USNTPS-FTM-No. 103

U.S. NAVAL TEST PILOT SCHOOL
FLIGHT TEST MANUAL

FIXED WING STABILITY AND CONTROL
Theory and Flight Test Techniques

NAVAL AIR WARFARE CENTER
AIRCRAFT DIVISION
PATUXENT RIVER, MARYLAND

Revised January 1997

U.S. NAVAL TEST PILOT SCHOOL

FLIGHT TEST MANUAL

USNTPS-FTM-No. 103

FIXED WING STABILITY AND CONTROL
Theory and Flight Test Techniques

This Flight Test Manual, published under the authority of the Commanding Officer, U.S. Naval Test Pilot School, is intended primarily as a text for the pilots, engineers and flight officers attending the school. Additionally, it is intended to serve as a reference document for those engaged in flight testing. Corrections and update recommendations to this manual are welcome and may be submitted to:

Commanding Officer
U.S. Naval Test Pilot School
22783 Cedar Point Road
Patuxent River, MD 20670-5304

January 1997

BACKGROUND AND PURPOSE

This manual is primarily a guide for pilots and engineers attending the U.S. Naval Test Pilot School. However, it may be used as a guide in any fixed wing flying qualities investigation. The text presents basic fixed wing stability and control theory, qualitative and quantitative test and evaluation techniques, and data presentation methods. In most sections, more than one technique is described for each test. Generally, the best technique for a particular investigation will depend on the purpose of the investigation, the amount of instrumentation available, and the personal preference of the individual test pilot. The approach of the qualitative stability and control testing presented herein is an attempt to associate all flying qualities tests with particular pilot tasks required in the performance of the total mission of the airplane. The pilot's opinion of a particular flying quality will consequently depend primarily on the pilot workload while performing the desired task. Quantitative evaluation techniques presented may be used to substantiate pilot opinion or gather data for documentation of airplane characteristics. The performance of both qualitative testing and quantitative evaluation is considered essential for any successful flying qualities investigation.

TABLE OF CONTENTS

FIXED WING STABILITY AND CONTROL
Theory and Flight Test Techniques

CHAPTER ONE

INTRODUCTION

CHAPTER ONE

FIGURES

CHAPTER ONE

EQUATIONS

CHAPTER ONE

INTRODUCTION

1.1 PHILOSOPHY OF FLYING QUALITIES TESTING

The flying qualities of a particular airplane cannot be discussed unless the total mission of the airplane and the multitude of individual tasks associated with that total mission are defined. The definition of flying qualities leaves no other choice: "Flying qualities are defined as those stability and control characteristics which influence the ease of safely flying an airplane during steady and maneuvering flight in the execution of the total mission." The "total mission" will be initially determined when the need for a new airplane is realized. However, the mission may be diminished, magnified, or completely changed during the service life of the airplane. Therefore, in the formulation of a test and evaluation program for any airplane, the total mission must be defined and clearly understood by all test pilots and engineers involved in the program.

The individual tasks associated with the accomplishment of a total mission must also be determined before the test and evaluation program can be formulated. Although the individual tasks may be further subdivided, a military mission will normally require the pilot to perform the following tasks:

1. Preflight ground or deck operations.
2. Take-off and climb.
3. Navigation to a predetermined point.
4. Strategic or tactical maneuvering.
5. Navigation to a landing point.
6. Approach and landing.
7. Postflight ground or deck operation.

Because this manual is strictly concerned with flying qualities, many ground, deck, or in-flight tasks necessary for mission accomplishment will not be discussed. These tasks include attachment of payloads, maintenance, servicing, engine start and operation of navigation and weapon systems. Under severe emergency conditions, the pilot tasks may involve engine airstart, fire extinguishment, jettison of equipment, or simply abandoning the airplane without serious injury. These areas must be investigated on every military airplane and their importance cannot be overemphasized.

The tasks for which the most favorable flying qualities are required are the “essential” or “critical” tasks required by the total mission. For an aircraft which must perform air-to-air, air-to-ground, and/or reconnaissance functions (and training for those functions), the greatest emphasis must be placed on the flying qualities exhibited while performing the maneuvers required to accomplish these critical tasks. These tasks will, of course, vary greatly with the total mission of the airplane. In any case, adequate flying qualities must be provided so that take-off, approach, wave-off, and landing maneuvers can be consistently accomplished safely and precisely.

The prime reason for conducting flying qualities investigations, then, is to determine if the pilot-airplane combination can safely and precisely perform the various tasks of the total mission of the airplane. This determination can generally be made by the pure qualitative approach to stability and control testing. However, this is only part of the complete test program. Quantitative testing must also be performed in order to:

1. Substantiate, if possible, the pilot's qualitative opinion.
2. Document those characteristics of the airplane which particularly enhance or derogate some flying quality.
3. Provide data for comparing airplane characteristics and for formulating future design changes.
4. Provide base data for determination of future expansion of flight and CG envelope or future expansion of total mission.
5. Determine conformance or nonconformance with appropriate test specifications.

A balance between qualitative and quantitative testing must be achieved in any stability and control test and evaluation program.

1.2 RESPONSIBILITIES OF TEST PILOT AND ENGINEER

Almost every flight test and evaluation team will be composed of one or more test pilots and one or more project engineers. The team concept provides the necessary balance between qualitative testing (the pilot's opinion) and quantitative evaluation (the engineer's knowledge of theory, instrumentation, and specifications). The team concept does not imply, however, that the test pilot should be only a “driver”. To perform the necessary tests and evaluations, the test pilot must also have at least conversational knowledge of theory, instrumentation, and specifications. Furthermore, the engineer must possess a thorough knowledge of the pilot tasks required in performing a total mission in order to participate fully in formulation and conduct of the test and evaluation program.

1.2.1 The Test Pilot

The competent, productive test pilot must be highly proficient with the stick and throttle if he is to obtain accurate data. He must be trained and have well-developed observation and perception powers if he is to recognize problems and adverse characteristics. He must have a keen ability to professionally analyze test results if he is to understand and explain the significance of his findings. To fulfill these expectations, he must possess a superior knowledge of:

1. The airplane undergoing evaluation and airplanes in general.
2. The total mission of the airplane and the individual <u>pilot tasks</u> required to accomplish the mission.
3. Test techniques and associated theory required for qualitative testing and quantitative evaluation.
4. Specifications relevant to the evaluation program.
5. Technical report writing.

The test pilot's knowledge of the airplane must exceed the knowledge required just to “mechanically” operate the engine-airframe combination. The test pilot must also consider the effects of internal and external configuration on flying qualities.

In particular, a thorough knowledge of the flight control system is essential if the test pilot is to do a creditable job of stability and control testing. Many of the characteristics which shape the pilot's opinion of the airplane in performing a particular task are the direct result of the flight control system.

The successful test pilot must possess more than a superior knowledge of the particular test vehicle. He also needs flight experience in many different types of aircraft. Only by seeing “in person” the widely varying characteristics exhibited by different design and mission concepts can he prepare himself for accurate and precise assessments of particular design and mission concepts. Further, by flying many different types, he develops the quality of adaptability - he can easily and quickly adapt himself to the characteristics of a new airplane. When flight test time is severely limited by monetary and time considerations, this quality or trait is invaluable.

The total mission of the airplane must be perfectly clear in the test pilot's mind. To obtain this clear concept of the total mission, the test pilot must review and study the specific operational requirements on which the design was based, the detail specification under which the design was developed, and other planning documents. Knowledge of the individual pilot tasks required for total mission accomplishment is derived most easily from recent operational experience. (Recent operational experience in missions similar to the design mission of the airplane under evaluation is particularly advantageous.) If the test pilot does not have the advantage of the recent operational experience, he can gain knowledge of the individual pilot tasks from talking with other pilots, studying operational and tactical manuals, and/or visiting replacement pilot training squadrons.

The test pilot's knowledge of theory, test techniques, relevant specifications, and technical report writing may be gained through formal education or practical experience. The most beneficial, rewarding, and easiest road to knowledge in these areas is through formal study with practicable application at an established test pilot school. This education allows the pilot to converse with the engineer in technical terms which are necessary to describe flying qualities phenomena.

1.2.2 The Project Engineer

The successful project engineer must have at least general knowledge of the same items for which the test pilot is mainly responsible. Additionally, he must possess superior knowledge of:

1. Instrumentation requirements.
2. Formulation and coordination aspects of the test and evaluation program.
3. Data acquisition, reduction, and presentation.
4. Technical report writing.

The project engineer will normally be responsible for the determination of instruments required to carry out the investigation. This also involves determination of the ranges and sensitivities required and formulation of an instrumentation “specification” or planning document. His responsibilities also include witnessing or conducting weight and balance tests, engine calibrations, and fuel quantity system calibrations.

Because the engineer does not normally fly in the test airplane, and therefore is usually available in the project office, he is in the best position to coordinate all aspects of the program. This involves aiding in preparation and, if necessary, revision of the “test plan” and coordinating the order in which flights will be conducted. Additionally, the project engineer will normally prepare all test flight cards and be present to assist in all flight briefings and debriefings.

A great deal of the engineer's time will be spent in working with flight and ground test data. He must review preliminary data from contractor wind tunnel studies and flights. From this data, critical areas may be determined prior to actual military flight tests. During the actual flight tests, the engineer may monitor and aid in the acquisition of data through telemetry facilities and radio, or by flying in the test airplane. Following completion of flight tests, the engineer coordinates data reduction, data analysis, and data presentation.

The engineer's knowledge of technical report writing allows him to participate fully in the preparation of the report. He will write many parts of the report which do not require pilot opinion information. The engineer usually is given the arduous tasks of proofreading the entire manuscript and approving (for distribution) the first printed copy of the technical report.

1.3 CONCEPTS OF STABILITY AND CONTROLLABILITY

In order to exhibit satisfactory flying qualities, the airplane must possess a certain measure of both stability and controllability. The optimum "blend" depends on the total mission of the airplane. A certain degree of stability is necessary if the airplane is to be easily controlled by a human pilot. However, too much stability can severely derogate the pilot's ability to perform maneuvering tasks. The attainment of an optimum blend of stability and controllability should be the goal of the airplane designer. When the optimum blend is attained, flying qualities greatly enhance the ability of the pilot to perform the intended mission.

1.3.1 Stability

The airplane is a <u>dynamic system</u>, i.e., it is a body in motion under the influence of forces and moments <u>producing</u> or <u>changing</u> that motion. In order to investigate the motion of the airplane, it is necessary to establish first that it can be brought into a condition of equilibrium, i.e., a condition of balance between opposing forces and moments (not necessarily a "force time" condition from the pilot's standpoint). Then the stability characteristics of the equilibrium condition can be determined. The airplane is <u>statically stable</u> if restoring forces and moments are created which tend to restore it to equilibrium when disturbed from equilibrium. Thus, static stability characteristics must be investigated from equilibrium flight conditions, in which all forces and moments are in balance. The direct in-flight measurement of certain static stability parameters is not feasible in many instances. Therefore, the flight test team must be content with measuring parameters which only give indications of static stability. However, these indications are usually adequate to establish conclusively the mission effectiveness of the airplane and are more meaningful to the pilot than the numerical value of the stability derivities.

The pilot makes changes from one equilibrium flight condition to another through one or more of the airplane's <u>modes of motion</u>. These changes are initiated by excitation of the modes by the pilot and terminated by suppression of the modes by the pilot. These modes of motion may also be excited by external perturbations. The study of the characteristics of these modes of motion is the study of <u>dynamic stability</u>. Dynamic stability may be classically defined as the ability of the airplane to eventually regain original flight conditions after being disturbed. Dynamic stability characteristics are measured from nonequilibrium flight conditions during which the forces and moments acting on the airplane are not in balance.

Static and dynamic stability determine the pilot's ability to control the airplane. While static instability about any axis is generally undesirable, if not completely unacceptable, excessively strong static stability about any axis may derogate controllability to an unacceptable degree. For some pilot tasks, neutral static stability may actually be desirable because of the increased controllability which results. Obviously, the optimum level of static stability depends on the mission of the airplane.

Here the characteristics of the <u>modes of motion</u> of the airplane determine its dynamic stability characteristics. The most important characteristics are the <u>frequency</u> and <u>damping</u> of the motion. The <u>frequency</u> of the motion is defined as the “number of cycles per unit time” and is a measure of the “quickness” of the motion. The term <u>undamped natural frequency</u> is often used in describing airplane motion. It is the frequency of the motion if the motion exhibited zero damping.

Damping of the motion is defined as a progressive diminishing of its amplitudes and is a measure of the subsidence of the motion. The term <u>damping ratio</u> is often used in describing airplane motion. It is the ratio of the <u>damping which exists</u> to <u>critical damping</u>. The damping ratio of the airplane modes of motion has a profound affect on flying qualities. If it is too <u>low</u>, the airplane motion is <u>too easily excited</u> by inadvertent pilot control inputs or by atmospheric turbulence. If it is too <u>high</u>, the airplane motion following a control input is slow to develop and the pilot may describe the airplane as “sluggish.” The <u>mission of the airplane</u> again determines the optimum dynamic stability characteristics. However, the pilot always desires <u>some level of positive damping</u> of all the airplane's modes of motion.

Static and dynamic stability prevent unintentional excursions into dangerous ranges (with regard to airplane strength) of dynamic pressure, normal acceleration, and sideforce. The stable airplane is resistant to deviations in angle of attack, sideslip, and bank angle without action by the pilot. These characteristics not only improve flight safety, but allow the pilot to perform maneuvering tasks with smoothness, precision, and a minimum of effort.

1.3.2 Controllability

Controllability may be defined as the capability of the airplane to perform, at the pilot's wish, any maneuvering required in total mission accomplishment. The characteristics of the airplane should be such that these maneuvers can be performed precisely and simply with a minimum of pilot effort.

The pilot's opinion of controllability is shaped by several factors. The most apparent of these factors are the <u>initial response</u> of the airplane to a control input and the <u>total attitude change</u> which results. In addition, the cockpit control forces and deflections required to accomplish necessary pilot tasks are extremely important. These factors depend on the static and dynamic stability of the airplane and the characteristics of the flight control system. The complexity or degree of difficulty which the pilot encounters during maneuvering tasks is directly dependent on the stability characteristics of the airplane (Figure 1.1).

The reversed-transitional control movements shown in (d) are never required when the airplane possesses adequate stability; therefore, the nature of the control movements required while maneuvering the stable airplane are greatly simplified. (Although Figure 1.1 uses the <u>longitudinal or lateral cockpit</u> controller as an example, the same analysis would, of course, apply to the directional cockpit control.) The simplicity of control movements required in maneuvering the stable airplane significantly reduces the pilot expenditure of effort devoted to directly <u>flying the airplane</u>. Thus, he can devote more of his attention to <u>mission tasks</u>, which may involve placing weapons precisely on a target, or merely navigating from point to point in space (Figure 1.2).

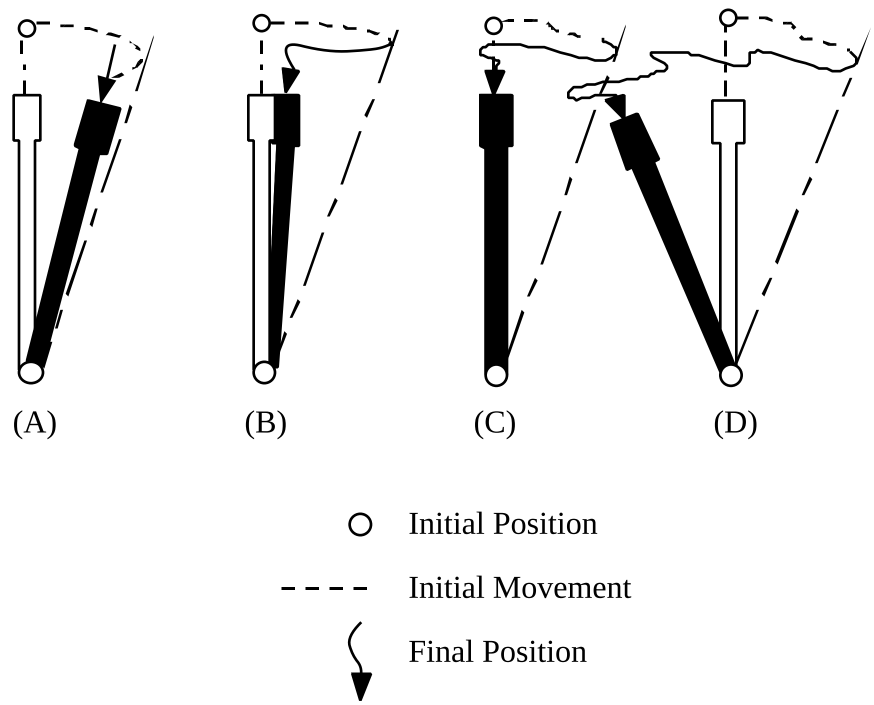


Figure 1.1
Control Movement Required in Changing from One Steady State Flight Condition to Another

(A) Stable Airplane
(B) Weakly Stable Airplane
(C) Neutrally Stable Airplane
(D) Unstable Aircraft

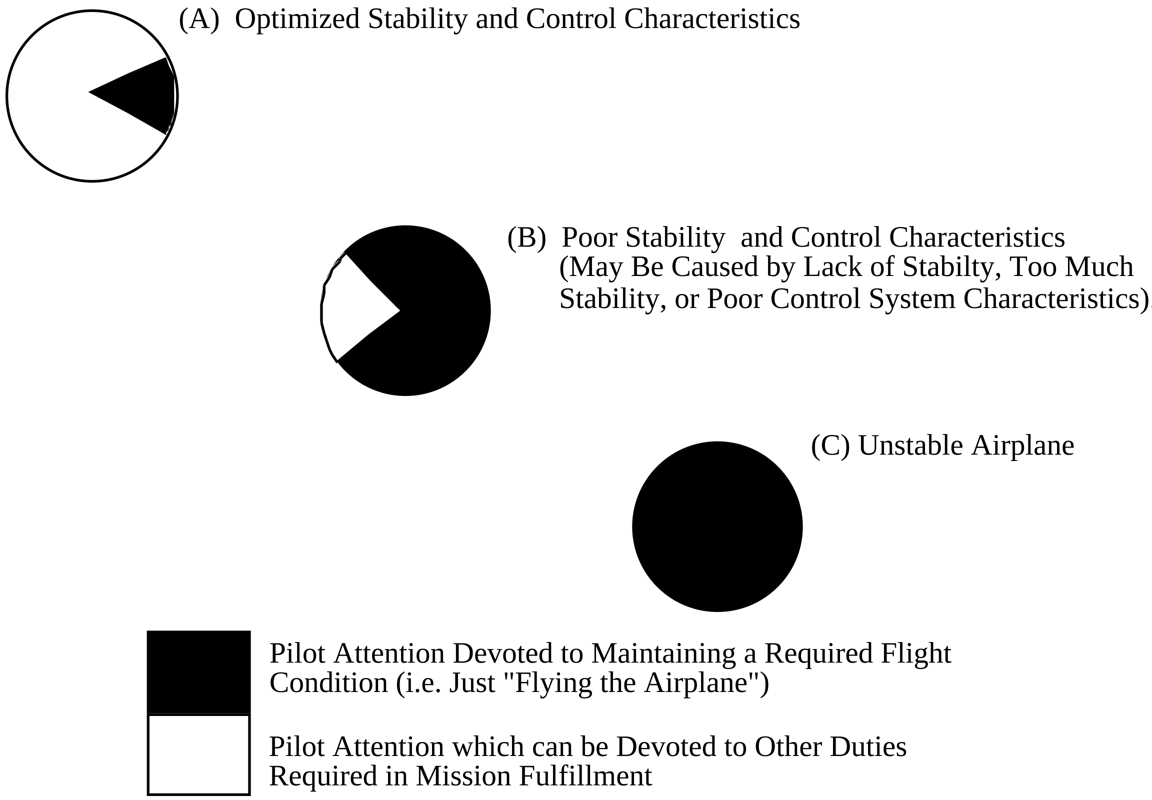


Figure 1.2
Typical Patterns of Pilot Attention and Expenditure of Energy Required as Functions of Airplane Stability and Control Characteristics

1.4 MECHANICS OF DYNAMICS

This section is designed to introduce the language of and provide some background for the dynamic stability discussions presented later.

1.4.1 The Spring-Mass-Damper System

An airplane in flight displays motion similar to the motion of a spring-mass-damper system (Figure 1.3). The static stability of the airplane is analogous to the spring; airflow interaction with the airplane components provides damping and the moment of inertia of the airplane is analogous to the mass of the spring-mass-damper system.

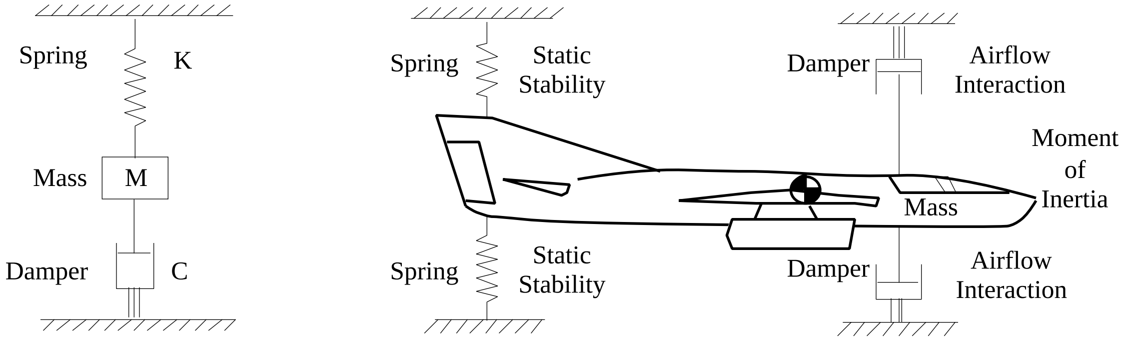


Figure 1.3
An Airplane in Flight is Similar
to a Spring-Mass-Damper System

Of course, the motions of the airplane are much more complicated than the motion of the simple spring-mass-damper system. However, the solution of the equation of motion for the spring-mass-damper system provides a useful analogy to the solution of the equations of motion of the airplane.

The homogeneous form of the second order linear differential equation of motion of the spring-mass-damper system may be written:

$$M\ddot{\psi} + C\dot{\psi} + K\psi = 0 \qquad \textit{eq 1.1}$$

Where:

M = mass of the body

C = damping constant, a measure of the strength of the viscous damper

K = spring constant, a measure of the stiffness of the spring

ψ = displacement of the mass from an equilibrium position.

$\dot{\psi}$ = velocity of the mass.

$\ddot{\psi}$ = acceleration of the mass.

The characteristics equation is of the following form (a trivial solution has been neglected):

$$\lambda^2 + \frac{C}{M}\lambda + \frac{K}{M} = 0 \qquad \text{eq 1.2}$$

This characteristic equation yields two roots which may be written as follows:

$$\lambda_{1,2} = -\frac{C}{2M} \pm \sqrt{\left(\frac{C}{2M}\right)^2 - \frac{K}{M}} \qquad \text{eq 1.3}$$

It is interesting to study the characteristics of these roots as the value of the spring constant, K, is increased from zero. The movement of these roots may be graphically shown on the complex plane. The significance of the positions of the roots is shown in Figure 1.4.

