong negative slope (stabilizing) that gives the overall airplane a pitching moment coefficient with C_L which has a negative (stable) slope. The airplane is said to be in trim at the point where the pitching moment curve crosses the horizontal axis ($C_m = 0$).

The equation that defines the slope of the pitching moment curve for the entire airplane in gliding flight is given as follows:[1,2]

$$\frac{\mathrm{d}C_{m_{c.g.}}}{\mathrm{d}C_L} = \frac{X_a}{\overline{C}} + \left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{fus} + \left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{nac} - \left(\frac{a_t}{a_w}\right)\overline{V_H}\eta_t\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right) \tag{20.3}$$

where

X_a/C = wing contribution, which is a measure of the location of the a.c. in relation to the c.g.

a_t = lift curve slope of the horizontal tail

a_w = lift curve slope of the wing

$d\varepsilon/d\alpha$ = the change in downwash with angle of attack change

All of the above terms are fixed except for the wing term, which can be varied by moving the center of gravity. If we move the center of gravity in a direction that will make the wing term become larger in the positive direction (c.g. moving aft), we will eventually reach a point where the dC_m/dC_L curve slope will be zero. This c.g. location is called the stick-fixed neutral point N_0 or the a.c. for the total airplane. With the stick-fixed neutral point defined we can determine the slope of the pitching moment curve by the following relation:[1,2]

$$\frac{\mathrm{d}C_{m_{c.g.}}}{\mathrm{d}C_L} = \frac{X_{c.g.}}{\overline{C}} - N_0 \tag{20.4}$$

where

$X_{c.g.}/\overline{C}$ = the location of the aircraft's c.g. in % MAC

The distance between the actual aircraft's c.g. and the neutral point as expressed in Eq. (20.4) is called the stick-fixed static margin.

20.3 Longitudinal Control

If we examine the $\mathrm{d}C_m/\mathrm{d}C_L$ curve in Fig. 20.2 for the total airplane, we can see that it is only in trim ($C_m = 0$) at one point, or one value of C_L. The airplane would not be a very useful vehicle if it could only fly at one value of C_L, so we must have some method of flying at a variety of lift coefficients. If

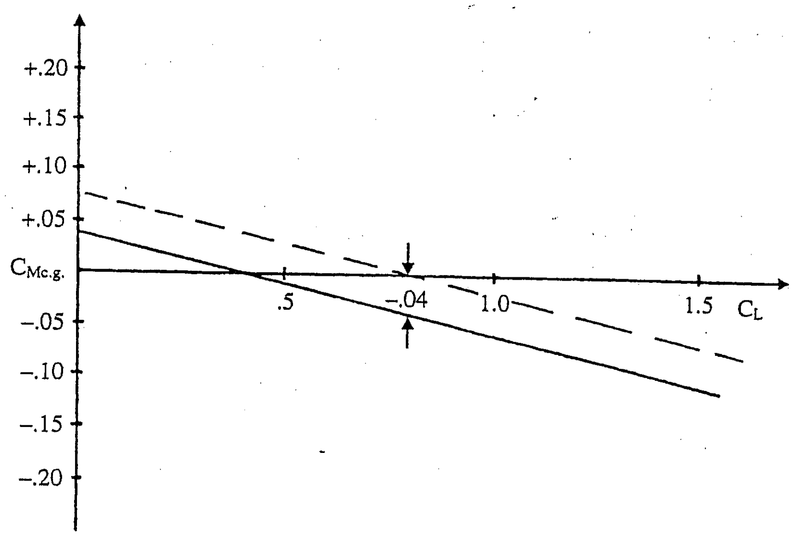

Fig. 20.5 Longitudinal control mechanics.[1]

we examine Fig. 20.5 (Ref. 1) we see the pitching moment curve for a given airplane with the airplane in trim at a lift coefficient of about 0.4. If the pilot wanted to slow the airplane down and operate at a trimmed lift coefficient of 0.8, he would have to provide an additional increment of pitching moment coefficient of + .04 to overcome the negative pitching moment coefficient of −.04 existing at a C_L of 0.8 (Ref. 1).

If we take another look at Eq. (20.3) and write it as follows:[1]

$$C_{m_{c.g.}} = C_{m_{a.c.}} + \frac{X_a}{\bar{C}} C_L + (C_{m_{c.g.}})_{fus} + (C_{m_{c.g.}})_{nac} - a_t \alpha_t \eta_t \overline{V_H} \qquad (20.5)$$

we can see that there are three terms in this equation that might be used to change the trim point. They are: 1) the wing pitching moment $C_{m_{a.c.}}$, 2) the c.g. location X_a/C, and 3) the tail angle of attack α_t.

The wing pitching moment about its aerodynamic center $C_{m_{a.c.}}$ is a function of the wing camber and wing twist. We can control wing camber with such devices as leading- and trailing-edge flaps, but since we wish to use these devices to control available lift coefficient we would prefer not to use them for aircraft control. This is the method used by tailless aircraft, however.

We can change the X_a/C term by shifting the c.g. This is the control method used by many of the current generation of hang gliders. There are several drawbacks to this method of control. First, there are physical limits on the amount that we can shift the c.g. Second, as we saw earlier, a shift in the c.g. changes the stability level of the airplane and this is not desirable.

The last of the three methods is to change the tail angle of attack. This can be done by moving the entire horizontal tail, as is the case with a stabilator, or

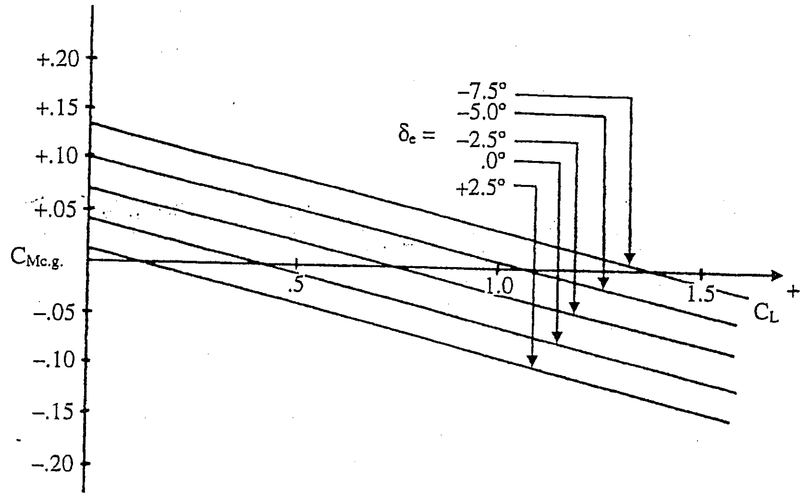

Fig. 20.6 Variation of $C_{m_{c.g.}}$ vs C_L for various elevator angles.[1]

by moving a hinged surface at the trailing edge of the horizontal tail called the elevator. Both of these methods produce large changes in pitching moments. The stabilator is the more powerful device. It is used in applications where the c.g. has considerable forward travel or the wing pitching moment about the a.c. is large, such as in supersonic flight. A very significant fact about this form of control is that, in addition to being powerful, it also does not significantly change the longitudinal stability. Therefore, we may have a series of C_m vs C_L curves corresponding to each elevator angle as is shown in Fig. 20.6 (Ref. 1). This allows us to have as many "trim points" in the unstable C_L range as we desire.

20.4 Elevator Position Stability

The in-flight measurement of pitching moments about the c.g. is a difficult task. This type of measurement is more suited to a wind tunnel. However, since there are few wind tunnels available large enough for full-scale airplanes and subscale tunnel models do not always provide correct answers due to Reynolds number and other effects, it is necessary to have a means to measure longitudinal stability in flight.

If we plot the elevator position vs lift coefficient for each value of trim point ($C_m = 0$) in Fig. 20.6, we will obtain a plot similar to that shown in Fig. 20.7 (Ref. 1).

This curve may be expressed by the equation:[1]

$$\delta_e = \delta_{e_{C_L=0}} - \frac{\left(\dfrac{dC_m}{dC_L}\right)_X}{C_{m_{\delta_e}}} C_L \tag{20.6}$$

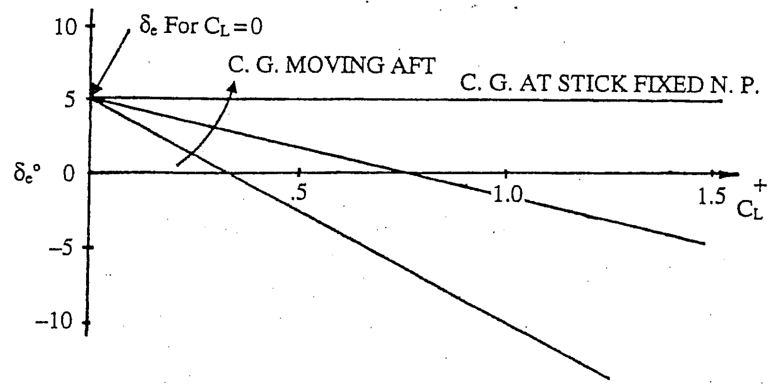

Fig. 20.7 Elevator position vs C_L for various c.g. positions.[1]

where

δ_e = the elevator deflection
$\delta_{e_{C_L=0}}$ = the elevator deflection for zero lift coefficient
$(dC_m/dC_L)_X$ = the slope of the pitching moment coefficient vs C_L curve for the airplane
$C_{m\delta e}$ = the pitching moment coefficient due to elevator deflection, or the control power coefficient

For a stabilizer, elevator configuration:[4]

$$C_{m_{\delta_e}} = -a_t \tau \eta_t \overline{V_H} \tag{20.7}$$

where
τ = the tail effectiveness factor which is equal to 1.0 for a stabilator

The term $\delta_{e_{C_L=0}}$ is a constant. Although this value cannot be obtained in flight, it is important to know this fact as it will help us in fairing δ_e vs C_L flight data. If we differentiate Eq. (20.6) with respect to C_L we have:[1]

$$\frac{d\delta_e}{dC_L} = \frac{\left(\dfrac{dC_m}{dC_L}\right)_X}{C_{m_{\delta_e}}} \tag{20.8}$$

This equation can be called the elevator position stability equation. From this equation we can see that when $dC_m/dC_L = 0$ (the term used to define the stick-fixed neutral point) then the slope of the elevator position vs C_L curve will be zero ($d\delta_e/dC_L = 0$). We may then use this relation to find the stick-fixed neutral point by flight test. We can also deduce that the slope of the elevator position vs C_L curve will give us the sign if not the magnitude of the stick-fixed stability. Also, if we know the value for the elevator control power coefficient, we can determine the magnitude of the stability.

20.5 Stick-Free Longitudinal Stability

In our previous discussion we only discussed the case in which the elevator was rigidly fixed in a trim position. We related the level of stability in that case to the variation of elevator position with airspeed or lift coefficient. We now turn our attention to what happens to the static longitudinal stability when the elevator is freed and allowed to "float." This case is called stick-free longitudinal stability. This type of stability is usually only associated with airplanes that have reversible control systems, since in an irreversible control system the elevator is never really free to float. However, some of these systems have stability augmentation devices that move the control surface without pilot action. This movement is sometimes considered control surface float.

In evaluating the stick-free longitudinal static stability of a reversible control system, the control surface float is a function of the moments generated by the elevator about its hinge line. These moments, called hinge moments, are caused by two factors. The first of these factors is the tendency of the elevator to streamline itself with the relative wind. This is essentially the variation of elevator hinge moment with horizontal tail angle of attack. This moment expressed in its coefficient form is known as $C_{h_{\alpha_t}}$, or the hinge moment coefficient due to horizontal tail angle of attack.

The second factor affecting the float of the elevator is the hinge moment generated by elevator deflection when the horizontal tail is at zero angle of attack. This moment expressed as a coefficient is $C_{h_{\delta_e}}$, or the hinge moment due to elevator deflection.

The total elevator hinge moment may be expressed by the coefficient equation:[4]

$$C_{h_e} = C_{h_0} + C_{h_{\alpha_t}}\alpha_t + C_{h_{\delta_e}}\delta_e + C_{h_{\delta_t}}\delta_t \tag{20.9}$$

where

C_{he} = the total elevator hinge moment coefficient
C_{ho} = the elevator hinge moment due to camber, zero for a symmetrical section
$C_{h_{\delta_t}}$ = the elevator hinge moment due to trim tab deflection
δ_t = the trim tab deflection

When the elevator is in equilibrium the total hinge moment is zero, and the floating tendency is canceled by the "restoring" tendency. When this condition exists we may find the float angle by the equation:[1]

$$\delta_{e_{float}} = -\frac{C_{h_{\alpha_t}}}{C_{h_{\delta_e}}}\alpha_t \tag{20.10}$$

If the elevator tends to float with the relative wind ($C_{h_{\alpha_t}}$ and $C_{h_{\delta_e}}$ both negative), then the static longitudinal stability of the airplane is reduced with respect to the stick-fixed case. The stick-free longitudinal static stability can be expressed by the equation:[1]

$$\left(\frac{\mathrm{d}C_{m_{c.g.}}}{\mathrm{d}C_L}\right)_{free} = \left(\frac{\mathrm{d}C_{m_{c.g.}}}{\mathrm{d}C_L}\right)_{fixed} + C_{m_{\delta_e}}\left(\frac{\mathrm{d}\delta_{e_{float}}}{\mathrm{d}C_L}\right) \tag{20.11}$$

The terms $C_{m_{\delta_e}}$ and $\mathrm{d}\delta_{e_{float}}/\mathrm{d}C_L$ are normally negative. This causes the stick-free static stability to be less than the stick-fixed static stability.

Since the stick-free stability contains the stick-fixed terms, it, too, is affected by center of gravity position. If we move the c.g. position far enough aft we reach a point where $(\mathrm{d}C_{m_{c.g.}}/\mathrm{d}C_L)_{free} = 0$. This point is called the stick-free or elevator-free neutral point N_0'. Once we have determined the stick-free neutral point we can determine the stick-free longitudinal static stability for any c.g. position from the equation:[1]

$$\left(\frac{\mathrm{d}C_{m_{c.g.}}}{\mathrm{d}C_L}\right)_{free} = \frac{X_{c.g.}}{\bar{C}} - N_0' \tag{20.12}$$

The distance between the c.g. location and the stick-free neutral point is called the stick-free static margin.

20.6 Control Force Stability

Since none of the above stick-free equations provides us with variables that we can readily measure in flight tests, we need some method to relate flight measurable variables to the stick-free stability level. If we examine a plot of airplane pitching moment coefficient stick free, vs lift coefficient (Fig. 20.8),[1] we can see that for the case given the aircraft is trimmed at a lift coefficient of 0.5. If it was necessary to slow the airplane down and fly at a lift coefficient of 0.8, the pilot would have to move the control so as to overcome the stabilizing pitching moment of 0.03 at that lift coefficient. If the pilot did not choose to change the longitudinal trim setting, then in order to continue to fly at a lift coefficient of 0.8 a force on the longitudinal control would need to be held. This force must be sufficient to move the elevator from its trimmed float position to the position for zero pitching moment coefficient at $0.8C_L$.

The magnitude of this force can then be related to the stick-free static stability by the equation:[1]

$$\frac{\mathrm{d}F_s}{\mathrm{d}V_e} = 2K\frac{W}{S_W}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{free}\frac{V_e}{V_{e_{trim}}^2} \tag{20.13}$$

where

$\mathrm{d}F_s/\mathrm{d}V_e$ = the longitudinal control force variation with equivalent airspeed about the trim airspeed

K = a constant dependent upon control system gearing, elevator size and the horizontal tail efficiency factor ($K = -GS_eC_e\eta_t$)

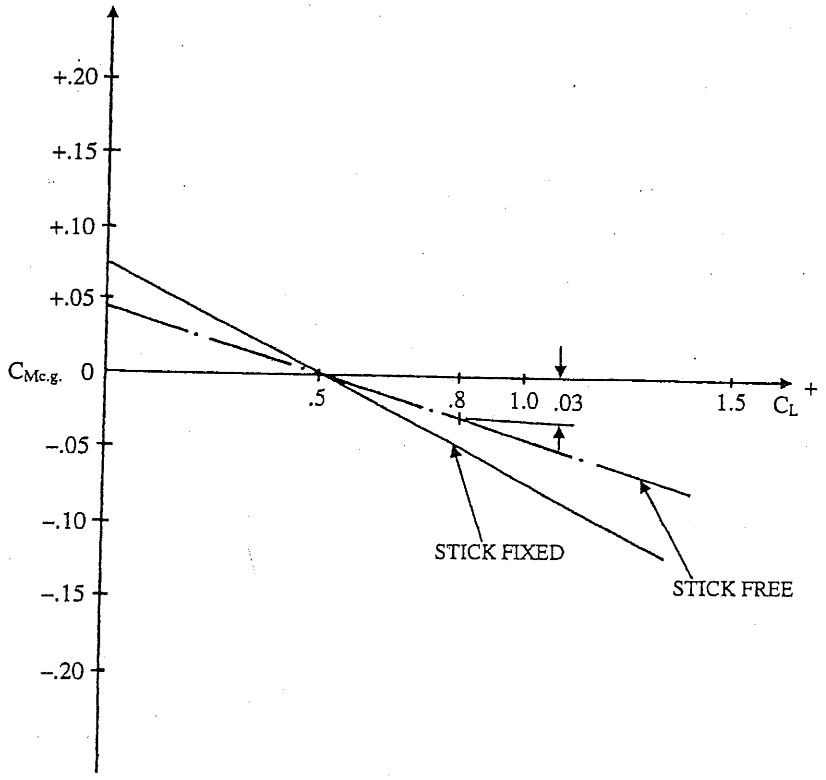

Fig. 20.8 $C_{m_{c.g.}}$ **vs** C_L **stick fixed and stick free.**[1]

As we can see from this equation, in order to determine the actual dC_m/dC_L free we must know the numerical values for $C_{h_{\delta_e}}$ and $C_{m_{\delta_e}}$. These values are not easily determined, so even with the use of Eq. (20.13) it may be difficult to determine the actual stick-free stability level. However, the variation of longitudinal control force with equivalent airspeed will give us an indication of whether the stick-free stability is positive, negative, or neutral.

The plot of dF_s/dV_e is often called stick-force stability or longitudinal control force stability. If we examine the equation we can see that when we are at the stick-free neutral point, $(dC_m/dC_L)_{free} = 0$, that dF_s/dV_e also equals zero. It is this fact we use in flight tests to estimate the stick-free neutral point of the airplane. $dF_s/dV_e = 0$ only corresponds to the stick-free neutral point when the control system does not incorporate control-feel gadgetry such as downsprings or bobweights. The neutral point determined when such devices are in the system is sometimes called the stick-force neutral point, and in such a case does not correspond to the stick-free neutral point.[1]

20.7 Stick-Free Longitudinal Static Stability for an Irreversible Control System[1]

In an irreversible control system the elevator does not float with or against the relative wind as it does in the reversible control system. Therefore, for these systems the term stick-free longitudinal stability is somewhat of a misnomer. It is necessary, however, for the pilot to have the control force sensations of stick-free stability. Several different systems are used to provide this control feel, with one of the most common ones being the extendable link. In this system, an airspeed and altitude sensor senses a change in the trimmed flight condition and signals the extendable link in the control system to expand or contract. This extension or contraction of the control system forces the elevator to move away from its equilibrium control position without moving the pilot's control stick. The pilot must then move the stick to bring the elevator back from its artificial float position to the equilibrium position. This movement provides a force and makes the pilot think a stable longitudinal control force variation about trim has been achieved.

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21 Static Longitudinal Stability Flight Test Methods

21.1 Introduction

From the pilot's standpoint, the static longitudinal stability of an aircraft may be divided into several characteristics. These characteristics include gust stability, speed stability, and flight path stability.

Both gust stability and speed stability are related to the classical stick-fixed and stick-free static longitudinal stability and are dependent on stability margins. They are also affected by friction in the longitudinal control system and by control system gimmicks such as downsprings, bobweights, or artificial stick-force systems.

Flight path stability is related to the pilot's opinion of the aircraft in the approach configuration.

Since gust stability and speed stability are dependent on stability margins, it is worthwhile to determine the neutral point locations in order to fix the aft c.g. limit. We would want this limit to provide us with a useable c.g. travel, while at the same time giving us adequate stability margins. Both of these items are airplane mission dependent.

21.2 Federal Aviation Administration Regulations

The FARs have had requirements for static longitudinal stability since their inception. However, their requirements are only for stick-free longitudinal static stability as demonstrated by longitudinal control force when the airplane is displaced from trim. There are no requirements for stick-fixed longitudinal static stability or flight path stability.

21.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

CAR 3.114 and 3.115 require that for specific flight conditions (which include climb, cruise, and landing), at a specified trim speed, a pull shall be required to obtain and maintain speeds below trim and a push shall be required to obtain and maintain speeds above that trim speed. In addition, the airspeed shall return to within 10% of the original trim speed when the control force is slowly released from any speed within the required range of airspeeds required to demonstrate static stability. The configurations are specified in 3.115 for each flight condition along with the trim airspeeds and ranges of airspeeds for which static stability must be demonstrated.

CAR 3.116 states that instrumented stick-force measurements need not be made when the changes in airspeed are clearly reflected by changes in stick force and these stick forces are not excessive.

21.2.2 Federal Air Regulations Part 23 (Ref. 2)

FAR 23.173 and 23.175 discuss longitudinal static stability and its demonstration. These regulations read much like CAR 3 except that the cruise condition has been expanded to several cruise conditions (high and low speeds) and the landing case has been expanded to include approach. The stable range in climb has changed to be ±15% of the trim speed rather than a multiple of the stalling speed as it is in CAR 3. This regulation makes no mention of stick-force measurements, but does require a stable slope of the stick-force curve. For commuter aircraft the free return speed has been reduced from the ±10% of the trim airspeed to ±7.5% of the trim airspeed. The landing and approach flight conditions are to be measured with both power off and with power for a 3 deg descent.

21.2.3 Advisory Circular 23-8A (Ref. 3)

Advisory Circular 23-8A provides some additional guidance for measurement of static longitudinal stability. It states that if autopilot, or other systems that connect to the longitudinal control system, increase the friction of the control system, then the test should be conducted with these systems installed. The AC also discusses a test method and discusses the determination of the stable control force slope. In this instance the AC mentions use of hand-held force gauges or other methods to measure the force in addition to qualitative measurements by the test pilot. The AC also suggest that data should be collected within a ±2000-ft altitude band.

21.3 Stick-Fixed Neutral Point Determination

21.3.1 Flight Test and Data Reduction Method[4,7]

As was discussed in the section on stick-fixed stability theory, the stick-fixed stability $(\mathrm{d}C_m/\mathrm{d}C_L)_{\text{fixed}}$ can be related to the elevator position δ_e through the relation:[4,7]

$$\frac{\mathrm{d}\delta_e}{\mathrm{d}C_L} = \frac{(\mathrm{d}C_m/\mathrm{d}C_L)_{\text{fixed}}}{C_{m_{\delta_e}}} \tag{21.1}$$

Since $\mathrm{d}\delta_e/\mathrm{d}C_L$ will be zero when $(\mathrm{d}C_m/\mathrm{d}C_L)_{\text{fixed}}$ is zero, the stick-fixed neutral point can be found by moving the aircraft c.g. aft until the plot of δ_e vs C_L has a zero slope. Although it is possible to determine the stick-fixed neutral point by this method, it is not a safe way to approach the problem.

A safer way to approach the problem is to measure the elevator position δ_e vs equivalent airspeed V_e, both above and below some specified trim airspeed, for a number of c.g. positions safely ahead of the neutral point. This should be accomplished for the configurations specified in the FAA Regulations at the trim airspeeds and power settings specified. Once these data have been taken they are plotted and reduced using the sequence shown in Fig. 21.1 (Refs. 4, 8).