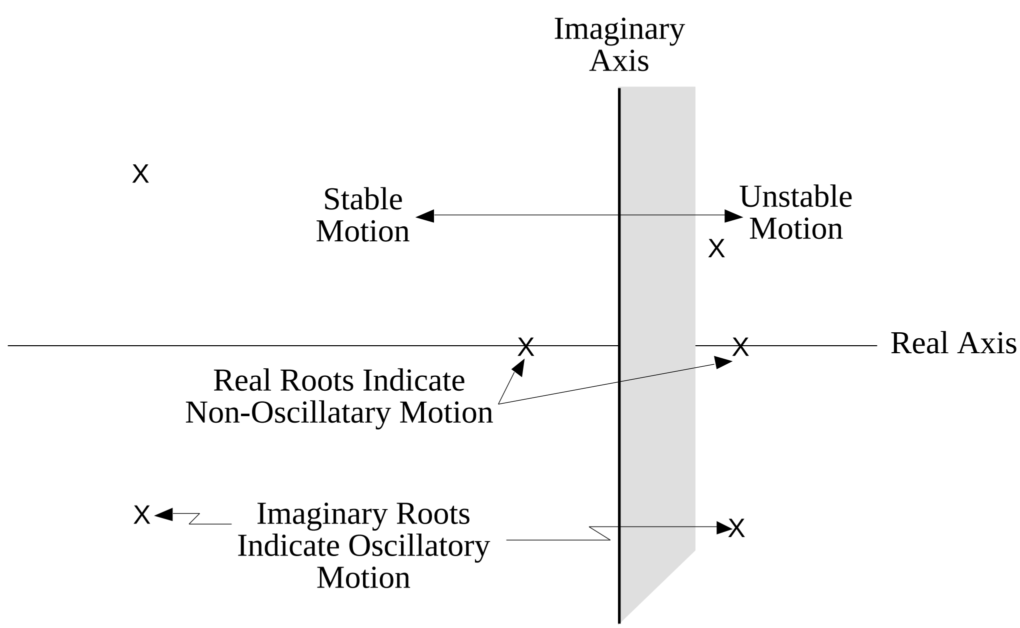


Figure 1.4
The Complex Plane

As the spring constant is increased from zero, the movement of the roots is shown in Figure 1.5. As long as the damping of the system is predominant, i.e., $(C/2M)^2 > K/M$, the roots will lie along the real axis and the motion of the system is described as <u>aperiodic</u> or <u>deadbeat</u> subsidence (the system is overdamped). When $K/M = (C/2M)^2$, the roots meet at point A on the real axis. The value of the damping of the system at this point is called critical damping, C_{CRIT}.

$$C_{CRIT} = 2M\sqrt{K/M} \qquad \textit{eq 1.4}$$

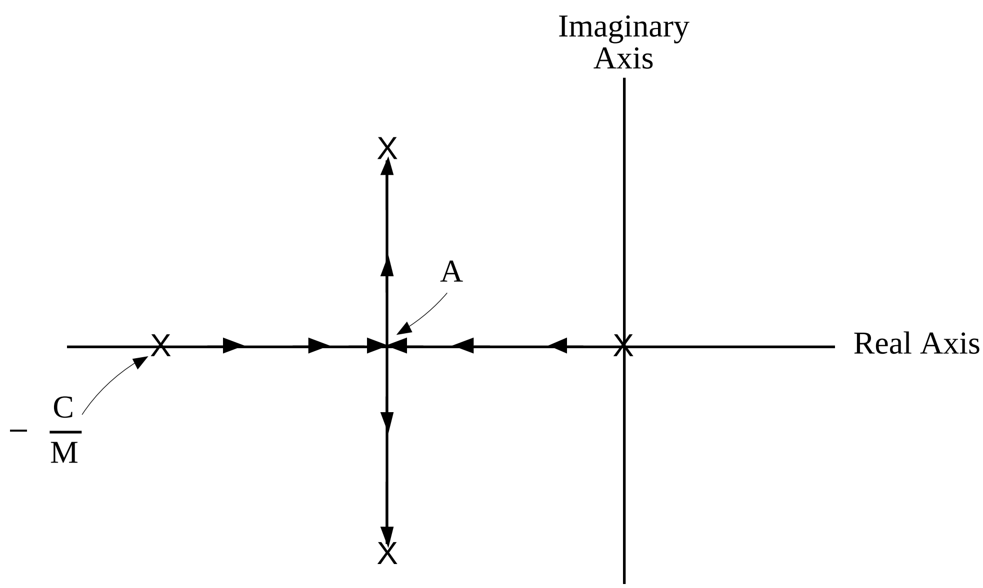


Figure 1.5
Effect of Increasing Spring Constant

When the roots are positioned at point A, the motion of the system is still described as <u>aperiodic</u> or <u>deadbeat</u> subsidence. However,, it is on the verge of becoming <u>oscillatory</u>, i.e, it is <u>critically damped</u>.

If the spring constant is increased further such that $K/M > (C/2M)^2$, the solutions to the equation of motion are composed of real and imaginary parts. The roots split at point A; the real part remains constant and as K increases, the imaginary part becomes larger. The motion of the system is now oscillatory and the frequency increases as K increases. However, for all values of K, the motion is damped after the disturbing force is removed.

The spring-mass-damper system is a second-order system since its describing differential equation contains the dependent variable (ψ) and the first and second derivatives of the variable. The measure of the strength of the system to seek an equilibrium condition is called the system stiffness, and is the square of the system frequency when damping is not present. This frequency is called the undamped natural frequency ω_n, of the system. (It is usually a computed number since most systems have damping and the measured system frequency will be the damped natural frequency, ω_d.) The undamped natural frequency for the spring-mass-damper system may be expressed as follows:

$$\omega_n = \sqrt{\frac{K}{M}} \qquad \textit{eq 1.5}$$

The degree of dynamic stability of a second order system is generally expressed in terms of the system damping ratio, ζ . It is the ratio of the real system damping constant to the damping constant which would make the system critically damped.

$$\zeta = \frac{C}{C_{CRIT}} \qquad \textit{eq 1.6}$$

The characteristic equation for the spring-mass-damper system may be written in terms of undamped natural frequency and damping ratio as follows:

$$\lambda^2 + 2\zeta\omega_n\lambda + \omega_n^2 = 0 \qquad \textit{eq 1.7}$$

The two roots of the equation then may be written:

$$\lambda_{1,2} = -\zeta\omega_n \pm i\ \omega_n\sqrt{1 - \zeta^2} \qquad \textit{eq 1.8}$$

These roots plotted on the complex plane are shown in Figure 1.6. Several important relationships are also presented.

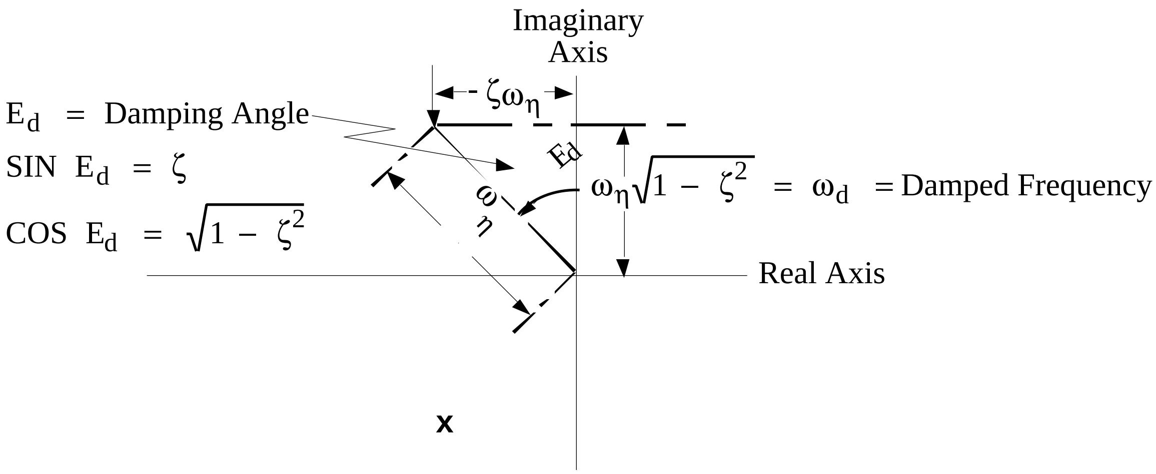


Figure 1.6
Relationship of Position of Roots on Complex Plane to Motion Characteristics

1.4.2 Response of a Second Order System to a Disturbance

The response of a second order system to a disturbing force which is instantaneously applied (step input) is shown in Figure 1.7. In this case, the motion is convergent to a steady state or equilibrium condition. The "quickness" of the response depends mainly on the undamped natural frequency of the system and the oscillatory nature of the response depends on the damping ratio. The amplitude of the steady state value of the response is quite dependent on the square of the undamped natural frequency or the system stiffness. The greater the system stiffness, the smaller is the steady state value of the response, if other factors remain constant.

The response of the second order system shown in Figure 1.7 is commonly called a "second order response," i.e., the response exhibits some oscillatory motion before reaching an equilibrium condition. If the damping ratio of a second order system is increased to a sufficient level, the response of a second order system may appear to be a "first order response," i.e., the response builds up smoothly to a steady state with no oscillatory motion (Figure 1.8). The time required to reach 63.2 percent of a steady state first order response is called the motion time constant, τ.

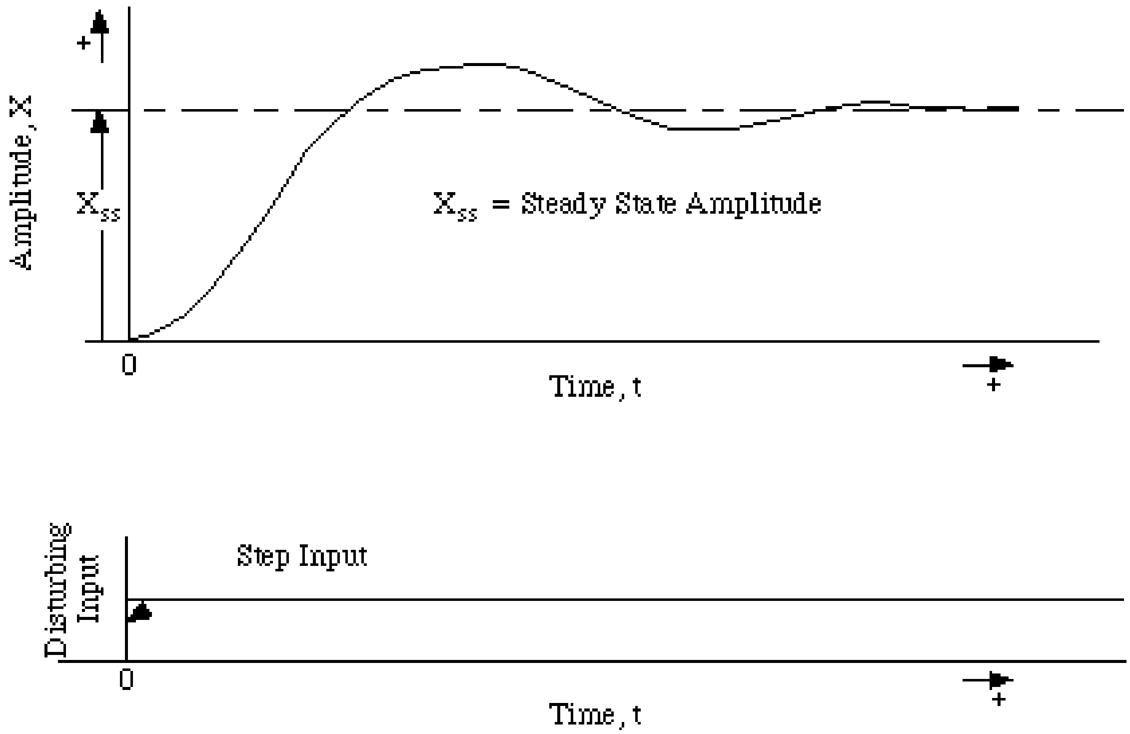


Figure 1.7
Time Response of a Second Order System to a Step Input

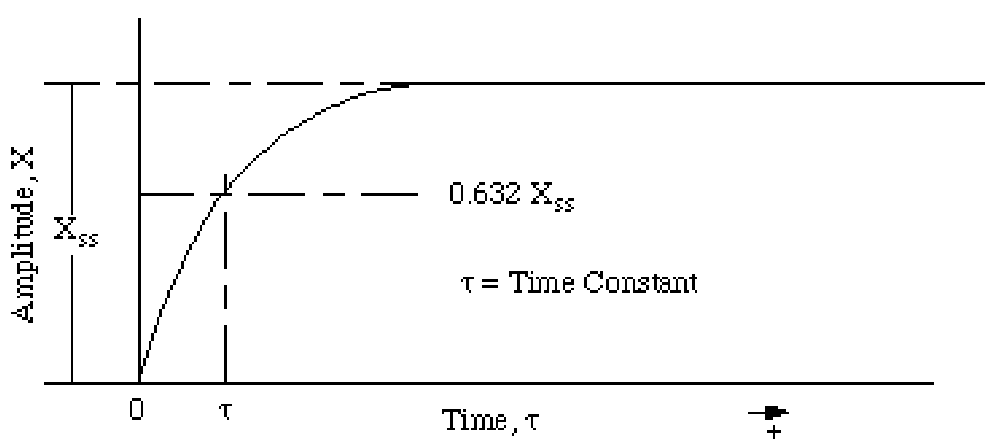


Figure 1.8
Typical First Order Response

1.4.3 Analysis of Second Order Responses

There are various methods for determining the characteristics of second order responses. The graphical methods presented herein are fairly simple and are considered to be of sufficient accuracy for most flight test work.

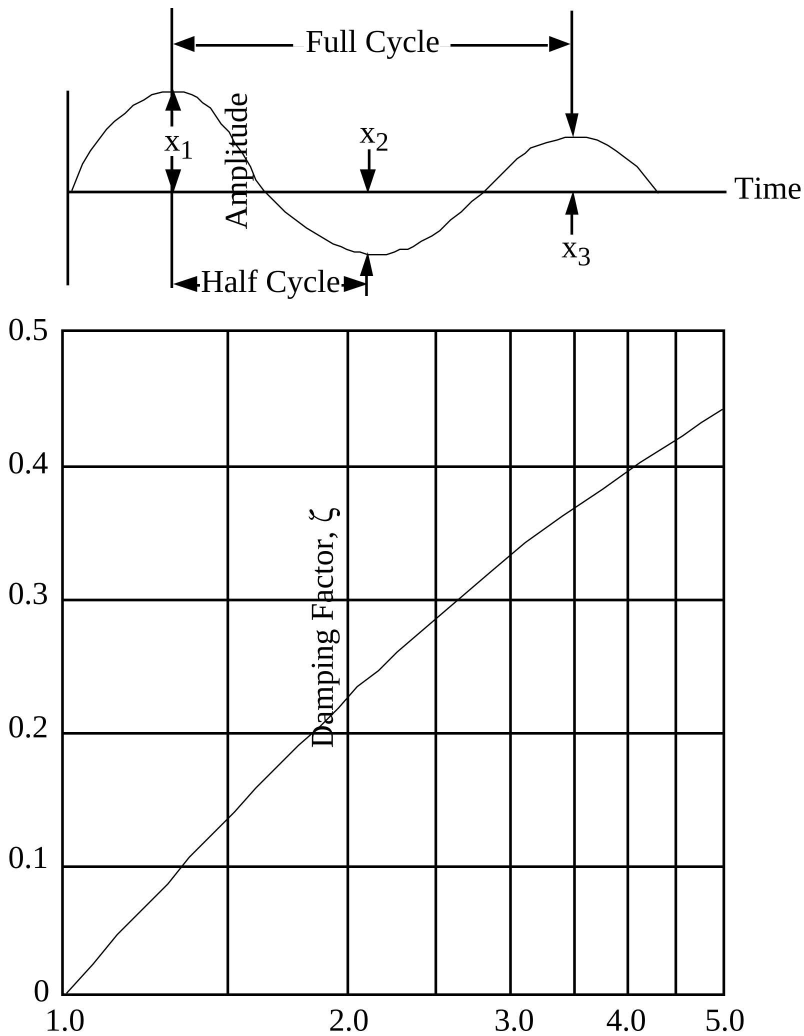
If the system exhibits a damping ratio less than about 0.5, the oscillatory motion will be significant enough to measure a half-cycle amplitude ratio and determine the damping ratio as shown in Figure 1.9. The undamped natural frequency may then be computed as follows:

$$\omega_n = \frac{\pi}{\Delta T_1 \sqrt{1 - \zeta^2}} \qquad \textit{eq 1.9}$$

Where:

ΔT_1 = time between the first two peaks, i.e., the time required for the first half-cycle.

If the system is heavily damped, determination of the motion parameters is more difficult. From a practical flight test standpoint, the pilot will probably not be able to detect visually any oscillatory tendency if the damping ratio is greater than 0.5. Therefore, it may suffice to call the motion "essentially deadbeat" in that case. However, if sufficient instrumentation is installed, the method shown in Figure 1.10 may be used to determine approximate values for damping ratio and undamped natural frequency. One of the most frustrating problems in the analysis of very heavily damped responses is the detection and selection of the proper "peaks" of the response curve. For the analysis shown in Figure 1.10, the first two response peaks after the control input has reached steady state should be used.



Half Cycle Amplitude Ratio, $\left(\dfrac{x_n}{x_{n+1}}\right)$

For Oscillatory Divergence $(\zeta < 0)$,

Merely Change Horizontal Scale to $\left(\dfrac{X_{n+1}}{X_n}\right)$

and Change Vertical Scale to Negative Sign.

Figure 1.9
Determination of Damping Ratio
for Lightly Damped System

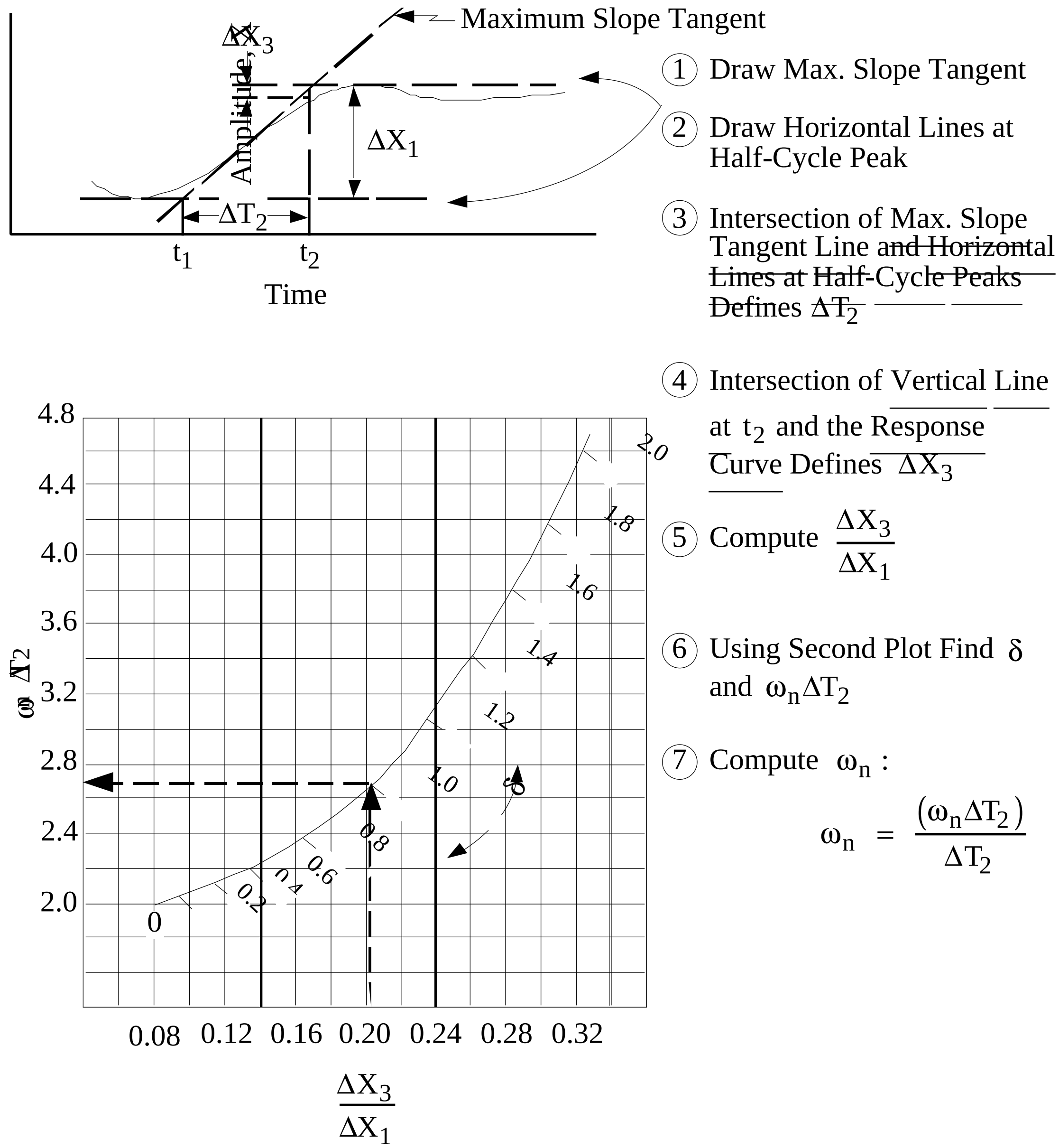


Figure 1.10
Determination of Second Order Response
Characteristics for Heavily Damped Systems

Other parameters used to describe the characteristics of second order responses include the following:

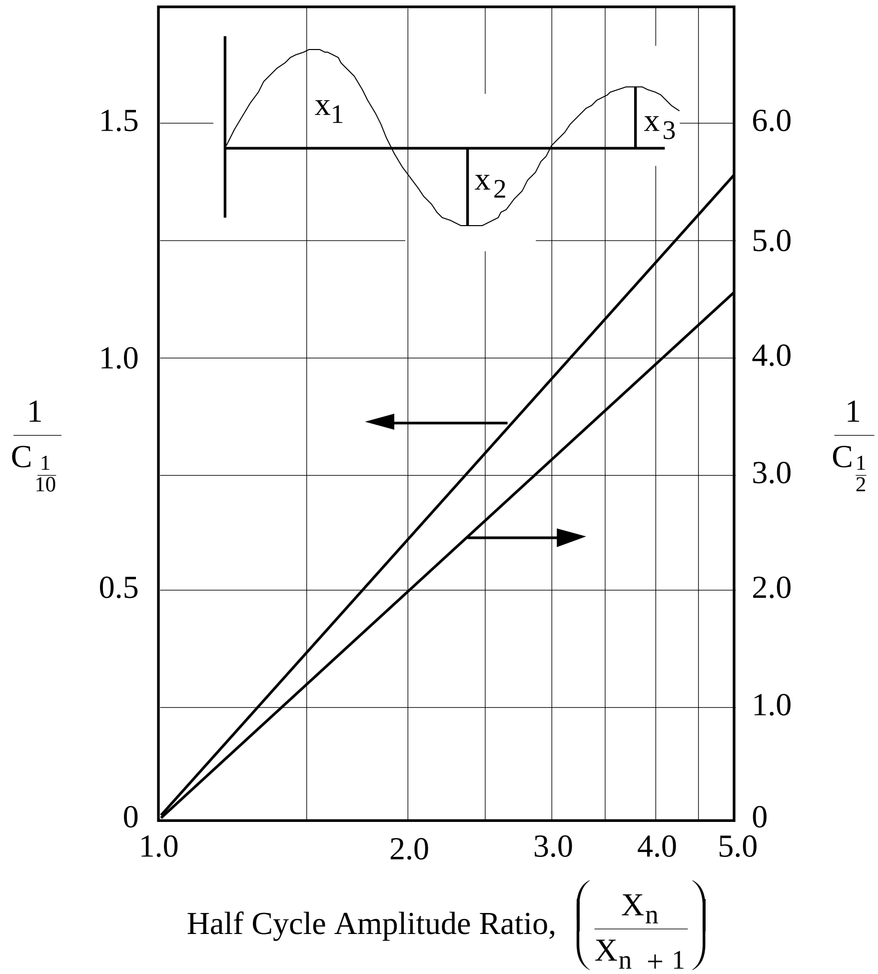
$T_{\frac{1}{2}}$ = time in seconds for the motion to subside to half its amplitude.

T_2 = time in seconds for the motion to double its amplitude.

$C_{\frac{1}{2}}$ = cycles required for the motion to subside to half its amplitude.

C_2 = cycles required for the motion to double its amplitude.

These parameters may be determined by the method shown in Figure 1.11 or Figure 1.12. (In determining certain flying qualities specification requirements, the parameters $\frac{1}{C_{\frac{1}{2}}}$ and $C_{\frac{1}{10}}$ are often utilized.)



For Oscillatory Divergence $(\zeta < 0)$,

Merely Change Horizontal Scale to $\left(\frac{X_{n+1}}{X_n}\right)$

and Change Vertical Scales to $\frac{1}{C_2}$ and $\frac{1}{C_{10}}$

Figure 1.11
Determination of $\frac{1}{C_{\frac{1}{2}}}$ and $\frac{1}{C_{\frac{1}{10}}}$ from Half-Cycle Amplitude Ratio

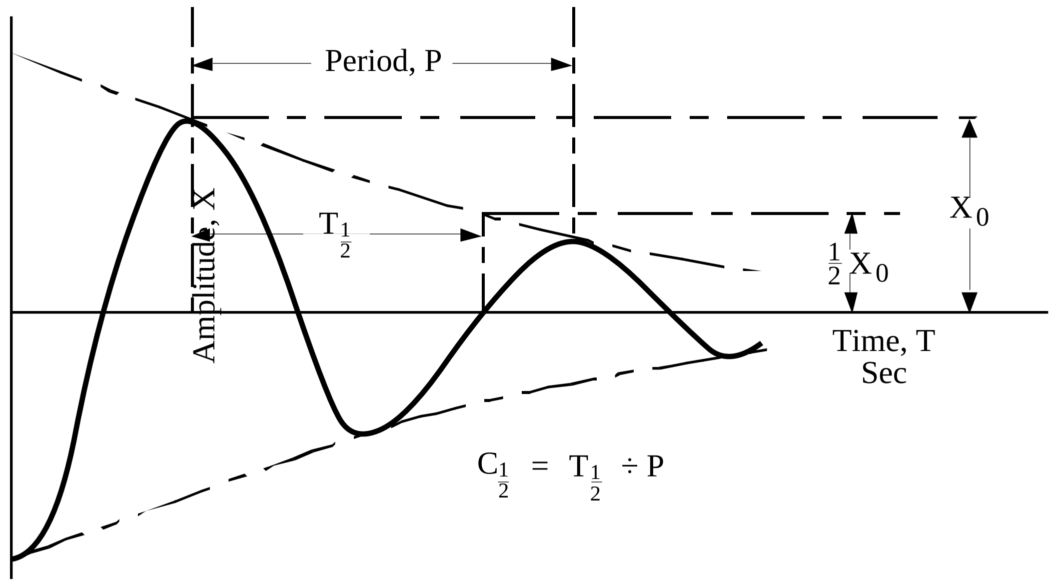


Figure 1.12
Graphical Method for Determining $T_{\frac{1}{2}}$ and $C_{\frac{1}{2}}$

1.4.4 Airplane Motion

The airplane in flight is a complicated dynamic system with six "degrees of freedom," or possible components of motion. However, for the simplified study of airplane dynamics, the motion of the airplane is considered to be restricted to a "plane of symmetry" and a "plane of asymmetry" with no "interaction" or "cross-coupling" between the planes of motion. Motion in the plane of symmetry is, of course, longitudinal motion; motion in the plane of asymmetry is lateral-directional motion. By separating the study of airplane dynamics in this manner, the analysis is greatly simplified and yields quite accurate results for most flight conditions. The effects of "cross-coupling" can be studied separately for special flight conditions.

The characteristic equations of motion for the longitudinal and lateral-directional cases are fourth order linear differential equations. At present, let if suffice to say that the difficulty in solving these equations by "normal" procedures is considerable. However, by use of an operational calculus technique called "Laplace Transformations," the solution can be determined quite easily. The equations of motion will not be derived in this text; nor will a great deal of the mathematical manipulations required to solve the equations be presented. These derivations and mathematical manipulations can be found in appropriate literature and will be presented in the academic syllabus of the U.S. Naval Test Pilot School.

The classical solution of the longitudinal characteristic equation for the airplane yields four roots having real and imaginary parts. Normally, these roots form “complex pairs” which describe two second order modes of motion - the airplane short period mode and the long period or “phugoid” mode.

The lateral-directional characteristic equation also yields four roots for the classical case. Two of the roots have real and imaginary parts and form a complex pair which describe a second order mode of motion commonly called the “Dutch roll mode.” Two of the roots have only real parts. One of the real roots describes an essentially first order, heavily damped motion - the “roll mode.” The second of the real roots describes another first order motion which may be convergent, divergent, or neutral. This mode of motion is called the “spiral mode.”

1.5 INFLUENCE OF FLIGHT CONTROL SYSTEM ON FLYING QUALITIES

A rigorous discussion of the numerous flight control system design concepts is beyond the scope of this text. However, some brief discussion of control system influence on the pilot's opinion of the airplane is appropriate.

All airplane flight control systems may be placed into one the following three categories:

1. Manual Control System: The pilot deflects the appropriate control surface through direct mechanical connections between the cockpit control and the aerodynamic control surface. The pilot force required is a function only of control surface hinge moments developed and pure mechanical design of the control system. No hydraulic, pneumatic, or electrical power boosting is employed. For control systems of this type, extensive use is made of aerodynamic and mass balancing and geared, spring, and servo tabs. Other control system “gadgetry” such as springs and bob weights may also be employed to improve basic airplane characteristics.

2. Power-Assisted Control Systems: The pilot deflects the appropriate control surface by direct mechanical connections between the cockpit control and the aerodynamic control surface. However, a suitable power unit (usually hydraulic) is appropriately placed in the control system to assist the pilot in moving the control surface. The pilot force required is thus a function of the ratio of power assist provided or "boost" as well as control surface hinge moments developed. Again, extensive use may be made of aerodynamic and mass balancing geared, spring, servo tabs, and other control system "gadgetry."

3. Fully Power-Operated Control Systems: Through cockpit control deflections, the pilot positions a valve of a power unit; the power unit in turn positions the control surface proportional to the pilot's cockpit control input. The pilot force required is purely a function of cockpit control deflection and does not depend on control surface hinge moments. It is apparent then that an artificial feel system must be provided to give the pilot the normal control force variations. Extensive use is made of springs, bob weights, dynamic pressure sensors, dashpots, and other electrical, mechanical, or hydraulic devices in order to provide satisfactory stability and control characteristics.

The manual and power-assisted control systems are reversible control systems; i.e., the pilot receives some control force feel by virtue of the hinge moments developed when the aerodynamic control surface is deflected. The fully power-operated control system is an irreversible control system; i.e., the pilot receives no control force feel from the development of control surface hinge moments.

No matter what type of flight control system is utilized, the requirements placed on the flight control system remain the same. The control system must give the pilot the ability to make simple and unhindered control deflections in any direction. Control deflections and forces required for maneuvering the airplane must be commensurate with the mission of the airplane, the structural limits of the airplane, and pilot strength limitations. The controls must exhibit good centering when released and must exhibit no tendency toward lightly damped or undamped free oscillatory motions. There should be no noticeable lag between the deflections of the cockpit controls and the movement of the corresponding control surface.

Since the flight control system is the implement by which the pilot is “mated” to the airplane, the importance of good control system characteristics cannot be overemphasized. The control system must be suitably matched to the stability, control, and inertial characteristics of the basic airplane, and to the requirements of the human pilot. Proper flight control system and basic airplane matching provide the pilot with the opportunity to fully utilized the maneuvering capabilities of the airplane for maximum mission performance.

1.6 THE U.S. NAVAL TEST PILOT SCHOOL DEMONSTRATION AND PROGRESS CHECK FLIGHTS

The U.S. Naval Test Pilot School utilizes actual flight instruction in the techniques of stability and control testing. At the beginning of each new phase of study, students are exposed to actual flight test techniques and methods through Demonstration Flights. At the completion of each phase, students demonstrate their proficiency in that phase of flight testing during Progress Check Flights. The purpose of the Demonstration Flight is to provide instruction in stability and control test techniques in a realistic environment; whereas the purpose of the Progress Check Flight is to evaluate the student's progress and render additional instruction in troublesome areas.

1.6.1 The Demonstration Flight

The Demonstration Flight will be preceded by thorough briefings which will present background theory, test techniques, analysis of test results in terms of mission accomplishment and specification requirements, and data presentation methods. It is the student's responsibility to prepare for the Demonstration Flight by thorough review of briefing notes, appropriate technical literature, and relevant specifications. Thorough preparation is essential for derivation of maximum benefits from this flight.

The performance and maneuvering longitudinal stability demonstration flights are flown in any of the school's jet fleet. The Lateral-Directional and Nonmaneuvering Longitudinal Stability demonstration flights are usually flown in an airplane with side-by-side seating. One or more students and one instructor comprise a normal flight crew with the students sharing equally in airborne instructional time. Since the students may not be qualified in the demonstration airplane, the instructor pilot usually handles all normal pre-flights, ground operations, takeoffs, and landings. The students are not required to know the demonstration airplane from an operational standpoint. During the actual instructional phase of the flight, the instructor will demonstrate both qualitative and quantitative test techniques, use of special instrumentation, and data recording procedures. After the student has observed and understands each technique, he is given an opportunity to practice until attaining a reasonable level of proficiency. Throughout the Demonstration Flight, the instructor will discuss the significance of each test, implications of certain characteristics exhibited, and slight variations in the test techniques which would be appropriate in other type airplanes. The student is encouraged to ask questions during the progress of the flight. Many points are made perfectly clear in only a few seconds in flight; to accomplish the same on the ground would probably require several minutes. A thorough postflight discussion between instructor and students completes the Demonstration Flight. During the debrief, the data which were obtained on the flight is plotted, discussed, and analyzed.

The student is required to plan the flight completely, giving due consideration to a real or simulated mission of the airplane and appropriate specification requirements. The student conducts the flight briefing, which must include a definition of the mission and a brief description of the flight control system, as well as discussions of test techniques and specification requirements.

As the student demonstrates his knowledge of qualitative and quantitative test techniques in flight, he is expected to comment on the importance and meaning of the tests with respect to the real or simulated mission. The instructor will comment on validity of the results obtained, errors or omissions in test procedures, and may demonstrate variations in test techniques which have not been previously introduced. The student will be expected to investigate qualitatively the pilot effort required in the performance of a typical mission task. This task may be a tracking maneuver or ground controlled approach; i.e., some task which requires precise control of the airplane. The student will be asked to rationalize the reasons for the simplicity or difficulty of the maneuver during the debrief following the flight.

The debrief consists of the student discussing and analyzing the results of the in-flight tests. The analysis must be oriented toward the influence of the characteristics exhibited on the mission effectiveness of the airplane.

CHAPTER TWO

STALLS

CHAPTER TWO

FIGURES

CHAPTER TWO

TABLES

CHAPTER TWO

EQUATIONS

		PAGE
$V_S = \sqrt{\dfrac{2nW}{\rho C_{L_{max}} S}}$	*eq 2.1*	2.14

CHAPTER TWO

STALLS

2.1 INTRODUCTION

All airplanes are subjected to stall investigations for the following reasons:

1. Safety and operational considerations.

2. Actual flight tests are the only means of precisely determining stall characteristics.

3. Expansion of the operational flight envelope.

4. Determination of trim airspeeds for future tests.

The investigation of stall characteristics is a phase of flying qualities which is difficult to associate with particular pilot tasks. There are no total missions in which stalls are required for mission accomplishment, although pilot training and familiarization in stall characteristics are considered an essential phase of the training mission. However, all airplanes will be stalled at one time or another in operational use if sufficient longitudinal control is available and if no stall prevention device is installed. Therefore, stall tests are an integral part of any flying qualities program.

The emphasis placed on the stall investigation depends on the total mission of the airplane. If mission accomplishment involves a great deal of maneuvering, the pilot is very likely to inadvertently stall; therefore, a thorough stall investigation must be carried out. If mission accomplishment involves a minimum of maneuvering, the pilot is not likely to inadvertently stall; therefore, the stall investigation may be less stringent.

Stall investigations encompass both normal and accelerated stalls. The normal stall is defined as a stall which occurs while the airplane is in an unaccelerated flight condition. The accelerated stall is defined as a stall which occurs while the airplane is in an accelerated flight condition, such as a pull-up or a level turn. Accelerated stalls usually exhibit more violent characteristics than normal stalls; therefore, normal stalls should be investigated thoroughly before commencement of accelerated stall tests. The <u>total mission</u> of the airplane dictates where the primary emphasis is placed during the stall investigation. For the airplane which will be maneuvered extensively, primary emphasis must be placed on accelerated stalls which could result from mission tasks. If mission accomplishment involves a minimum of maneuvering, primary emphasis should be placed on normal stall characteristics. The large passenger, transport, or heavy bomber type airplane will most likely be inadvertently stalled in unaccelerated flight during transitions associated with instrument departures or approaches.

Normal and accelerated stalls may be further classified as “positive g” or “negative g” stalls. This discussion of stall characteristics will be concerned only with “positive g” normal and accelerated stalls because:

1. Normal “negative g” stalls are difficult to obtain in most operational airplanes due to insufficient longitudinal control effectiveness.

2. Precise pilot technique is required to perform “negative g” accelerated stalls (stalls entered at less than -1.0g).

3. Pilot discomfort discourages entry into “negative g” normal or accelerated stalls.

“Negative g” normal and accelerated stalls maybe investigated in a build-up program for spin testing, which will be discussed in a subsequent section.

Normal and accelerated stall characteristics indirectly affect mission performance of the pilot - airplane combination. Satisfactory stall characteristics greatly increase pilot confidence in his airplane. When assurance can be provided that violent departures into uncontrolled flight will not result from inadvertent stalls, the pilot will utilize fully the maneuvering capabilities of the airplane for maximum mission effectiveness.

2.2 THEORY

The classical stall may be defined as a condition in which the airplane wing is subjected to an angle of attack greater than the angle for maximum lift coefficient. Stall speed can be defined as the minimum steady airspeed attainable in unaccelerated flight or the minimum usable airspeed. However, characteristics exhibited by many airplanes in the region of the stall preclude attainment of the classic aerodynamic stall. These characteristics vary widely among different airplanes and are greatly affected by a multitude of factors. The major factors affecting stall characteristics are discussed herein. A resumä of stall warning and stall prevention devices is also presented.

2.2.1 Wing Design

2.2.1.1 WING SECTION CHARACTERISTICS

Wing section design determines the value of the maximum lift coefficient, the angle of attack at which it is achieved, and the rate of change of lift coefficient with angle of attack in the region of the stall. The most influential wing section parameters are the wing thickness and position of maximum thickness, the amount of camber, and the leading edge radius.

The influence of airfoil thickness and camber on maximum lift coefficient is quite pronounced. A thin symmetrical airfoil (thickness ratio less than .08) has such a small leading edge radius that large adverse pressure gradients induce leading edge flow separations at low angles of attack. The thick (thickness ratio greater than .12) or highly cambered airfoil creates large adverse pressure gradients near the upper surface trailing edge which causes separation near the trailing edge at low angles of attack. An airfoil of moderate thickness (thickness ratio from .08 to .12) and camber may exhibit a tendency for separation to occur simultaneously at both leading edge and trailing edge (Figure 2.1). (Note: Positive cambering of a thin symmetrical airfoil generally reduces the tendency for early separation and increases maximum lift coefficient. However, too much cambering of thick sections can produce the adverse characteristics discussed above.)

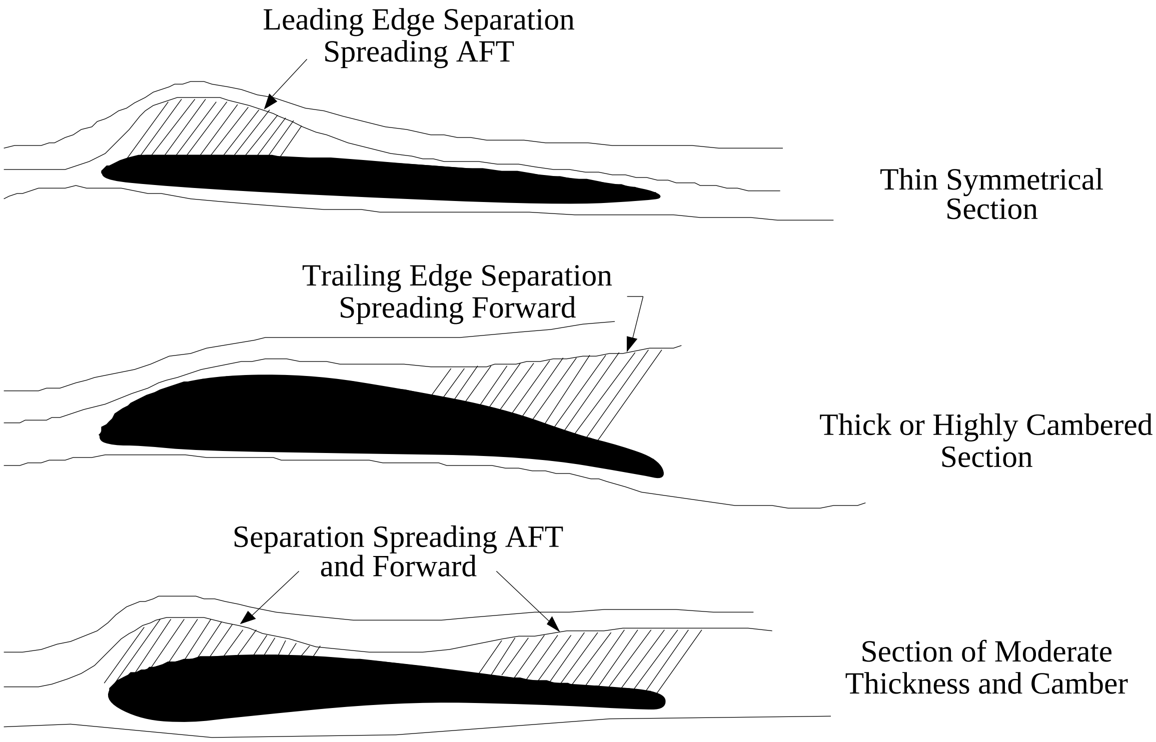


Figure 2.1
Types of Section Stall

The type of section stall has a great deal of influence on stall characteristics. If separation occurs first near the trailing edge of the airfoil, the spread of separation forward is fairly slow and gradual until the maximum lift coefficient is attained. This type of separation progression indicates that the lift curve would exhibit a smooth, gradual change in slope near the stall, although the stall would be rather well-defined. An airplane having this type of lift coefficient - angle of attack relationship would probably exhibit satisfactory stall warning and a well-defined aerodynamic stall (Figure 2.2).

The second lift curve of Figure 2.2 also exhibits a well-defined peak, however, the peak is followed by a very rapid, even discontinuous, decrease in lift coefficient for a small increase in angle of attack. This type of lift curve can result from leading edge flow separation spreading rapidly aft on the airfoil or simultaneous leading edge-trailing edge separation. The airplane with this type of lift curve would exhibit little or no aerodynamic stall warning and a sudden, abrupt stall. This stall may be quite violent because the sharpness and discontinuity of the lift curve indicate that one wing can easily stall prior to the other generating rolling motion at the stall. This "asymmetric" stall can be caused by small difference in wing sections along the span or small differences in local flow direction due to vertical gusts or yawing motion. In any case, the downgoing wing experiences an increase in angle of attack, while the upgoing wing experiences a decrease. This situation may result in autorotation, a motion in which the rolling is self-sustaining and which may result in the airplane entering a spin. The abrupt, "asymmetric" stall tendency may be overcome by increasing the radius of the leading edge of the wing and/or by cambering the wing judiciously. If this approach is not practical, some improvement in stall characteristics may be realized by installing devices on the wing leading edge to introduce turbulence into the boundary layer. However, this correction is usually a "trial and error" process.

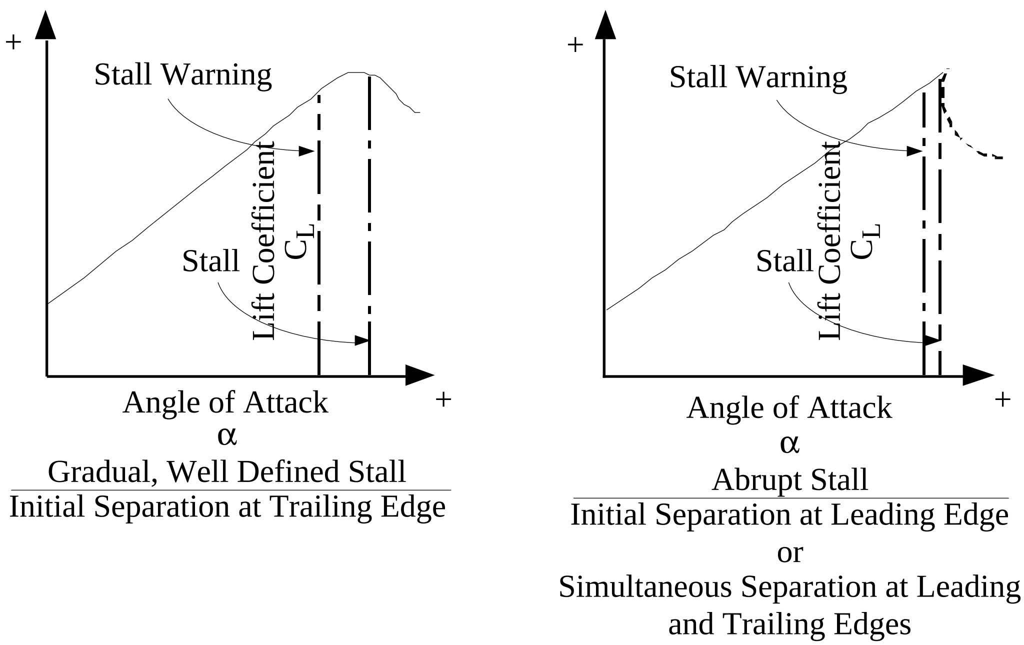


Figure 2.2
Influence of Shape of Lift Curve on Stall Characteristics

2.2.1.2 WING PLANFORM CHARACTERISTICS

Wing planform design influences the slope of the lift curve - angle of attack relationship, downwash pattern, and the portion of the wing span which stalls first. The most influential planform parameters are aspect ratio, sweep, and taper.

The slope of the lift curve at airspeeds near stall is determined primarily by aspect ratio and sweep angle. An increase in aspect ratio[1] increases the slope of the lift curve, while an increase in sweepback decreases the slope (Figure 2.3).