Positive stick-fixed, or elevator position, longitudinal stability is not required by the federal air regulations, but is important in determining if the stick-free longitudinal stability can be improved through gimmicks like down-springs or bobweights.

21.4 Stick-Free Neutral Point Determination

As was discussed earlier, the stick-free longitudinal stability $(dC_m/dC_L)_{free}$ can be related to the elevator control force by the relation:[4]

$$\frac{dF_s}{dV_e} = 2K\frac{W}{S_W}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free}\frac{V_e}{V_{e_{trim}}^2} \tag{21.2}$$

From this relation we can see that when we are at the stick-free neutral point $(dC_m/dC_L)_{free} = 0$ then the derivative dF_s/dV_e is also equal to zero. Again, we would prefer not to test at the actual neutral point. Therefore, while we are collecting the data for stick-fixed stability, we also record elevator control force. We then plot elevator control force F_s vs equivalent airspeed V_e as is shown in the first plot of Fig. 21.2 (Refs. 4, 8).

For FAA certification it is not necessary to determine the stick-free, or control force, neutral point. For FAA testing one only needs to plot elevator control force at the control wheel, or control stick, and plot it vs calibrated airspeed as is shown in the first plot of Fig. 21.2. This plot must have a stable slope, as shown, to satisfy the regulations.

As can be seen in Eq. (21.2), the derivative dF_s/dV_s is a function of aircraft trim as well as stability. This fact reduces the value of a neutral point extracted from this derivative. If we divide stick force by dynamic pressure, the derivative of this quantity $d(F_s/q)/dC_L$, is a function of stability only.[7]

$$\frac{d(F_s/q)}{dC_L} = -A\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free} \tag{21.3}$$

where

$A = -KS_eC_e$

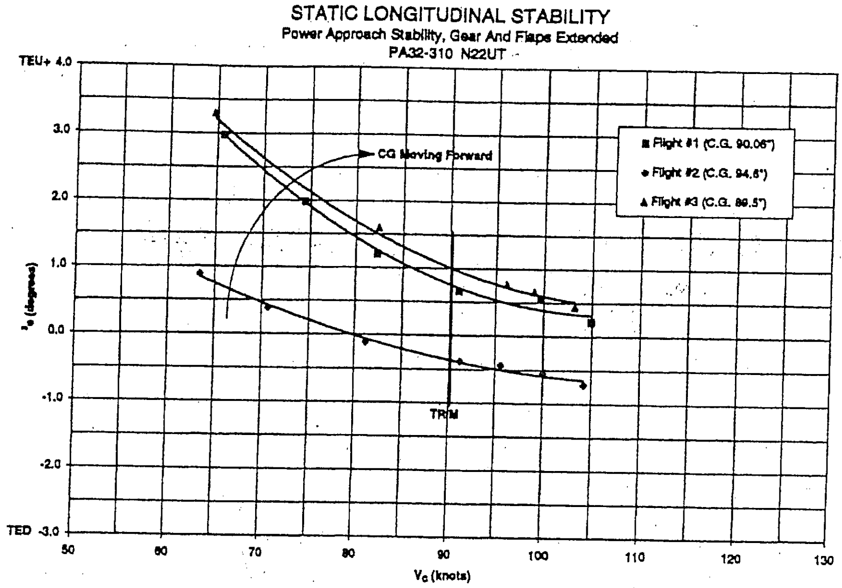
S_e = elevator area

C_e = elevator MAC

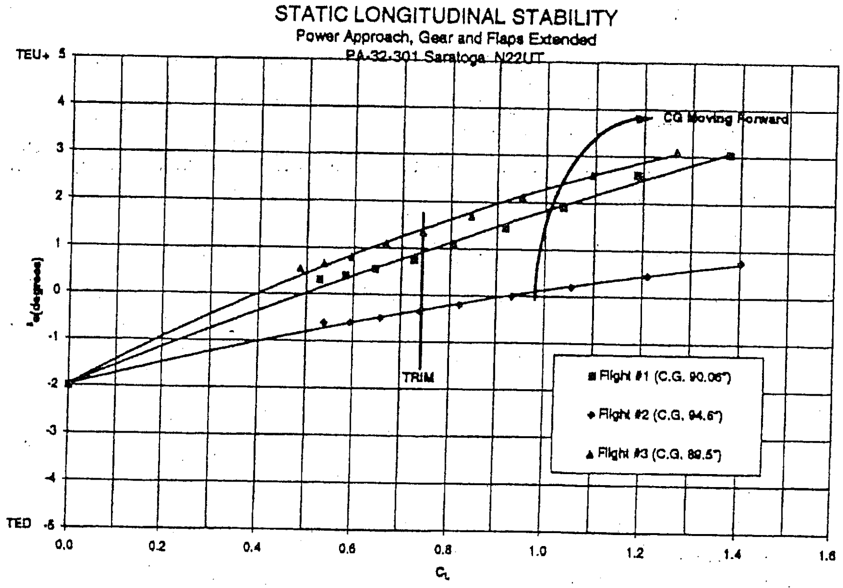
K = control system gearing constant

The next step in the data reduction process is to divide the stick force by dynamic pressure and plot this vs lift coefficient (see second plot of Fig. 21.2.). In order to extract the neutral point, the sequence of Fig. 21.2 is continued.

Again, it should be cautioned that this method will not give the stick-free neutral point if there are springs or other "force feel" systems in the

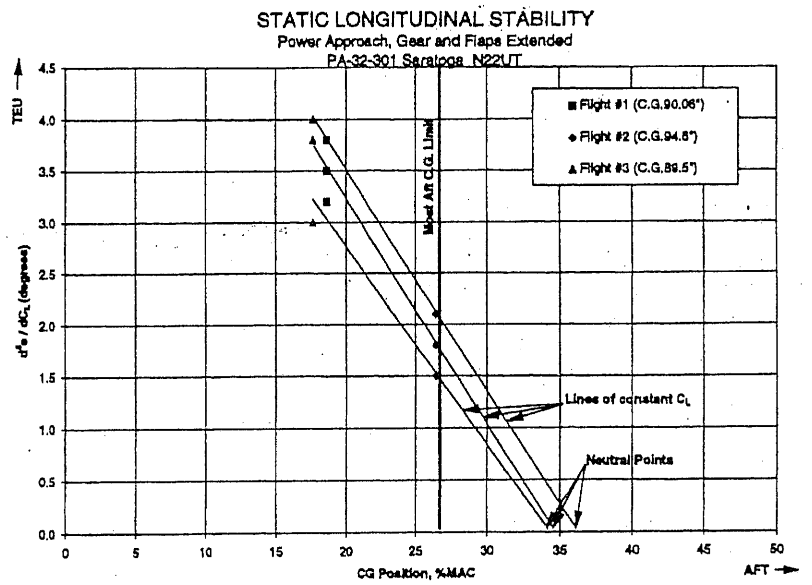


Step 1. Plot data of elevator position (δ_e) vs calibrated airspeed (V_C) for each flight at different c.g. positions and fair a smooth curve through the data. Mark the trim airspeed on the plot.

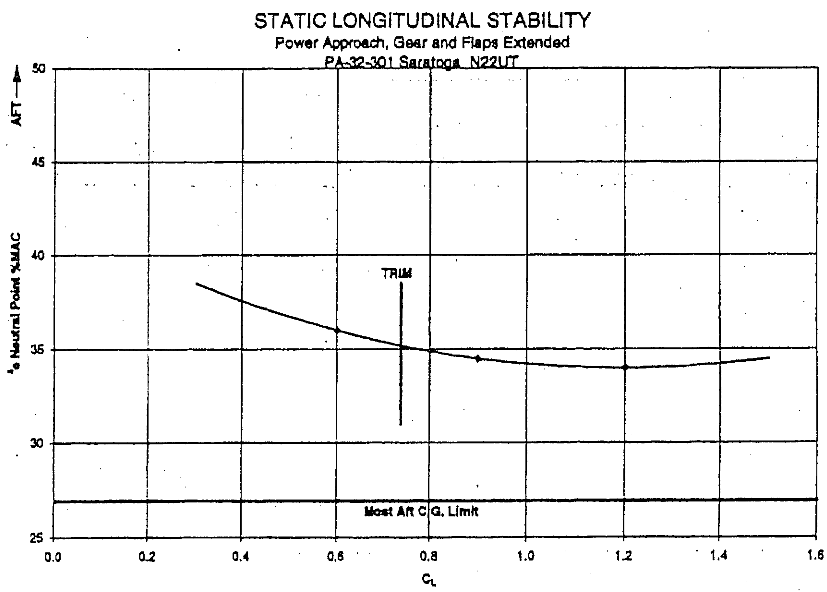


Step 2. From the smooth curves of Step 1 plot elevator position δ_e vs lift coefficient C_L. Select airspeed at which the flight test data were obtained to calculate C_L but obtain the δ_e from the faired lines. Note that these curves all ray from the δ_e for $C_L = 0$.

Fig. 21.1 Graphical determination of elevator position neutral point.[3,8]

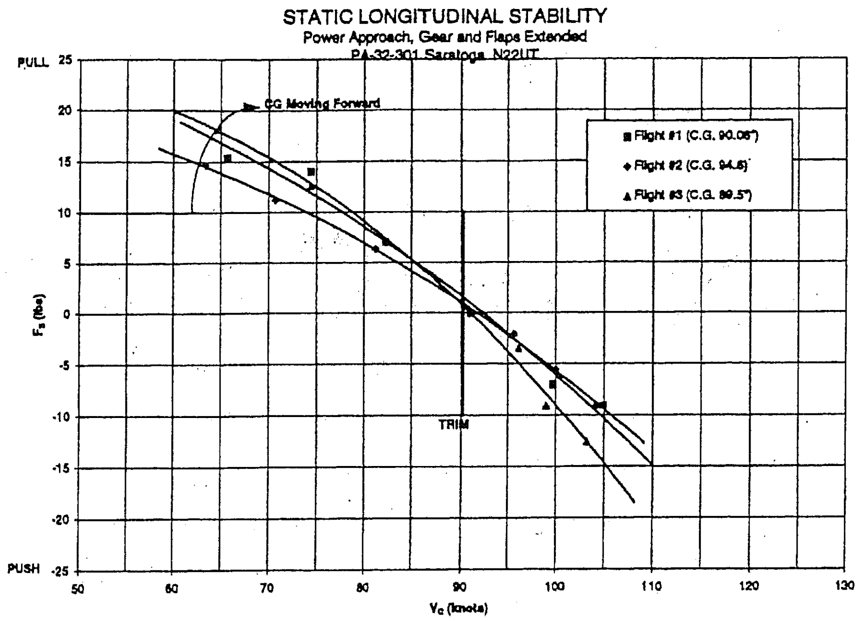


Step 3. Take slopes $d\delta_e/dC_L$ at even increments of C_L from each of the curves and plot c.g. position. Fair curves through the points for each respective C_L and extrapolate to zero. This is the c.g. position of the neutral point for that lift coefficient.

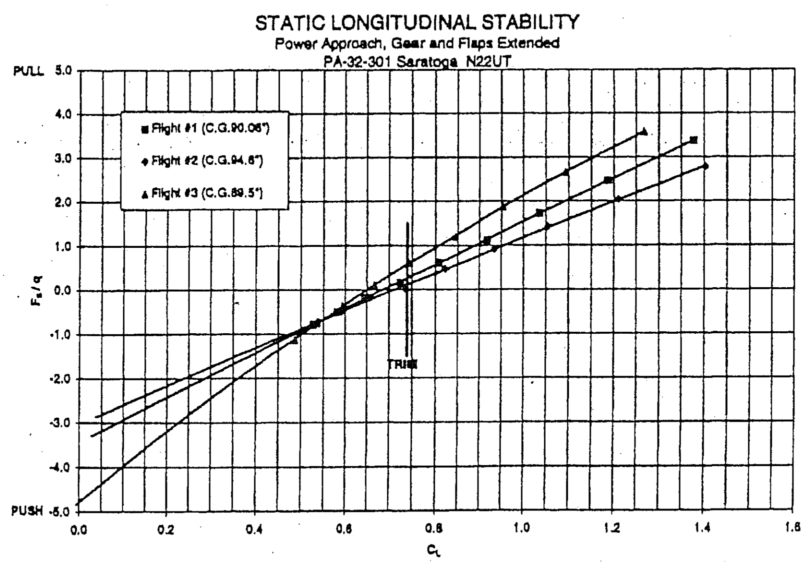


Step 4. Plot the locus of neutral points for each C_L vs C_L and compare with the desired most aft c.g. position. Mark trim C_L neutral point since this is most important neutral point.

Fig. 21.1 Graphical determination of elevator position neutral point (continued).[3,8]

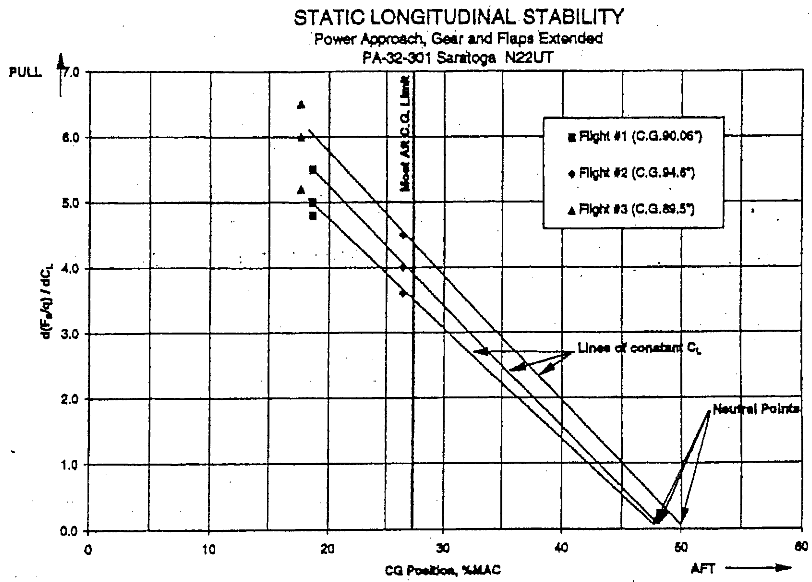


Step 1. Plot elevator control force F_S vs calibrated airspeed V_C for each c.g. position tested and fair a smooth curve through the data points. Mark the trim airspeed on the plot. For FAA testing this is all that is required by the regulations.

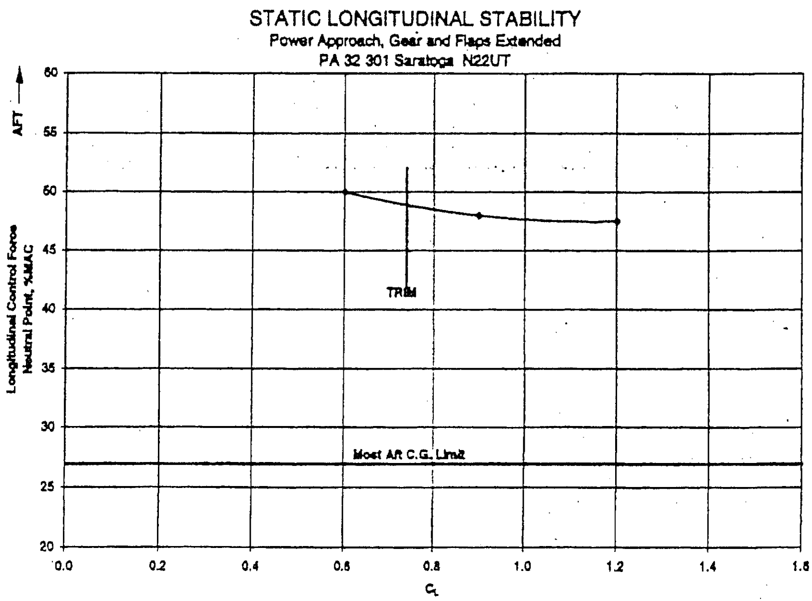


Step 2. Using even increments of airspeed obtain the F_S from the previous plot using the faired lines and not the data points and plot F_s/q vs C_L for each c.g. position tested. These lines should cross at or near the trim C_L.

Fig. 21.2 Graphical method for determining control force neutral point (continued).[3,8]



Step 3. At even increments of C_L take slopes $dF_s/q/dC_L$ for each c.g. position tested and plot these slopes vs c.g. as shown. Fair straight lines connecting the slopes for each C_L and extrapolate to a zero slope. This point is the control force neutral point for that C_L.



Step 4. Plot the locus of neutral points vs C_L and mark the trim C_L. This is the most important control force neutral point since the pilot spends most of his time flying at or near trim.

Fig. 21.2 Graphical method for determining control force neutral point (continued).[3,8]

longitudinal control system. In such a case, you will only have an apparent or stick-force neutral point.

21.5 Flight Test Method for Determination of Neutral Points

Both the stick-fixed and stick-free neutral point data are collected at the same time. First, the pilot trims the aircraft to the trim airspeed and power setting required by the regulation for the flight condition (climb, cruise, or power approach). The following data are then recorded:

1) observed trim airspeed
2) elevator position (Note: It will not be zero.)
3) longitudinal control force (It should be zero.)
4) fuel consumed (for test weight calculation)
5) power setting
6) altitude
7) ambient air temperature

Once the trim data are obtained the airspeed is either increased or decreased by use of the longitudinal control without retrimming the aircraft and the new value of airspeed is held constant by exerting a force upon the longitudinal control. Items 1 through 4 of the above data set are read again at this new speed. Whether one uses a speed above or below the trim airspeed for the first point depends upon the flight condition being measured. If it is a climb condition then the first point should be above trim, if power approach it should be below trim, if cruise it makes no difference. The reason for doing this is to reduce the altitude gain or loss during the measurement. This procedure is then repeated at an airspeed on the opposite side of the trim airspeed. Once that data is obtained, the airspeed is then again moved to the opposite side of the trim speed to some value that is at least 5 kn higher or lower than the previous measurement. This alternating procedure with data points 5–10 kn apart is continued until the required stable range is covered. Data items 1 through 4 are collected at each airspeed.

After completing the last point above and below the trim airspeed, the longitudinal control is gradually released toward trim until the pilot's hands can be removed without any further airspeed change. This airspeed is then recorded as the "free return airspeed." It is an indication of control system friction and the FARs require that it not be more than + or − 10% of the trim airspeed.

Once the data have been collected, the instrument corrections are applied and the data plotted as shown in Figs. 21.1 and 21.2.

21.6 Other Static Longitudinal Stability Tests

21.6.1 Speed Stability[7]

The military specifications require that aircraft have a stable stick force throughout its speed range. This requirement addresses itself to the irreversible control system since these systems do not have classical stick-free stability.

The test for speed stability is quite simple. A series of overlapping stick force vs equivalent airspeed plots are obtained across the operating envelope as shown in Fig. 21.3 (Ref. 7).

The military specifications do allow for some instability in the transonic range. This instability may not be of such nature as to be objectionable to the pilot.

In measuring speed stability, one must be careful to take into account the control system friction and breakout forces. The normal procedure is to measure the force on the back side, or low force side, of the friction band for speeds below trim, and on the high side (low force side) at speeds above trim.

21.6.2 Flight Path Stability[5,7]

An airplane is said to exhibit positive flight path stability if an increase in airspeed by elevator alone decreases the flight path angle, while a decrease in airspeed by this method increases the flight path angle. Flight path stability is directly related to the speed at which an airplane flies its approach and the relationship of this speed to the thrust required curve. For instance, an airplane

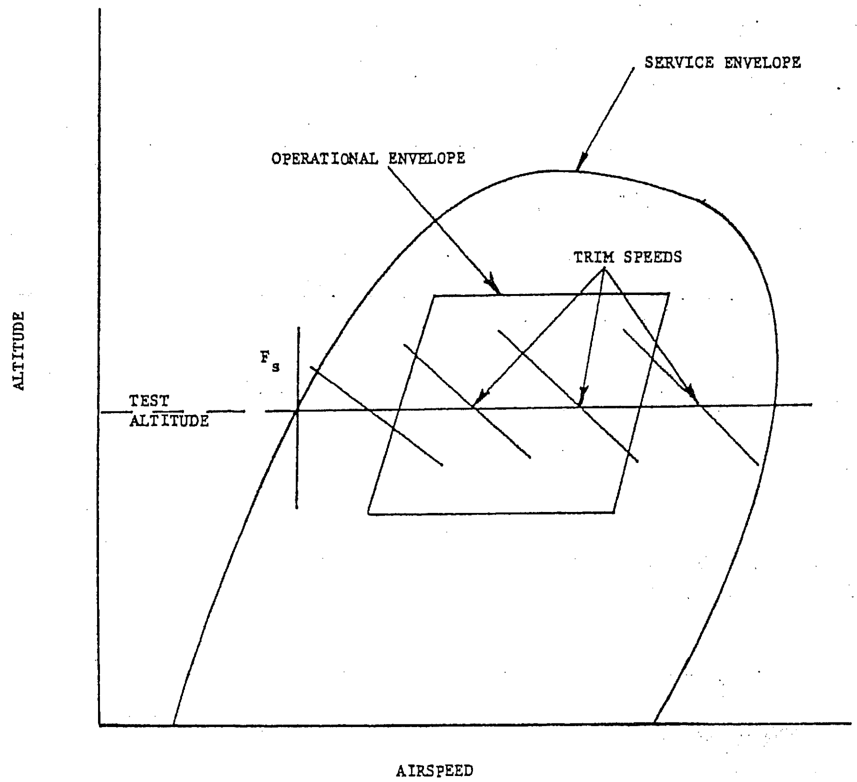


Fig. 21.3 Speed stability measurement.[7]

that flies its approach at an airspeed that is in the flat portion of the thrust required curve generally has better flight path stability than one with its approach speed on the back side of the thrust required curve. Therefore, flight path stability may sometimes be improved by increasing the approach speed.

Although the items that affect flight path stability are more nearly related to airplane performance than to airplane stability, they do affect the pilot's opinion of the airplane's handling qualities and affect workload during an approach. It is for this reason that flight path stability is included as a longitudinal stability test.

To analyze flight path stability in the power approach configuration, we need a plot of flight path angle vs true airspeed, such as is shown in Fig. 21.4 (Ref. 5). To obtain this plot we need to measure the rate of descent in the power approach configuration through a range of -10 to $+10$ kn of the approach speed. The airplane should be trimmed at the approach speed and the speed variations made with elevator only. In certain cases it may be possible to conduct this test at the same time data is being gathered for the power approach longitudinal stability test since pilot techniques for the two tests are nearly the same.

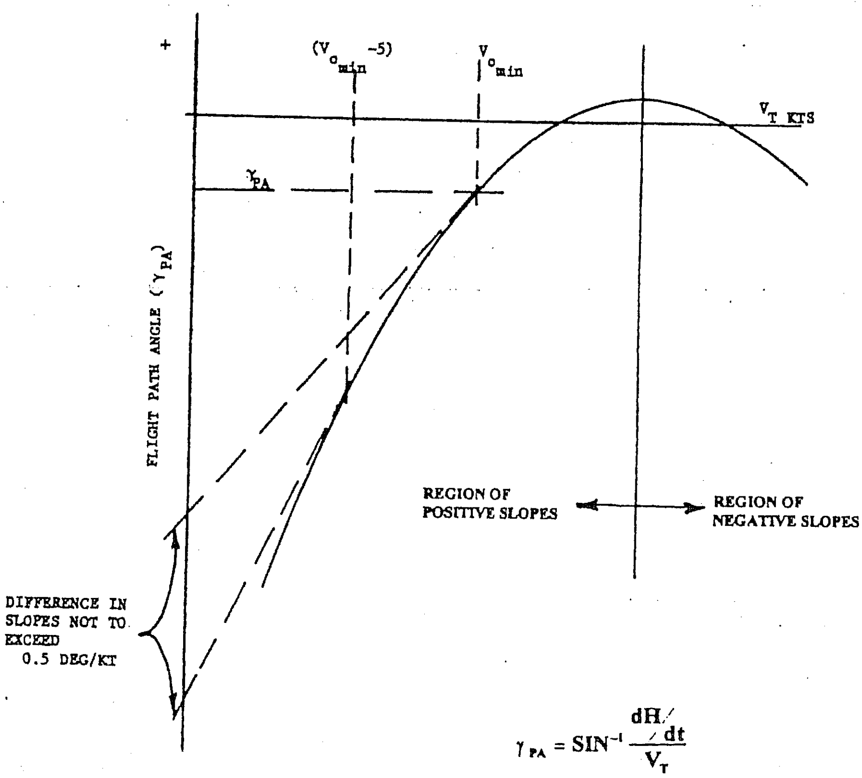


Fig. 21.4 Flight path stability measurement.[5]

Once we have obtained the data it should be corrected for weight and other nonstandard performance factors and plotted as shown in Fig. 21.4.