The slope of the lift curve influences the angle of attack and pitch attitude at which the aerodynamic stall is encountered. If the slope of the lift curve is shallow, the angle of attack for the stall may be attained only at a very high airplane nose attitude and with a very large rate of descent. Furthermore, adverse stability and control characteristics may be encountered before the attainment of the maximum lift coefficient. Therefore, airplanes with low aspect ratio and highly swept wings generally do not exhibit a true aerodynamic stall and a "minimum flying speed" would be determined based on other criteria.

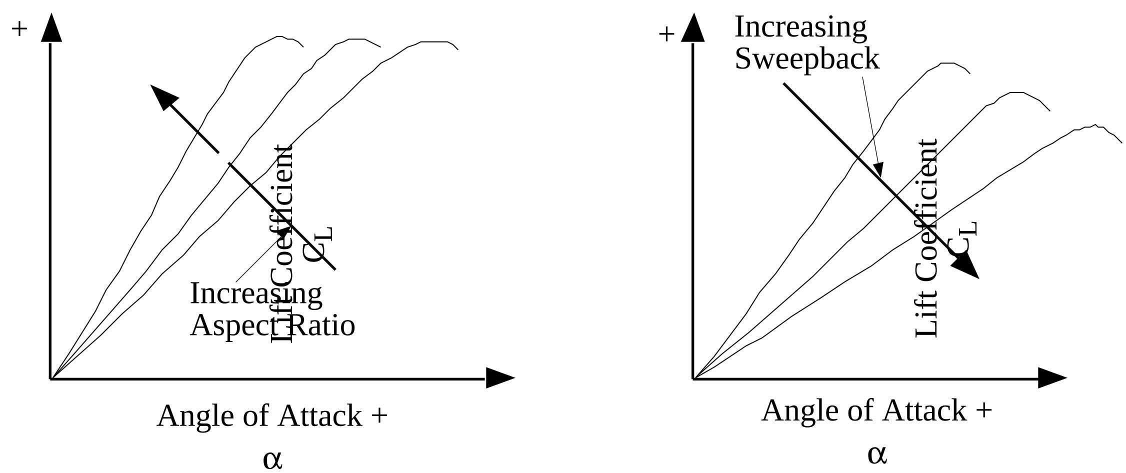


Figure 2.3
Typical Influence of Aspect Ratio and Sweepback on Lift Curve Slope

Downwash is the unavoidable result of lift production by a real wing. It reduces the angle of attack at which individual wing sections operate (Figure 2.4).

---

[1] Aspect ratios of 3 to 6 are considered "medium," above 6 are considered "high", less than 3 are considered "low".

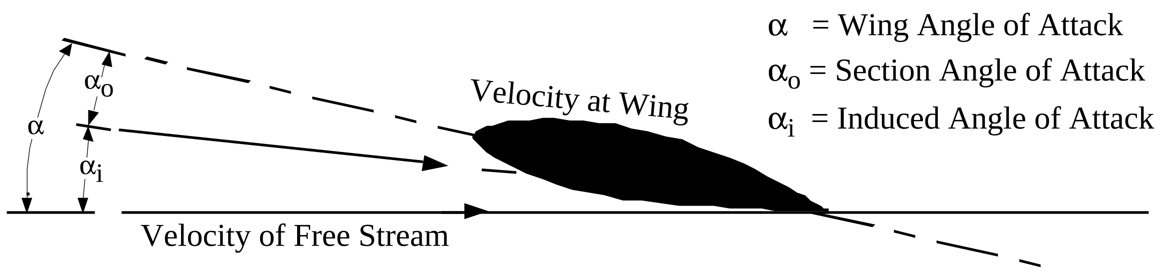


Figure 2.4
Downwash Influence on Section Angle of Attack

The spanwise distribution of downwash dictates the section angle of attack and hence the section lift-coefficient distribution along the span. This distribution is extremely important because of its influence on the part of the span to first reach a stalled condition. The spanwise downwash distribution depends on wing taper and sweep, if the wing has zero twist and the same section from root to tip. As the degree of taper increases, the area of first stall on the span moves from root to tip (Figure 2.5). An increase in sweepback has a similar effect as the increase in taper. The tendency of the wing to stall first at the tips seriously derogates stall characteristics. While the root stall is generally preceded by buffeting of the fuselage and tail caused by turbulent air shed from the root section, the tip stall generally occurs with little or no stall warning. Since the lateral control surfaces are usually positioned near the wing-tips, loss of roll control is often experienced when the stall occurs first at the tips.

The swept wing has an inherent tendency toward tip stall because sweep back changes the spanwise downwash distribution such that the wing area near the tip operates at larger section angles of attack than other wing areas. This generates a pressure gradient along the span of the wing with pressure decreasing from root to tip. As a consequence, considerable spanwise flow of the boundary layer occurs. This spanwise flow from root to tip may be considered a form of "natural" boundary-layer control for the inboard area of the wing and increases the already inherent tendency toward tip stall. (Note: It should be remembered that spanwise flow occurs on any wing planform. However, the swept wing is particularly prone to spanwise flow).

Tip stalling of the swept wing results in an additional factor which tends to derogate stall characteristics. Since sweepback places the tips aft of inboard sections, tip stalling precipitates a forward shift of the wing center of pressure. This cases the wing to become more longitudinally destabilizing; if the destabilizing influence is greater than the stabilizing influence of the horizontal tail, the airplane tends to pitch nose-up at the stall. This characteristic makes the airplane prone to inadvertent stalling and "deep stall" penetrations.

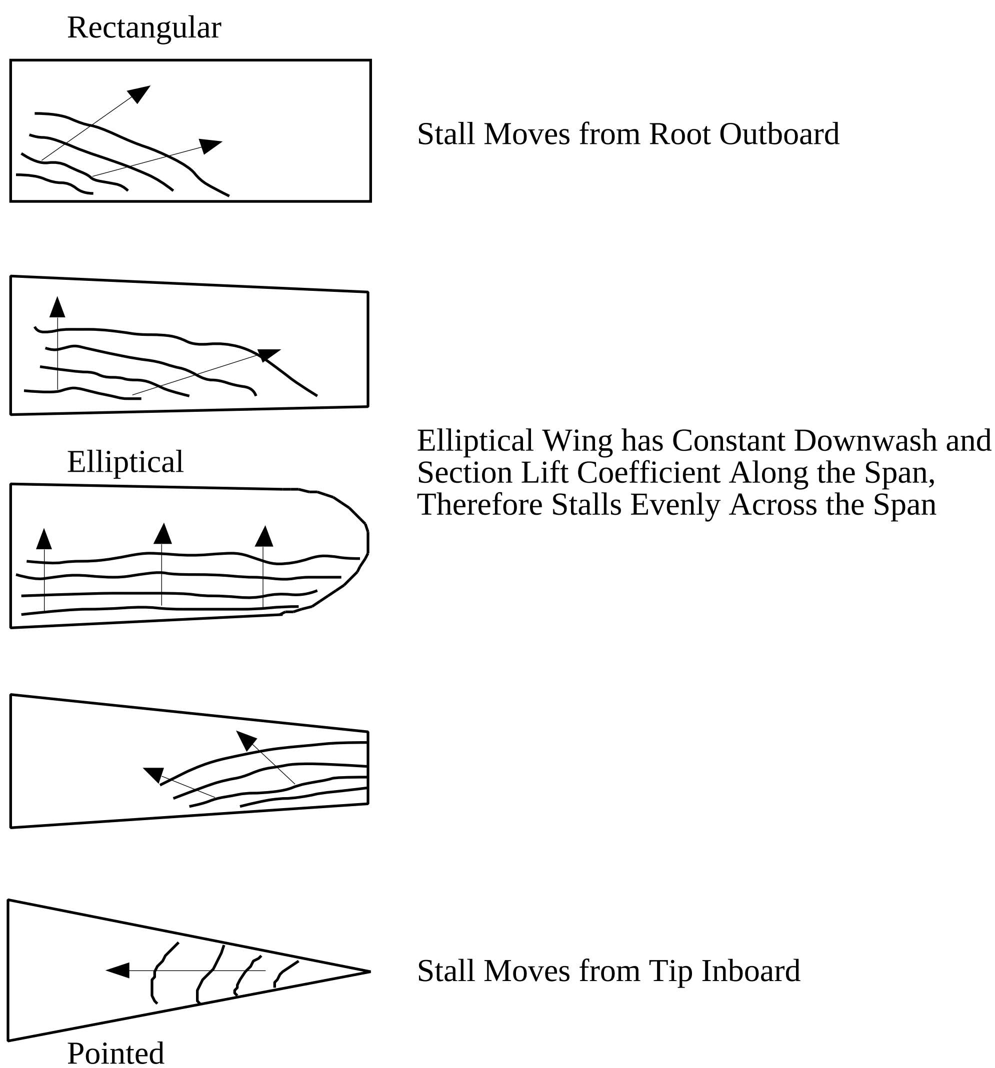


Figure 2.5
Typical Influence of Wing Taper on Stall

There are several means by which tip stalling tendencies may be decreased or eliminated. The most common are listed below. They may be used singly or in combination.

1. Twist: The wing is gradually twisted from root to tip so that outboard sections are always at a lower angle of attack than inboard sections (sometime called washout).

2. Incorporation in the wing tip area of an airfoil section of higher maximum lift coefficient than inboard sections.

3. Wing tip slots or slats: Spanwise passages near the leading edge to delay separation at high angles of attack.

4. Wing tip vortex generators: Small spanwise airfoils which introduce a higher energy level in the boundary layer.

5. Inboard stall strips: Spanwise leading edge protrusions which cause flow separation at the wing root at high angles of attack.

6. Fences: Thin chordwise strips which inhibit spanwise flow.

7. Leading edge discontinuities: A device which creates a vortex just above the wing surface to inhibit spanwise flow.

2.2.1.3 EFFECTS OF HIGH LIFT DEVICES

High lift devices are used to increase the maximum lift coefficient of the wing, allowing stall-free flight at slower airspeeds. Their main influence on stall characteristics is indirect. With high lift devices operating, the airplane stalls at slower speeds; therefore, the <u>effectiveness</u> of the aerodynamic control surfaces for controlling airplane attitude in the stall region is weakened. In addition to this indirect effect common to all high lift devices, some direct effects of particular devices are discussed below.

Flap deflection changes the spanwise distribution of downwash and hence the section angles of attack. This change in section angles of attack may cause significantly different stall characteristics when flaps are deflected.

Boundary layer control (BLC) tends to change the shape of the lift curve near stalling angles of attack (Figure 2.6). The sharper peaks of the lift curve, when boundary layer control is used, make the stall more abrupt and also create the tendency for an abrupt roll at the stall. In addition, a very large reduction in angle of attack may be necessary to effect stall recovery.

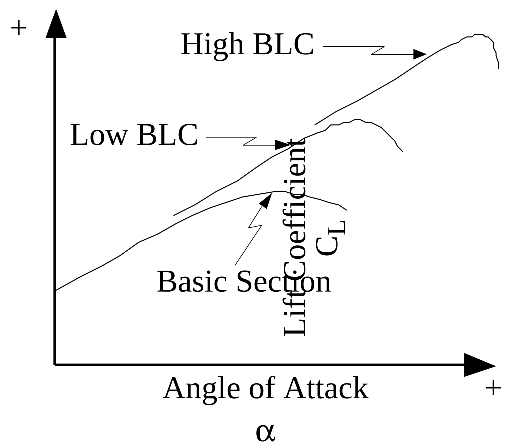


Figure 2.6
Effect of Boundary Layer Control on Lift Curve

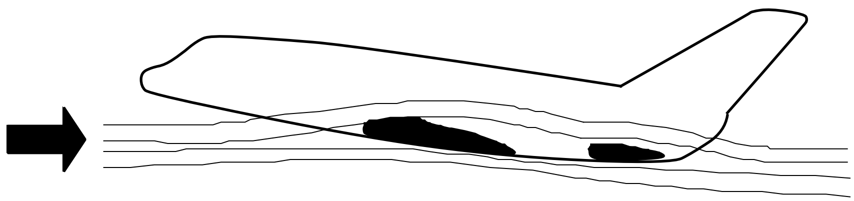
Slots or slats may be used to improve airflow conditions at high angles of attack. One means of utilizing these high lift devices is through use of the “automatic slot.” The automatic slot is a slot in the leading edge of the wing created by the movement of a slat which is retained in the leading-edge contour of the wing at low angles of attack, but extends to create the slot as the stalling angle of attack is approached. The slats operate without action by the pilot and, unless design precautions are taken, have an inherent tendency to extend and retract asymmetrically. Leading edge slats have exhibited an annoying propensity toward asymmetric extension during approaches to accelerated stalls. In this flight regime, their asymmetric extension may generate violent, uncontrollable rolling motion. Asymmetric extension can be eliminated by incorporation of slat interconnects or a hydraulic device to hold the slats on the leading edge of the wing until the landing gear or flaps are extended.

2.2.2 Horizontal Tail Design and Location

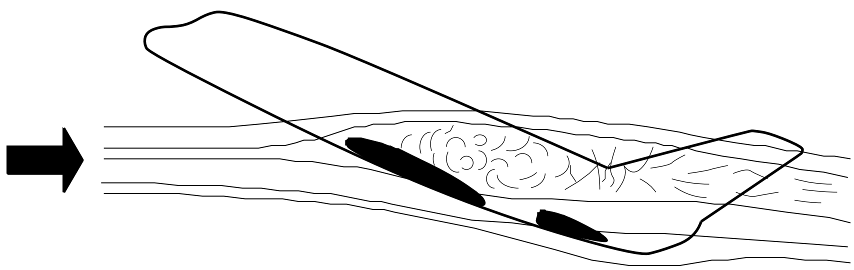
Horizontal tail design and location have a major influence on stall characteristics. Since the contributions of the fuselage and wing to longitudinal stability are generally destabilizing in the stall region, the horizontal tail must provide the necessary stabilizing pitching moments if the airplane is to remain longitudinally controllable. The vertical location of the horizontal tail with respect to the wing is of extreme importance for this dictates the airflow characteristics at the horizontal tail at high angles of attack. A rigorous discussion of all possible vertical tail locations and associated influences on stall characteristics is beyond the scope of this text. However, two examples are presented to demonstrate the problems which exist.

First, consider an airplane design which incorporates a horizontal stabilizer mounted low on the empennage. At low angles of attack, this tailplane is immersed in airflow which has been altered by the wing (Figure 2.7). However, at low angles of attack, there is little loss of stream velocity behind the wing, although the stream is deflected downward by the downwash angle. The horizontal tail, therefore, maintains its effectiveness at low angles of attack since the flow field is not too greatly disturbed. As the angle of attack is increased, airflow begins to breakdown on the wing and loading distribution and associated changes in downwash occur. The wake behind the wing becomes more and more nonstreamlined and turbulent. Very low values of dynamic pressure may exist over an extensive region aft of the wing. If the angle of attack is increased sufficiently, a complete breakdown of flow spreads over the entire wing and the stall occurs. However, if the horizontal stabilizer is mounted low on the empennage, the stabilizer emerges from the wing wake at high angles of attack (Figure 2.7). This causes the horizontal tail to maintain a strong longitudinally stabilizing influence at the stall, generating large nose-down pitching moments. In addition, the longitudinal control surface maintains a high degree of effectiveness throughout the stall, allowing the pilot close control over pitch attitude.

The placement of the horizontal stabilizer high up on the vertical fin (T-tail) has become increasingly popular in recent years, particularly for passenger and transport airplanes. With the appearance of aft fuselage mounted engines (which allowed a structurally simple and aerodynamically clean wing), the horizontal stabilizer was placed higher to avoid interference flow and structural fatigue from engine exhaust. The T-tail also realizes other advantages such as an increase in effectiveness at low angles of attack since, in that flight regime, it does not operate in the wake of the wing. In addition, it has an "endplate effect" on the vertical tail, and thereby increases the effectiveness of that surface. Unfortunately, the T-tail design causes severe problems at high angles of attack, particularly at stalling angles of attack.



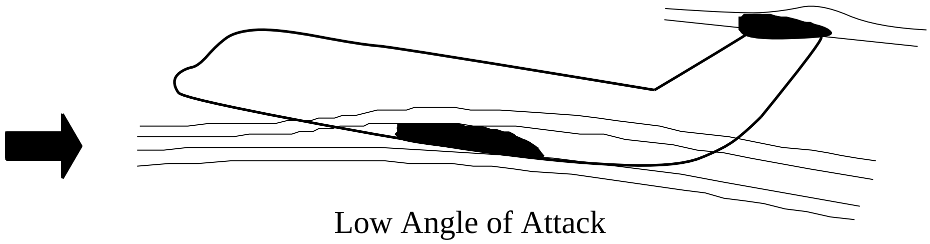
Low Angle of Attack



High Angle of Attack

Figure 2.7
Typical Flow Patterns About
the Low-Mounted Horizontal Tail

Insight into the T-tail stall problem can be gained by a study of Figure 2.8. At low angles of attack, the T-tail receives little or no influence from the downwash caused by lift production of the wing. However, as the airplane is rotated to higher and higher angles of attack, the high mounted horizontal stabilizer is moved closer and closer to the now nonstreamlined, turbulent wake from the wing. In the region of stall, the T-tail may be engulfed in the wing wake; this results in a drastic reduction in horizontal tail and longitudinal control effectiveness. The reduction in stabilizing effect from the horizontal tail causes a severe pitch-up tendency which the pilot may not be able to counteract even by applying full nose-down longitudinal control. This stall, from which recovery is impossible without an unconventional recovery technique or a "recovery augmentor," such as a tail parachute, is referred to as a "super stall" or "deep stall" and has been experienced by T-tail aircraft flying at an aft center of gravity position.



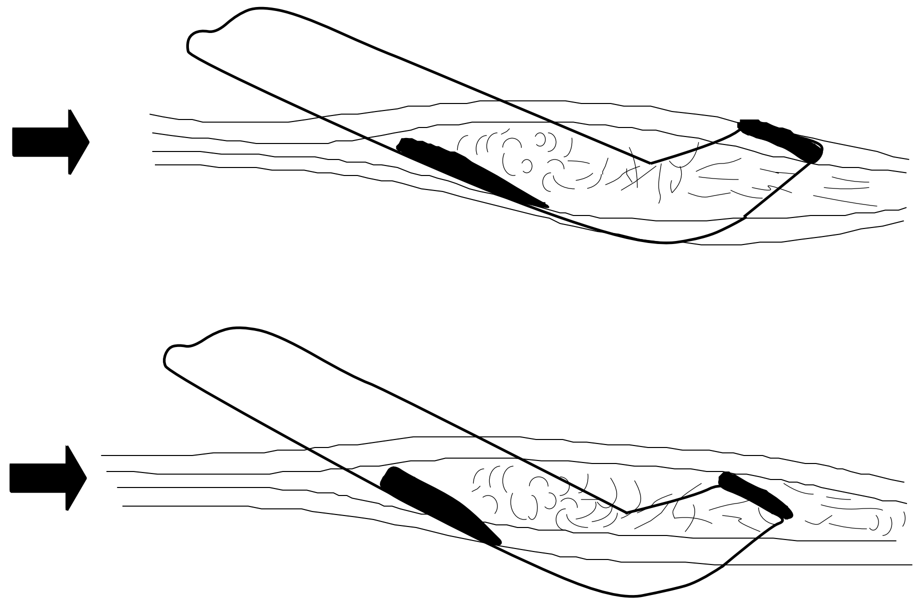


Figure 2.8
Typical Flow Patterns About the T-Tail

The problem of the T-tail entering disturbed airflow at high angles of attack can be complicated by aft mounted engine nacelles (Figure 2.9). The associated increase in airflow disturbance may increase the severity of the loss in horizontal tail effectiveness or cause the loss in effectiveness to occur at lower angles of attack.

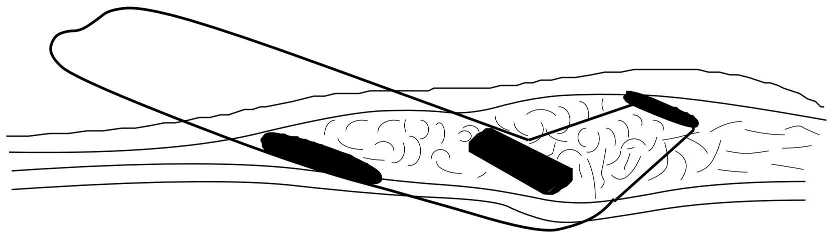


Figure 2.9
Aft-Fuselage Mounted Engines Complicate the T-Tail Airflow Disturbance at High Angles of Attack

If an airplane experiences a “deep-stall” or “super-stall” problem, it may be necessary to incorporate a stall prevention device, such as a “stick-pusher.” Such systems must be reliable and must not cause dangerous flight conditions if accidentally activated during take-off or landing. In stall testing an airplane which may experience the “super-stall,” it may be necessary to install “recovery augmentation” devices, such as tail parachutes or rockets mounted in the nose or tail. The incorporation of an angle of attack indicator is absolutely essential for these stall tests.

2.2.3 Acceleration

Maneuvering produces an effect on stall speed which is similar to the effect of weight. As an example, an airplane in a steady level turn requires a higher lift coefficient, thus increased angle of attack, for a given airspeed; therefore, stall speed is higher in level turning flight.

$$V_S = \sqrt{\frac{2nW}{\rho C_{L_{max}} S}} \qquad \textit{eq 2.1}$$

Where:

V_S = true stall airspeed in feet/seconds

n = normal acceleration in g

W = airplane gross weight in pounds

ρ = air density in slugs/ft 3

S = wing area in square feet

$C_{L_{max}}$ = maximum lift coefficient, dimensionless

Since the maximum lift coefficient is dependent only on angle of attack for a given configuration, the <u>angle of attack at stall is the same for any value of normal acceleration</u>. Note: The effects of Reynolds number and Mach number on maximum lift coefficient are neglected here to simplify the discussion.

Accelerated stall characteristics are more violent than normal stall characteristics for a given configuration because the accelerated stall always occurs <u>at a higher airspeed</u> and may occur at a much higher rate of entry. Adverse characteristics noted during normal stalls are magnified by the increased airspeed at the accelerated stall. Therefore, accelerated stalls should be investigated with caution. Rigorous normal stall tests must precede any accelerated stall evaluation.

Accelerated stall warning may vary with <u>rate of entry</u> into the stall. Rapid rotations generating rapid increases in acceleration (greater than one g per second) may result in virtually no aerodynamic stall warning. Rapid rotations may also result in abrupt accelerated stalls at <u>indicated</u> angles of attack less than <u>actual</u> angles of attack due to lag in the angle of attack indicator.

If operational considerations require that the airplane be flown well into the buffet regime to obtain optimum turning performance, airframe buffet may lose significance as accelerated stall warning unless there is a noticeable increase in buffet intensity just prior to the stall. This situation may result in other pilot cues being used for accelerated stall warning.

Poststall gyrations may be induced by intentionally maintaining an accelerated stall condition. The pattern and severity of the motions are generally dependent upon the energy level (airspeed and altitude) at entry. The investigation of poststall gyrations is usually performed in a build-up program for a spin investigation. However, these gyrations may be experienced during intentional or inadvertent accelerated stalls.

2.2.4 Power

The influence of power on stall characteristics depends upon the type, number, location, and rating of the engine or engines installed.

If the airplane is equipped with a jet power plant (or power plants), the only major effect of power will be that stall airspeed will be less with increased power for a given configuration. Stall characteristics will be indirectly influenced in this situation in that the airplane response to pilot control inputs will be lessened at lower airspeeds.

The stall characteristics of airplanes equipped with reciprocating or turboprop engines may be greatly influenced by the power setting existing at the stall.

If the wing is partially or completely immersed in the propeller slipstream, stall speed will vary markedly with power setting. Full power stalls may occur at extremely low airspeeds and the weakened effectiveness of the control surfaces at these low speeds may severely degrade stall recovery characteristics. Full power stalls in airplanes of this type must be approached with due caution. If the control surfaces are immersed in slipstream, the effectiveness of the surfaces will vary with the amount of power output.

The high powered, single-engine, single-rotation propeller airplane may exhibit a severe “torque-roll” tendency if power is applied rapidly at low airspeeds in the region of the stall. This characteristic may dictate a stall recovery procedure which involves maintaining a fairly low power setting until airspeed increases to a predetermined value.

2.2.5 Stability and Control Augmentation

Stability and control augmentation systems may introduce large control inputs (independent of the pilot) at or near stalling angles of attack which may be detrimental to stall characteristics. This will be most apparent if the augmentation system possesses a high degree of sensitivity and control authority. For illustrative purposes, two examples are presented which emphasize the possible influence of these systems on stall characteristics.

The first example is extracted from accelerated stall tests of a light jet attack airplane equipped with longitudinal and lateral control augmentation and directional stability augmentation. The time history of an accelerated stall which was aggravated by the roll damper mode of the control augmentation system is shown in Figure 2.10. The stall was entered from a left turn with 2g normal acceleration; approach to the stall was characterized by increasing airframe buffet. Just prior to the stall (at 8 seconds on the time history), note that the pilot was required to hold right aileron position to keep the airplane from entering a tighter left turn. The stall was marked by a "directional slice" to the left, at which time the pilot neutralized the controls (at 10 seconds on the time history). At this time, the roll damper portion of the control augmentation mode, sensing a left roll rate without a pilot control input, applied a large right lateral control input. Note that the cockpit control stick was essentially neutral at this time. The aileron input of the roll damper was in the pro-spin direction and the airplane entered a left spin. After approximately two turns of the spin, the pilot deactivated control augmentation and effected recovery by applying aileron into the spin, rudder against the spin, and full aft longitudinal control.

The second example is extracted from normal stall tests of a twin-engine turboprop transport airplane equipped with directional stability augmentation. This augmentation system was composed of yaw damping, directional trim follow-up, and a turn coordination feature. The time history of a normal stall (Power approach configuration) which was aggravated by the turn coordination feature of the stability augmentation system is shown in Figure 2.11. Power approach configuration stalls in this airplane were characterized by abrupt rolls (note the bank angle change at the stall). The turn coordination feature of the stability augmentation sensed the rolling motion and attempted to coordinate with a large left rudder input. Note that about 10 degrees of left rudder deflection was introduced by the stability augmentation system while the pilot was holding right rudder pedal deflection. The large left rudder input increased the left bank angle and sideslip excursions and the airplane entered a series of uncontrollable snaprolls. Recovery was initiated by deactivating stability augmentation. During the recovery, airspeed and normal acceleration limitation of the airframe were exceeded.

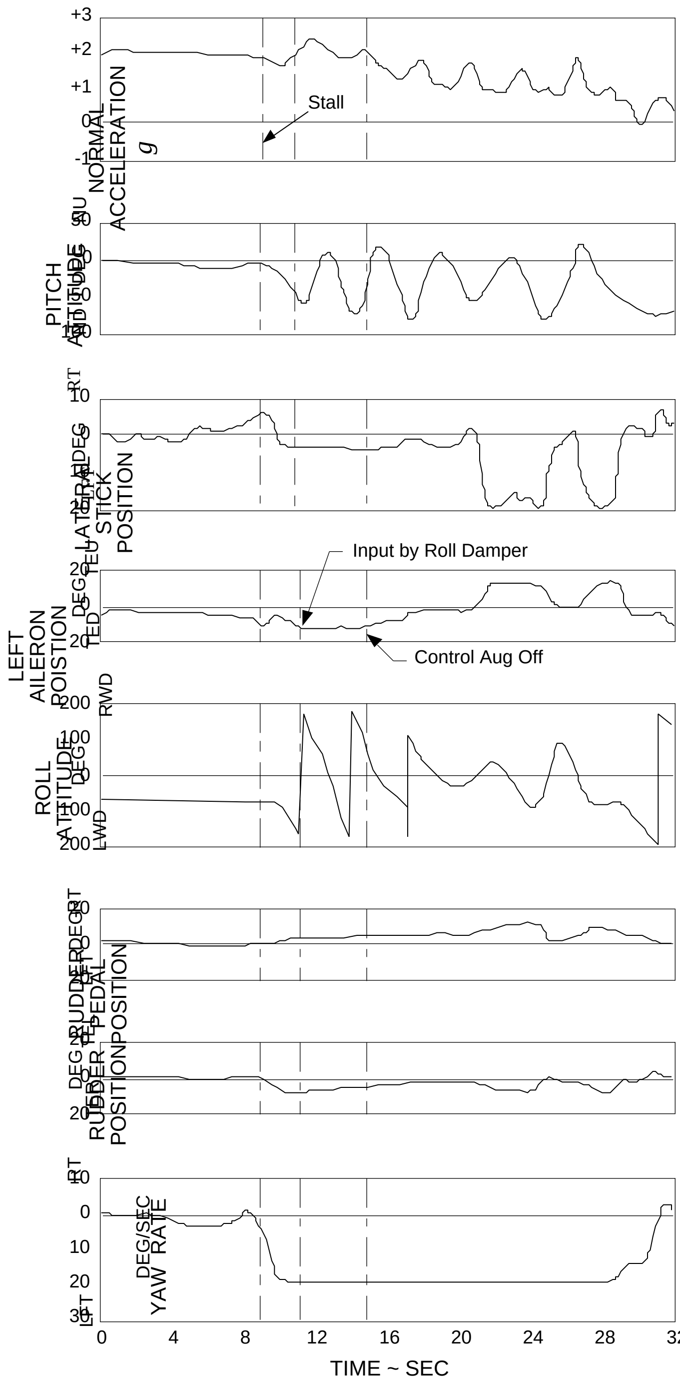


Figure 2.10
Accelerated Stall Time History

Configuration: Power (Mil Thrust)
Loading: Normal Attack
Altitude at Stall: 28,000 FT
Airspeed at Stall: 200 KIAS

CG: 26.5% MAC
Gross Wt: 28,490 LB
Yaw Stab: On
Control Aug: On

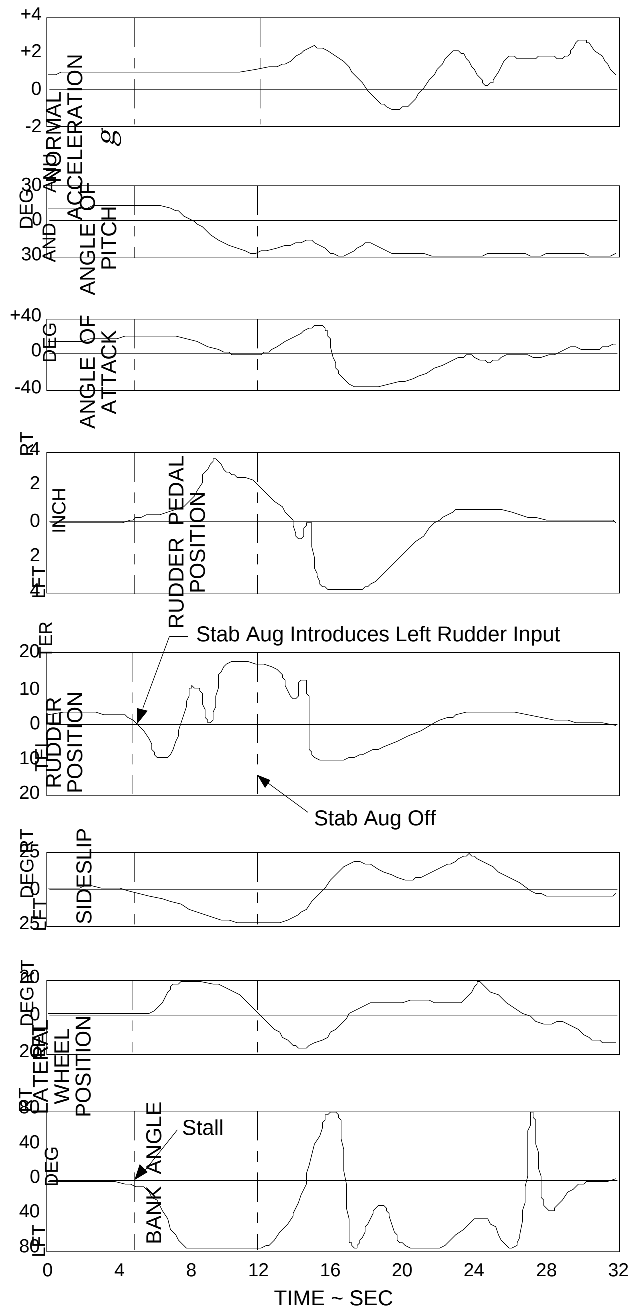


Figure 2.11
Normal Stall Time History

Configuration: Power Approach	CG: 33.6% MAC
Altitude at Stall: 10,000 FT	Gross Wt: 45,900 LB
Airspeed at Stall: 80 KIAS	Stab Aug: On

It should be emphasized that stability and control augmentation systems do not always degrade stall characteristics. Some systems may have no influence; other systems may have significant influence on airplane behavior in the region of the stall. Knowledge of the various modes and functions and the control authority of the augmentation system in the airplane being tested is essential if the stall investigation is to be conducted rigorously and safely.

2.2.6 Miscellaneous Factors

Additional factors influencing the behavior of the airplane in the region of the stall are listed below.

1. Location of Control Surfaces - If the control surfaces are immersed in low energy separated airflow at the stall, the controllability of the airplane will be decreased. The lateral control surfaces are particularly susceptible to immersion in separated flow.

2. Configuration. The extension of wing flaps, wing leading edge slats, speed brakes, landing gear, etc., will have some influence on stall characteristics. This influence may be estimated by consideration of the location of various devices in relation to control surfaces and stabilizers. Some configuration changes, such as flap extension, may result in airframe buffet which masks the prestall aerodynamic buffett, decreasing its value as a stall warning.

3. External Stores. Stall characteristics may be altered by various combinations of external stores. Asymmetric store loadings may severely degrade stall characteristics, particularly during accelerated entries. The investigation of stall characteristics under asymmetric loading conditions should be accomplished on any airplane which may carry asymmetric loads in operational use.

4. Center of Gravity. Stall characteristics may be markedly influenced by airplane center of gravity (CG) if the airplane exhibits a deficiency in longitudinal control effectiveness. At forward CG positions in some airplanes, nose-up longitudinal control effectiveness may not be sufficient to attain maximum lift coefficient. The minimum attainable airspeeds for these airplanes would be marked by steady flight with full nose-up longitudinal control; minimum attainable speed

would, of course, vary with CG position, decreasing as the CG moves aft. At aft CG positions in other airplanes, nose-down longitudinal control effectiveness may not be sufficient to quickly reduce angle of attack after attaining the stall. This situation would seriously compromise, and might preclude, stall recovery.

5. Shock-Induced Separation. Shock-induced separation or a "shock stall" may cause the stall to occur at a lower angle of attack than might be predicted through incompressible flow considerations. Tendencies toward shock-induced separation would, of course, increase with increasing subsonic airplane Mach number; however, shock stalls can occur at Mach numbers well below the "normal transonic region." The phenomenon of shock-induced separation may be particularly evident during accelerated stalls.

2.2.7 Characteristics Which May Limit Minimum Steady Airspeed

For some airplanes, the attainment of maximum lift coefficient may not be possible or feasible. This may be caused by a loss of directional control without a reduction of lift, lack of longitudinal control effectiveness, or an extremely large increase in drag coefficient.

2.2.7.1 LOSS OF CONTROL WITHOUT REDUCTION OF LIFT

During approaches to normal or accelerated stalls, directional stability may be reduced significantly through the deterioration of airflow around the vertical stabilizer. At high angles of attack, the vertical tail may become immersed in nonstreamlined, low energy flow generated by flow separation on the wing and interference effects from aft-fuselage mounted engines, speedbrakes, or other protrusions (Figure 2.12).

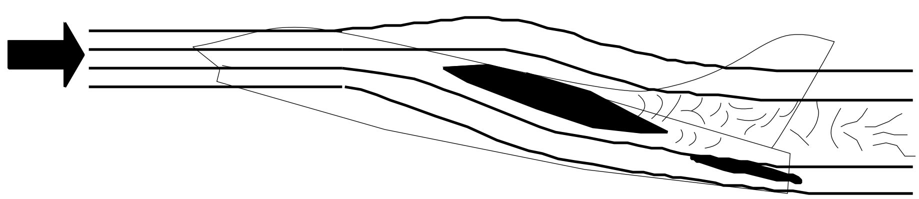


Figure 2.12
Typical Flow Pattern Around the Vertical Tail at High Angle of Attack

The deterioration in effectiveness of the vertical tail generally results in increasing yaw excursions with increases in angle of attack. The airplane may diverge directionally prior to attaining maximum lift coefficient if the destabilizing action progresses sufficiently. Directional divergence can be "triggered" or aggravated by lateral control inputs if these control inputs generate significant yawing moments. High angle of attack directional divergence is sometimes referred to as "slicing" and would limit minimum steady airspeed and preclude attainment of the maximum lift coefficient.

2.2.7.2 LACK OF LONGITUDINAL CONTROL EFFECTIVENESS

The longitudinal control surfaces on some airplanes may not be sufficiently effective to rotate the airplane to the angle of attack corresponding to maximum lift coefficient. These airplanes are sometimes referred to as "elevator-limited" airplanes. Minimum steady airspeed or maximum angle of attack in this situation is that which is attained with full nose-up longitudinal control. Since elevator effectiveness is a function of center of gravity (CG) position, stalling airspeed and angle of attack for these airplanes will vary with CG position.

2.2.7.3 "ZERO RATE OF CLIMB SPEED"

The very low aspect ratio (less than two) airplane exhibits practically no aerodynamic stall; however, its minimum practical airspeed will be limited by performance considerations, if not by adverse stability and control characteristics. The variation of lift and drag coefficients for the low aspect ratio or "slender delta" design gives insight into the problem which may exist (Figure 2.13).

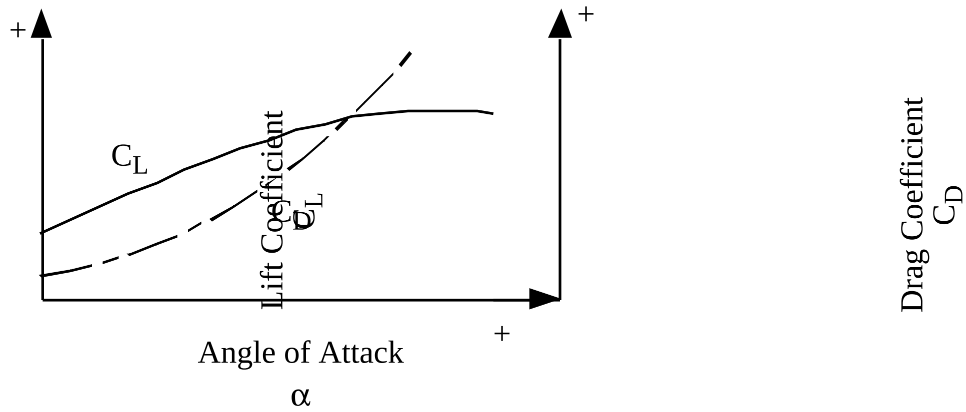


Figure 2.13
Typical Variations of Lift and Drag Coefficient
for the Low Aspect Ratio Airplane

While the lift curve exhibits no definite peak which would define maximum lift coefficient, the drag curve may exhibit a tendency to slope upward sharply at high angles of attack. An angle of attack, corresponding to an airspeed, would be attained at which the airplane could not maintain a rate of climb with maximum engine power. This airspeed is defined as the "zero rate of climb speed" (ZRCS). Of course, it will change with configuration, altitude, engine output, and gross weight. The only hazard directly associated with flight at airspeeds less than ZRCS is loss of performance. For example, if an airplane decelerates below ZRCS during the approach, a sacrifice in altitude (possibly a significant one) must be made in order to execute a wave-off. A disturbing feature of an airplane capable of steady flight at airspeeds below that at which it has sufficient power to maintain level flight is the long "settling time" needed to establish a final flight path. For instance, it may be possible to fly at speeds slightly below ZRCS with a slight rate of climb for periods as long as 1 minute. The slight rate of climb is caused by the inertia of the airplane as is settles down on its final flight path. The pilot might deduce that he is above ZRCS due to this phenomenon. However, he eventually finds that the airplane begins a shallow descent. Increasing angle of attack at this stage only increases the rate of descent and some height must be sacrificed for recovery. Recovery from airspeeds below ZRCS can only be accomplished by pushing the nose over to decrease angle of attack, then reestablishing a climb at an airspeed above ZRCS.

For some airplanes, "zero rate of climb speed" may constitute the extreme limit of safe flight, and operational speeds must be chosen which provide adequate margins against accidental exposure to irrecoverable situations.

NOTE: It must be emphasized the ZRCS is expected to limit minimum airspeed only for airplanes with very low aspect ratio and very slender wing designs. During stall investigations of any airplane, certain flight conditions will be encountered where the airplane will be descending at significant rates, such as landing configuration with idle power or power on stalls at high altitude. However, the high rate of descent does not necessarily indicate a minimum airspeed limit above aerodynamic stalling airspeed and should not be reported as such. Whenever safety considerations/risk management permit, the stall investigation should probe into the stall region as deeply as possible.

2.2.8 Stall Warning and Stall Prevention Devices

2.2.8.1 ARTIFICIAL STALL WARNING

Airplanes which do not exhibit adequate aerodynamic stall warning, such as airframe buffet, are frequently equipped with devices which detect the approach of the stall and transmit a warning to the pilot. Artificial stall warning is, at best, a poor substitute for aerodynamic stall warning since the detection device is never absolutely reliable.

Any artificial stall warning system should satisfy the following requirements:

1. The system should be capable of stall warning for any airplane configuration, airspeed, altitude, normal accelerations, sideslip, bank angle, and power setting. In addition, the system should not be susceptible to atmospheric influence, such as temperature and pressure variations, precipitation, and icing.

2. The warning provided the pilot should be unmistakable and sufficiently in advance of the stall to allow avoidance of the stall without undue pilot effort.

3. The system should be easy to maintain and easy to calibrate on the ground.

Some of the devices used to detect approach of the stall and their principle of operation are listed below.

Table 2.I
Summary of Stall Warning Devices

Device	Principle of Operation
Free Floating Probe or Vane	Airflow direction (angle of attack)
Drag Sensing Probe	Airflow direction (angle of attack)
Differential Pressure Head	Airflow direction (quantity proportional to angle of attack)
Null Pressure Probe	Airflow direction (angle of attack)
Leading Edge Tab	Wing dynamic pressure
Trailing Edge Tab	Wing dynamic pressure
Trailing Edge Pitot Tube	Wing dynamic pressure
Pitot Tube with Local Spoiler	Wing dynamic pressure
Flush-Mounted Wing Port	Static pressure at wing surface
Trailing Edge "Blister"	Static pressure at wing surface
Boundary Layer Pitot Tube	Boundary layer pressure fluctuation

The means by which the pilot is warned of the approaching stall may be visual (warning light), oral (sound in earphones), or physical (shaking or vibrating of rudder pedals or control stick). The most suitable artificial forms of cockpit warning are probably the "stick shaker" and vibrating stick grip; these warning signals are similar to aerodynamic buffeting of the controls and are difficult to misinterpret.

2.2.8.2 ARTIFICIAL OR AUTOMATIC STALL PREVENTION

For some airplanes, particularly large transport and passenger types, stalling maneuvers may be structurally or aerodynamically unsafe. In order to guarantee adequate flight safety even under abnormal flight conditions, such as strong, sudden pull-ups or abrupt longitudinal attitude changes caused by gusts, these airplanes may be equipped with a "stall prevention" system. Stall prevention systems are used quite commonly in "T-tail" airplanes.

Any artificial stall prevention system should satisfy the following requirements:

1. The system should be capable of stall prevention for any airplane configuration, airspeed, altitude, normal acceleration, sideslip, bank angle, and power setting. The system should not be susceptible to atmospheric influence, such as temperature or pressure variations, precipitation, and icing.

2. The system should provide a large nose-down pitching moment at the stall or just after the stall; however, the pilot should be able to "override" the system if he desires. The "override" force should be large enough to discourage inadvertent "override" and associated "deep-stall" penetration.

3. Inadvertent operation of the system should not lead to dangerous flight conditions. This is particularly applicable to the take-off and landing evolutions.

4. The system should be easy to maintain and easy to calibrate on the ground.

A commonly used stall prevention device is a "stick pusher" arrangement which is activated through a signal from an angle of attack or pressure sensor.

No matter how well-designed and how reliable it may be, a stall prevention system represents added complexity in the airplane. Unless safety or overriding design considerations dictate otherwise, stall prevention systems should be avoided.

2.3 TEST PROCEDURES AND TECHNIQUES

2.3.1 Preflight Procedures

Successful stall investigations can be accomplished only after thorough preflight planning. During preflight planing, the purpose and scope of the tests must be clearly defined. After purpose and scope are clearly understood, a "plan of attack" or test method can be formulated.

Preflight planing should start with research. This includes a study of the airplane - many stall characteristics can be predicated by studying various design parameters of the airplane. All available information on stall characteristics should be reviewed. Much useful information may be gained by conversations with pilots and engineers familiar with the airplane.

The test conditions - altitude, configuration, center of gravity, and trim airspeeds - must be determined. Test conditions should be commensurate, as much as possible, with the mission environment of the airplane. However, safety considerations/risk management dictate that investigations of stall characteristics be performed in such a manner that the most critical conditions are tested only after a reasonable build-up program. Altitude at stall entry should never be lower than 10,000 feet above ground level; however, a higher minimum altitude may be used if unusual characteristics are expected. Although center of gravity (CG) position may affect both the stall and the recovery, tests at the most forward and most aft operational CG positions are generally adequate. However, if a lack of nose-down longitudinal control or "pitch-up" at high angles of attack are suspected, forward CG positions should be used for initial investigations. Because of possible adverse stall characteristics resulting from high power settings and extension or activation of high-lift devices, a "clean" airplane configuration with low engine power settings should be chosen for initial stall tests. Appropriate trim airspeeds should be chosen for each configuration to be evaluated. For example, appropriate trim conditions for an investigation of power approach configuration stalls would be those corresponding to normal approach airspeed and angle of attack. Of course, the effects of "trimming" into the stall and "out of trim" entries into the stall should be determined also.

The amount and sophistication of instrumentation required will depend on the purpose and scope of the evaluation. A pure qualitative investigation can be accomplished with only cockpit and hand-held instruments. A portable tape recorder for pilot comments is especially useful. If accurate quantitative information is needed, or if preliminary studies indicate very adverse stall characteristics, automatic recording devices, such as oscillograph, photopanel, and telemetry, should be utilized. The parameters to be recorded and ranges and sensitivity of test instrumentation will vary somewhat with each test program.