We then need to take slopes at points along the curve, including a slope at the approach speed for reference. We then may compare these slopes with applicable requirements to determine if they comply.

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22
Dynamic Longitudinal Stability Theory

22.1 Introduction

In our discussions of static longitudinal stability we have been interested in the initial tendency of the airplane to return to the equilibrium condition after being disturbed. Dynamic longitudinal stability is a study of how one equilibrium flight condition is changed to another equilibrium condition, or how the airplane responds to a disturbance with time.[1]

As we mentioned in a previous discussion, positive dynamic stability exists when the amplitude of the displacement decreases with time. The level of positive dynamic stability required, if such a requirement exists, is usually specified by the time for the oscillation to damp to half-amplitude.[3]

For small perturbations, dynamic longitudinal stability like static longitudinal stability does not normally couple with motion about another axes, and we can consider it in the two-dimensional sense. This being the case, the principal variables in longitudinal dynamics stability will be:[3]

1) the pitch attitude of the airplane
2) the angle of attack
3) the flight velocity
4) the elevator deflection when considering the stick free case

Aircraft dynamic motion can be divided into two sets. The first set that we will discuss in this chapter is the longitudinal set. The second set is the lateral-directional set that will be discussed in a later chapter. Both sets of motion are described by quartic differential equations.

The longitudinal quartic equation can be factored into a pair of second order differential equations. One of these second order differential equations describes the longitudinal short period motion which on most airplanes is a well-damped motion of fairly high frequency, normally with a period of under three seconds.

The other second order differential equation factored from the longitudinal quartic equation describes a motion called the phugoid or long period motion. It is a lightly damped motion of low frequency with a period on the order of 30 s or more.

A third set of motion called the short period elevator motion may exist for airplanes with reversible control systems. It resembles the longitudinal short period but is driven by the elevator.

Dynamic stability is better classified as a part of handling qualities since each of the modes mentioned above is related to the flying task at hand. For instance, the pilot uses the phugoid or long period mode to make airspeed changes, and this mode is more closely related with static stability. In maneuvering tasks the pilots initial inputs are pitch and angle of attack changes without changes in airspeed. The short period modes respond to these changes, and we may say that short period motion more closely aligns itself with maneuvering stability.

22.2 Theory

To study dynamic stability we must use an axis system other than the body axis used in the previous chapters because we must determine the aircraft's motion as referenced to an axis system not rigidly attached to the airplane. In essence, we must determine the motion of the body axis relative to another axis system that is not attached to the airplane. The axis system most often used is the moving Earth axis system. This system assumes the Earth is flat, that the axis system moves with the airplane, and that true north is the principle reference.[4]

Both dynamic longitudinal motions behave, as do all second order systems, like the spring-mass-damper system that most of us who took higher mathematics studied in our first course in differential equations. Figure 22.1 shows the typical spring-mass-damper system as described in most differential equations textbooks.

Assuming zero friction, the equation of motion for this system is shown in Eq. (22.1).

$$\ddot{x} + \frac{c}{m}\dot{x} + \frac{k}{m}x = \frac{F(t)}{m} \tag{22.1}$$

The standard form of a second order ordinary differential equation with constant coefficients is shown in Eq. (22.2).

$$\ddot{x} + 2\zeta\omega_n\dot{x} + \omega_n^2 x = f(t) \tag{22.2}$$

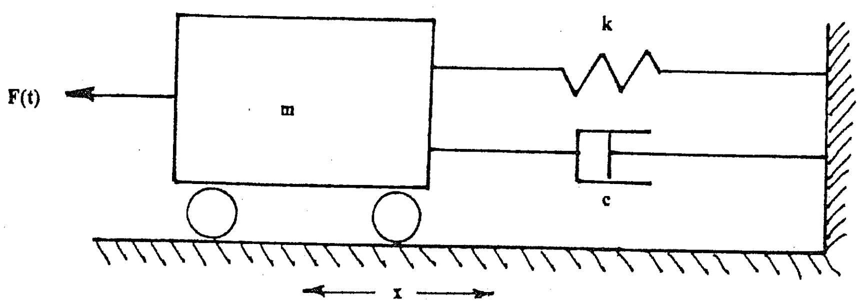

Fig. 22.1 Spring-mass-damper system.

When we compare the two equations we see that the natural frequency, ω_n, is described by the equation:

$$\omega_n = \sqrt{\frac{k}{m}} \tag{22.3}$$

and the damping ratio, ζ, is expressed by the equation:

$$\zeta = \frac{c}{2\sqrt{km}} \tag{22.4}$$

Since we cannot solve second order differential equations directly, we must use some mathematical tricks. One such mathematical trick is called the Laplace transform.

The Laplace transform changes a higher order differential equation that we cannot solve into an algebraic equation that we can solve. The Laplace transform uses a complex variable "S," sometimes called the Laplace operator, which can be described by its real and imaginary parts in the form:

$$S = \sigma + j\omega \tag{22.5}$$

This complex variable is often described by plotting it in the complex plane (or S plane). In this plane the real part, σ, is plotted on the X axis while the imaginary part, ω, is plotted on the Y axis. From the standpoint of dynamic stability, values for roots of the characteristic equation that fall to the right of the imaginary axis represent a stable system, while values that fall to the left of the imaginary axis are unstable. Roots of the equation that fall on the real axis represent a motion of the system that is non-oscillatory, or deadbeat. If the roots have an imaginary component they represent a system that is oscillatory. The frequency of the oscillation increases as the roots move away from the real axis.

Most important for our purposes is that by use of a Laplace transform a differentiation in time is transformed into a multiplication by the Laplace operator "S" and an integration in time is transformed into a multiplication by "$1/S$."

In the spring-mass-damper system, such a transformation of the characteristic equation results in a second order polynomial in "S" such that:

$$S^2 + 2\zeta\omega_n S + \omega_n^2 = 0 \tag{22.6}$$

In this equation the term $2\zeta\omega_n$ is called the damping term and the term ω_n^2 is called the frequency term. The Greek letter ζ is called the "damping ratio." One should note the difference between the damping ratio and the damping term as the difference is important. In dynamic motion the ζ can be described by the number of overshoots that occur during the damped oscillation. The damping term can be described as inversely proportional to the time to damp.

Now that we have examined oscillatory systems and Laplace transformations, let us return to the airplane and dynamic longitudinal motion which can be described by two equations that are similar to the spring-mass-damper equation just discussed. First, let us examine the long period motion or longitudinal phugoid.

22.3 Long Period or Phugoid

The phugoid mode is essentially an airspeed and altitude oscillation at a near constant angle of attack. This mode has such a long period that even large changes in the frequency of the oscillation do not make a significant difference to the pilot, providing there is a natural horizon with which to detect pitch changes. However, during instrument flight a lightly damped phugoid presents a problem because of the attention required to keep the airplane on the pilot's selected airspeed and altitude.

The phugoid mode is characterized by an alternately climbing and diving of the airplane, with airspeeds higher than trim at the bottom and lower than trim at the top of the oscillation. During these oscillations the airplane trades potential energy for kinetic energy and vice versa. If we fly along side in another airplane at a constant airspeed, the test aircraft appears to rise and fall like a mass on a spring. At a constant angle of attack the high speed at the bottom of the oscillation produces excess lift. The low speed at the top of the oscillation causes a reduction in lift that produces a net down force. The up force at the bottom and the down force at the top act like the spring constant in the spring-mass-damper system. The airplane drag acts like the damper in the system since it increases with increasing airspeed and decreases with decreasing airspeed. This action tends to return the aircraft to the neutral or trim airspeed condition.[1]

Since the phugoid acts like a spring-mass-damper system we may use the techniques for solving the equations of such a system to solve the phugoid related longitudinal equations to determine the important parameters of the phugoid motion.

If we write longitudinal equations in determinant form we have:[1]

$$\begin{array}{l} \text{drag characteristics} \\ \text{lift characteristics} \\ \text{pitching moment characteristics} \end{array} \begin{vmatrix} S + D_u & D_\alpha - g & g \\ L_u/u_o & S + L_\alpha/u_o & -S \\ -M_u & -M_{\dot{\alpha}}S - M_\alpha & S^2 - M_{\dot{\theta}}S \end{vmatrix} = 0$$

where

S = Laplace operator
g = acceleration due to gravity
u = horizontal velocity
u_o = initial horizontal velocity or trim airspeed
α = angle of attack
$\dot{\alpha}$ = rate of change of angle of attack
θ = pitch attitude
$\dot{\theta}$ = pitch rate

$D_u = \dfrac{\partial D/\partial u}{m}$ = change in drag with change in horizontal velocity divided by airplane mass

$D_\alpha = \dfrac{\partial D/\partial \alpha}{m}$ = change in drag with change in angle of attack divided by airplane mass

$L_u = \dfrac{\partial L/\partial u}{m}$ = change in lift with change in horizontal velocity divided by airplane mass

$L_\alpha = \dfrac{\partial L/\partial \alpha}{m}$ = change in lift with change in angle of attack divided by the airplane mass

$M_u = \dfrac{\partial M/\partial u}{I_{yy}}$ = change in pitching moment with change in horizontal velocity divided by the moment of inertia in pitch, a speed stability term

$M_\alpha = \dfrac{\partial M/\partial \alpha}{I_{yy}}$ = change in pitching moment with change in angle of attack divided by the moment of inertia in pitch, an angle of attack stability term

$M_{\dot{\alpha}} = \dfrac{\partial M/\partial \dot{\alpha}}{I_{yy}}$ = change in pitching moment with rate of change of angle of attack divided by the moment of inertia in pitch, a downwash lag term

$M_{\dot{\theta}} = \dfrac{\partial M/\partial \dot{\theta}}{I_{yy}}$ = change in pitching moment with pitch rate divided by the moment of inertia in pitch, a pitch rate damping term

If we make simplifying assumptions to this determinant for the phugoid case,[2] (that angle of attack remains constant and that the pitching moment characteristics cancel one another) then the determinant simplifies to what is called the phugoid minor.[1,2]

$$\begin{vmatrix} S + D_u & g \\ L_u/u_o & -S \end{vmatrix} = 0$$

By solving this determinant we obtain the characteristic equation for the phugoid:[1,2]

$$S^2 + D_u S + g(L_u/u_o) = 0 \qquad (22.7)$$

The roots of this equation yield considerable information about the phugoid and we will discuss that in more detail later. In addition, evaluating the equation using the methods for spring-mass-damper systems will yield several other important parameters. These parameters are:[1,2]

1) Natural frequency ω_p

$$\omega_p = \sqrt{2}(g/u_o) \tag{22.8}$$

2) Period P_p

$$P_p = 1.38u_o \tag{22.9}$$

where
u_o is in ft/s

3) Damping ratio ζ_p

$$\zeta_p = 1/\sqrt{2}\,\frac{C_D}{C_L} \quad \text{or} \quad \frac{0.707}{(L/D)} \tag{22.10}$$

The equations just given only provide an approximation of the phugoid parameters. They may be used, however, to provide estimates of the parameters prior to actual measurement in flight tests.

As mentioned earlier, we may also evaluate the phugoid by use of the roots of Eq. (22.7). To do this we use the complex plane to plot the roots since some will have imaginary components. In the case where there are imaginary components to the roots the phugoid will be oscillatory. If the roots are real numbers then the phugoid motion will not oscillate. If the roots fall upon the right hand or positive side of the imaginary axis the phugoid motion will be unstable and if they fall on the negative side the motion will be stable or subside. So, it is easy to see that much can be learned about the phugoid motion by plotting the roots. Fig. 22.2 (Ref. 1) shows the classic phugoid roots for a condition where the c.g. is somewhere forward of the stick-fixed neutral point. (The short period roots are also shown and will be discussed later.)

In the case shown in Fig. 22.2 the phugoid is stable, oscillatory, and lightly damped. As we move the c.g. aft toward the neutral point, the frequency of the phugoid decreases, the period becomes longer, and the damping remains nearly constant. The roots move toward the real axis as is shown in Fig. 22.3 (Ref. 1).

If we continue moving the c.g. aft the oscillation will cease and the motion will become aperiodic. This is represented in Fig. 22.4 (Ref. 1) by the roots becoming real. This usually occurs just forward of the neutral point.

Once the c.g. is moved aft of the neutral point, one root becomes real and positive as is shown in Fig. 22.5 (Ref. 1). This indicates a pure divergence and is what we might expect from a statically unstable airplane.

22.4 Short Period

The airplane short period motion may be further divided into: 1) airplane short period—stick fixed; and 2) airplane short period—stick free.

Both of these cases can be said to take place with the airplane at a constant velocity with changes occurring in pitch attitudes and angle of attack. The

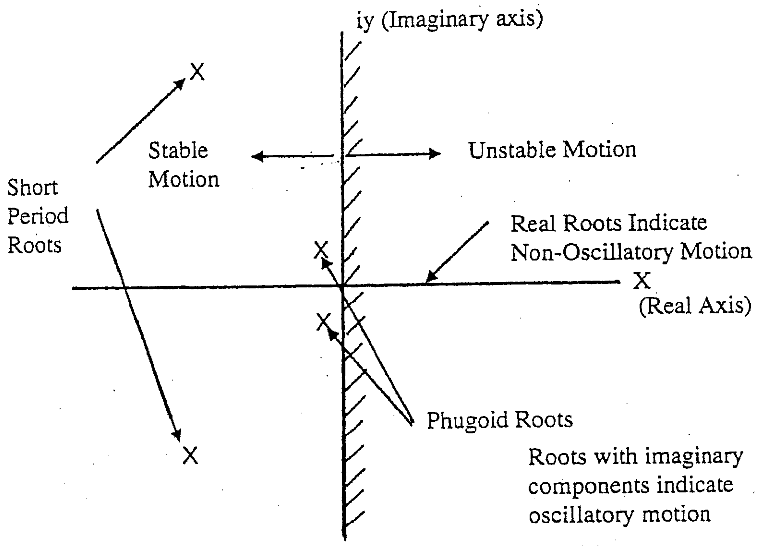

Fig. 22.2 Root locus plot for dynamic longitudinal stability.[1]

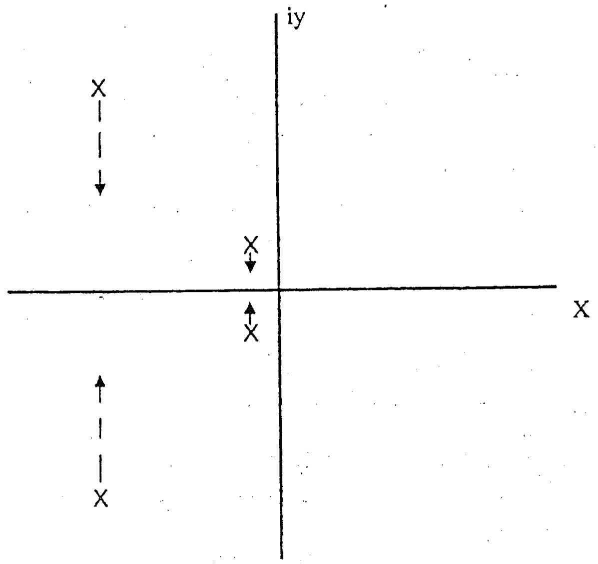

Fig. 22.3 Effects of aft movement of the c.g.[1]

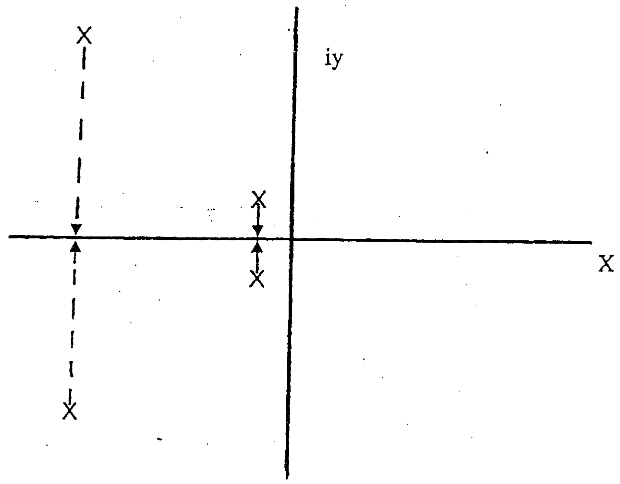

Fig. 22.4 Longitudinal dynamic stability roots when the c.g. is just ahead of the neutral point.[1]

restoring tendency for the pitch oscillation is provided by static stability while the amplitude of the oscillation is decreased by pitch damping. Due to the nature and frequency of the short period motion it is more closely related to the maneuvering tasks. The short period frequency is relatively high having a period that generally ranges between 0.5–5 s (Ref. 3). Fortunately, for

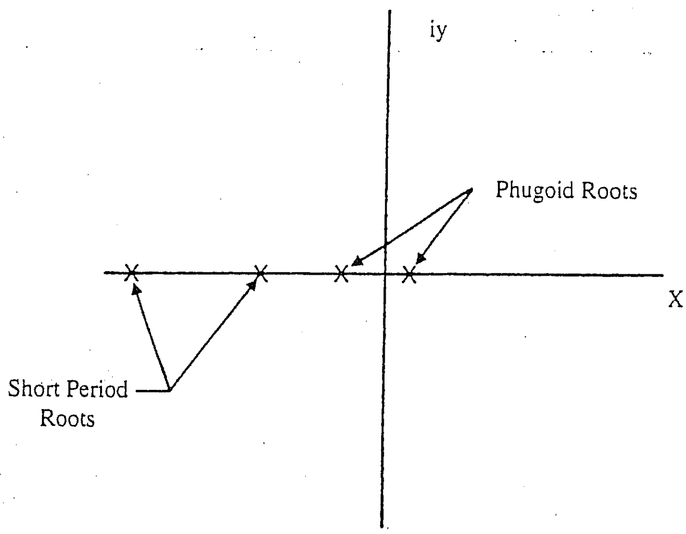

Fig. 22.5 Effects upon dynamic longitudinal stability roots when the c.g. is moved aft of neutral point.[1]

conventional subsonic airplanes the airplane short period motion is normally heavily damped. Times to damp to half amplitude for such aircraft are about 0.5 s (Ref. 3). However, for subsonic airplanes with reversible control systems, it is possible to have a stick-free airplane short period with weak damping or even unstable oscillations. This may occur because of a coupling of motion between the airplane short period pitching and elevator rotation about the hinge line.[3] Such short period motion can be quite dangerous since its period varies between one and 2 s. This time corresponds closely with pilot response time, and rather than damping the motion the pilot may actually increase it.

As we have just discussed, both the damping and frequency (or period) of the airplane short period have a profound effect on the longitudinal flying qualities, particularly when related to maneuvering tasks. Therefore, short period data is normally presented in plots of undamped natural frequency $\omega_{n_{SP}}$, and the damping ratio ζ_{SP} (see Fig. 22.6).[1] This plot is sometimes called the short period thumbprint.[1] However, in flight the pilot does not sense the undamped natural frequency $\omega_{n_{SP}}$, but senses the damped frequency $\omega_{d_{SP}}$. This frequency is dependent upon the damping ratio ζ_{SP} as well as the undamped natural frequency $\omega_{n_{SP}}$. However, it is easier to understand the effects of short period frequency and damping if we discuss varying only one parameter at a time.

First, let us assume that the damping is satisfactory and constant. In such a case, medium to high values of $\omega_{n_{SP}}$ provide a satisfactory airplane response for maneuvering tasks. If the natural frequency is low, then the pilot may think that the airplane is sluggish or tends to dig in. However, once it does respond it may respond too much. The pilot may also find the airplane hard to trim. If the short period natural frequency is very high the airplane tends to respond too quickly, making any precise tracking task difficult. If the frequency tends toward the low end of the very high range, it may be possible to improve handling by increasing F_s/g gradients.[1,2]

Now let us hold the short period natural frequency $\omega_{n_{SP}}$ constant and vary the damping. The damping strongly affects the time of response and the airplane to longitudinal control inputs or external disturbances. When the damping decreases to a value less than 0.5 the pilot becomes aware of the short period oscillation. Even though the pilot is aware of the motion it is still heavily damped. At very low values of damping the short period is easily excited by pilot inputs. Maneuvering forces will feel lighter than they actually are because the airplane will respond faster than the pilot thinks it should. At moderate-to-good values of damping the short period motion is no longer apparent to the pilot, and maneuvering tasks may be performed without undue effort. When short period damping is increased into the heavy range the airplane response becomes slower and slower. The pilot must force the initial response by large control inputs and will usually describe the airplane as sluggish. Therefore, at a constant value of $\omega_{n_{SP}}$ the pilot's opinion may be varied from overly responsive to sluggish by changing short period damping. This, of course, is similar to the opinions obtained when we held damping constant and varied natural frequency, and points out the usefulness of the thumbprint plot.[1,2,4]

We have been discussing the effects of short period natural frequency and damping. Now let us determine the components of the longitudinal equations

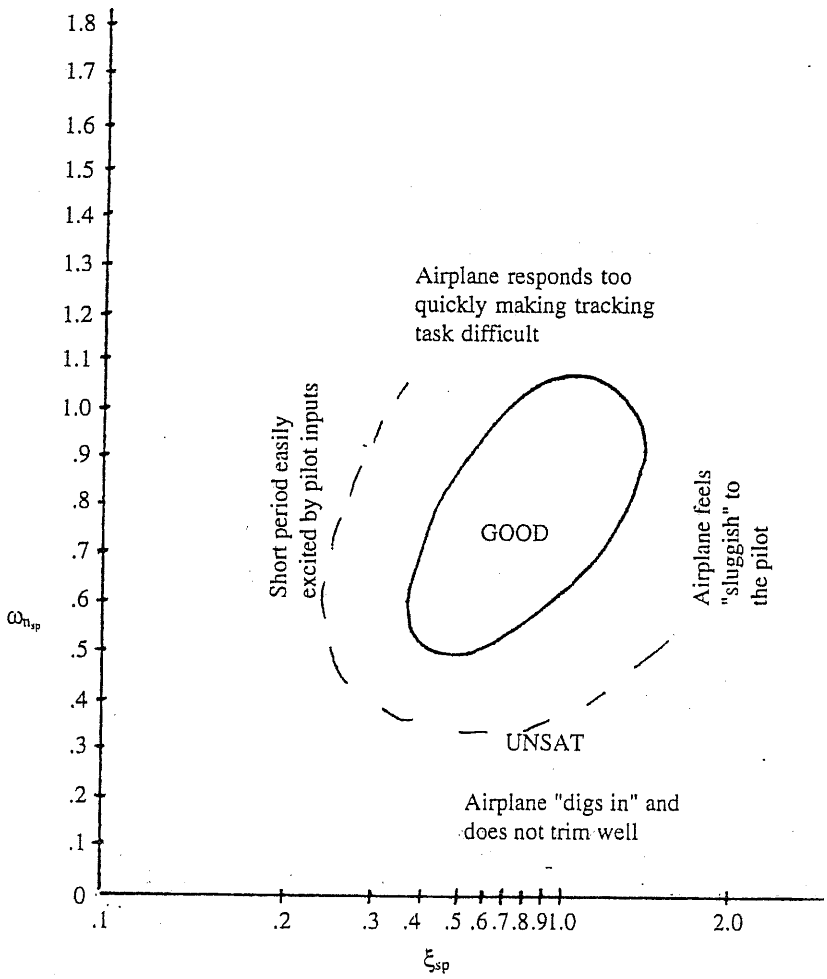

Fig. 22.6 Short period thumbprint.[1]

that affect these values. For the short period we can assume that the airspeed remains constant during the motion. In addition, we can assume that the drag does not have an effect upon the short period and that it is not affected by pitch attitude. We will also ignore the effects of compressibility. When we make these assumptions, the longitudinal determinant reduces to:[1]

$$\begin{vmatrix} S + L_\alpha/u_o & -1 \\ -M_{\dot{\alpha}}S - M_\alpha & S - M_\theta \end{vmatrix} = 0$$

or what might be described as the short period minor. When we solve this determinant we have the characteristics equation for the short period motion:[1]

$$S^2 + (L_\alpha/u_o - M_{\dot{q}} - M_{\dot{\alpha}})S + [M_\alpha + (L_\alpha/u_o)M_{\dot{q}}] = 0 \tag{22.11}$$

In the same manner as used for the phugoid we can determine the undamped natural frequency of the short period.[1]

$$\omega_{n_{SP}} = \sqrt{\frac{\gamma/2P_aM_2}{I_{yy}}\,S\bar{c}C_{L_\alpha}\left(\frac{X_{cg}}{\bar{c}} - N_m\right)} \tag{22.12}$$

where

γ = ratio of specific heats (1.4 for air)
P_a = absolute pressure in pounds per square foot
M = Mach number
C_{L_α} = lift curve slope
$\frac{X_{cg}}{\bar{c}} - N_m$ = nondimensional distance between the airplane c.g. and the stick-fixed maneuver point
S = reference or wing area
$\bar{c}$ = mean aerodynamic chord

An equation for the damped natural frequency of the short period may also be derived if we assume:[1]

$$M_{\dot{\alpha}} = 0 \quad \text{and} \quad L_\alpha/u_o = -M_{\dot{\theta}} \tag{22.13}$$

With these assumptions the equation for damped natural frequency is:[1]

$$\omega_{d_{SP}} = \sqrt{-M_\alpha} \tag{22.14}$$

The damped short period natural frequency is only a function of angle of attack stability.