The final step in preflight planning is the preparation of pilot data cards. An example of a stall data card is presented in Figure 2.14. However, most test pilots desire to modify data cards to their own needs or construct data cards for each test. At any rate, the data cards should list all quantitative information desired and should be easy to interpret in flight. For stall investigations in particular, several data cards with adequate space for pilot comments should be provided.

WEATHER			CARD NO. PIR RIS
AIRPLANE TYPE	BU NO	TIME T. O. TIME LAND	DATE
PILOT		T O C G GEAR DOWN %	UP %
CONDITION			T. O. GROS WT.
EXTERNAL CONFIGURATION			
TEST CONFIGURATION		TEST ALTITUDE	

APPROACH	TRIM SPEED		
	TRIM TABS		
	POWER		
	FUEL QUANTITY		
	WRN		
	BUFFET		
	CONTROL EFFECTIVENESS		
	CONTROL FORCES		
STALL			
	LONG STICK POSITION		
	BUFFET		
	ROLL		
	PITCH		
	CONTROL EFFECTIVENESS		
	CONTROL FORCES		
RECOVERY	ALT. LOST		
	CONTROL EFFECTIVENESS		
	CONTROL FORCES		
	PROGRESSIVE STALL TENDENCIES		

STALL DATA **PRNC-NATC-3900/4** GPO 929-452

Figure 2.14
Typical Stall Data Card

2.3.2 Flight Test Techniques

Stall characteristics must be evaluated in relation to their influence on mission accomplishment. Thus, both normal and accelerated stalls must be performed under entry conditions which could result from various mission tasks. However, prior to evaluating stalls entered from these conditions, a more controlled testing approach should be employed. This approach allows lower deceleration rates into the stall and lower pitch attitudes at the stall, thereby reducing changes for “deep-stall” penetration without adequate buildup. After the controlled stall investigation, if stall characteristics permit, simulated inadvertent stalls should be investigated under conditions representative of operational procedures.

2.3.2.1 THE CONTROLLED STALL TEST TECHNIQUE

The easiest and safest approach to controlled stall testing is to divide the investigation into three distinct parts:

1. Approach to the stall

2. Fully developed stall

3. Stall recovery

2.3.2.2 APPROACH TO THE STALL

During this phase of the investigation, adequacy of stall warning and retention of reasonable airplane controllability are the primary items of interest. Assessment of stall warning requires subjective judgment by the pilot. Only the pilot can decide when he has been adequately warned. Warning must occur sufficiently in advance of the stall to allow prevention of the stall by normal control applications after a reasonable pilot reaction time. However, stall warning should not occur too far in advance of the stall. For example, it is essential that stall warning for approach configuration occur below normal approach speed. Stall warning which occurs too early is not only annoying to the pilot but is meaningless as an indication of proximity to the stall.

The type of stall warning is very important. Primary stall warning is generally in the form of airframe buffet, control shaking, or small amplitude airplane oscillations in roll, yaw, or pitch. Other secondary cues to the approach to the stall may be high pitch attitude, large longitudinal control pull forces (of course, this cue can be destroyed by "trimming into the stall"), large control deflections, or sluggish control response. In any case, stall warning, whether natural or artificial, should be unmistakable, even under conditions of high pilot workload and stress and under conditions of atmospheric turbulence. If an artificial stall warning device is installed, approach to the stall should be evaluated with the device operative and inoperative to determine if the device is really required for normal operations.

During this phase of the evaluation, the test pilot must evaluate stall warning with the intended use and operational environment in mind. He must remember that he is specifically looking for the stall warning under controlled conditions. The operational pilot probably will not be. This question must be answered: will the operational pilot, preoccupied by other tasks and not concentrating on stalls, recognize approach of the stall and be able to prevent the stall?

The general flying qualities of the airplane should be investigated during the approach to the stall as well as stall warning characteristics. Longitudinal, lateral, and directional control effectiveness for maintaining a desired attitude may deteriorate significantly during the approach to the stall. Loss of control about any axis such as uncontrollable pitch-up or pitch-down, "wing drop," or directional "slicing" may define the actual stall. During the approach to the stall, the test pilot should be particularly aware of the amount of longitudinal nose-down control available because of the obvious influence of this characteristic on the ability to "break" the stalled condition and make a successful recovery.

This phase of stall investigation usually begins with onset of stall warning and ends at the stall; therefore, the test pilot will certainly be concerned with the manner in which the airplane stalls and the ease of recovery. However, <u>primary emphasis</u> is placed on obtaining an accurate assessment of stall warning and general flying qualities during the approach to the stall. During initial investigations, it may be prudent to terminate the approach short of the actual stall, penetrating deeper and deeper with each succeeding approach until limiting conditions or the actual stall are reached. In addition, the rate of approach should be low initially, less than 1 knot per second for normal stalls. Investigations of accelerated stalls should be made by using the “constant normal acceleration” technique or the “constant airspeed” technique. The constant normal acceleration technique is performed by selecting and holding a desired g level while allowing the airplane to decelerate until the stall is encountered. Slow deceleration rates (typically 2 knots/second) are used for initial investigations. As experience is gained, faster deceleration rates should be performed unless safety considerations dictate otherwise. The constant airspeed technique will be discussed in the STALL TEST TECHNIQUES section.

The test pilot should record at least the following cockpit data during the approach to the stall:

1. Airspeed and angle of attack at stall warning.

2. Type and adequacy of stall warning.

3. Longitudinal control force at stall warning (either measured or estimated).

4. Qualitative comments regarding controllability and control effectiveness.

2.3.2.3 FULLY DEVELOPED STALL

During this phase of the investigation, the primary objective is to accurately define the stall and the associated airplane behavior. The stall should be well-marked by some characteristic, such as pitch-up or pitch-down or lateral or directional divergence. In general, any pitch-up or directional divergence at the stall is undesirable because pitch-up may precipitate a deep stall penetration and directional divergence may lead to a spin. Pitch-down at the stall and lateral divergence may be acceptable; however, severe rolling, pitching, or yawing or any combination of the three are obviously poor characteristics.

Control effectiveness as evidenced by the pilot's ability to control or induce roll, pitch, or yaw should be evaluated in the stall, if airplane behavior permits this to be done safely. Obviously, control effectiveness should be evaluated with a suitable build-up program. Initially, control inputs only large enough to effect an immediate coordinated recovery should be used. As experience is gained, the airplane should be maintained in the stalled condition for longer and longer periods of time, and the effectiveness of all controls evaluated with larger and larger control deflections.

The test pilot should record at least the following cockpit data regarding the stall:

1. Airspeed and angle of attack at stall.

2. Load factor (accelerated stalls only).

3. Characteristic which defines the stall.

4. Longitudinal control force at the stall (either measured or estimated). The ratio of longitudinal control forces at stall and stall warning is a rough indication of longitudinal stability in the high angle of attack region and an indication of the ease of inadvertent stalling.

5. Qualitative descriptive comments.

2.3.2.4 STALL RECOVERY

During this phase of the investigation, primary items of interest are the ease of recovery (the pilot's task), general flying qualities during the recovery, altitude required for recovery, and the determination of an optimum recovery technique. The definition of stall recovery may vary with the configuration under investigation. For example, the goal of recovery for configurations commensurate with combat maneuvering may be to regain sufficient control effectiveness about all three axes to perform offensive or defensive maneuvering tasks; the attainment of level flight may not be critical in these configurations. The goal of recovery for take-off and approach configurations should be the attainment of level flight with a minimum loss of altitude <u>and</u> the regaining of sufficient control effectiveness to safely maintain stall-free conditions. In each case, the test pilot must clearly define “stall recovery.”

During initial investigation, the stall recovery procedures specified in pertinent publications should be utilized and the ease of effecting recovery evaluated. If no procedure has been developed, initial recovery must be accomplished with a preliminary technique formulated from all available technical information. As experience is gained, various modifications to the recovery procedure should be made until an optimum procedure is determined. In arriving at an optimum procedure for use by the operational pilot, the test pilot must not only consider the effectiveness of the technique (in terms of altitude loss or maneuverability regained) but must also consider the simplicity of the technique.

The test pilot should record at least the following data regarding stall recovery:

1. Qualitative comments on ease of recovery

2. Optimum recovery technique

3. Altitude loss in recovery

4. Qualitative comments on control effectiveness

2.3.2.5 PROFILE OF THE CONTROLLED STALL TEST TECHNIQUE

The general flight profile of the controlled stall investigation is presented in Figure 2.15. Points along the profile are further explained on the following page. It should be remembered that until familiarity with stall behavior of the airplane is gained, the profile may be broken off at any point.

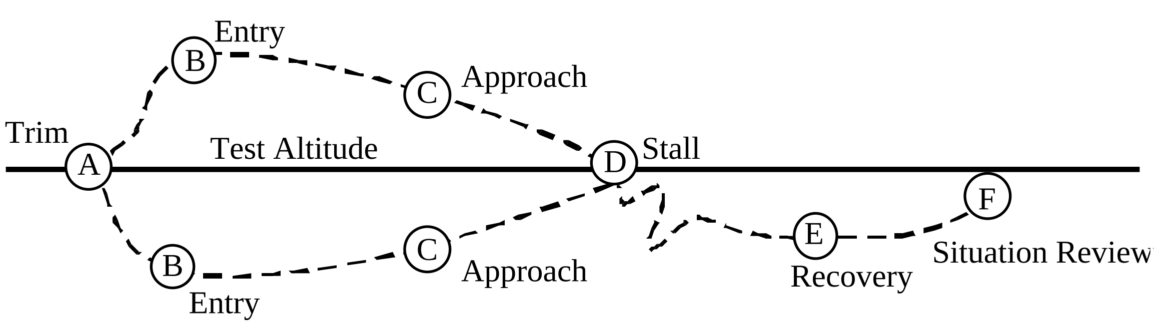


Figure 2.15
General Profile of the Controlled Stall Investigation

A. Trim Point. The configuration under investigation should be established. At least the following items should be recorded in the cockpit:

1. Trim speed

2. Trim tab setting

3. Power setting

4. Fuel quantity

If automatic recording devices are installed, a “trim shot” should be made.

B. Entry Point. Decide on an entry point which will result in the stall occurring near the test altitude (+1000 feet). The entry procedures will be different for normal and accelerated stall investigations.

<u>Normal Stalls</u>. Slow the airplane rapidly to about 20 KIAS above the estimated stall warning speed. Power reduction or speed brake extension may be utilized. <u>Reestablish trim configuration at this new airspeed</u>. Make a slight pitch increase to start the deceleration toward the stall. Using the visual horizon as a primary cue and airspeed indicator as a crosscheck, establish the desired deceleration rate. Deceleration rate should be one knot per second or less initially, but may be increased as experience is gained.

<u>Accelerated Stalls</u>. For initial investigations, the constant normal acceleration technique is normally used. Select an entry normal acceleration commensurate with configuration, flight conditions, and familiarity with the accelerated stall characteristics. If appropriate and feasible, slow the airplane to about 40 KIAS above the estimated stall warning speed for the selected load factor. Entry normal acceleration should be increased to maximum allowable or attainable as familiarity is gained. Establish a roughly level turn at entry normal acceleration. Maintaining normal acceleration constant, establish the desired deceleration rate. The primary reference should be the visual horizon, although the normal accelerometer, angle of attack indicator, and airspeed indicator will have to be crosschecked frequently. Deceleration rate should be approximately 2 knots per second or less initially, but may be increased as experience is gained.

C. Approach to the Stall. If automatic recording devices are utilized, they should be activated at some convenient point prior to stall warning. The event marker may be used to mark stall warning on the recording traces. In order to aid in remembering data, the pilot should call out the airspeed and angle of attack at stall warning onset and mentally note the type and adequacy of the warning. For approaches to normal stalls, utilize pitch control to maintain 1.0 g normal acceleration and the predetermined deceleration rate. During approaches to accelerated stalls, a combination of bank angle and pitch attitude are used to maintain normal acceleration and deceleration rates at predetermined values. An increase in bank angle will slow the deceleration rate and a decrease in bank angle will speed it up, providing <u>the normal acceleration is maintained constant</u>.

D. The Stall. There is a natural tendency to relax nose-up longitudinal control as the stall is approached in unaccelerated or accelerated entries. This tendency should be overcome by maintaining deceleration rate and normal acceleration into the stall with positive pitch attitude control. If the stall is marked by pitch-down, pitch attitude and normal acceleration should be closely monitored for accurate detection of the stall. At the stall, actuate the event marker if automatic recording devices are used and call out the airspeed, angle of attack, and altitude at the stall. Mentally note the airplane behavior at the stall and initiate recovery control inputs and configuration changes.

E. The Recovery. Follow the predetermined recovery procedure and effect recovery. Qualitatively evaluate recovery characteristics. Call out final recovery altitude and actuate the event marker if utilized. The automatic recording devices should be deactivated when convenient.

F. The Situation Review. As the airplane is started toward the next stall test point, the pilot should record at least the following cockpit data from the last stall:

1. Stall warning speed and angle of attack

2. Type and adequacy of stall warning

3. Stall speed and angle of attack

4. Stall characteristics

5. Recovery characteristics

6. Altitude lost and airspeed buildup during recovery

2.3.2.6 ALTERNATE TECHNIQUE FOR ACCELERATED STALL INVESTIGATIONS

It is recommended that initial accelerated stall tests be performed utilizing the "constant normal acceleration" technique described above; this technique allows a gradual build-up to accelerated stalls at high levels of normal acceleration. After experience is gained in the accelerated stall characteristics of the airplane, the wind-up turn or "constant airspeed" technique may be utilized; this technique is more expeditious and somewhat simulates inadvertent stalls in operational use. The technique merely involves gradually increasing angle of attack or normal acceleration at <u>constant airspeed or Mach number</u> in a wind-up turn until the airplane stalls. The difference between the angle of attack or normal acceleration at stall warning onset and at stall is an additional measure of the adequacy of the stall warning.

2.3.2.7 SIMULATED INADVERTENT STALLS

If the results of the controlled stall investigation indicate that inadvertent stalls will produce no dangerous flight conditions, <u>simulated inadvertent stalls</u> should be investigated from entry conditions which could result from various operational procedures. These entry conditions will generally involve more rapid deceleration rates during normal stalls and more rapid increases in normal acceleration during accelerated stalls.

The mission tasks most likely to result in inadvertent stalls should be used as entry conditions. These mission tasks may be those required in air combat maneuvering, gunnery exercises, missile attacks and reattacks, and conventional and nuclear weapons deliveries. Other tasks peculiar to take-off and approach conditions must also be used as entry conditions; these may include simulated catapult launches, field take-offs, wave-offs, or "bolters," and field or carrier approaches. Of course, the mission tasks will vary widely in all test programs; these are presented as examples for illustration. No matter what tasks are selected, all stalls should be performed at a safe altitude (at least 10,000 feet).

By performing simulated inadvertent stalls under conditions representative of operational procedure, more complete knowledge is gained of the adequacy of stall warning, the characteristics of the actual stall, and the ease of recovery from the stall. No rigorous stall investigation would be complete without this type of evaluation.

2.3.3 Postflight Procedures

As soon as possible after returning from the flight, the test pilot should write a brief, rough qualitative report of the airplane behavior in the region of the stall. This report should be written while the events of the flight are fresh in his mind. The qualitative opinion will be the most important part of the final report of the stall characteristics.

Appropriate data should be selected to substantiate the pilot's opinion. If automatic recording devices have been utilized, stall time histories will be presented in the stall report. The time histories should be of particularly well flown stalls, or of stalls during which some unusual characteristics were observed. Examples of stall time histories are presented earlier in this section (Figure 2.10 and 2.11). Stall data also may be effectively presented in tabular form. An example is presented in Figure 2.16.

CONF	TRIM		CG	ALT	IAS		V_W/V_S	AOA		N_Z		Longitudinal Cont. Forces (LBS)		ALT. LOST IN RECOVERY (FT)
	M	IAS	(%MAG)	(FT)	Warn	Stall		Warn	Stall	Warn	Stall	Warn	Stall	

Figure 2.16
Typical Stall Data Table

2.4 SPECIFICATION REQUIREMENTS

Requirements for stall characteristics are contained in Section 3.4.2 of Military Specification MIL-F-8785C of 5 November 1980, hereafter referred to as the Specification. The requirements of Section 3.4.2 may be modified by the applicable airplane Detail Specification. Comments concerning individual paragraphs are presented below.

3.4.2 <u>Flight at High Angles of Attack.</u> The requirements of 3.4.2 through 3.4.2.2 are intended to assure safety and the absence of any compromise in the performance of any mission task due to stall warning, stall, and stall recovery characteristics.

3.4.2.1 <u>Stalls.</u> The stall may be defined by either airflow separation with increasing angle of attack causing loss of lift, control difficulty, or excessive buffet/vibration (see 6.2.2 and 6.2.5) or by a minimum permissible airspeed for safe execution of a specific mission task (see 3.1.9.2.1). The maximum obtainable angle of attack may be control limited; i.e., full aft stick applied, in which case the maximum obtainable angle of attack defines the stall (see 6.2.5 a). The stall may be defined in terms of airspeed or angle of attack, but the definition must clearly state which of the above conditions exist.

3.4.2.1.1 Stall Approach. For normal stalls, deceleration rates of up to 1 knot per second should be used to determine compliance with the Specification requirements. For accelerated stalls, the approach rate should be a function of angle of attack rather than airspeed. Rates of increase of wing incidence angle of attack of 2 degrees per second or less should be used to determine Specification compliance. For both normal and accelerated stalls, greater deceleration or angle of attack rates could be utilized during simulated inadvertent stalls under conditions representative of operational procedures.

3.4.2.1.1.2 Warning Range for Accelerated Stalls. Conflict could arise here between the defined Operational Flight Envelope and the minimum angle of attack values at which onset of stall warning is permitted. For TPS purposes, this requirement will be considered met if onset of stall warning occurs within the angle of attack limits stated.

The value of α_o may be determined in flight by recording α over a range of normal load factors from -1 to +3g and plotting the results. The intercept of the curve with the α axis at zero load factor gives α_o. In most cases, oscillograph data will be required to obtain accurate results. Ships service AOA gauges will probably give a fair approximation but may contain nonlinearities.

3.4.2.1.3 Stall Prevention and Recovery. The requirement here which needs particular attention is that stall recovery technique shall be simple and easy to apply and that there shall be no excessive altitude loss.

3.4.2.2 Post-Stall Gyrations and Spins

3.4.2.2.1 Resistance of Loss of Control

3.4.2.2.2 Recovery from Post-Stall Gyrations and Spins. Tests to determine compliance with these requirements will only be conducted at TPS if specifically briefed. Tests of this nature require a cautious and progressive approach which is time-consuming and requires special safety precautions. An indication of the probability of meeting the intent of these requirements may possibly be obtained without investigating the entries and control applications specified.

2.5 GLOSSARY

Camber — The curvature of the mean line of an airfoil section from leading edge to trailing edge.

Thickness Ratio — The ratio of the maximum thickness of an airfoil section to its chord length.

Autorotation — Uncontrolled rolling or rotating, as in a spin.

Aspect Ratio — The ratio of the span of the wing to the mean chord.

Taper — A gradual reduction in chord length from wing root to wing tip.

Deep Stall — A flight condition in which the airplane has attained an angle of attack far higher than the angle of maximum lift coefficient.

Slat — Any of certain long narrow vanes or auxiliary airfoils. The vane used in an automatic slot.

Slot — A long and narrow opening, as between a wing and a deflected Fowler flap. A long and narrow spanwise passage in a wing, usually near the leading edge, for improvement of airflow conditions at high angles of attack.

Endplate — A plate or surface at the end of an airfoil attached in a plane normal to the airfoil that inhibits the formation of tip vortex, thus producing an effect similar to that of increased aspect ratio.

Reynolds Number — A nondimensional parameter representing the ratio of the momentum forces to the viscous forces about a body in motion. Reynolds number decreases with increase in altitude and increases with increase in true velocity, if the dimensions of the body remain constant.

Post-Stall Gyrations — Random oscillations of the airplane about all axes following departure from controlled flight.

Shock Stall — A stall brought on by compressibility burble; i.e., by separation aft of a shock wave.

2.6 REFERENCES

1. *Advisory Group for Aeronautical Research and Development*, Conference Proceedings No. 17, Stability and Control, September 1966.

2. *Advisory Group for Aeronautical Research and Development*, Flight Test Manual, Volume II, Stability and Control, by Courtland D. Perfins, Professor, Aeronautical Engineering Department, Princeton University.

3. *Flight Testing of Aircraft*, by M.G. Kotick, et al., NASA Technical Translation F-442, April 1967.

4. *Handling and Performance Exercises Test Techniques*, Empire Test Pilots' School, Flying Department Notes.

5. *Military Specification, Flying Qualities of Piloted Airplanes*, MIL-F-8785C, of 5 November 1980.

6. *National Aeronautics and Space Administration Aeronautical Dictionary*, by Frank D. Adams, United States Government Printing Office, Washington, 1959.

7. *A Piloted Simulation Study of Operational Aspects of the Stall Pitch-Up*, by Maurice D. White and George E. Cooper, Ames Research Center, Moffett Field, California, NASA Technical Note D-4071, July 1967.

8. *Principles of Aerodynamics*, by James H. Dwinnell, Boeing Airplane Company, Seattle, Washington. Publisher: McGraw-Hill Book Company, Inc., New York, New York, 1949.

9. *A Review of the Stall Characteristics of Swept Wings*, by Charles W. Harper and Ralph L. Maki, Ames Research Center, Moffett Field, California, NASA Technical Note D-2373, July 1964.

10. *Stability and Control*, Empire Test Pilots' School Ground School Notes, Book "C".

11. *Stability and Control*, USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

12. *Supersonic and Subsonic Airplane Design*, by Gerald Corning, Professor, Aeronautical Engineering Department, University of Maryland. Publisher: Edwards Brothers, inc., Ann Arbor, Michigan, 1953.

13. *"Zero Rate of Climb Speed" as a Low Speed Limitation for the Stall-Free Aircraft*, by W.J.G. Pinsker, British Ministry of Technology Current Paper No. 931, May 1966.

CHAPTER THREE

SPINS

CHAPTER THREE

FIGURES

CHAPTER THREE

SPINS

3.1 INTRODUCTION

In order to obtain the maximum capability from tactical airplanes, it is necessary to fly them near the limit of their flight envelopes. This includes lift boundaries, structural limits, and minimum and maximum airspeed limits. This type of flying will often result in inadvertent stalls and occasionally in inadvertent spins. If the tactical pilot has confidence in his capability to successfully recover from any uncontrolled flight maneuver which may be inadvertently entered, he will not hesitate to fly the airplane near the limits of its flight envelope. If, however, the pilot does not have this confidence, and does not know whether or not he can recover his airplane from a spin, he will probably allow himself a greater margin of safety and not fly his airplane to the extreme edges of the tactical envelope. On the other hand, any airplane which cannot be consistently recovered from a spin or a departure after an accelerated stall will also not be flown to its utmost tactical advantage. In both cases, a significant and extremely important portion of the airplane's tactical capability will be negated. It is of primary importance, therefore, that all U.S. Navy tactical and training airplanes be evaluated by Navy pilots in comprehensive spin programs. In this manner, spin recovery techniques and optimum spin avoidance maneuvers can be determined, thus providing the fleet pilot with the information he needs to gain confidence in his airplane's tactical capabilities. Even if an airplane is never cleared for intentional spins, the results of a good spin investigation will provide important data to the fleet and to flight handbooks which will show tactical pilots that the airplane has been spun and recovered successfully.