The equation for the short period damping ratio ζ_{SP} is:[1]

$$\zeta_{SP} = \frac{\sqrt{\frac{\rho S}{2}}}{2\sqrt{-\frac{\bar{c}}{I_{yy}}C_{L_\alpha}\left(\frac{X_{cg}}{\bar{c}} - N_m\right)}}\left\{\frac{C_{L_\alpha}}{W/g} - \frac{C_{M\theta}\bar{c^2} - C_{M_\alpha}\bar{c^2}}{2I_{yy}}\right\} \tag{22.15}$$

where
W = aircraft weight

A detailed study of Eqs. 22.12 and 22.15 will reveal much about the short period motion. As with the phugoid, the short period motion may also be evaluated by use of the root locus plot in the complex plane. Effects of shifting the c.g. aft are also shown for the short period in Figs. 22.2–22.5.

22.5 Elevator Short Period

The elevator short period motion is essentially a flapping motion of the elevator about the hinge line. It is sometimes mistakenly thought to be control surface flutter. In most cases the motion is heavily damped and has a typical period of from 0.3 to 1.5 s with times to one half amplitude of approximately 0.1 s (Ref. 3). Cases of poorly, or neutrally, damped elevator short period have occurred on airplanes with both elevators and stabilators. In the case of elevator controlled airplanes, the cause is usually a convex control surface that does not have equal curvature on both sides. On stabilator controlled airplanes it may occur when geared tabs, which are used to provide control force, have their gearing ratio set too high.

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23
Dynamic Longitudinal Stability Flight Test Methods and Data Reduction

23.1 Introduction

Since dynamic longitudinal stability involves evaluating the response of the airplane over a period of time, the flight test methods must, by necessity, differ from those required for static stability. In addition, we must also consider more sophisticated methods of data collection since the aircraft responses may be oscillatory and of short duration. Let us, again, separate the discussion into methods to evaluate the phugoid and methods to evaluate the short period.

23.2 Federal Aviation Administration Regulations

The FARs only address the short period mode of dynamic longitudinal stability. They do not address the long period longitudinal dynamic stability—or phugoid—mode of motion. The reason behind this is that the long period mode of motion can be easily damped by the pilot under visual flight conditions. However, some more recent research has shown that an undamped phugoid may create problems during instrument flight. The FAA has in recent amendments revised the regulation to reflect the change in thinking on the phugoid. In any case, an airplane will possess better flying qualities if the phugoid motion is damped, so it should not be ignored just because there is no regulation to cover it.

23.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

CAR 3.117 provides the requirements for dynamic longitudinal stability for airplanes certified under this regulation. It says that any short period oscillation that occurs between stalling speed and the maximum permissible speed shall be heavily damped with the controls held fixed and with the controls free.

23.2.2 Federal Aviation Regulations Part 23 (Ref. 2)

FAR 23.181 through amendment 14 reads essentially the same as CAR 3.117. However, later amendments to FAR Part 23 have added two provisions. One of these provisions deals with the use of a stability augmentation system. If such a system is used then the requirement for short period damping with controls fixed is eliminated. The second provision of the later FAR Part 23 deals with the phugoid motion when the controls are released from a displacement of ±15% of the trim speed. It requires that the airplane should not exhibit a dangerous characteristic and that the phugoid motion should not be so unstable as to increase the pilot's workload or otherwise endanger the airplane.

23.2.3 Advisory Circular 23-8A (Ref. 3)

Advisory Circular 23-8A discusses what is meant by a heavily damped short period motion in the regulation. It says that qualitatively the motion should appear to the pilot to be "deadbeat," which means no apparent overshoots. If there are apparent overshoots, then the aircraft should be instrumented and the airplane shown to be damped within two cycles.

The advisory circular discusses flight test techniques for evaluating the short period including the pulse input and the doublet input, which are discussed later in this chapter. The advisory circular also states that the short period should be evaluated for all of the flight conditions where static longitudinal stability is evaluated.

A discussion is also included regarding stability augmentation systems (SAS) and how to evaluate the aircraft when one of these systems is installed.

23.3 Flight Test Methods for Evaluating the Phugoid

The flight test method for measuring the phugoid motion is quite simple. First, the aircraft is trimmed to the test trim speed and the test configuration of power, gear, and flaps established. Once in configuration and trim, the airspeed is displaced 10–15 kn from trim by use of the elevator control. The elevator is then returned to the trim elevator position using control movement near the aircraft's long period frequency and the resultant airplane oscillation recorded.

The airspeed may be displaced either above or below the trim speed. Normal procedure is to observe phugoid motions from displacements both above and below trim.

Also, once returned to the trim position the control stick may be either held fixed or released. Again, both approaches should be tested since differences between the stick-fixed and stick-free cases normally exist.

Data may be recorded by a data collection system or, since the frequency is low, by hand.

23.4 Phugoid Data Reduction

To reduce the data, first make a plot of equivalent airspeed vs time, as is shown in Fig. 23.1, for all cases tested. On top of this plot, plot a subsistence envelope (also shown in Fig. 23.1).

From the plot in Fig. 23.1 determine the amplitude ratio X_n/X_{n+1}. With the resultant amplitude ratio, enter Fig. 23.2 (Ref. 4) and determine the damping ratio ζ. If the phugoid is erratic it may be necessary to measure several subsistence ratios and determine resulting damping ratios. The damping ratios may then be averaged to come up with an average damping ratio for the motion.

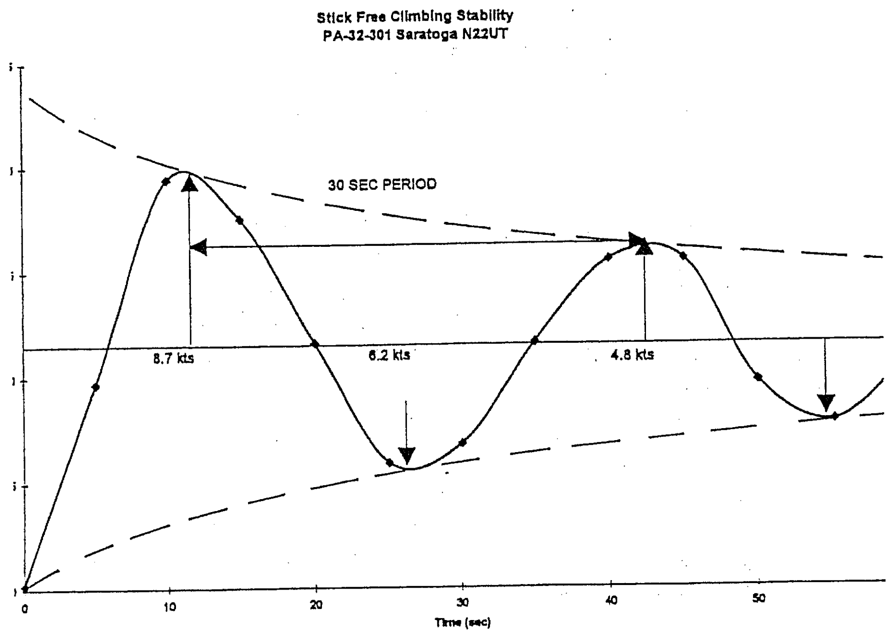

Fig. 23.1 Plot of climb phugoid airspeed vs time.

Once we have damping ratio we may determine the undamped natural frequency ω_p from the equation:[7]

$$\omega_p = \frac{2\pi f}{\sqrt{1-\zeta^2}} \tag{23.1}$$

where
$f = \Delta\text{cycles}/\Delta\text{time}$

We may then wish to make plots of phugoid frequency vs airspeed and damping ratio vs airspeed for comparison with applicable specifications or regulations. We may also make plots of the phugoid roots using the natural frequency and damping and by knowing that:

$$\zeta = \cos\theta \tag{23.2}$$

and using Fig. 23.3. Knowing the value and location of these roots may be useful in tailoring autopilots or in later modifications to the airplane.

23.5 Short Period Flight Test Methods

As might be expected, different techniques are used to test the airplane and elevator short periods.

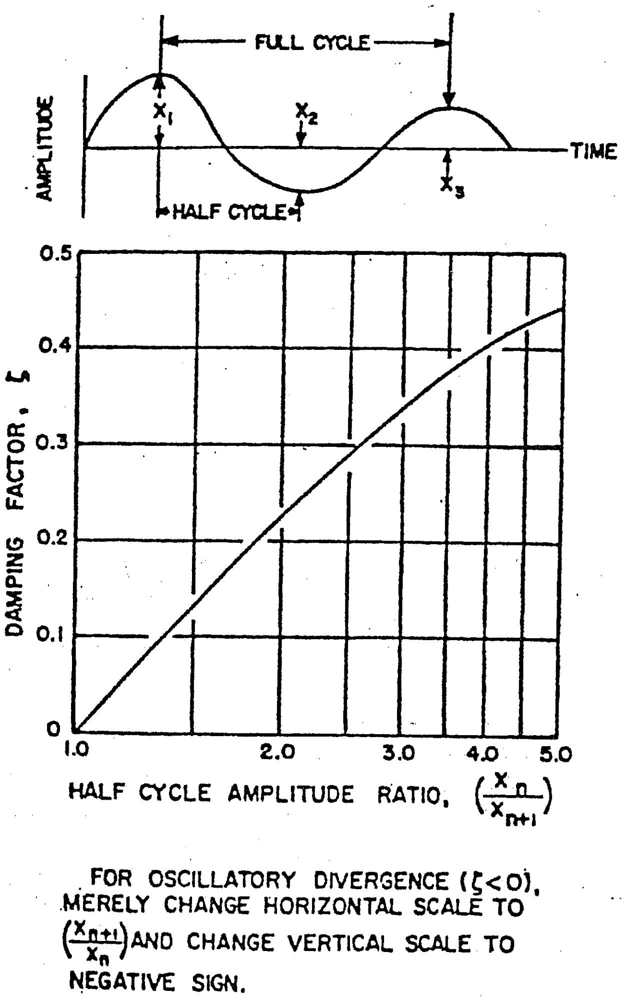

Fig. 23.2 Determination of damping ratio for lightly damped system.[4]

23.5.1 Airplane Short Period[3-5,7]

First, let us discuss the methods to evaluate the airplane short period. There are three methods used to excite the short period. They are 1) doublet input; 2) pulse input; and 3) 2-*g* pull-up.

The doublet input is a very good method for evaluating the short period, because in addition to exciting the short period motion it tends to suppress the phugoid. To perform the doublet input, the pilot first trims the aircraft to the test condition. The test instrumentation is started, and the pilot rapidly moves the control nose down, then nose up, then back to trim. Once the control is

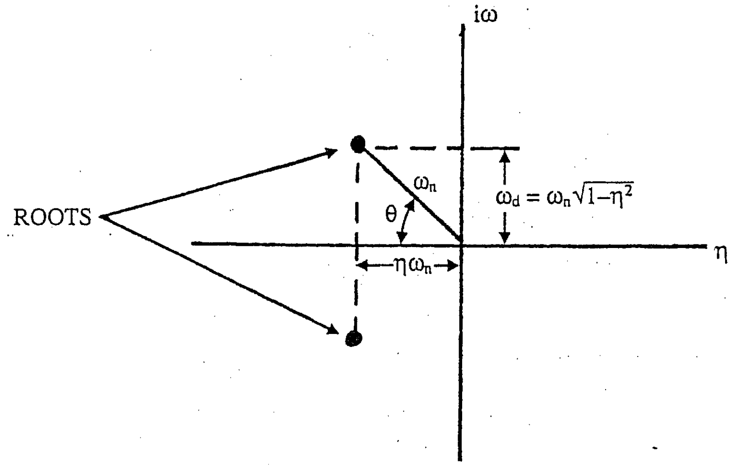

Fig. 23.3 Root locus plot.

returned to trim, it may be either held in the trim position or released, depending upon the type of short period (stick-fixed or stick-free) to be evaluated. Data recording should continue until all short period motion has subsided. In testing the short period, the pilot should try several different doublets, in which the frequency of the input is varied, until the frequency that best excites the aircraft's short period frequency is found. Due to the shortness of the motion, data recording will need to be done using an automatic recording device, unless only the number of overshoots is recorded.

The pulse input might be described as one half of a doublet input. In performing the pulse input, the control is only moved forward, or aft, of trim, but not both as in the doublet. The pulse input is not as good a method for evaluating the airplane short period as is the doublet input, because it tends to also excite the phugoid. This makes it difficult to reduce the data since it may be hard to separate the short period motion from the phugoid motion. However, it may be necessary to use the pulse method for airplanes that have a very high short period frequency.

The 2-*g* pull-up method is also a good method for evaluating the short period, since it too suppresses the phugoid. It is a very good method for airplanes that have a low short period frequency. To perform the 2-*g* pull-up method the pilot first trims the aircraft to the test condition, records the trim data, and then starts a pull-up decreasing airspeed and increasing altitude. The pilot then pushes the nose over and enters a dive in a fairly steep nose down attitude. As trim airspeed and altitude are approached the aircraft is smoothly rotated so as to achieve trim airspeed and attitude at the same time. When this occurs the control stick is rapidly returned to trim and released or held fixed depending upon the short period to be evaluated. The 2-*g* pull-up method provides a large amplitude input for testing short periods with heavy damping. However, it does require considerable pilot skill and proficiency in performing the maneuver.

23.5.2 Elevator Short Period

To evaluate the elevator short period the pulse method described in the preceding section is used. As might be expected the elevator short period is always evaluated stick free and should normally be heavily damped. Also, as mentioned in the theory, it only has meaning for a reversible control system.

23.6 Short Period Data Reduction

The airplane short period natural frequency and damping ratio may be found using the procedure shown in Fig. 23.4 (Ref. 4). Once these values have been obtained they may be plotted on the short period thumbprint for evaluation as is shown in Fig. 22.6. Short period roots may also be plotted using the methods shown for the phugoid.

For well-damped systems, the elevator short period is evaluated by counting the number of overshoots of the trim position. A more detailed investigation of

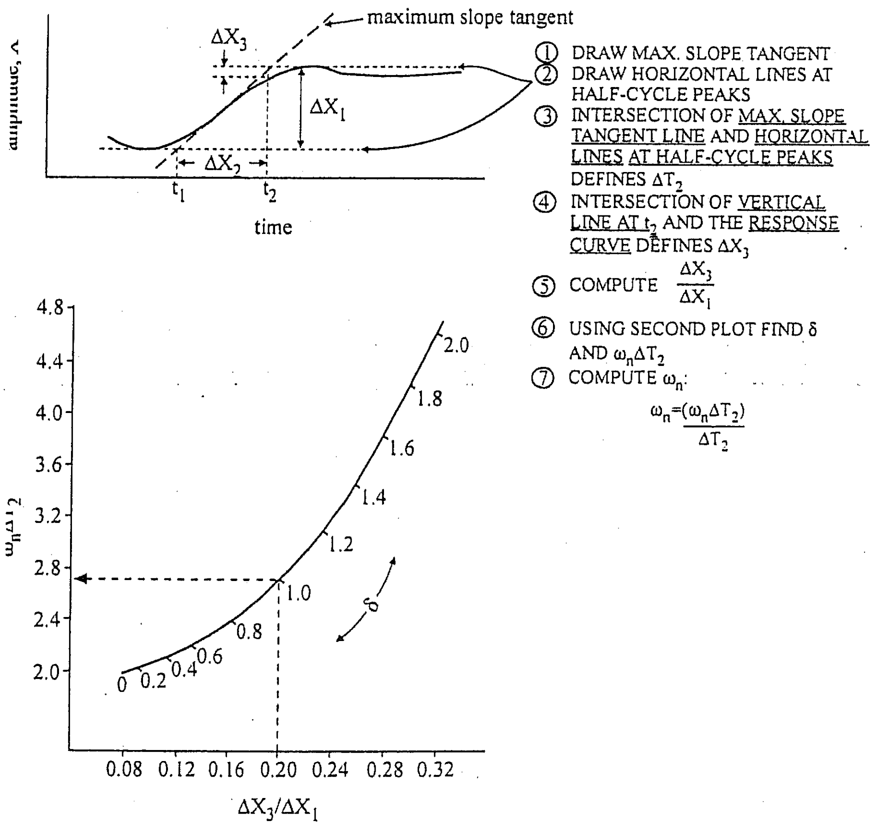

Fig. 23.4 Determination of second order response characteristics for a heavily damped system.[4]

the elevator short period is not warranted unless low damping exists or data are needed for handling qualities improvements.

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24
Longitudinal Maneuvering Stability Theory

24.1 Introduction

In order to perform their mission all airplanes must maneuver. We can say then that maneuvering stability is important for all airplanes, with the degree of importance relating to the mission of the airplane. The level of positive maneuvering stability also depends on the mission of the airplane much in the same way that the level of static longitudinal stability depends on the airplane's mission. When the airplane maneuvers it has a curved flight path and is subject to a normal acceleration. The amount of normal acceleration an airplane can withstand is a function of its structural design load factor. The lower the design load factor the more positive maneuvering stability the airplane should exhibit, or the more difficult it should be for the pilot to over-g the airplane. Using this logic, we would expect the transport or bomber aircraft to have a high level of positive maneuvering stability, while the fighter has a lower level. This makes sense because we want the pilot of a fighter to be able to easily maneuver the airplane.

Both the military specifications and the FAA regulations contain some requirements for a level of positive maneuvering stability. This is only a recent requirement with the FAA regulations, however, as earlier regulations did not contain any requirements for maneuvering stability.

The pilot may maneuver the airplane longitudinally in a number of manners, performing wings level pull-ups, push-overs, or may bank the aircraft and turn. The pilot may also use any combination of these maneuvers.[1,2] For the purposes of this discussion, we will confine our remarks to a discussion of steady pull-ups and steady turns at a constant airspeed. In both of these maneuvers, the airplane has an increased angle of attack, and lift coefficient, over the same trim airspeed in level flight. The airplane also exhibits a rate of rotation about its center of gravity.[1] This rate of rotation about the aircraft c.g. causes an airflow at the horizontal tail that is in the same direction as the direction in which the nose is pitching (see Fig. 24.1).[3] This apparent flow changes the relative wind at the horizontal tail which in turn changes the tail angle of attack as is shown in Fig. 24.1. This change in effective angle of attack contributes significantly to the stability of the airplane in maneuvering flight. This contribution is directly dependent on the pitch rate of the airplane if we hold the airspeed constant.[1] If we hold the airspeed constant, then the pitch rate is a function of the normal acceleration.

The pitch rate θ relationships for a steady pull-up and steady level turn are shown below:[1]

Steady pull-up

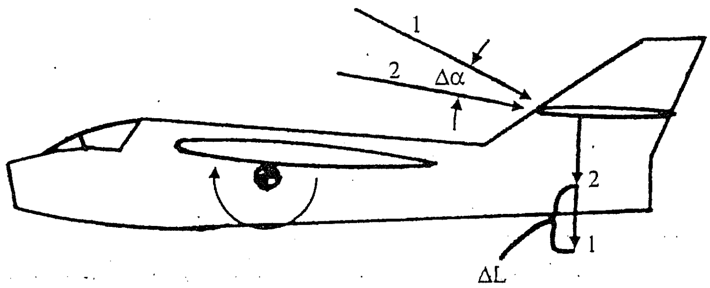
$$\dot{\theta}_{PU} = \frac{g(n-1)}{V} \tag{24.1}$$

where
$\dot{\theta}$ = pitch rate in rad/s
g = acceleration due to gravity, ft/s^2
n = normal acceleration, g
V = true airspeed, ft/s

Steady level turn

$$\dot{\theta}_{ST} = (g/V)(n - 1/n) \tag{24.2}$$

Since the pitch rate is a function of the normal acceleration, and it is the pitch rate that causes the pilot to use more or less elevator deflection and longitudinal control force during maneuvering flight than during nonmaneuvering flight, then the normal acceleration n is generally used as the independent variable for maneuvering stability. Therefore, in flight testing, the parameters "stick force



1. Relative wind at horizontal tail prior to maneuvering and original tail down load

2. Relative wind at horizontal tail during maneuvering causing reduced tail down load

Fig. 24.1 Change in relative wind at the horizontal tail due to maneuvering.[3]

per g" and "elevator position per g" are used as indexes of maneuvering stability.[1]

As we would suspect, the elevator positions required to maintain a given airspeed in maneuvering flight are not the same as those required for the same airspeed in level, unaccelerated flight. This difference in elevator angle can be translated into a pitching moment due to pitch rate.[1] The equation for this moment is:[1]

$$M_{c.g_{\dot{\theta}}} = -a_t \frac{l_t^2 \dot{\theta}}{V_T} q_t S_t \tag{24.3}$$

where
a_t = lift curve slope of the horizontal tail
l_t = length of the "tail arm" in feet
q_t = dynamic pressure at the tail in pounds per square foot
S_t = horizontal tail area in square feet
V_T = true airspeed in feet per second

If we write this equation in coefficient form, we have the pitch rate damping coefficient, which is expressed as:[1]

$$C_{m_{\dot{\theta}}} = -2a_t \eta_t \bar{V} \frac{l_t}{\bar{c}} \tag{24.4}$$

where
η_t = tail efficiency factor
$\bar{V}$ = tail volume coefficient
$\bar{c}$ = MAC of wing in feet

In light of this past discussion and in keeping with the terminology used for static stability, we discuss maneuvering stability as either 1) elevator position maneuvering stability or 2) stick-force maneuvering stability.

24.2 Elevator Position Maneuvering Stability

First let us turn our attention to what might be described as stick-fixed or elevator position maneuvering stability. The elevator position for steady wings level pull-ups at a constant airspeed can be stated as:[1]

$$\delta_{e_{PU}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}} \frac{W/S}{1/2\rho_0 V_e^2} \left\{ \left(\frac{dC_m}{dC_L}\right)_{fixed} n + \frac{C_{m_{\dot{\theta}}} \rho g \bar{c}}{4(W/S)}(n-1) \right\} \tag{24.5}$$

where
δ_{e_0} = elevator angle for $C_L = 0$
$C_{m_{\delta_e}}$ = elevator control power

If we differentiate Eq. (24.5) with respect to the normal acceleration, we have:[1]

$$\left(\frac{\mathrm{d}\delta_e}{\mathrm{d}n}\right)_{PU} = -\frac{1}{C_{m_{\delta_e}}}\frac{W/S}{1/2\rho_0 V_e^2}\left\{\left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{fixed} + \frac{\rho g \bar{c}}{4(W/S)}C_{m_{\dot\theta}}\right\} \tag{24.6}$$

An examination of Eq. (24.6) shows that the $\mathrm{d}\delta_e/\mathrm{d}n$ term carries a negative sign. This is because more up elevator is required to stabilize the airplane at load factors above $n = 1$.

Another interesting item shown by Eqs. (24.5) and (24.6) is that the maneuvering stability equations contain two terms. The first term is essentially the stick-fixed static stability of the airplane, while the second term is provided by the pitch rate of the airplane. It is called the pitch damping term.[1]

The equations corresponding to Eqs. (24.5) and (24.6) for the steady turn case are as follows:[1]

$$\delta_{e_{ST}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}}\frac{W/S}{1/2\rho_0 V_e^2}\left\{\left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{fixed} n + \frac{C_{m_{\dot\theta}}\rho g \bar{c}}{4(W/S)}(n - 1/n)\right\} \tag{24.7}$$

$$\left(\frac{\mathrm{d}\delta_e}{\mathrm{d}n}\right)_{st} = -\frac{1}{C_{m_{\delta_e}}}\frac{W/S}{1/2\rho_0 V_e^2}\left\{\left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{fixed} + \frac{C_{m_{\dot\theta}}\rho g \bar{c}}{4(W/S)}(1 + 1/n^2)\right\} \tag{24.8}$$

It is interesting to note that the only difference in Eqs. (24.5) and (24.6) and Eqs. (24.7) and (24.8) is in the damping term. It can also be seen that at high normal accelerations this difference becomes insignificant.

Aft movement of the c.g. affects the stick-fixed maneuvering stability in much the same way that it affects longitudinal stability. However, in addition to affecting the stability term $(\mathrm{d}C_m/\mathrm{d}C_L)_{fixed}$ it also affects the damping term. This is because the tail arm changes as the c.g. moves aft, causing the damping term to decrease also. When the c.g. reaches the stick-fixed neutral point, $\mathrm{d}\delta_e/\mathrm{d}n$ is only dependent upon the damping. If we continue aft movement of the c.g. the damping term also reaches zero and $\mathrm{d}\delta_e/\mathrm{d}n = 0$. This c.g. position is called the stick-fixed maneuvering neutral point or stick-fixed maneuver point N_M. If the stick-fixed static longitudinal stability is the same in maneuvering flight as in level flight, then the stick-fixed maneuver point should always be aft of the stick-fixed neutral point.[1] Fig. 24.2 (Ref. 1) shows the effects of c.g. shift on the elevator position vs load factor plot.

Altitude effects on the stick-fixed maneuvering stability must be evaluated in two ways. First, the effects are evaluated at a constant equivalent airspeed. This is shown in Fig. 24.3 (Ref. 1). If V_e is held constant, the $(\mathrm{d}C_m/\mathrm{d}C_L)_{fixed}$ term of Eq. (24.8) does not change. However, the damping term is affected by density and decreases with increasing altitude. This causes the stick-fixed maneuvering stability to decrease also.[1]

If we now evaluate the altitude effects at a constant Mach number, we find that the stick-fixed maneuvering stability increases with increasing altitude as is shown in Fig. 24.4 (Ref. 1). This is because there is a slight increase in the damping term of Eq. (24.8) and a large increase in the stability term when we hold a constant Mach number with increasing altitude.[1]

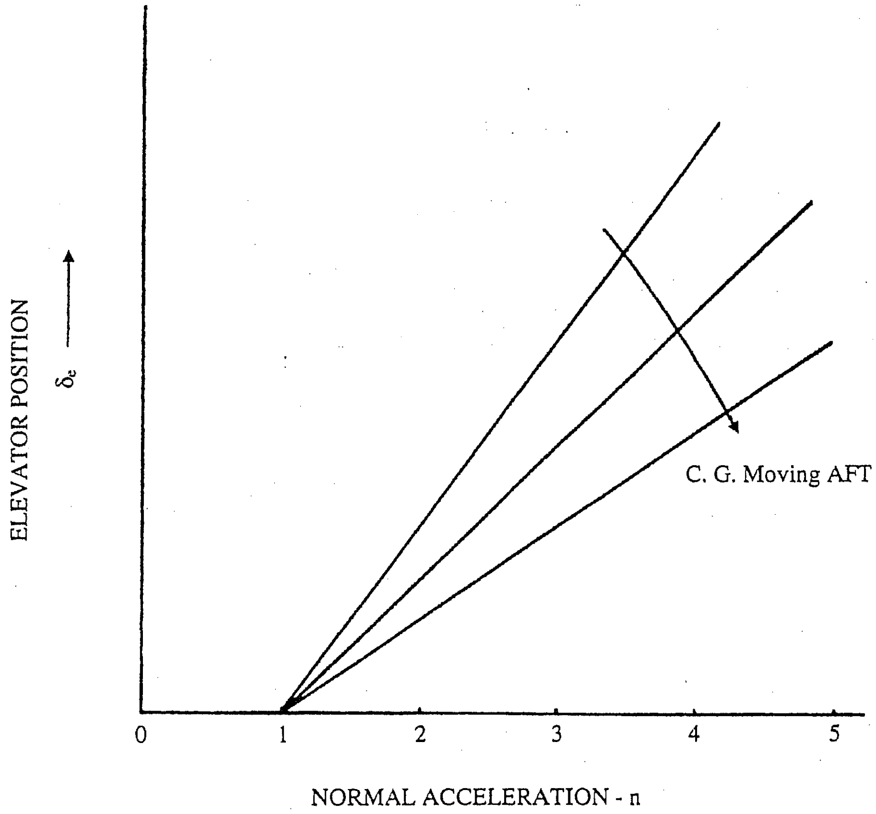


Fig. 24.2 Effects of c.g. movement on $d\delta_e/dn$ (Ref. 1).

If we examine Eq. (24.8) we can see that when V_e is varied, there is a large effect on $d\delta_e/dn$. This is because V_e appears as a squared term in the equation. The effects of varying V_e are shown in Fig. 24.5 (Ref. 1).

24.3 Stick-Force Maneuvering Stability

In our previous discussion, we talked about one of the criteria for maneuvering stability, that of elevator position per *g*. Now let us turn our attention to the second and more important criterion of stick force per *g*. This criterion is important for all aircraft, but for aircraft that must do considerable maneuvering, such as fighters and agricultural aircraft, it may very well be the most important single stability and control parameter.[1]

As was the case with stick-free static stability, we must also consider stick-force maneuvering stability for reversible and irreversible control systems. Since, in a reversible control system, freeing the elevator will cause some float angle due to elevator hinge moments, the stick force required to hold some level of normal acceleration will also be affected by this float angle.

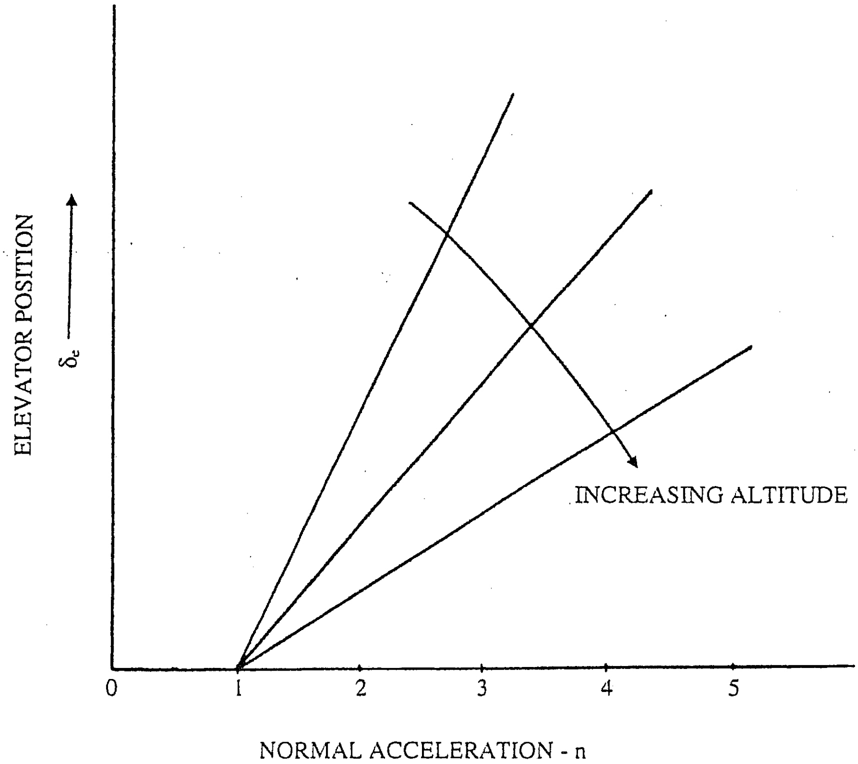


Fig. 24.3 Effects of altitude on $d\delta_e/dn$ at a constant equivalent airspeed.[1]

The irreversible control system is not affected by elevator float since it is held rigidly fixed by its servo mechanism. This mechanism may introduce an artificial float due to system design, or may have designed in some mechanism to vary stick force per g, so they too must be considered.

24.3.1 Stick-Force Maneuvering Stability—Reversible Control System

First let us consider the reversible control system. The equations for the stick force during a wings level pull-up or a level turn at constant airspeed can be expressed as follows:[1]

$$F_{s_{PU}} = K\frac{W}{S}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free}\left\{\frac{V_e^2}{V_{e_{trim}}^2} - n\right\} + K\frac{1}{2}\rho l_t g(n-1)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\} \tag{24.9}$$

$$F_{s_{ST}} = K\frac{W}{S}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free}\left\{\frac{V_e^2}{V_{e_{trim}}^2} - n\right\} + K\frac{1}{2}\rho l_t g(n-1/n)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\} \tag{24.10}$$

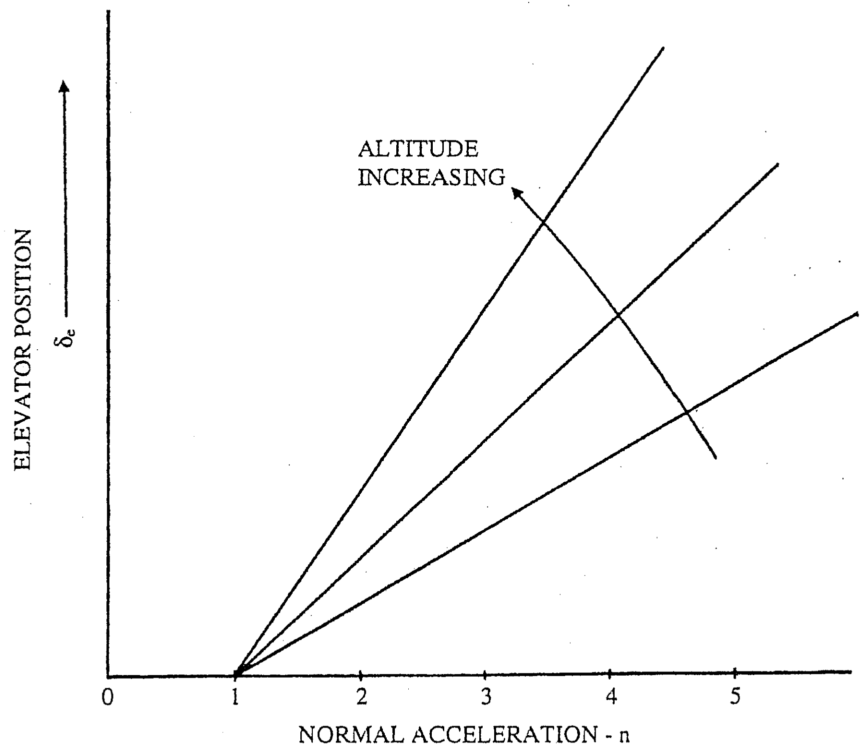


Fig. 24.4 Effects of altitude on $d\delta_e dn$ at a constant Mach number.[1]

where
K = control system gearing constant
τ = rate of change of effective angle of attack with change of elevator deflection

To determine the stick force maneuvering stability, we differentiate Eqs. (24.9) and (24.10) with respect to the normal acceleration and have:[1]

$$\left(\frac{dF_s}{dn}\right)_{PU} = -K\frac{W}{S}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free} + K\frac{1}{2}\rho l_t g\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\} \tag{24.11}$$

$$\left(\frac{dF_s}{dn}\right)_{ST} = -K\frac{W}{S}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free} + K\frac{1}{2}\rho l_t g(1 + 1/n^2)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\} \tag{24.12}$$

Again, we can see from these equations that the only difference between the pull-ups case and the level turn case is the pitch rate difference. This difference also becomes less significant as normal acceleration increases. These equations, like their elevator position per g counterparts, also contain two terms. The first

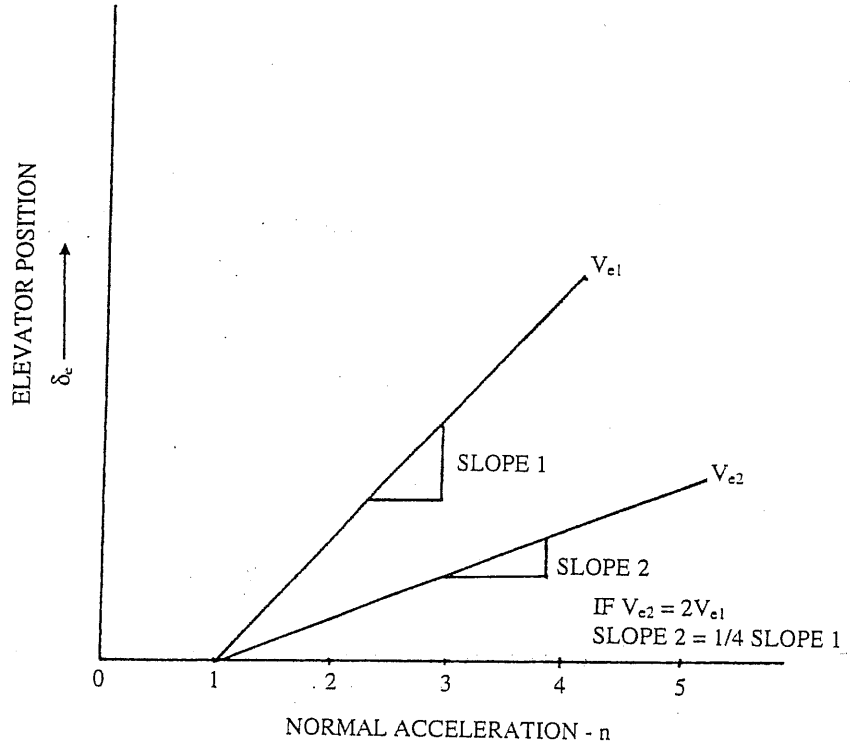


Fig. 24.5 Effects of varying equivalent airspeed on $d\delta_e/dn$ (Ref. 1).

term is the stick-free stability term while the second term is a pitch damping term.[1]

The c.g. location will affect the stick-force maneuvering stability much in the same way it affects the elevator position case. As the c.g. moves aft, both the stability term and the damping term become smaller.[1] As we would expect, the stick-free maneuver point is aft of the stick-free neutral point but, in most cases, ahead of the stick-fixed maneuver point.

If we increase altitude at a constant c.g. position, we find that the stick force per *g* decreases with both constant equivalent airspeed and constant Mach number. For the constant Mach number case, this is the reverse of what happens for elevator position per *g*.

If we vary the equivalent airspeed from that of trim, the stick force in maneuvering flight will vary considerably due to the V_e^2 term in the stick-free stability portion of Eqs. (24.9) and (24.10). The stick force per *g* will not vary, however, due to the fact that the V_e^2 term has dropped out of Eqs. (24.11) and (24.12). This fact can still cause problems in the measurement of stick force per *g* if airspeed is not held at a constant value during measurement.[1] See Fig. 24.6 (Ref. 1).

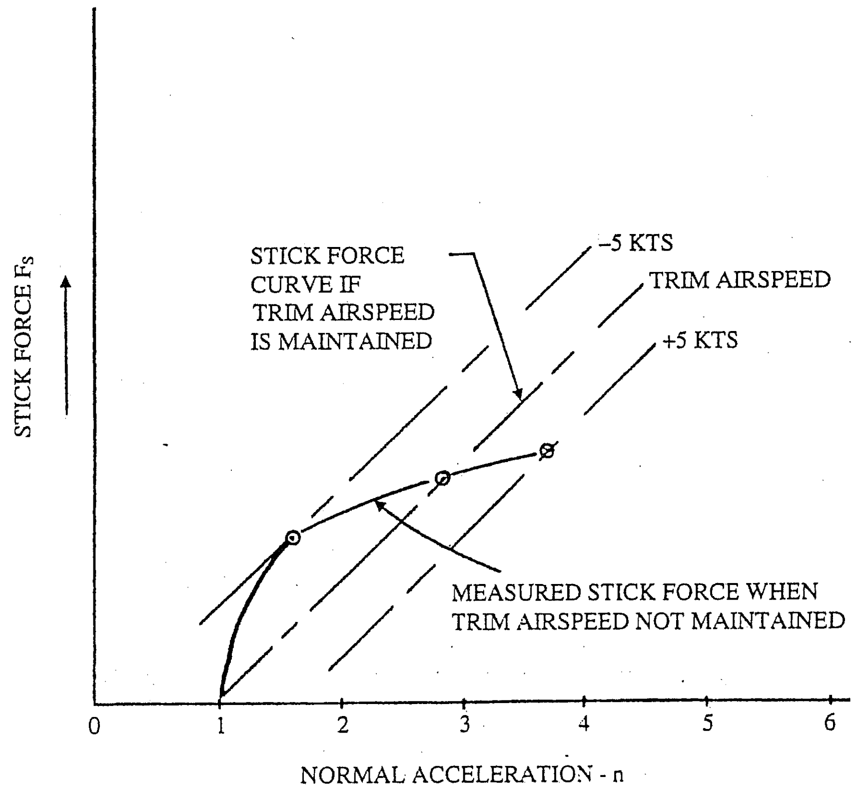


Fig. 24.6 Effects of not maintaining trim airspeed while collecting F_s vs n data.[1]

24.3.2 Stick-Force Maneuvering Stability—Irreversible Control Systems

Now let's turn our attention to the irreversible control system. There are several items for irreversible systems that are the same as reversible systems.[1] For instance: the same relations of control force exist between steady turns and pull-ups: i.e., more force is required in steady turns; the stick force per g decreases as c.g. moves aft; and poor speed control during measurement of stick force per g can result in erroneous data.[1]

First, we shall examine an irreversible control system in which the control force is proportional to the elevator deflection, or:[1]

$$F_s = K_1 \Delta\delta_e \tag{24.13}$$

where

K_1 = a constant that describes the characteristics of the system

For this type of system the stick force per g equation in steady turns can be written as:[1]

$$\left(\frac{dF_s}{dn}\right)_{ST} = -\frac{K_1}{C_{m_{\delta_e}}}\frac{W/S}{1/2\rho_0 V^2_{e_{trim}}}\left\{\left(\frac{dC_m}{dC_L}\right)_{fixed} + \frac{C_{m_{\dot\theta}}\rho g\bar{c}}{4(W/S)}(1 - 1/n^2)\right\} \quad (24.14)$$

It is interesting to note from this equation that stick force per g is dependent on stick-fixed stability rather than stick-free stability.[1]

If we have an irreversible system that incorporates a dynamic pressure sensor so that:[1]

$$F_s = K_2 q \Delta\delta_e \quad (24.15)$$

where
K_2 = the constant describing the characteristics of the control system

The stick force per g for this "q-feel" system may be written as:[1]

$$\left(\frac{dF_s}{dn}\right)_{ST} = -\frac{K_2(W/S)}{C_{m_{\delta_e}}}\left\{\left(\frac{dC_m}{dC_L}\right)_{fixed} + \frac{C_{m_{\dot\theta}}\rho g\bar{c}}{4(W/S)}(1 - 1/n^2)\right\} \quad (24.16)$$

For this system the influence of the trim airspeed on stick force per g is the same as that for the reversible control system.[1]

24.4 Compressibility Effects

When the airplane enters the regime of compressibility, two things happen that affect the maneuvering stability. First, the wing aerodynamic center shifts aft. This causes a large increase in static longitudinal stability. In addition, shock waves may form on the horizontal tail and limit control effectiveness. Both of these effects tend to increase maneuvering stability.[1]

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[1]Langdon, S. D., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-TTM No. 103, 1 Aug. 1969.

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25
Maneuvering Stability Methods and Data Reduction

25.1 Introduction

Like other areas of stability and control flight testing, evaluation of maneuvering stability requires significant inputs from the test pilot on the suitability of the aircraft to perform the maneuvering task of its mission. These inputs should be in addition to the qualitative evaluation of the airplane's ability to meet applicable regulations or specifications. The test pilot may form an opinion of the aircraft's suitability to perform the maneuvering task of its mission from flights other than the quantitative maneuvering flight test. Because of this we will discuss the factors making up pilot opinion and the quantitative evaluation separately.

25.2 Federal Aviation Administration Regulations

FAA regulations prior to FAR Part 23, Amendment 14, did not contain any requirements for maneuvering stability. This lack of requirements led to a number of accidents with airplanes that had low values of maneuvering stability and a requirement was added to the later versions of FAR Part 23.

25.2.1 Federal Aviation Regulation Part 23.155 Elevator Control Force in Maneuvers[1]

With Amendment 14, FAR 23.155 was added to the regulation. It states that the elevator control force in pounds necessary to achieve the positive limit maneuvering load factor must not be less than the greater of the takeoff gross weight/100, or 20 lb, for airplanes with wheel controls. For airplanes with stick controls these values change to the takeoff gross weight/140, or 15 lb. However, the force need not be greater than 50 lb for wheel controls or 35 lb for stick controls. These forces are measured in a turn with the trim setting used for level flight at a trim airspeed of V_{NO} and 75% power for reciprocating engines or maximum continuous power for turbine engines. The trim airspeed should not exceed V_{NE} or V_{MO}/M_{MO}, whichever is appropriate. The regulation also states that there should be no excessive decrease in the slope of the stick force per g curve as the load factor is increased.