3.2 THEORY

3.2.1 General

There is probably no other aerodynamic maneuver about which exists more misinformation and confusion than the spin. While the interaction of aerodynamic and inertia forces in a spin involves long and complex equations of motion, the factors which cause spins are in themselves relatively simple. It would be well at the start of this discussion to dispel any confusion by defining clearly those terms which will be used in this discussion of stalls and spins.

3.2.2 Stalls

An aerodynamic stall is defined as a condition in which the wing attains an angle of attack greater than the angle of attack for maximum lift, resulting in a loss of lift and an increase in drag. Stalls may be either erect, inverted, normal, or accelerated. For purposes of this discussion, normal erect stalls are those stalls entered in positive angle of attack flight at a load factor of one g or less by decreasing airspeed (including both slow and rapid deceleration). Normal inverted stalls are similar with the exception that the angle of attack and load factor have negative values. Erect accelerated stalls are those stalls entered at load factors greater than 1.0 g and inverted accelerated stalls are those entered at less than -1.0 g. Inverted accelerated stalls are rarely seen due to difficulties in pilot technique, pilot discomfort, and control effectiveness limitations which usually restrict or prevent these maneuvers.

3.2.3 Post-Stall Gyrations

Post-stall gyrations are maneuvers entered after stalls which are different from spins. These gyrations are often extremely violent and result in random pitch, roll, and yaw oscillations. In many cases, the characteristics of post-stall gyrations are determined by the steady state spin speed of an airplane. The steady state spin speed is that speed attained by the airplane in a steady state spin (defined below). Post-stall gyrations which occur at airspeeds faster than steady state spin speeds are those which normally occur after accelerated stalls. In these entries, the post-stall gyrations acts to dissipate the kinetic energy prior to the airplane entering an incipient spin. Post-stall gyrations occurring below steady state spin speeds are normally associated with nose high, low airspeed conditions during which inertia forces become more powerful than aerodynamic forces. Low speed post-stall gyrations are usually more unpredictable and cause the pilot the most concern due to the fact that his aerodynamic controls are less powerful than the inertia forces acting on the airplane. Erratic angle of attack and random, unpredictable airplane motion are the most pronounced characteristics of a post-stall gyration.

3.2.4 Spin Definition

The spin is a maneuver during which the airplane descends rapidly toward the earth in a helical movement about a vertical axis (called the spin axis) at an angle of attack between the stall and 90 degrees. The steady state spin is always characterized by autorotation (defined below). Spins are of two distinct types, erect and inverted. Erect spins differ from inverted spins in the sign of the angle of attack and load factor; that is, in an erect spin, there is a positive angle of attack and load factor, whereas in an inverted spin, there is a negative angle of attack and negative load factor.

Each spin is divided into two phases; incipient and steady state. The incipient phase of the spin is that portion of a spin occurring after post-stall gyrations, if any, have ceased and the airplane commences a spin-like motion; however, the aerodynamic and inertia forces have not yet achieved a balance. In a steady state, or fully developed spin, the aerodynamic and inertia forces have reached a balance. The pitch attitude, angle of attack, vertical velocity, and yaw rate reach constant, average values, or changes in any of these parameters are uniformly repetitive. Some airplanes never reach true steady state or fully developed spins, but attain only partially developed spins. The difference is that in the partially developed spins, stabilization is lacked in one of the parameters listed above. For example, pitch attitude or yaw rate might fluctuate in a random, nonrepetitive fashion.

3.2.5 Factors Causing Spins

Spins are caused by a combination of two primary factors: exceeding stall angle of attack and sideslip. These two factors result in a phenomena known as autorotation. Autorotation is defined as rotation which occurs without lateral control input. It is a result of unequal angle of attack distribution between the wings of the airplane. Figure 3.1 shows a lift coefficient (C_L) angle of attack (α) curve for a typical airplane.

At angles of attack lower than the stall (Point A), any change in angle of attack between the wings due to sideslip tends to raise the lower wing. This is called dihedral effect or lateral stability. Once the stall angle of attack is exceeded (Point B), any sideslip which induces an apparent change in angle of attack between the wings results in the opposite restoring moments and causes the airplane to rotate in the direction of the low wing. This rotation in turn increases the angle of attack difference between the two wings and the maneuver becomes self-sustaining. The drag on the downgoing wing also becomes greater due to the increased angle of attack and in turn causes yawing moments in the direction of rotation. The motions in roll, yaw, and pitch are opposed by, or coupled with, inertia moments until eventually a balance of forces and equilibrium is achieved. Figure 3.2 shows an example of this sort of aerodynamic and inertia balance.

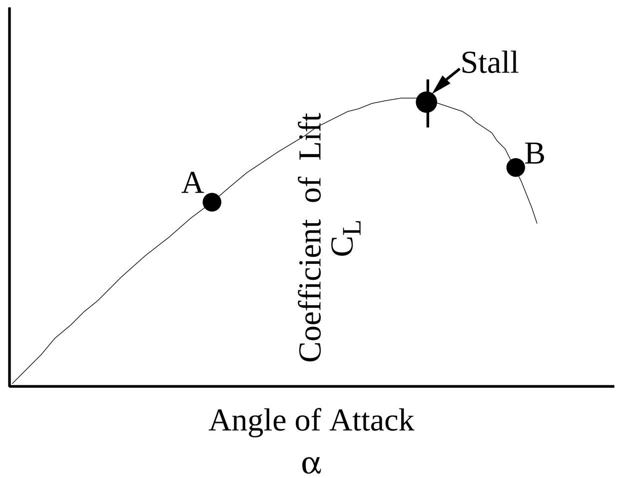


Figure 3.1
Typical Lift Slope Curve

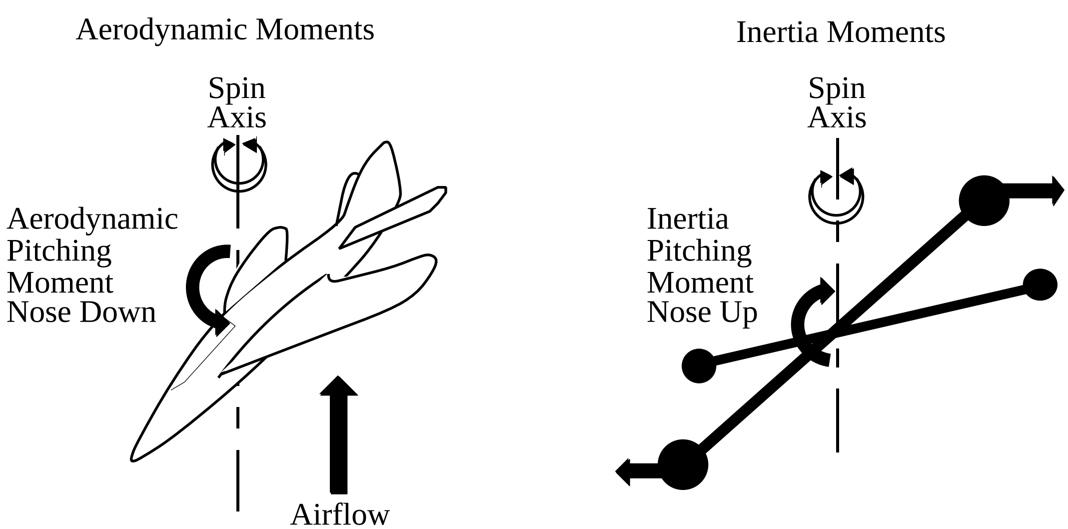


Figure 3.2
Balance of Aerodynamic and Inertia Pitching Moments in a Spin

The airplane mass may be illustrated by fly balls or a series of weights. As this series of weights rotate about the spin axis, a nose-up inertia pitching moment is caused. This moment balances out the aerodynamic nose-down pitching moment. The other axes of motion contain corresponding balances of aerodynamic and inertia moments. A discussion of spin tunnel research on this subject is found in Reference 1.

It is important to realize that the airplane mass distribution has an extremely strong effect on the spin and spin recovery characteristics. This mass distribution is normally discussed in terms of the inertia yawing moment parameter, IYMP. This term is equal to $I_x - I_y \div mb^2$, I_x and I_y being the moments of inertia about the x and y body axis, respectively; m, the mass of the airplane; and b, the wing span. Present trends in modern aircraft usually result in large negative values of inertia yawing moment parameter (i.e., fuselage-heavy airplanes). This is the result of thin wings, high density jet engines, and fuel cells in the fuselage of the airplane. Inertia yawing moment parameters will change greatly in many airplanes by fuel consumption, addition or release of external stores, etc., The effects of mass distribution should be determined prior to actual spin testing and initial tests should be performed in those loadings considered least critical from the inertia yawing moment parameter standpoint. There may well be some loadings in which spin recovery will be impossible. For example, spin recovery in the A-1 was impossible or unacceptable with heavy wing loadings. However, recovery could be accomplished easily when wing stores were jettisoned.

Sufficient spin tunnel data has been accumulated to show strong trends in the capability of various control combinations to recover airplanes versus the magnitude of the inertia yawing moment parameter. In general, airplanes with fuselage-heavy loadings will require use of lateral, as well as directional and longitudinal control, for spin recovery. In airplanes of this type, lateral control often becomes a more powerful spin recovery control than the rudder. Additional information on this theory may be found in References 2 and 3.

3.3 TEST PROCEDURES AND TECHNIQUES

3.3.1 Preliminary Data

Navy spin programs occur only after the contractor has demonstrated satisfactory spin recovery characteristics of the particular airplane involved. During the spin demonstrations, a wealth of important information is obtained which should be fully exploited and utilized by the Navy test pilots. In addition, any areas which are not clear or need further amplification should be discussed with the contractor pilot who flew the spin demonstration. A great deal of information is obtained in spin tunnel evaluations which are performed on scale models of nearly all new airplanes prior to the contractor's spin work. The spin tunnel reports are available to the project pilot and should be studied thoroughly.

It is extremely important for the project pilot to pay particular attention to any changes that are made to the test airplane configuration between the demonstration by the contractor and the Navy spin evaluation. Changes to the demonstration configuration are often made as a result of deficiencies found during NPE and BIS trials. These changes may appear to be entirely unrelated to the spin evaluation but may nevertheless seriously affect the airplane spin characteristics. If, in the opinion of the project pilot, these changes are significant, the contractor should be required to conduct additional spin demonstrations in the most recent configuration. Examples of these may be changes in canopy design, speed brake extension limits, addition of various stores, modification of high lift devices, revision of CG limits, and other obvious changes which would affect the stability and mass distribution of the airplane. Unless the project pilot is positive that the changes are inconsequential, additional demonstrations in the modified configurations should be required.

3.3.2 Test Instrumentation

Having a properly instrumented airplane is of maximum importance in spin testing due to the rapidity with which parameters change. External and internal photography will also be extremely useful in both analyzing the spin characteristics and in subsequent training of squadron pilots.

Primary internal instrumentation may include magnetic tape, oscillograph, and photopanel. Pertinent parameters include control positions and forces, angle of attack, sideslip, airplane attitudes, rates, and so one. One unique parameter for spin tests is turn count or azimuth angle. Measurement of this parameter requires installation of a turn count gyro or photoelectric cell. A list of typical spin test instrumentation parameters is presented in Figure 3.3.

The test airplane cockpit should be instrumented to provide the pilot with controls for activation of data records and primary and secondary emergency antispin device actuation. In addition, there are several devices which may be installed to provide the pilot with orientation cues and warning signals. These devices include oversized, centered, turn needle (or roll and yaw lights), single pointer altimeter (such as a glare shield mounted cabin pressure altimeter hooked to the regular static source), low altitude warning lights and aural warnings, and direct readouts of any parameters considered critical. The pilot should be provided with a kneeboard or cockpit mounted tape recorder or telemetry voice recorder channel. The recording device permits the pilot to make a running commentary of the spin as it progresses through its various phases. Because so much is happening in a short period of time, the pilot is not usually able to write down all his observations and comments. The recorder is particularly valuable for mission suitability observations.

Internal motion picture or pilot's-eye cameras and externally mounted cameras should record the relative motion of the outside world. These films are useful in reconstructing airplane motions and the relative violence of the maneuvers in the cockpit.

External instrumentation usually includes telemetry, photo theodolite, and chase plane film/TV coverage. Real-time telemetry will permit a project engineer to monitor various critical parameters such as angle of attack, engine operation, altitude, turn direction, control position, etc. A ground-based safety pilot may usefully be employed in the telemetry receiving station, linked to the test pilot by duplex (two-way) continuous voice radio communications.

Motion picture films are, of course, very useful in showing the real life sequence of the spin and provide the capability of slow motion analysis of airplane motions. The films may subsequently be used along with cockpit films for spin training films and other presentations.

Quantities Measured			
	Mag Tape/ Oscillograph	Photo Panel	Pilot's Panel
Film Frame Counter	X	X	X
Oscillograph Burst Counter	X	X	X
Pilot Signal	X	X	
Running Time	X		
Longitudinal Stick Force	X		
Lateral Stick Force	X		
Left and Right Rudder Pedal Force	X		
Longitudinal Stick Position	X		
Lateral Stick Position	X		
Rudder Pedal Position	X		
Left Elevator Position	X		
Left Aileron Position	X		
Rudder Position	X		
Left and Right Elevator Trim Tab Positions	X		
Rudder Trim Tab Position	X		
Normal Acceleration	X		
Pilot's Seat Acceleration	X		
Angle of Attack	X		
Angle of Pitch	X		
Angle of Bank	X		
Angle of Sideslip	X		
Rate of Pitch	X		
Rate of Roll	X		
Rate of Yaw	X		
Spin Turn Count	X		
Left and Right Engine Oil Pressure		X	
Left and Right Engine Fuel Used		X	X
Noseboom Airspeed		X	X
Noseboom Altitude		X	X
Production Airspeed		X	
Production Altitude		X	
Critical Structural Loads	X		

Figure 3.3
Typical Test Instrumentation

3.3.3 Chase Plane Requirements

A chase plane is mandatory on all spin flights in a Navy spin program. Consideration should be given to assure that the chase plane is compatible with the test airplane. Using a chase airplane that has large disparities in performance with the test airplane can result in unnecessary flight delays waiting for the chase plane to get in position. On the other hand, a chase airplane with inferior low speed handling qualities (to the test airplane) may result in inability to closely monitor the test airplane in slow speed flight and in the spin. The chase pilot is used to maintain surveillance of the test area, count spin turns, and act as a safety backup by monitoring altitude and inspecting the test airplane at frequent intervals for any external signs of damage or stress.

3.3.4 Anti-Spin Devices

Each airplane to be utilized in a Navy spin program must be fitted with an appropriate anti-spin device. This is an emergency device utilized by the pilot when aerodynamic controls are ineffective in recovering the airplane from the spin. Most commonly used is an anti-spin parachute which is deployed behind the airplane to slow the yaw rate, lower angle of attack, and thereby recover from the spin. Anti-spin parachutes are of many types and sizes and the requirements for a particular chute are usually predicted on spin tunnel research. Chutes may be deployed and opened ballistically in certain cases. Other devices utilized are anti-spin rockets which can be used either as anti-yaw devices or as pitch devices to lower the angle of attack and subsequently stop a spin. Another possible consideration is use of vectored thrust as a method of lowering angle of attack and breaking a spin. In any event, the anti-spin device decided upon should be demonstrated by the contractor during the spin demonstration. The device should be tested on the ground in flight prior to the commencement of the spin tests.

3.3.5 Spin Flight Testing

The evaluation of deep normal and accelerated stall characteristics should be the start of the spin program. In some cases, this will be the initial Navy evaluation of deep stalls. This will occur whenever an initial investigation of stall characteristics indicates that the airplane has strong pro-spin tendencies in deep stall penetrations. In any spin program, however, the buildup program for the spins should start with a thorough investigation of deep stall characteristics. It is important for the test pilot to remember that various criteria other than the actual aerodynamic stall may have been used in previous evaluations to define stall speeds. In these cases, the criterion will be some limiting factor based on the flying qualities, performance stall speeds, or carrier suitability minimum usable speeds. It is quite possible that true aerodynamic stalls may not have been fully investigated in previous tests. The deep stall penetrations should, therefore, proceed in a logical buildup sequence starting with normal stalls at high altitude using low power settings.

It is important to keep the power settings low initially so that the nose of the airplane will be as low as possible at the stall. Thus, when the airplane stalls, the best possible conditions for regaining flying speed will exist because the nose will already be low. The pilot should build up to full control deflection in the stall with both lateral and directional controls. In many cases, these large control inputs will induce post-stall gyrations or incipient phase spins. The gyrations and spins produced in this fashion should not be permitted to build up to steady state conditions but should be recovered from immediately.

The incipient phase spin testing should be commenced from one g, power off stalls, in the loading which has been predicted to be the least critical insofar as center of gravity and IYMP are concerned. Intentional spins are usually entered by application of full pro-spin control deflections at or after the stall. Specific entry procedures are described in the military spin specification, Reference 4. Entries and types of spins may be modified by the detail specification for the airplane. It is important to emphasize that the majority of testing should be concerned with recovering from post-stall gyrations and the incipient phase spin characteristics. These regions of uncontrolled flight are those which the fleet pilot will see most frequently in tactical use.

The test pilot should build up to steady state spins very slowly. A good schedule for a buildup would be to initially look at spins for 1/2 turn, 1 turn, 1 1/2 turns, and so on, until steady state spins have been attained. Once the steady state spin has been attained and the pilot has ascertained that he can recover consistently from the spin, the evaluation should proceed to investigate the variables of configuration and control changes. Included here are the effects of control positions and configuration variations such as speed brakes, power, and flaps - both in the spin and for recovery. Since lateral control position may be a powerful variant, the inputs should be made in an incremental buildup.

After a thorough and complete one g entry evaluation in the least critical loading, the evaluation should move on to entries from accelerated stalls, then into vertical entries. A gradual buildup in load factor and pitch attitude, respectively, is again warranted. Occasionally, inverted spins may result from nose high entries. The pilot should be aware of this. This point in the evaluation may, in fact, be the optimum place to evaluate inverted spins. Following this buildup, tactical entries such as from high g reversals, improper nose high recoveries, and improperly executed aerobatics should be investigated. Again, the emphasis should be placed on defining the capability and requirements to recover the airplane in post-stall gyrations and incipient phase maneuvers. Finally, the effects of inertia coupling at low speeds should be investigated. These maneuvers will probably present the most violent post-stall gyrations and often result in inverted spins even though erect pro-spin controls are utilized. Low speed inertia coupled maneuvers are entered by applying abrupt pro-spin control deflections during low speed rolling maneuvers.

Once the complete spectrum of entries, control positions, and tests have been completed in the least critical loading, the data should be spot-checked in a buildup program at the other loadings concentrating on normal service utilization. It is important to remember that changing the external loading of the airplane may well change the spin and spin recovery characteristics radically. A logical buildup for each new loading is again warranted, especially in the area of asymmetric loads.

3.3.6 Miscellaneous Tests

A complete spin evaluation will require the investigation of several miscellaneous areas. Some of these areas will be unique to only one airplane and the project pilot will be required to use his imagination to insure that he has considered all logical conditions. Some examples are discussed below.

3.3.6.1 POWER EFFECTS AND ENGINE OPERATION

Power effects may be negligible or extensive and engine operation may vary drastically between various power settings. Pro and anti-spin asymmetric power should be investigated on multiengine airplanes. Asymmetric power may be an aid or hindrance to spin recovery. The various combinations of tests and possible ramifications should be obvious. In many cases, high angle of attack and/or sideslip will cause erratic engine performance, stalls, chugs, and flameouts.

3.3.6.2 CONFIGURATION CHANGES AND NONOPTIMUM RECOVERIES

The project pilot should evaluate the effects of various configuration changes on spin and recovery characteristics. Items here include effects of flaps, speed brakes, and so on. Past experience has shown that actuation of these items may either prevent or produce spin recovery. In many cases, they will have little or no effect. In any event, it is important to know these effects and the evaluation should not overlook this important area of investigation.

Nonoptimum recovery variations should also be investigated in detail. This will assist in determining critical recovery parameters. Nonoptimum recovery variations include utilizing less than full recovery control deflections, various combinations of recovery controls, control releases, control neutralization, etc. The timing of anti-spin control application should also be investigated. For example, application of steady state spin recovery controls in a post-stall gyration or incipient phase spin may actually act as pro-spin controls. These data are obviously of importance to the fleet pilot.

3.3.6.3 DEGRADED SYSTEMS OPERATIONS

Insofar as logical and feasible, the project pilot should investigate spin recoveries under conditions of degraded systems operations. Loss of flight control boost, stability augmentation, and so on, could produce significant variations in spin recovery capability. Spin maneuvers may have side effects on systems operations and it is appropriate for the project pilot to comment on the mission suitability aspects involved. Examples include tumbling of attitude gyros, illumination of various warning lights, loss of fuel through venting, adequacy of pilot restraint system, and so on.

3.3.7 Required Data

There is myriad of data pertinent to any spin program. These data are normally presented in the report as time histories of various important rates, positions, control deflections and forces, altitude, etc. Qualitative data presented by the pilot, however, are the most important data presented. The description of how it feels in the cockpit, the ability of the pilot to stay orientated in the spin recovery, and so on, are the most important portions of the evaluation from the pilot's point of view. For example, use of angle of attack for dive pull out following spin recovery may be extremely critical. If this is true, it is important for the project pilot to define the limitation in terms of cues available. Use of a cockpit tape recorder with which the pilot can make a running commentary of his spin is invaluable for this purpose. Nose position, turn needle position, altitude loss per turn, yaw rate, various recovery techniques, and so one, can be discussed as they occur. Occasionally, various engine parameters may not be instrumented and it will be important for the pilot to observe engine operating characteristics during the spin. It is up to the project pilot to insure that his important cockpit observations are not lost in a maze of quantitative data. A pilot-oriented, qualitative assessment of the spin entry characteristics, the spin characteristics, and the spin recovery characteristics must be foremost. A checklist of some of the important data to obtain in spin testing is presented at the end of this section.

3.3.8 Safety Considerations/Risk Management in Spin Programs

There is probably no other type of flight testing which requires a more comprehensive and logical buildup program than spin evaluations. This buildup program should begin by a complete pilot study of all previous spin data on the plane he will be evaluating, as well as a look at earlier spin reports to observe and look for various problem areas which occurred in these previous evaluations. The project pilot should provide himself with several spin familiarization flights in airplanes cleared for intentional spins.

As stated earlier, prior to commencing a spin, it is necessary for the pilot to do stall work which logically and reasonably proceeds in a planned buildup to a complete spin evaluation of the airplane in the least critical loading. Following this initial series of tests, additional loadings may be evaluated in a reasonable buildup program.

Prior to commencement of actual spin tests, the project pilot should devote some flights to dive pull-out data at various airspeeds, angles of attack, and power settings with and without speed brake. A dive pull-out table should then be prepared and utilized by the project pilots. From this dive pull-out table, decision altitudes should be established. Decision altitudes should include: altitude at which to stop other than optimum recovery tests, altitude for anti-spin device deployment, ejection or bailout altitude. It is important for the pilot to have these altitudes fixed firmly in his mind prior to doing any spin testing. If a certain critical altitude is reached, the pilot will have a preplanned course of action to follow and will not delay in making the proper decision as to emergency spin recovery actuation or ejection, if necessary.