25.2.2 Advisory Circular No. 23-8A (Ref. 2)

Although FAR 23.155 specifies that the F_s/g is to be measured in a turn at the level flight trim speed it does not specify a method. The advisory circular also does not specify a method. Therefore, either the steady turn or windup turn methods discussed later could be used. The advisory circular does give some cautions regarding stick-force lightening with potential for pitch up and discusses that the control force can be qualitatively evaluated if the aircraft enters the buffet boundary during the test. The advisory circular calls out two trim airspeeds for conducting the test. One at the maximum level flight speed and another at the maneuvering airspeed, V_A. It also specifies the power setting to be 75% MCP or the maximum power setting selected by the applicant as a limitation. The advisory circular states that compliance with the regulation may be shown in one of two ways. One way is to measure the load factor at the maximum allowable control force of 50 or 35 lb, depending upon the type of control. This method could only be used for aircraft that had high values of F_s/g. The second method, which is the more normal method, is to read the control force in even increments of load factor and plot a curve of the results. The advisory circular allows extrapolating this curve for 0.5 g to obtain the maximum load factor if the curve is linear, but only 0.2 g if the curve shows force lightening.

25.3 Evaluation by Pilot Opinion

A number of factors contribute to the pilot's opinion of the airplane during maneuvering tasks. Two items that help to form the pilot's opinion of the aircraft during maneuvering flight are longitudinal control system breakout force and friction. Friction in a control system is always undesirable. High friction will cause poor control feel during maneuvering, and may mask the airplane's actual stick force per g gradient at low values of normal acceleration. A reasonable amount of breakout force is desirable during maneuvering since it can reduce sensitivity in control feel about trim, and prevent pilot induced oscillations. However, excessive breakout force may cause the pilot to feel a lag in the control system and cause overcontrolling.[3]

Another control system item that may affect the pilot's opinion of the airplane during maneuvering is free-play. Free-play in the control system makes precise tracking task at low values of normal acceleration difficult. This causes the pilot to fly slightly out of trim so that the plane is always on one side or the other of the free-play dead-band. Attempts should be made to keep control system free-play to a minimum.[3]

Residual control system oscillations after a longitudinal control input are undesirable. During rapid maneuvering they produce an objectionable control feedback.[3]

Positive stick centering is a desirable feature during maneuvering since it allows the pilot to return to trim by releasing control pressure.[3]

The primary factor in the pilot's opinion of the airplane during maneuvering is the variation in stick force with normal acceleration called stick force per g. The gradient of the stick force per g curve should be a function of the airplane's mission, its design load factor, and the type of longitudinal control in the cockpit.[3,4]

First if the airplane's mission is such that it requires extensive maneuvering then the stick force per g gradient should not be so high as to tire the pilot. It should be high enough, however, to prevent inadvertent over-stressing of the airplane.[4]

If the airplane is designed with a low load factor such as a bomber or transport aircraft, then the stick force per g gradient should be large. Such aircraft will not be maneuvered extensively, and will normally have a control wheel that allows for the pilot to accept larger stick force per g gradients. The control wheel gives the pilot more leverage than does a stick control, and also allows him to use both hands.[3]

An airplane with a side stick controller, such as is appearing on more recent fighter designs, would require a very low stick force per g gradient since the pilot cannot exert great force on such devices.

Stick force per g is normally measured in a steady state condition. However, the transient stick force per g gradient should also be sufficient to prevent overstress of the airplane due to a rapid longitudinal control input. Since these transient forces are difficult to measure, pilot opinion is normally relied on for this information.[3]

Elevator position per g is also an important parameter in the pilot's opinion of the airplane. The criteria for this parameter is that trailing edge up elevator should increase with increasing load factor. Although elevator position per g is an important parameter to the pilot, it is not as important as stick force per g (Ref. 3).

It has also been found that some stick motion with increasing load factor is important to the pilot's opinion of the airplane.[3] This control motion improves control feel for the pilot and allows him to determine when he has reached control stops.

What has just been described are some of the factors that affect the pilot's opinion of the airplane during maneuvering. It is important to seek out the pilot's opinion on the maneuvering mission effectiveness, since not all of the above factors are measurable. We would hope, however, that the factors that are measurable would verify the pilot's opinion.

25.4 Flight Test Methods for Quantitative Evaluation

Now let us turn our attention to the measurable quantities of maneuvering stability and the methods we use to measure them. The most common parameters measured are stick force per g and elevator position per g, stick position per g and n/α. The data obtained from these tests may be used for comparison with regulatory requirements or for extrapolating neutral points. There are five methods that may be used to obtain maneuvering stability data. They are:

1) steady pull-ups
2) steady pushovers
3) wind-up turns (slowly varying g method)
4) steady turns (stabilized g method)
5) constant g

25.4.1 *Steady Pull-Ups*[3,4]

This method involves obtaining maneuvering stability data by varying normal acceleration with pitch rate during wings level pull-ups. To perform this method one first establishes the trim condition at the test altitude and records the trim data. Once this is accomplished a zoom climb should be entered, without changing trim or power settings, followed by a push-over to enter a shallow dive toward the trim altitude. When the airspeed approaches the trim airspeed up elevator is applied to establish a pitch rate that will place the aircraft back on the trim airspeed at the desired load factor. During the short period of time that the aircraft is stabilized in this condition data should be recorded. The magnitude of the zoom climb, push-over, and pull-up will depend upon the desired load factor, with larger maneuvers being required for larger load factors.

Airspeed control is critical on this maneuver, and any data on which the airspeed was more than ± 5 kn from the trim airspeed should be discarded. Altitude should be within ± 200 ft of the trim altitude, and pitch attitude should be within $\pm 15°$ of the trim attitude.

The normal acceleration should be increased in even steps up to the maximum acceleration desired, or the onset of stall buffet.

25.4.2 *Steady Pushovers*[3]

This maneuver is used to obtain maneuvering stability data at less than one g. It is essentially the reverse of the steady pull-up and is performed in that manner. The minimum normal acceleration obtainable by this maneuver is limited by the design negative load factor and the amount of down elevator available. In most cases the down elevator limit will be reached prior to achieving the maximum negative load factor.

25.4.3 *Wind-Up Turns (Slowly Varying g Method)*[3,4]

The wind-up turn is an easy method to obtain a large amount of data in a single test maneuver. To perform the wind-up turn one must first trim the aircraft to the desired conditions at the test altitude and record the trim data. The aircraft is then climbed 500–1000 ft above the trim altitude, trim power reset, and trim airspeed reobtained. The aircraft is then smoothly and slowly rolled into the windup turn while maintaining trim airspeed. If an automatic data recording device is installed, data may be recorded from the initiation of the turn. If not, data should be collected in even increments up to maximum acceleration or stall buffet. Airspeed and altitude limitations for this method are the same as those for the steady pull-ups method. Wind-up turns would be performed to the left and right to check for any turn direction effects on the aircraft maneuvering stability.

25.4.4 *Steady Turns (Stabilized g Method)*[3,4]

This method is used primarily for testing transport and bomber aircraft, and for fighters in the power approach configuration. The method is performed by first trimming the aircraft at the test altitude, and recording the trim data. The next step is to climb the aircraft above the test altitude and reset the trim power. The aircraft is then rolled into a 15 deg bank, and the nose is lowered to obtain and maintain the trim airspeed. Once the airspeed and bank angle are stabilized the data should be recorded. The bank angle is increased another 15 deg and the procedure repeated. Data points are obtained every 15 deg up to 60 deg, and at 0.5 g increments up to the limit load factor or stall buffet after 60 deg has been reached. Airspeed and altitude limitations are the same for this method as for the other methods.

25.4.5 Constant g Method[4]

This method may also be used to determine the buffet or stall envelope of the airplane. To perform the method the aircraft is trimmed at the test altitude and the maximum airspeed for the test. The aircraft is then placed in a constant

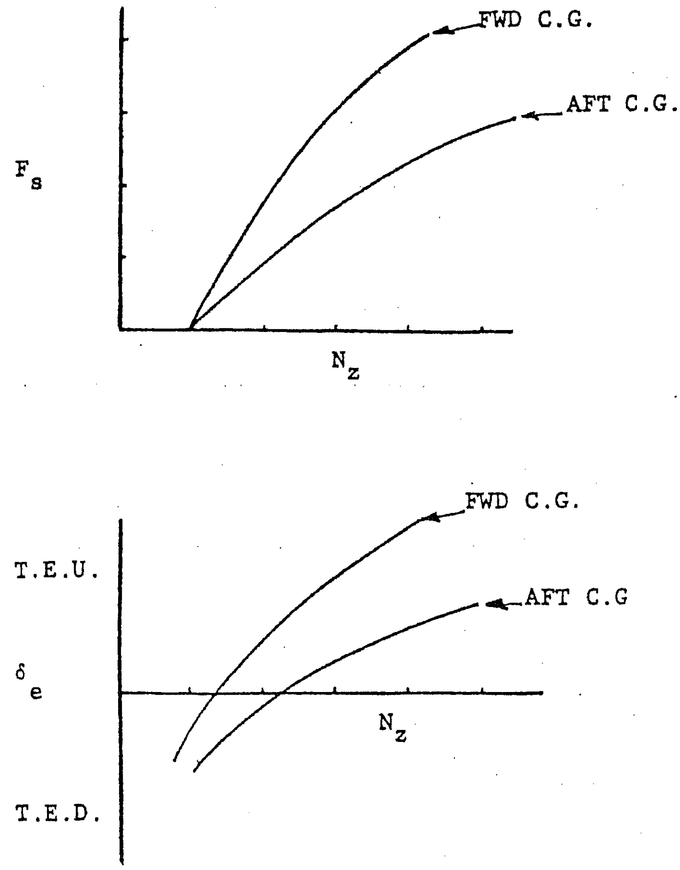

Fig. 25.1 Maneuvering stability data plots.[4]

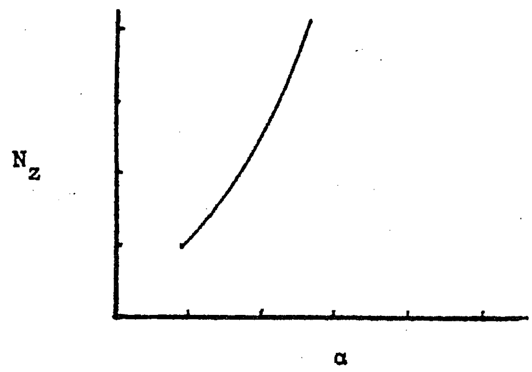

Fig. 25.2 Load factor vs angle of attack.[4]

g turn, data recording started, and the aircraft is climbed or descended to obtain a 2–5 kn/s airspeed bleed rate. The primary parameter to maintain during the test is the constant load factor. The airspeed bleed rate is a secondary parameter. The test altitude should be maintained within ±200 ft. Should the aircraft go outside this band the test should be discontinued, and started again within the band at an airspeed slightly above where it was discontinued. Due to the rapidly changing airspeed, this method requires the use of an automatic data recording device.

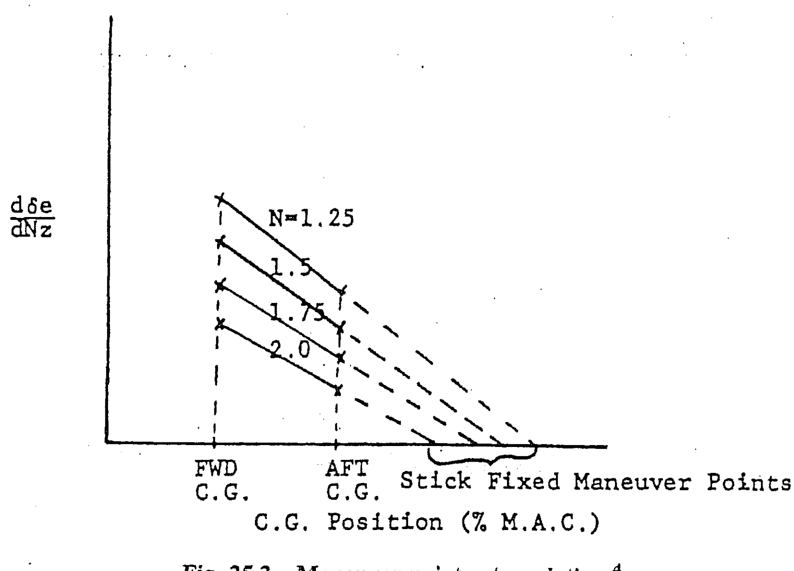

Fig. 25.3 Maneuver point extrapolation.[4]

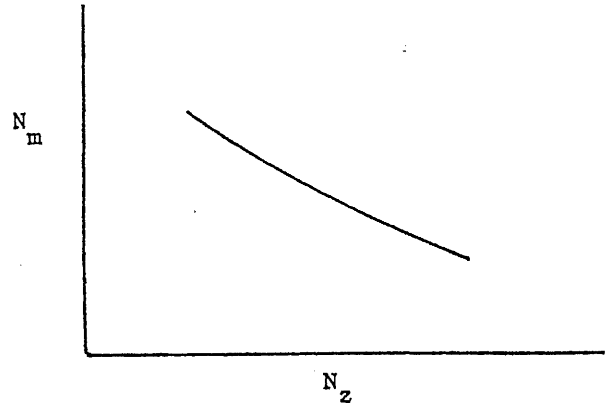

Fig. 25.4 Stick-fixed maneuver points vs load factor.[4]

25.5 Data Reduction Techniques

Once the data has been obtained by use of one of the methods just described, we must present it in some meaningful form.

The first step in any data reduction sequence is to correct the observed data for instrument and other errors from the calibration curves.

Next we plot stick force F_s, and elevator position δ_e vs load factor N_Z; see Fig. 25.1 (Ref. 4). Then plot load factor vs angle of attack α; see Fig. 25.2 (Ref. 4). If all we are concerned about is comparison with specifications or

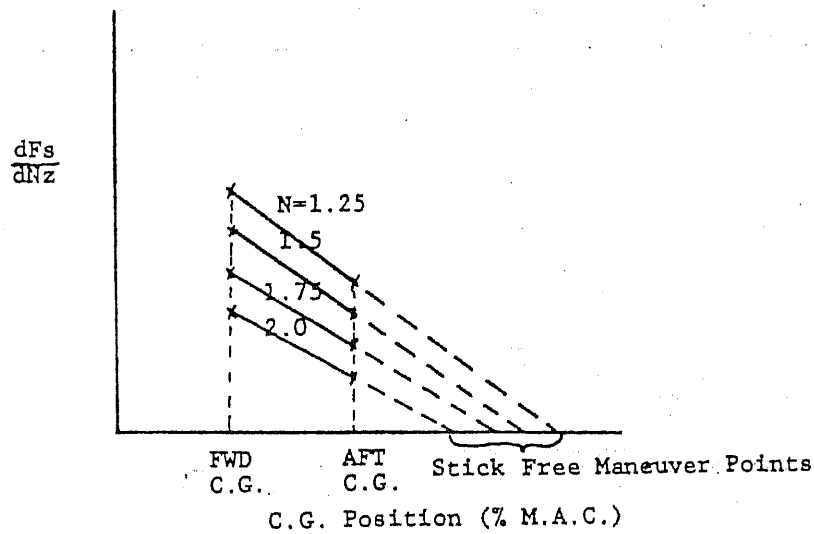

Fig. 25.5 Stick-free maneuver point extrapolation.[4]

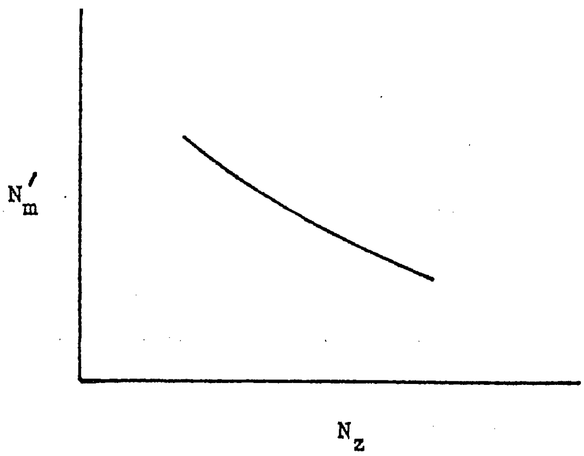

Fig. 25.6 Stick-free maneuver point vs load factor.[6]

regulations, then this may be as far as we need to go. However, if we wish to determine maneuvering stability margin, maneuver points, and other aerodynamic data then we must perform other steps.

To determine the stick-fixed maneuver point N_M we need to take slopes of the δ_e vs N_Z curve at several values of N_Z for each c.g. tested. We then plot

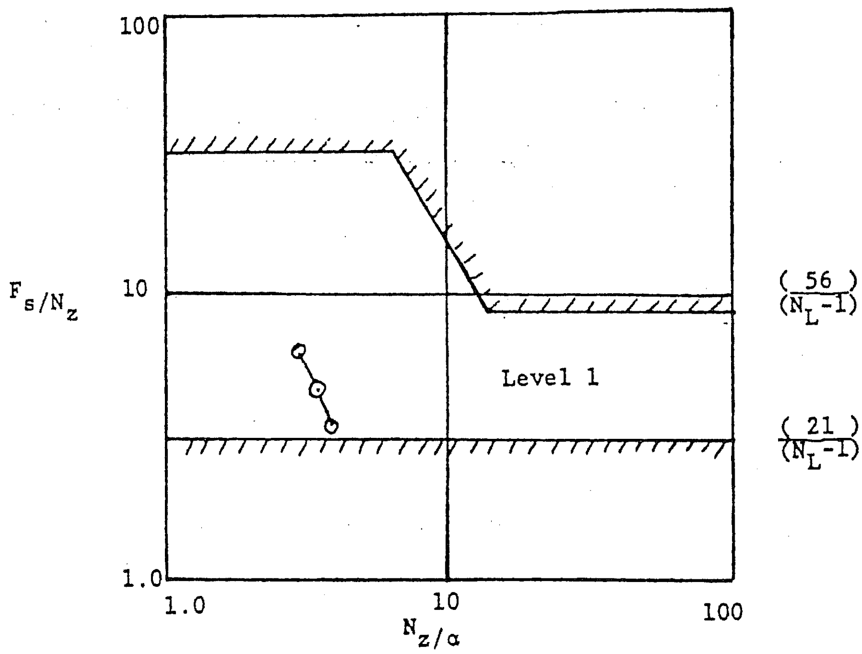

Fig. 25.7 Stick-force per g vs N_Z/α (Ref. 5).

the values of $d\delta_e/dN_Z$ vs c.g. positions as is shown in Fig. 25.3 (Ref. 4) and extrapolate the values of $d\delta_e/dN_Z = 0$ to determine stick-fixed maneuver points at each value of N_Z. We may then obtain plots of how maneuver point varies with load factor as is shown in Fig. 25.4 (Ref. 4).

The stick-free maneuver point may also be determined in a similar manner as is shown in Figs. 25.5 and 25.6 (Ref. 4).

If we need to know if the local stick force per g gradient meets the requirements of MIL-F-8785B, then we may wish to construct a log-log plot of F_s/N_Z vs N_Z/α such as is shown in Fig. 25.7 (Ref. 5).

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26
Longitudinal Control and Trim Theory and Flight Test Methods

26.1 Introduction

In our past discussions we have spoken mostly of the various types of longitudinal stability and how stability is affected by center of gravity travel. In every case we can say that stability is critical at the most aft c.g. and that the most aft c.g. limit is usually set by longitudinal stability considerations. All of the forms of longitudinal stability become more positive as the c.g. moves forward.

Elevator control power $C_{m_{\delta_e}}$ does not increase as rapidly as longitudinal stability does as the c.g. moves forward, and as a result the forward c.g. is limited by control power. This is especially true when the airplane is operating in ground effect, such as during takeoff and landing. When a wing operates in ground effect the downwash at the tail is reduced due to a decrease in the wing tip vortices.[1] This decrease in downwash at the tail creates a requirement for additional up elevator in order that the airplane can be rotated to the angle of attack for $C_{L\max}$. These two factors make the airplane control power limited at forward c.g. in the takeoff or landing configuration.

The ability of the trimming device to trim out longitudinal control forces throughout the speed range of the airplane is also subject to c.g. location. In this case the aft center of gravity is critical for high speed trim while the forward c.g. is critical for the low speed and large flap deflection cases.

Trim change forces with configuration change are another item that should be evaluated during longitudinal testing. It is desirable to keep trim changes with configuration change to a minimum in order to improve the longitudinal flying qualities of the airplane.

The regulatory requirements for longitudinal control are stated as:[2]

1) a maneuvering control requirement
2) a takeoff control requirement
3) a landing control requirement

The military specifications contain all three of these requirements, since many military airplanes are concerned with adequate maneuvering.

The FAA regulations do not contain a specific maneuvering control requirement but do contain requirements for takeoff and landing control.

Both the military specifications and the FAA regulations contain requirements for longitudinal trim.

26.2 Federal Aviation Administration Regulations

Both the Civil Air Regulations and Federal Aviation Regulations have recognized the need for requirements for longitudinal control, trim, and trim change during configuration change. For instance, rather than have requirements for takeoff distance for small airplanes, the CARs and early FARs had instead a requirement for nosewheel liftoff airspeed and rate of climb in the takeoff configuration that insured takeoff within a reasonable distance. Therefore, the regulations looked at controllability and trimability as necessary for a reasonable flying airplane.

26.2.1 Civil Aeronautics Regulation 3 (Ref. 3)

CAR 3.106 discusses controllability. It states that the airplane shall be satisfactorily controllable during takeoff, climb, level flight, dive, and landing with or without power. It also states that it shall be possible to make transitions from one flight condition to another without requiring exceptional piloting ability, or strength, and without exceeding any limits. The strength of pilot limits are not required to be measured unless they are near the specified limit, which for temporary application is 60 lb for stick controls and 75 lb for wheel controls. For prolonged application, this limit is only 10 lb.

CAR 3.109 covers longitudinal control and the requirements for trim change forces with configuration change. CAR 3.109(a), the longitudinal control portion, requires that in both clean and landing configurations, it shall be possible to pitch the airplanes nose down to accelerate to the best angle of climb speed, V_x, from airspeeds down to the stall, both power on and power off, with the airplane trimmed to $1.4V_{S1}$ if its takeoff gross weight is in excess of 6000 lb or $1.5V_{S1}$ for airplanes with TOGW less than that value. CAR 3.109(b)(c) are the control forces with configuration change portion of the regulation. There are seven configuration changes that require testing and the control forces shall not exceed the force requirements stated in 3.106. Items 1, 2, and 3 require extending and retracting the flaps while trimmed at 1.4 times the instantaneous value of the stalling speed. The out of trim control force shall be evaluated after each action. This is to be performed both power on and power off. Items 4 and 5 require the addition of takeoff power from a condition of idle power with the airplane trimmed as in the other items. This is to be performed from a clean configuration and a landing configuration. Item 6 requires obtaining airspeeds of $1.1V_{S1}$ and the lesser of $1.7V_{S1}$ or maximum flap extension speed with the power off and the landing gear and flaps down. The seventh item is called out as CAR 3.109(c). It requires that it shall be possible to maintain essentially level flight when flap retraction from any position is initiated during steady horizontal flight at $1.1V_{S1}$ with simultaneous application of maximum continuous power. It shall be possible to accomplish this without exceptional piloting skill.

CAR 3.112 states the requirements for trim. It requires for single engine airplanes that the airplane be capable of being trimmed to hands-off flight in three different flight conditions in several different configurations. The first flight condition is climb at maximum continuous power. In that condition it shall be possible to trim the airplane to an airspeed between V_X and $1.4V_{S1}$ with the landing gear retracted and the wing flaps up and in the takeoff position. The second flight condition is in a power-off glide with the landing gear extended at an airspeed not in excess of $1.4V_{S1}$ ($1.5V_{S1}$ for airplanes under 6000 lb maximum gross weight). In this case, it should be possible to trim the aircraft to hands off with the flaps retracted and with them extended at the forward c.g. at maximum gross weight, and at the most forward c.g. position approved regardless of weight. The third flight condition is during level flight with gear and flaps up at any airspeed from $0.9V_H$ to V_X or $1.4V_{S1}$.

Multiengine airplanes have an additional requirement stated in 3.112(b). It requires that it be possible with the critical engine inoperative to trim the aircraft in a climb in a clean configuration with the operative engine at maximum continuous power to an airspeed between V_Y and a speed of $1.4V_{S1}$.

26.2.2 Federal Aviation Regulation Part 23 (Ref. 4)

FAR Part 23.145 provides the requirements for longitudinal control. FAR 23.145(a) corresponds to CAR 3.109(a) and is essentially the same in the early FAR Part 23. However, later amendments to FAR Part 23 changed the trim airspeed from V_X to $1.3V_{S1}$. FAR 23.145(b) covers the control force with configuration change. It compares with CAR 3.109(b)(c), however there are changes in trim airspeed requirements and in some cases power settings. There are variances in these items between early versions of FAR Part 23 and later versions so attention should be paid to the regulation under which the airplane was originally certified if the flight test involve a Supplemental Type Certificate or an Amended Type Certificate. At this writing, FAR Part 23 has added three requirements to this section that were not covered in CAR 3. They include: a requirement to be able to maintain a power-off glide with no more than 10 lb of force for any combination of weight or c.g. with the landing gear and wing flaps extended; another requirement to be able to bring the airplane to a landing attitude by normal use of all controls except the primary longitudinal control; and third a requirement to demonstrate a control capability sufficient to achieve a load factor of $1.5\,g$ up to an airspeed of V_{MO}/M_{MO} to counter an inadvertent upset.

FAR Part 23.161 discusses the trim requirements and is the replacement for CAR 3.112. FAR Part 23.161(c)(1) and (2) concern the trim in the climb condition and the early versions of FAR Part 23 are essentially the same as CAR 3.112. The current version of FAR Part 23 have changed the trim speed requirement to that used in determining climb performance and have eliminated the requirement to perform the test with the flaps retracted. FAR Part 23 changed the power-off glide condition of CAR 3 to a power approach condition with power for a 3 deg descent. Early versions of the regulation required a trim speed between 1.3 and $1.5V_{S1}$, while later versions require a trim speed of $1.4V_{S1}$ and the approach speed V_{ref}. For level flight trim, early versions of FAR Part 23 read much the same as CAR 3. Current versions change the requirement from $0.9V_H$ to V_H and the minimum speed to just $1.4V_{S1}$ rather than either $1.4V_{S1}$ or V_X. The trim requirement for multiengine airplanes remains the same as in CAR 3 for early versions of FAR Part 23, but has changed to no more than a 5-lb out of trim control force in the current version. Current versions of FAR Part 23 also require that commuter category aircraft have no more than a 10-lb out of trim force for the longitudinal control in advent of an engine failure with the landing gear up and the flaps in the takeoff configuration if the takeoff path at an airspeed of V_2 extends beyond 400 ft above the runway.

26.2.3 Advisory Circular 23-8A (Ref. 5)

The advisory circular addresses longitudinal control by clarifying some of the language of the regulation. It states for the case of pushing the nose down from a low airspeed, that it is left to the test pilot to determine if there is sufficient control power to accomplish the task and if that occurs rapidly enough to suit the test pilot. It also states that the term "speeds below the trim speed" mean speeds down to the stalling speed. It also defines the "exertion of more control force than can readily be applied with one hand for a sort period of time" to mean the force for temporary application (75 lb for the longitudinal control). However, change 1 to this advisory circular says that there may be circumstances where pitch forces less than 75 lb may be necessary for safety. It allows that if such a case is found that the lower force should be established under FAR Part 21.21(b)(2). Prolonged forces are for forces that cannot be totally trimmed out.

The advisory circular discusses the weights and centers of gravity that should be tested. It suggests that the test be conducted at all corners of the c.g. envelope.

Regarding instrumentation the advisory circular states that special instrumentation is not required except for the measurement of the 10 lb maximum force required by 23.145(d). In that case a force gauge is necessary if this force cannot be trimmed to zero.

For the required tests for longitudinal trim, the advisory circular only discusses the center of gravity for testing and specifies that the most critical combinations of weight and c.g. should be tested.

26.3 Longitudinal Control

As was stated earlier, longitudinal control is a function of elevator deflection available and the longitudinal stability of the airplane. Since longitudinal stability is a function of c.g. position, the longitudinal control is also a function of c.g. position. One of the requirements for an airplane is that it be able to achieve $C_{L\max}$ at its forward c.g. position.[2] This is in order that it may achieve the lowest stall speed possible upon which to base other performance requirements. In addition, military fighters must achieve $C_{L\max}$ during maneuvering at forward c.g., which is a more severe requirement. The airplane is more stable during maneuvering, which causes a requirement for more up elevator. By making a plot such as Fig. 26.1 (Ref. 2) of elevator position vs C_L for various c.g. positions, we can determine the most forward c.g. position where $C_{L\max}$ can be achieved.

Since ground effect modifies the C_L capabilities of the airplane and creates less downwash at the tail, we cannot use the out-of-ground effect chart (Fig.

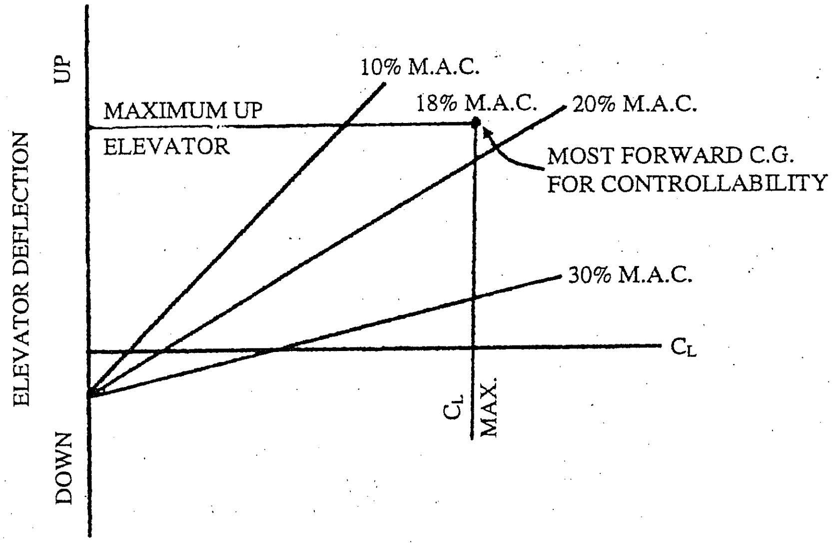

Fig. 26.1 Most forward c.g. determination.[2]

26.1) to determine the capabilities of control in ground effect. The requirements for control in ground effect are usually stated as a takeoff control requirement and a landing control requirement.

In order for an airplane to transition smoothly to flight, it must have sufficient control power to assume a takeoff attitude prior to reaching the liftoff speed. For tricycle gear airplanes this requirement is stated as a maximum speed for lifting the nosewheel from the runway. This speed is usually from 80 to 90% of the power-off stalling speed depending on the type airplane and the regulatory requirement, civilian or military.[2]

Fig. 26.2 (Ref. 2) shows the forces acting on an airplane during the takeoff roll. As can be seen from this figure, the rolling friction creates an additional

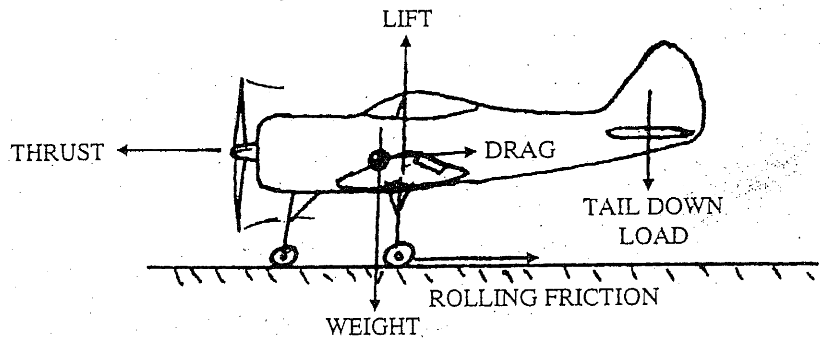

Fig. 26.2 Forces acting on aircraft during takeoff.[2]

nose down moment that also must be overcome by the horizontal tail. This additional moment may be counteracted on a propeller-driven airplane by the slipstream, which reduces the downwash at the tail and increases the dynamic pressure at the tail.[2] This is not the case for a jet aircraft or a T-tail airplane where the tail is out of the slipstream. In such cases, nosewheel liftoff may be the most critical control requirement. If flaps are used for takeoff this will also create an additional nose down moment that must be overcome by the horizontal tail.[2] In addition, flap deflection will also lower the stalling speed, which lowers the liftoff speed requirement.

For tail-wheel airplanes the requirement to lift the tail from the ground is critical at aft c.g. loadings. Since the stability of the airplane is the smallest at aft c.g., and any nose down moment helps, this requirement is generally not severe.

Controllability may also present a problem in the landing case. Again, the problem is most severe at forward centers of gravity.[2] Like takeoff, the airplane is operating in ground effect and the downwash at the tail is reduced.[2] However, this is where the similarity ends, since in the landing case the flaps are extended to their fullest extent and the power is off, so the slipstream is reduced. Although there is not any rolling friction to create an additional nose down moment, the other factors may be more powerful for an airplane with a conventional tail arrangement, and the landing case usually sets the up elevator requirement for this type airplane. For an airplane with a T-tail the landing requirement is not so severe, since the dynamic pressure at the tail is not as affected by the wing as in the conventional arrangement. Also, the reduction in downwash in ground effect does not affect this configuration as much as the conventional arrangement since the T-tail already operates in a reduced downwash environment.

From these discussions it can be seen that the aft c.g. limit is set by the minimum acceptable longitudinal stability, while the forward c.g. limit is set by the minimum acceptable controllability.[6]

26.4 Longitudinal Trim

For an airplane to have acceptable longitudinal handling qualities it must be possible for the pilot to reduce the control force to zero throughout most of the operating envelope. In addition, the control force generated by configuration change should be low enough so as to be easily handled by the pilot.[6] These trim changes should be in a direction so as to not cause an upset should the pilot be distracted by other piloting chores.[1] For instance, the nose should pitch down with flap extension rather than up into a potential stall.

As was stated earlier, it should be possible to trim the airplane to hands-off flight throughout most of its operating range. This is usually considered to be from 90% of the maximum level flight speed to the landing approach speed with the airplane in landing configuration.[3,4] Both the military specifications and the FAA regulations have specific values to which the airplane must trim in various configurations and c.g. positions. As might be suspected, the landing trim requirement is critical at the most forward c.g. condition just as is controllability. Also, as stated earlier the maximum speed trim requirement is critical at the aft c.g. condition.

For trim changes, both the magnitude and direction may vary with c.g. location, and for this reason are tested at both forward and aft c.g. locations. Figs. 26.3 and 26.4 (Ref. 7) show tables that are the trim change tests required for FAA certification of a light airplane. As can be seen from these tables, the trim changes to be measured are changes that the pilot will encounter in normal flying. The FAA regulations state that the control force for these changes should not be more than the pilot can handle with one hand.[3,4] The regulations

SECTION V, FLIGHT CHARACTERISTICS

A. **Most Forward C.G.**

23.141 3. **Longitudinal Control**
23.145

Configuration (Gear extended each case).	Information: Observed Data	Information: Trim Speed (see note below)	Does it require a change in the trim control or the exertion of more control force than can be readily applied with one hand for a short period to....
A. Glide Throttle Closed ( ) Gear Extended ( ) Flaps Retracted	Hp___Ft OAT___°F RPM___	___-IAS ___-CAS ___V_{S1}	Extend the flaps as rapidly as possible to landing position___° while maintaining approx. 40 percent above the instantaneous stall speed? No Yes F_r___Lbs.
B. Glide Throttle Closed ( ) Gear Extended ( ) Landing Flaps___°	Hp___Ft OAT___°F RPM___	___-IAS ___-CAS ___V_{S0}	Retract the flaps as rapidly as possible, while maintaining approx. 40 percent above the instantaneous stall speed? No Yes F_r___Lbs.
C. Climb M.C. Power ( ) Gear Extended ( ) Landing Flaps___°	Hp___Ft OAT___°F Power Setting___	___-IAS ___-CAS ___V_{S0}	Retract the flaps as rapidly as possible, while maintaining approx. same percent margin above the stall speed? No Yes F_r___Lbs.
D. Glide Throttle Closed ( ) Gear Extended ( ) Flaps Retracted ( )	Hp___Ft OAT___°F RPM___	___-IAS ___-CAS ___V_{S1}	Apply T.O. power quickly while maintaining the same speed? No Yes F_r___Lbs.
E. Glide Throttle Closed ( ) Gear Extended ( ) Landing Flaps___°	Hp___Ft OAT___°F RPM___	___Kts.-IAS ___Kts.-CAS ___V_{S0}	Apply T.O. power quickly while maintaining the same speed? No Yes F_r___Lbs.
F. Glide Throttle Closed ( ) Gear Extended ( ) Landing Flaps___°	Hp___Ft OAT___°F RPM___	___-IAS ___-CAS ___V_{S0}	Obtain and maintain speed at $1.1V_{S1}$? ___-IAS ___-CAS No Yes F_r___Lbs.
G. Glide Throttle Closed ( ) Gear Extended ( ) Landing Flaps___°	Hp___Ft OAT___°F RPM___	___-IAS ___-CAS ___V_{S0}	Obtain and maintain speed at the lower of $1.7V_S$ or V_F ___-IAS ___-CAS No Yes F_r___Lbs.

Trim Speed - Maximum weights of more than 6000 pounds use the speed that was used to determine the landing distances under § 23.75(a) or minimum trim speed whichever is higher.

Maximum weights of 6000 pounds or less use a speed between 1.3 V_{s1} and 1.5 V_{s1} or at the minimum trim speed whichever is higher.

Fig. 26.3 Longitudinal trim change with configuration change requirements at forward c.g. and TOGW.[7]

SECTION V, FLIGHT CHARACTERISTICS

B. Most Rearward C.G.

23.145 5. Controllability - Longitudinal:

CONFIGURATION (GEAR EXTENDED EACH CASE)	INFORMATION		DOES IT REQUIRE A CHANGE IN THE TRIM CONTROL OR THE EXERTION OF MORE CONTROL FORCE THAN CAN BE READILY APPLIED WITH ONE HAND FOR A SHORT PERIOD TO.......
	OBSERVED DATA	TRIM SPEED	
A. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) FLAPS RETRACTED ( )	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s1} ____ -IAS ____ -CAS	EXTEND THE FLAPS AS RAPIDLY AS POSSIBLE TO LANDING POSITION ____ ° WHILE MAINTAINING APPROX. 40 PERCENT ABOVE THE INSTANTANEOUS STALL SPEED? NO YES F_c ____ LBS.
B. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS ____ °	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s0} ____ -IAS ____ -CAS	RETRACT THE FLAPS AS RAPIDLY AS POSSIBLE, WHILE MAINTAINING APPROX. 40 PERCENT ABOVE THE INSTANTANEOUS STALL SPEED? NO YES F_c ____ LBS.
C. CLIMB M.C. POWER ( ) GEAR EXTENDED ( ) LANDING FLAPS ____ °	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s0} ____ -IAS ____ -CAS	RETRACT THE FLAPS AS RAPIDLY AS POSSIBLE, WHILE MAINTAINING APPROX. 40 PERCENT ABOVE THE INSTANTANEOUS STALL SPEED? NO YES F_c ____ LBS.
D. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) FLAPS RETRACTED ( )	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s1} ____ -IAS ____ -CAS	APPLY T.O. POWER QUICKLY WHILE MAINTAINING THE SAME SPEED? NO YES F_c ____ LBS.
E. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS ____ °	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s0} ____ -IAS ____ -CAS	APPLY T.O. POWER QUICKLY WHILE MAINTAINING THE SAME SPEED? NO YES F_c ____ LBS.
F. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS ____ °	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s0} ____ -IAS ____ - CAS	OBTAIN AND MAINTAIN SPEED AT $1.1V_{s1}$? ____ -IAS ____ CAS NO YES F_c ____ LBS.
G. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS ____ °	H_p ____ FT. OAT ____ °F RPM ____	____ V_{s0} ____ -IAS ____ -CAS	OBTAIN AND MAINTAIN SPEED AT THE LOWER OF $1.7V_{s1}$ OR V_F NO YES F_c ____ LBS.

*Trim Speed - Maximum weights of more than 6000 pounds use the speed that was used to determine the landing distances or minimum trim speed, if greater.

Maximum weights of 6000 pounds or less use a speed between $1.3V_{s_1}$ and $1.5V_{s_1}$ or minimum trim speed, if greater.

Fig. 26.4 Longitudinal trim change with configuration change requirements at aft c.g. and TOGW.[7]

say that this value is 75 lb for the light plane pilot[3,4] and 50 lb for the transport pilot.[8] For military airplanes the values are 10–20 lb (Ref. 9). Although these allowable forces are quite high, especially in the case of the light airplane, any test pilot worth his salt will press his designers to keep these forces as low as possible.

Trim changes are caused by a change in downwash at the horizontal tail due to the configuration change, such as flap extension or retraction and power addition or removal. Airplanes with conventional tail locations generally suffer large trim changes due to large changes in downwash and dynamic pressure at the tail with changes in flap position and power. On some airplanes the changes are so great as to require the addition of interconnects between the flap extension mechanism and the elevator trim.

It is in the area of trim changes that the T-tail may have its greatest advantage. Since the T-tail is out of the slipstream and the area of high downwash of the wing, it is not as sensitive to configuration or power changes as are conventional tail airplanes. Also, the trim changes generated by the T-tail airplane are generally in the correct direction and of small magnitude. Therefore, from the handling qualities standpoint T-tail airplanes are easy to fly.

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27 Methods for Improving Longitudinal Stability and Control

27.1 Introduction

Now that we have some idea as to the effects of various parameters on longitudinal stability and control, what can we do as a test pilot or flight test engineer to improve longitudinal stability and control once the airplane has reached the flight test stage?

It is quite obvious that once the airplane has reached the flight test stage a major design change is no longer desired due to schedule delays and cost factors. Also, if a change is required it is due to the fact that the designers were unable to predict with any degree of accuracy the control hinge moments and control coefficients. Therefore, what says they will be able to do better the next time around? Hinge moments and control effectiveness are difficult to predict even with the best information, so we usually find ourselves in flight test trying to fix some deficiency with minor changes.

Fortunately there are several minor things we can do to improve longitudinal stability and control. First, let us take a look at some of the simpler things and their effects on various parameters.

27.2 Control System Gadgets

The items that require the least change to the airplane are the things we would normally try first. These items fall into a category that can be called control system gadgets or gimmicks.[1] In making these changes we generally do not really change the natural stability and control, but, in essence, try to fool the pilot into thinking we have.[2] It is worthwhile to note that some of the items to be mentioned here work for both reversible and irreversible control systems.

27.2.1 Stick Centering Springs[1–4]

One method of adding an increment of control force is the stick centering spring (Fig. 27.1).[3] This spring tends to center the stick and adds a force increment which depends on stick displacement. This spring improves both apparent stick-free static and maneuvering stability at low speeds where control deflections are high, but the effect diminishes with an increase in airspeed, since control deflection also diminishes with airspeed. Also, since control deflections are high at forward c.g. and low at aft c.g., the centering spring increases stick force at forward c.g. more than aft c.g. This is in reverse to what we want

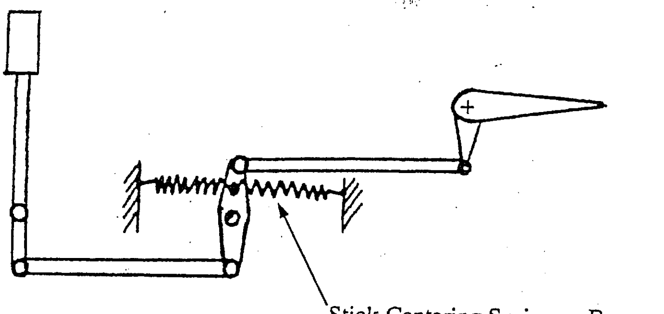

Fig. 27.1 Stick centering spring.[3]

and is one of the reasons the centering spring does not find much use with reversible control systems.

Another problem with this system for reversible control systems is that for various flight conditions the trim position of the stick is not always centered. This may result in unwanted force reversals. Due to these problems it is unlikely that you will see a stick centering spring used with a reversible longitudinal control system.

Such is not the case for the irreversible control system. In these systems the stick can always come back to the same place no matter what the trim condition. Also, the problem of diminishing force with increasing airspeed can be countered by an airspeed sensor that drives a control to stretch the spring as airspeed increases. For these reasons and its simplicity the stick centering spring is used quite frequently with irreversible control systems.

27.2.2 Elevator Down-Spring[1–4]

The elevator down-spring (Fig. 27.2)[3] is a method used quite frequently to improve longitudinal static stability. Like its name, the down-spring tends to make the elevator float down. With the spring installed, additional trim tab deflection must be added in order to overcome the spring force and cause the

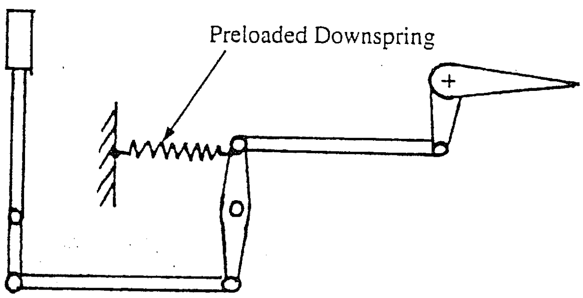

Fig. 27.2 Elevator down-spring.[3]

elevator to assume its trimmed position. This causes an increased stick-force gradient at all airspeeds. Since the force increment is not a function of stick position or normal acceleration the down-spring does not improve maneuvering stability.

The down-spring has a destabilizing effect on the long period dynamic longitudinal stability. This is because at airspeeds below trim the elevator is floated down, causing a steeper dive after the aircraft has reached the minimum speed. Then, with the trim tab adjusted further down (nose up) the airplane pitches nose up more. This steeper dive and steeper climb increases with each successive oscillation creating the dynamic instability.

Since the down-spring does not improve maneuvering stability and is destabilizing to dynamic longitudinal stability, it should not be used alone as a control force gadget.

As might be expected the down-spring is only used with reversible control systems.

27.2.3 Bobweight[1–4]

The bobweight is the device used most frequently in conjunction with a down-spring to improve longitudinal stability and control (Fig. 27.3).[3] The bobweight is a mass on an arm placed in the control system so as to provide a constant force. The bobweight is very versatile. From the standpoint of longitudinal static stability it reacts much like the down-spring since it, too, causes the elevator to float down.

When the airplane is maneuvered the force input provided by the bobweight is multiplied by the load factor, so a bobweight is a very effective device for improving maneuvering stability. The bobweight has the effect of actually shifting the maneuver points due to the effects it has on the pitch damping term.

Bobweights are also effective in damping out the phugoid motion. Since the airplane experiences load factors in excess of 1 g at the bottom of the phugoid motion, and less than 1 g at the top, the bobweight tends to restore the airplane to trim.

However, bobweights do have their problems. First, they add weight to the aircraft, which is undesirable. Second, they may adversely affect handling qualities in rough air or rapid maneuvering. This is because the inertia of the

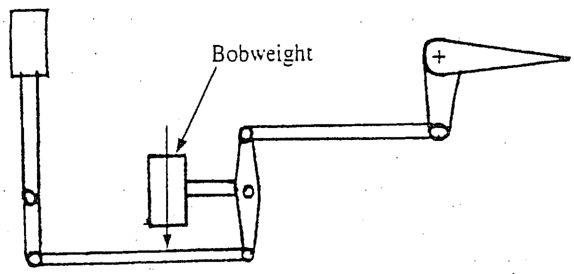

Fig. 27.3 Elevator bobweight.[3]

bobweight mass does not allow for instantaneous reaction to a change in acceleration, and in the case of turbulence or rapid maneuvering, may provide stick pumping or forces opposite to that desired. Although methods to counteract the turbulence problem, such as that shown in Fig. 27.4 (Ref. 1), have been developed, they only increase the weight and complexity of the control system. For this reason systems like that shown in Fig. 27.4 are not in widespread use.

The bobweight is a quite versatile device. It may be used to increase or reduce longitudinal control forces, depending upon its placement in the control system and relation to the aircraft c.g.

27.2.4 Spring-Weight Combinations

In order to reduce the weight penalty of the bobweight, while eliminating the dynamic and maneuvering problems of the down-spring, the two devices are used in combination. By tailoring the size of the bobweight with the spring constant of the down-spring it is usually possible to adjust static, dynamic, and maneuvering longitudinal stabilities to desired values.

The combination of weight and spring can be designed to meet space limitation problems. An example of such an arrangement is shown in Fig. 27.5.

27.3 Elevator Tabs

If springs and weights are not sufficient for the task, or if they increase weight or control system complexity more than desired, then we may need to move up in the complexity of the change. The next step up in complexity would be one of the several varieties of elevator tabs. These tabs consist of:

1) balance tab
2) servo tab
3) spring tab
4) spring-loaded tab

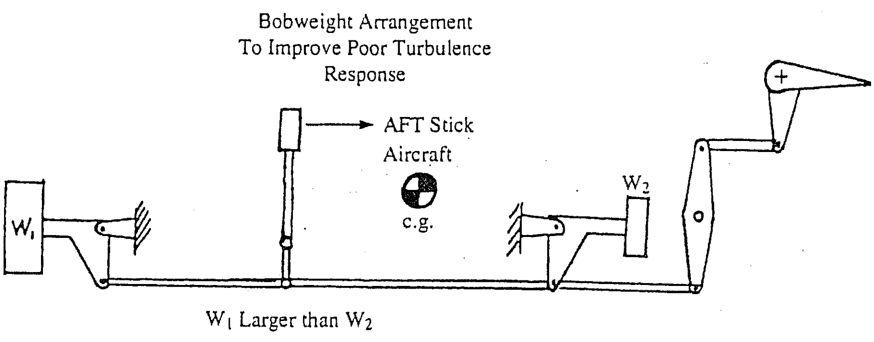

Fig. 27.4 Bobweight arrangement to improve turbulence response.[3]

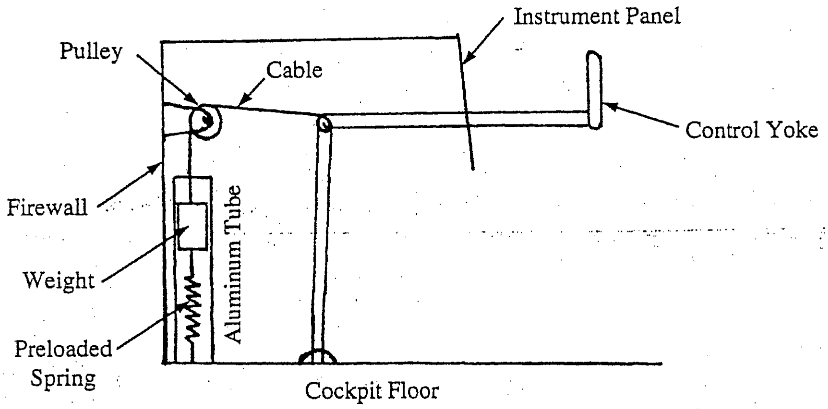

Fig. 27.5 Down-spring/bobweight combination.

27.3.1 Balance Tab[1–4]

The balance tab is a geared tab that can be made to either lead or lag elevator movement. This is accomplished by the method in which the tab is connected to the stabilizer as is shown in Fig. 27.6 (Ref. 3). The gearing ratio of the tab may be adjusted by the size of the control horn or push-rod as shown in Fig. 27.6. The leading tab is used to increase control forces, while the lagging tab will decrease forces. As a result, lagging tabs are not used very frequently in longitudinal control systems, since nearly always a force increase is needed.

The leading balance tab is a very effective device for "fixing" longitudinal stability problems. The leading tab reduced the tendency of the elevator to float, and as a result actually shifts the stick-free neutral point aft. It is also effective for increasing stick-free maneuvering stability since its input is a

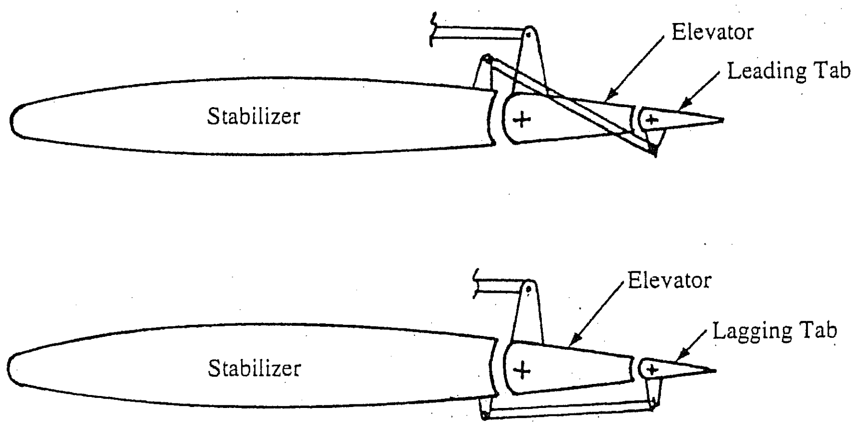

Fig. 27.6 Leading and lagging balance tabs.[3]

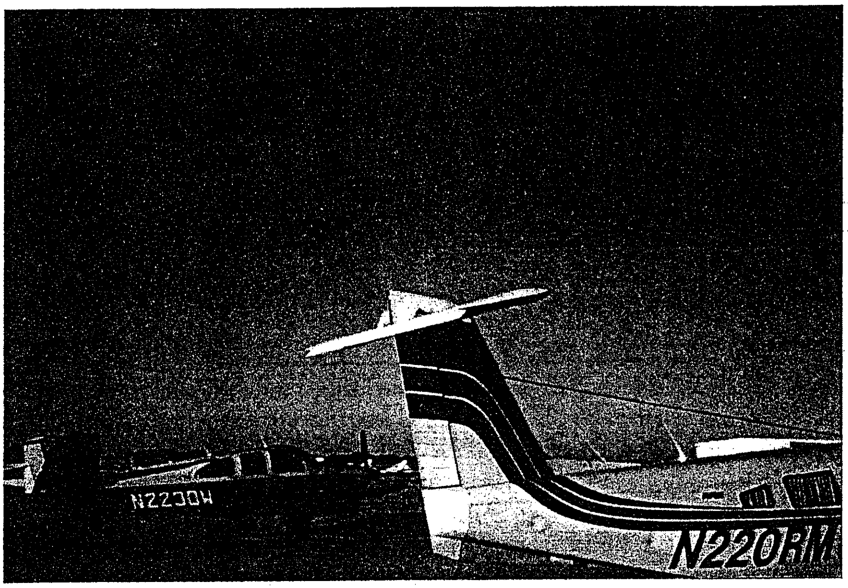

Fig. 27.7 Piper T-tail Lance with leading balance tab.

function of elevator deflection. The leading tab also works well for improving dynamics since it reduces the floating tendency. However, particular attention should be paid to flutter considerations when using these devices. Figs. 27.7 and 27.8 show the application of balance tabs on a general aviation aircraft and upon a WW II fighter.

27.3.2 Servo Tab[1–4]

The servo tab is a tab that actually drives the control surface such as is shown in Fig. 27.9 (Ref. 3). The control system is connected to the servo tab. Moving the control deflects the tab which moves the surface. In this manner large control surfaces may be deflected with reasonable control forces. Servo tabs found use on large transport and bomber aircraft prior to the advent of hydraulicly boosted controls.

27.3.3 Spring Tab[1–4]

A spring tab is a variation of the servo tab. A spring is added to the systems as is shown in Fig. 27.10 (Ref. 3). This spring restores some of the control force removed by the servo tab. By varying the spring constant the control forces may be tailored to desirable levels.

The F-80 Shooting Star jet fighter and its two-place T-33 trainer have a spring tab on the elevator to reduce control forces at high speed. The ailerons of this aircraft were irreversible hydraulicly actuated surfaces.

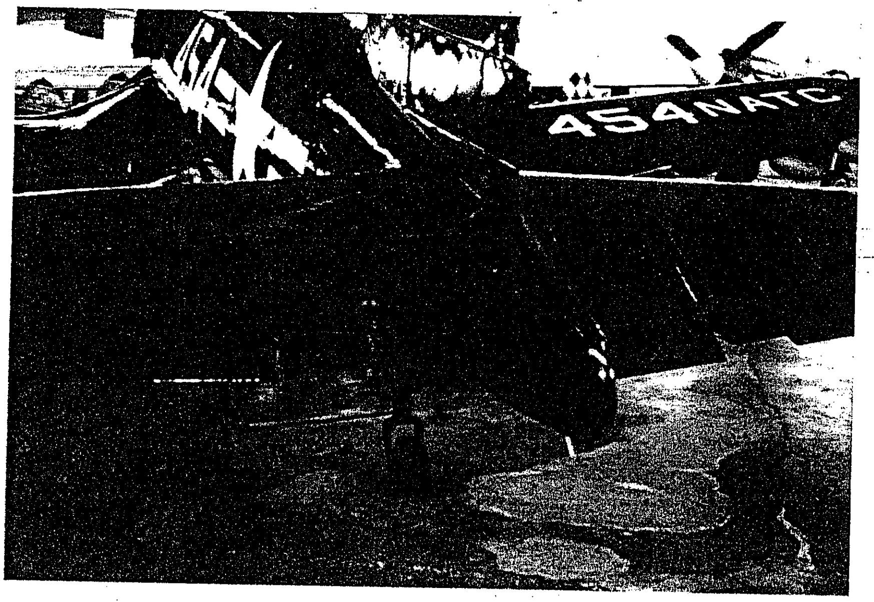

Fig. 27.8 F4U Corsair horizontal tail with leading balance tab.

27.3.4 Spring-Loaded Tab[1,3,4]

The spring-loaded tab is a device that, in effect, acts very much like a down-spring. The reason for this is that the tab has a preloaded spring about its hinge that causes it to deflect upward fully against its stop when at rest. When the aircraft moves, and the tab develops an aerodynamic hinge movement, the tab will begin to streamline. When the airspeed is increased to the point where the aerodynamic hinge moment equals the hinge moment of the spring then the tab will be streamlined. Above the speed where the tab comes off the up stop it provides a constant force gradient to float the elevator down, providing the down-spring effect. If flutter problems associated with such a tab can be solved it may be a more desirable "fix" than a down-spring, since it does not provide undesirable control forces during ground operation. A spring-loaded tab arrangement is shown in Fig. 27.11 (Ref. 3).

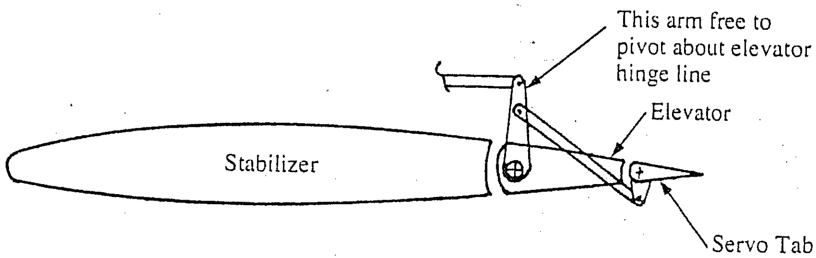

Fig. 27.9 The servo tab.[3]

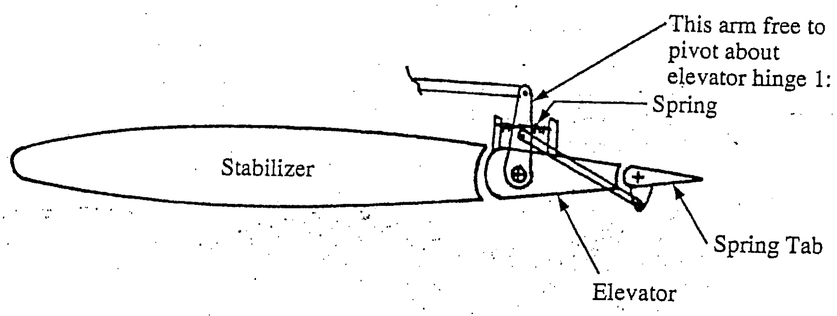

Fig. 27.10 The spring tab.[3]

27.4 Aerodynamic Balance

If our actions so far have not been able to cure our control force problems we may have to consider additional aerodynamic changes. Again, we would look to the simple changes first.

Like tabs, there are five aerodynamic balance changes that may be made to tailor control forces. They are:

1) overhang balance or a set-back hinge line
2) horn balance
3) internal balance with a flexible seal
4) blunt trailing edge or trailing-edge strips
5) beveled trailing edge

27.4.1 Overhang or Set-Back Hinge Line Balance[2,3]

Overhang balance is control surface area ahead of the hinge line that is distributed along the control surface. This is shown in Fig. 27.12 (Ref. 3). All aerodynamic balance must be considered with respect to the elevator hinge moments due to angle of attack $C_{h\alpha}$, or floating tendency, and the hinge moments due to elevator deflection $C_{h\delta}$, or restoring tendency. Aerodynamic balance such as overhang balance tends to reduce the floating tendency, due to the chordwise pressure distribution, and as a result increases the stick-free stability. The hinge moments due to elevator deflection, or restoring tendency, is also reduced by overhang balance. However, it is usually difficult to obtain a

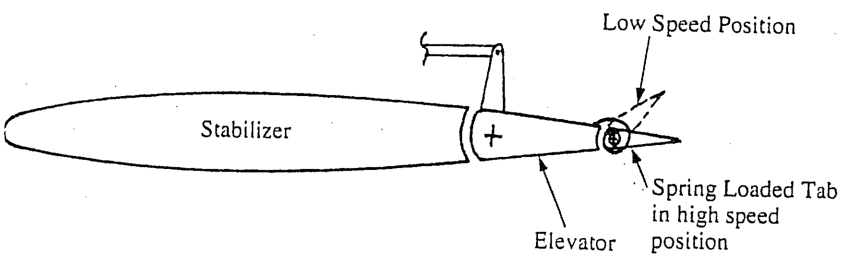

Fig. 27.11 Spring-loaded tab.[3]

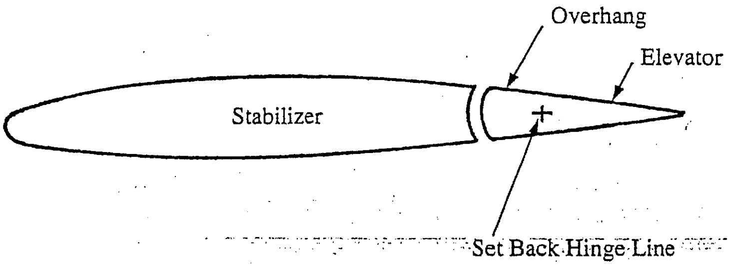

Fig. 27.12 Overhang or set-back hinge line balance.[3]

large amount of deflection balance without overbalancing the surface for angle of attack change. Since a change in overhang balance is a difficult change, both from the standpoint of estimating the effects and the physical change itself, it is probably one of the last things we would use to try and fix our problem.

27.4.2 Horn Balance[2,3]

Horn balance concentrates the balance area ahead of the hinge line. It is usually located at the tip of the elevator, as shown in Fig. 27.13 (Ref. 3), and may be of two types. Unshielded horn balance extends all the way to the leading edge of the stabilizer. Shielded balance only extends part way to the leading edge. Both are shown in Fig. 27.13.

The theory and effects of horn balance are the same as for overhang balance. It is used more often as a flight test "fix" than is overhang balance since it is structurally easier to change. Fig. 27.14 shows such a use on a general aviation airplane.

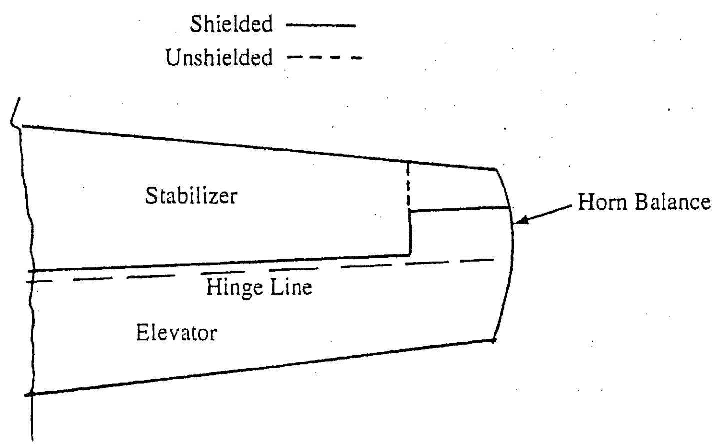

Fig. 27.13 Shielded and unshielded horn balance.[3]

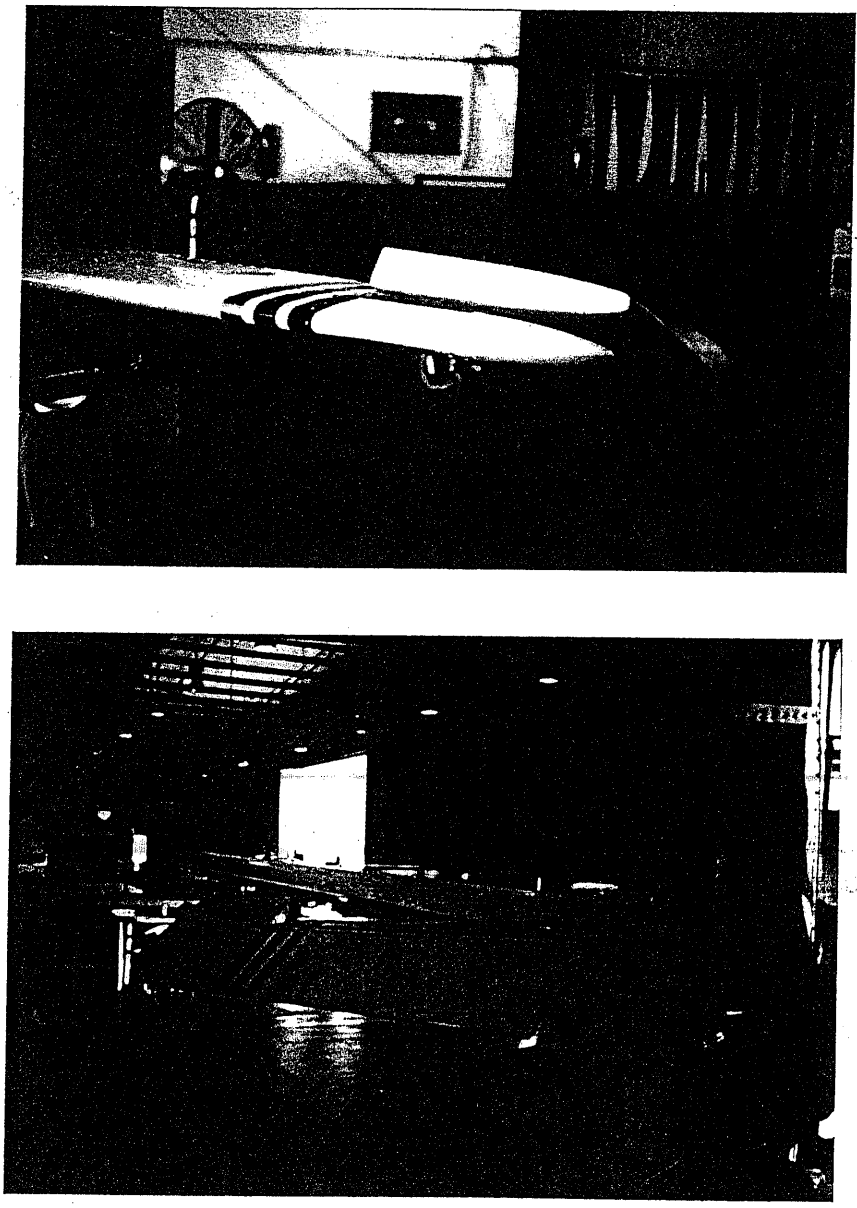

Fig. 27.14 Unshielded horn balance on the Micco 145A.

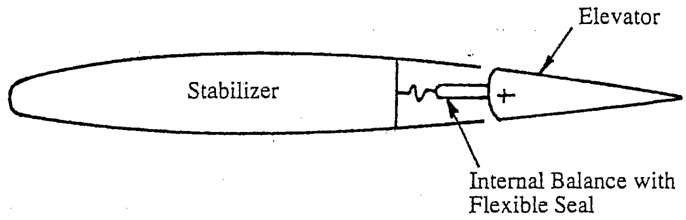

Fig. 27.15 Internal balance with a flexible seal.[3]

27.4.3 Internal Balance with Flexible Seal[2,3]

This type of aerodynamic balance is shown in Fig. 27.15 (Ref. 3). This type of balance works in the same manner as the other types of balance. It would generally be used as an original design item rather than a flight test fix, but the flexible gap seal may be used quite easily during flight test.

27.4.4 Blunt Trailing Edge or Trailing-Edge Strips[2]

The blunt trailing edge is a form of aerodynamic balance that tends to increase control forces through a stronger restoring tendency. It has been used successfully as a fix for elevator short period problems. It is a fairly simple change to make during flight testing, but may be difficult to tailor. The blunt trailing edge is shown in Fig. 27.16.

27.4.5 Beveled Trailing Edge[2,3]

The beveled trailing edge may be used to reduce control forces. It is the opposite of the blunt trailing edge. Beveling the trailing edge, as shown in Fig. 27.17 (Ref. 3), is not as simple a change as is blunting it.

All of the previously described fixes or gimmicks may be used separately or in combination. The objective of their use is to arrive at suitable control forces and handling qualities of the airplane based upon its mission.

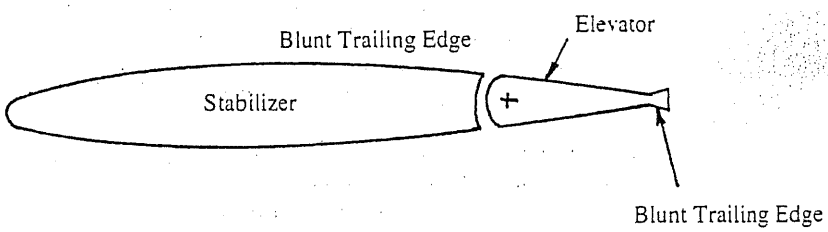

Fig. 27.16 Blunt elevator trailing edge.

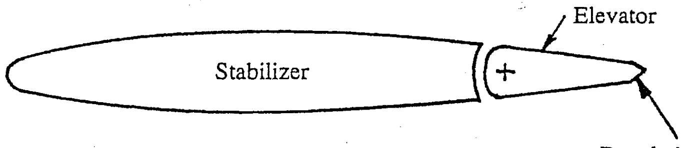

Fig. 27.17 Beveled elevator trailing edge.[3]

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