In many cases, unusual or erratic engine operation may occur during spin testing. It is not unreasonable to expect this because of the extreme high angles of attack, high yaw rates, and sideslip angles associated with spin flights. The pilot must, therefore, be very familiar with procedures for clearing compressor stalls, airstarts, flame-out landing pattern, etc. In some cases, it may be necessary to fit the airplane with some auxiliary power devices. For example, an air driven ram air turbine for emergency electrical power may not operate if deployed in a spin. Therefore, it may be necessary to install a battery to provide for ignition, critical electrical demands, or possibly even run auxiliary hydraulic pumps to provide adequate flight control. If the airplane is prone to engine flameouts in a spin, it may be valuable to fit it with a continuous ignition circuit. In any event, these factors should be considered. The pilot should practice flameout landing procedures, air starts, and various forms of degraded systems flight. It is logical to assume, therefore, that the spin testing should be done near a field to which an emergency or flameout landing can be made quickly.

Use of the chase plane for data and safety purposes was discussed previously. It also serves as a very important extra pair of eyes to maintain surveillance of the spin area and warn the spin pilot of any possible intruders. Additionally, radar coverage may be used to assist in keeping the area clear.

Finally, project pilots of spin airplanes should have a reasonable amount of time in the test airplane prior to commencing any spin test.

3.4 THE INVERTED SPIN

3.4.1 Introduction

Inverted spins have always provided an interesting and frequently spectacular realm of flight; however, it is also a realm of relative unfamiliarity to most pilots. It has been well documented that spins cannot be prevented by a handbook entry that “intentional spins are prohibited.” Also, inverted spins cannot be prevented by handbook entries that “the airplane resists inverted spins.” Someone will always find a way to end up inverted in uncontrolled flight. Because of this, spin testing, including inverted spins, will always remain an important part of the tests programs for new tactical airplanes.

The same general rules that apply for upright spins also apply when flight testing inverted spins. However, a few other considerations should be taken into account. The first area to consider is the disorientation that occurs to most pilots when initially exposed to inverted spins. In some airplanes it may be difficult to tell whether the spin is upright or inverted, particularly if significant pitch oscillations are superimposed on the spin (yaw) motion. The direction of the spin may also be difficult to determine. Specifically, this may be caused by the fact that the airplane rolls in the opposite direction from the spin; i.e., the airplane rolls left when in a right inverted spin. In an erect spin, if the roll rate is not oscillatory, the spin direction and roll direction are the same.

This problem of disorientation can be reduced or eliminated by several means. A solid buildup program in a spin trainer is essential before spinning a new airplane. The number of flights required to become acclimated to the spinning flight regime, and the inverted regime in particular, will vary with the experience and ability of the test pilot. The cockpit instrumentation of the tests airplane is also important and should include a turn or turn rate indicator. This instrument is similar in function to the turn needle in the turn and slip indicator. In every case, whether erect or inverted, the direction of the spin is indicated by the turn needle. Additionally, the pilot should be able, after the proper buildup, to determine the spin direction by the movement of the nose across the ground. The airplane should also be equipped with an angle of attack indicator. In the standard Navy AOA systems, the indicator will be pegged at zero during an inverted spin. In an upright spin, the indicator will be pegged at 30 units. If the airplane is equipped with a flight test sensitive angle of attack indicator, the readings will vary, depending upon the measurable range of the system.

The standard pilot restraint systems in most Navy airplanes are totally inadequate in the inverted flight/spin regimes. Additional means must be supplied to hold the pilot in the seat during negative g flight conditions. Besides being uncomfortable when not properly restrained, full rudder throw can be denied the pilot if he is not held in his seat and is allowed to float to the top of the cockpit. The usual method to provide adequate restraint is to install an additional lap belt or negative g strap which is attached either to the seat or to the seat pan.

Under negative or zero g conditions, many aircraft systems become degraded or are severely limited and this must be taken into account during inverted spin testing. Typically, oil pressure on many jet engines goes to zero during inverted spins. In addition, most airplanes have negative and zero g time limits due to the limited fuel tank capacity. During violent maneuvers sometimes encountered during spins, gyros may tumble and present misleading information to the pilot and the negative g structural limits of many airplanes can be easily exceeded. These limitations should be taken into account and approached cautiously through an appropriate buildup program.

3.4.2 Inverted Spin Entry Techniques

3.4.2.1 CONVENTIONAL ENTRY

The conventional method of entering an inverted spin consists of stalling the airplane inverted and applying pro-spin controls. If the inverted stall cannot be achieved due to inadequate elevator effectiveness, pro-spin controls are applied as the nose begins to fall through with full forward stick. Pulsing the rudders back and forth during deceleration may be somewhat effective in aggravating the yaw at spin entry. This conventional entry involves primarily the use of aerodynamic forces to enter the spin, although some inertial effects may also be present. In the case of the elevator-limited airplane, an inverted spiral vice the inverted spin may result and some airplanes simply will not spin inverted using this entry technique.

Recovery controls will vary for different airplanes but will always include full rudder opposite to the spin direction (opposite to turn needle deflection). Unlike erect spins in many of our current airplanes in which the rudder is only marginally effective in spin recovery, the rudder in an inverted spin will (except possible for a few T-tail types) be highly effective since it is in “clean airflow;” i.e., undisturbed by the wing, fuselage, and horizontal tail. This is illustrated in Figure 3.4. However, this should not be misinterpreted to infer that any airplane can be recovered from an inverted spin by use of opposite rudder only. Anti-spin aileron is required in many high inertia types.

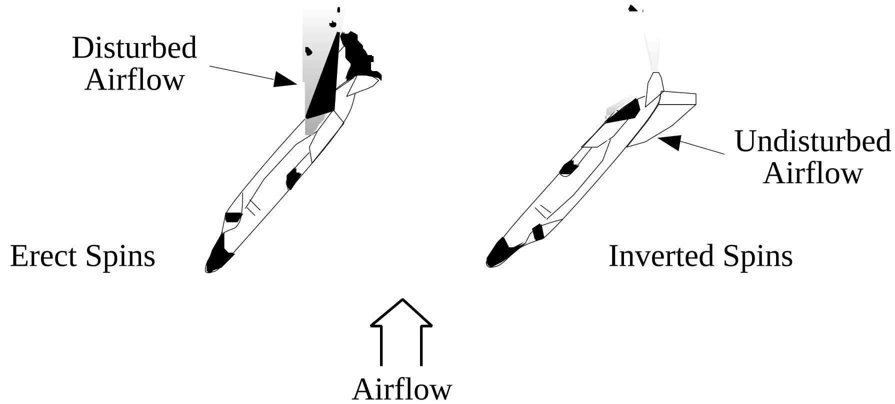


Figure 3.4
Airflow Disturbance in Erect/Inverted Spins

3.4.2.2 ROLL COUPLING ENTRY

Some airplanes are elevator limited in inverted flight and attempts to spin them inverted using conventional entry techniques often meet with failure; i.e., inverted spiral vice the inverted spin. However, the same airplane might spin inverted very readily using a rolling entry executed in a manner to take advantage of inertial coupling moments in pitch.

Use of inertia characteristics for spin recovery in high performance jet airplanes has been commonplace for years. In fact, most supersonic and many subsonic airplanes will not recover from a steady state spin unless recovery inertia moments are generated in yaw to augment weak aerodynamic yawing moments produced by the rudder. It logically follows that inertia coupling can be used for spin entry. For elevator-limited airplanes, the desired coupling moment is usually in pitch to compensate for limited inverted flight elevator effectiveness. The simplified equation of motion in pitch is:

$$\dot{q} = \frac{M}{I_y} + pr\frac{\left(I_z - I_x\right)}{\left(I_y\right)}$$

Where:

$\dot{q}$ = pitch acceleration

p = roll rate

r = yaw rate

q = pitch rate

M = aerodynamic pitching moment

I_x = moment of inertia in roll

I_y = moment of inertia in pitch

I_z = moment of inertia in yaw

Analysis of the inertia term, pr $\frac{(I_z - I_x)}{(I_y)}$, in the equation reveals that roll rate opposite to yaw rate (opposite signs) produces a negative or nose down pitching moment since $I_z - I_x$ is always positive. This pitching moment then is in the desired direction for an inverted spin entry. Now it can readily be seen that if the spin is entered with some roll momentum opposite to the direction of yaw, or spin direction, nose down (negative) aerodynamic pitching moments, as viewed from the cockpit, are augmented with inertial coupling effects.

It is important that the roll be made opposite to the direction of the intended spin; i.e., right roll for a left spin and left roll for a right spin. Roll in the direction of spin will have the opposite effect and produce pitching moments in the wrong direction.

The roll-coupled entry can be exaggerated by increasing the roll rate and creating as much roll inertia as possible. This roll inertia will then be dissipated in the post-stall gyrations. These post-stall gyrations may be quite mild or they can be totally spectacular and include pitching, rolling, and yawing to such a degree as to be indescribable. Needless to say, a suitable buildup program must be accomplished prior to any roll-coupled entries. If the controls are maintained in the pro-spin positions, the airplane will probably enter an inverted spin; however, experience has shown that it also might enter an upright spin with the same controls applied.

In summary, spin testing is one of the most interesting and challenging fields of test flying. In order to safely and effectively accomplish the objectives of any spin program, total preparation and a thorough understanding of the principles and test techniques involved is a necessity.

3.5 REFERENCES

1. *AGARD Flight Test Manual*, Volume II, Chapter 8, Parts I and II.

2. MIL-S-25015(ASG), *Military Specification; Spinning Requirements for Airplanes.*

3. NACA TN 1045, *Tail-Design Requirements for Satisfactory Spin Recovery*, of April 1946.

4. NASA Langley Research Center Paper *"Airplane Spinning"* by James S. Bowman.

5. NASA TR R-57, 1960, *Status of Spin Research for Recent Airplane Designs.*

6. NASA TT F 422, *Flight Testing of Aircraft*, Chapter 9.

7. *"Spin Testing United States Navy High Performance Airplanes"* by John A. Nial, NAWCAD Patuxent River, Maryland.

8. *U. S. Air Force Flight Test Center Stability and Control Techniques*, Section 4.

3.6 PILOT CHECKLIST OF SIGNIFICANT SPIN DATA

I. Stall Characteristics

A. Normal Stalls
1. Stall warning
2. Configuration effects
3. Power effects
4. Angle of attack trends
5. Control effects
6. Deep penetration effects
7. Stall recovery (include optimum recovery)

B. Accelerated Stalls (as above)

C. Inverted Stalls (as above)

D. Validity of NATOPS Manual Stall Information

II. Post-Stall Gyrations

A. High Speed (Accelerated Stall) Entries
1. Description of maneuvers
2. Do maneuvers progress to incipient spins or are they "non-spin?"
3. Pilot orientation
4. Recovery and avoidance maneuvers

III. Erect Spins

A. Entries

B. Incipient Phase

1. Description
    a. Turns (duration)
    b. Yaw rates
    c. Alt lost
    d. Recovery
    e. Orientation
    f. Control forces, positions, and effectiveness
    g. angle of attack and turn needle indications

2. Recovery from incipient phase
    a. Recover by letting go?
    b. Recover by neutralizing?
    c. Optimum recovery

C. Steady State Phase
1. Data as above

D. Recovery Phase
1. Recovery Variations
    a. Varying positions of longitudinal, lateral, and rudder controls
    b. Varying power and auxiliary aerodynamic devices
    c. Optimum recovery procedure
        (1) Critical recovery parameters (if any)
        (2) Progressive stall/spin tendencies
    d. Spin recovery capability under simulated IFR conditions

IV. Inverted Spins

A. Data as for Erect Spins and Recoveries

V. Inertial Coupled Maneuvers

A. Susceptibility of Entering

B. Techniques for Entering

C. Recovery and Avoidance Procedures

VI. Spin Avoidance

A. In Normal Stalls

B. In Accelerated Stalls

C. Inverted

D. Nose High Attitudes

VII. Miscellaneous Parameters

A. Power Effects

B. Engine Operation

C. Configuration Effects
1. Speed brakes
2. Flaps
3. Slats
4. Spoilers
5. Etc.

D. Artificial Stabilizer or Damper Effects

E. Structural Integrity

F. Pilot Restraint Provisions

G. Side Effects

1. On attitude gyro
2. On warning and caution systems
3. On radio/ICS fidelity
4. On fuel/vent system

CHAPTER FOUR

LONGITUDINAL FLYING QUALITIES

CHAPTER FOUR

FIGURES

FIGURES

FIGURES

FIGURES

FIGURES

CHAPTER FOUR

EQUATIONS

$$C_{m_{CG}} = C_{m_{ac}} + \frac{X_a}{\bar{c}} C_L + C_{m_{CG_{\substack{Fus\\Nac}}}} - a_t \left(\alpha_w - \varepsilon - i_w + i_t + \tau\delta_e\right) \bar{V}\eta_t$$

eq 4.10 4.14

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

eq 4.11 4.15

$$\frac{d\delta_e}{dC_L} = \frac{-\left(\frac{dC_m}{dC_L}\right)_x}{C_{m_{\delta_e}}}$$

eq 4.12 4.16

$$C_{h_e} = C_{h_{\alpha_t}} \alpha_t + C_{h_{\delta_e}} \delta_e$$

eq 4.13 4.20

$$\delta_{e_{Float}} = -\frac{C_{h_{\alpha t}}}{C_{h_{\delta_e}}} \alpha_t$$

eq 4.14 4.20

$$\frac{dC_{m_{CG}}}{dC_L}_{Free} = \frac{dC_{m_{CG}}}{dC_L}_{Fixed} + C_{m_{\delta_e}} \frac{d\delta_{e_{Float}}}{dC_L}$$

eq 4.15 4.20

$$\frac{dC_{m_{CG}}}{dC_L}_{Free} = X_{CG} - N_0'$$

eq 4.16 4.21

$$\frac{dF_s}{dV_e} = 2K \frac{W}{S} \frac{Ch_{\delta_e}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{Free} \frac{V_e}{V_{e_{Trim}}^2}$$

eq 4.17 4.24

$$\delta_{e_{Float}} = -\frac{C_{h_{\alpha t}}}{C_{h_{\delta_e}}} \alpha_t$$

eq 4.18 4.26

$\frac{dC_m}{dC_L} = \frac{\partial C_m / \partial \alpha}{\partial C_L / \partial \alpha}$	*eq 4.19*	4.35
$\begin{vmatrix} S + D_u & g \\ L_u / u_0 & -S \end{vmatrix} = 0$	*eq 4.20*	4.36
$S^2 + D_u S + g \frac{L_u}{u_0} = 0$	*eq 4.21*	4.37
ω_{n_p} = undamped phugoid frequency $= \sqrt{2}\, \frac{g}{u_0}$	*eq 4.22*	4.37
ζ_p = phugoid damping ratio $= \frac{1}{\sqrt{2}} \frac{C_D}{C_L}$	*eq 4.23*	4.37
ω_p = damped natural frequency $\approx \sqrt{2}\, \frac{g}{u_0}$	*eq 4.24*	4.37
p_p (sec) $= .138\, u_0$ (where u_0 is in feet per sec.)	*eq 4.25*	4.37
$\zeta_p \approx \frac{.707}{L/D}$	*eq 4.26*	4.37
$\dot{\theta}_{\text{pull-up}} = \frac{g\,(n-1)}{V}$	*eq 4.27*	4.87
$\dot{\theta}_{\text{steady level turn}} = \frac{g}{V}\left(n - \frac{1}{n}\right)$	*eq 4.28*	4.87
$M_{CG_{\text{Due to } \dot{\theta}}} = -a_t \frac{\ell_t^2\, \dot{\theta}}{V} q_t S_t$	*eq 4.29*	4.88

$$C_{m_{\dot{\theta}}} = \frac{\partial C_{m_{CG}}}{\partial\left(\frac{\dot{\theta}\bar{c}}{2V}\right)} = -2a_t \ \eta_t \bar{V} \ \frac{\ell_t}{\bar{c}}$$

eq 4.30 4.89

$$\delta_{e_{Pull-Ups}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL} V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} n + \frac{C_{m_{\dot{\theta}}}\rho g \bar{c}}{4 \ W/S} (n - 1)\right\}$$

eq 4.31 4.89

$$\left(\frac{d\delta_e}{dn}\right)_{Pull-Ups} = -\frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL} V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} + \frac{\rho g \bar{c}}{4 \ W/S} C_{m_{\dot{\theta}}}\right\}$$

eq 4.32 4.90

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

eq 4.33 4.90

$$\left(\frac{d\delta_e}{dn}\right)_{Steady\ Turns} = -\frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL} V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} + \frac{C_{m_{\dot{\theta}}}\rho g \bar{c}}{4 \ W/S}\left(n + \frac{1}{n^2}\right)\right\}$$

eq 4.34 4.90

$$F_{s_{Pull-Up}} = K\frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{Free} \left\{\frac{V_e^2}{V_{e_{Trim}}^2} - n\right\} + K\frac{1}{2}\rho \ell_t g(n-1)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.35 4.97

$$F_{s_{Steady\ Turns}} = K\frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{Free} \left\{\frac{V_e^2}{V_{e_{Trim}}^2} - n\right\} + K\frac{1}{2}\rho \ell_t g\left(n - \frac{1}{n}\right)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.36 4.98

$$\left(\frac{dF_s}{dn}\right)_{\text{Pull-Up}} = -K \; \frac{W}{S} \; \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} + K \; \tfrac{1}{2} \; \rho \ell_t g \left\{ C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau} \right\}$$

eq 4.37 4.98

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turn}} = -K \; \frac{W}{S} \; \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} + K \; \tfrac{1}{2} \; \rho \ell_t g \left(1 + \frac{1}{n^2}\right) \left\{ C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau} \right\}$$

eq 4.38 4.98

$$F_s = K_1 \Delta \delta e$$

eq 4.39 4.105

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turns}} = -\frac{K_1}{C_{m_{\delta_e}}} \; \frac{W/S}{\frac{1}{2}\rho_{SSL} \; V_{e_{Trim}}^2} \left\{ \left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot{\theta}}} \rho g \bar{c}}{4 \; W/S} \left(1 - \frac{1}{n^2}\right) \right\}$$

eq 4.40 4.106

$$Fs = K_2 q \Delta \delta_e$$

eq 4.41 4.107

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turns}} = -\frac{K_2 \; W/S}{C_{m_{\delta_e}}} \left\{ \left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot{\theta}}} \rho g \bar{c}}{4 \, W/S} \left(1 - \frac{1}{n^2}\right) \right\}$$

eq 4.42 4.107

$$\begin{vmatrix} S + \frac{L_\alpha}{u_0} & -1 \\ -M_{\dot{\alpha}} S - M_\alpha & S - M_{\dot{\theta}} \end{vmatrix} = 0$$

eq 4.43 4.118

$$S^2 + \left(\frac{L_\alpha}{u_0} - M_{\dot{\theta}} - M_{\dot{\alpha}}\right) S - \left(M_\alpha + \frac{L_\alpha}{u_0} M_{\dot{\theta}}\right) = 0$$

eq 4.44 4.118

$\omega_{n_{sp}}$ = undamped short period frequency

$$= \sqrt{\frac{\frac{1}{2} \; P_a \; M^2}{I_{yy}} \; S\bar{c} \; C_{L_\alpha} \left(\frac{X_{CG}}{\bar{c}} - N_M\right)}$$

eq 4.45 4.119

$$M_{\dot{\alpha}} \doteq 0$$

$$\frac{L_\alpha}{u_0} \doteq -M_{\dot{\theta}}$$ *eq 4.46* 4.119

$$\omega_{sp} = \sqrt{-M_\alpha}$$ *eq 4.47* 4.120

$$\zeta_{sp} = \frac{\sqrt{\frac{\rho S}{2}}}{2\sqrt{-\frac{\bar{c}}{I_{yy}} C_{L_\alpha} \left(\frac{X_{CG}}{\bar{c}} - N_M\right)}} \left\{\frac{C_{L_\alpha}}{w/g} - \frac{C_{m_{\dot{\theta}}} \bar{c}^2}{2I_{yy}} - \frac{C_{m\dot{\alpha}} \bar{c}^2}{2I_{yy}}\right\}$$

eq 4.48 4.121

$$\frac{F_s}{\alpha} = \left(\frac{F_s}{n}\right)\left(\frac{n}{\alpha}\right)$$ *eq 4.49* 4.135

CHAPTER FOUR

LONGITUDINAL FLYING QUALITIES

4.1 INTRODUCTION

The investigation of longitudinal stability and control involves the study of characteristics exhibited in the airplane's plane of symmetry. This plane of symmetry divides the airplane into two essentially symmetrical halves and contains components of motion only along the X and Z axes and about the Y axis (see Figure 4.1).

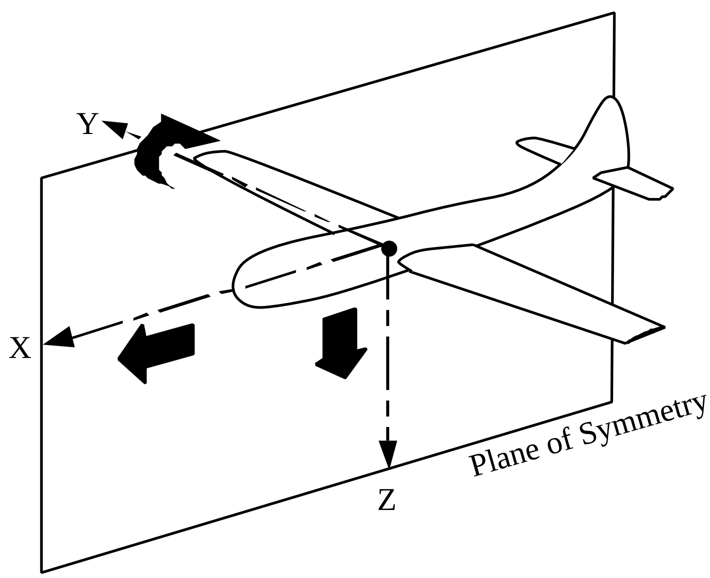


Figure 4.1
Airplane Axis System and Plane of Symmetry

Airplane motion in the plane of symmetry, i.e., longitudinal motion, generally results in insignificant motion in the plane of asymmetry, i.e., lateral and directional motion. (There are important exceptions to the last statement which will be discussed in a subsequent section on coupled motions.) Therefore, longitudinal stability and control can be investigated apart form lateral-directional stability and control.

Longitudinal flying qualities must be investigated from equilibrium and nonequilibrium flight conditions. From equilibrium flight conditions, the static longitudinal stability characteristics may be determined. These characteristics are:

1. Variation of longitudinal control forces and elevator positions with airspeed variations from trim in unaccelerated flight (longitudinal control force and elevator position stability).

2. Variation of longitudinal control forces and elevator positions with normal acceleration at a constant airspeed (longitudinal maneuvering stability, or "stick force per g" and "elevator position per g").

3. Variation of normal acceleration with angle of attack at a constant speed.

In order to change from one equilibrium flight condition to another, the pilot excites two longitudinal modes of motion which are suppressed in equilibrium flight. The characteristics of these modes of motion greatly influence the dynamic longitudinal stability characteristics of the airplane; these characteristics are determined from nonequilibrium flight conditions. The longitudinal modes of motion are called the "airplane short period" and the "long period" or "phugoid" motions. The characteristics of these motions to be investigated are:

1. Frequency or period of the motions.

2. Damping of the motions or lack of it.

The pilot's opinion of longitudinal flying qualities depends on all the static and dynamic longitudinal stability characteristics mentioned above plus the characteristics of the longitudinal control system. Therefore, it is not possible to state conclusively that one or two of the characteristics are overwhelmingly dominant in a particular flight condition. However, it is possible to rationalize that certain characteristics will affect flying qualities more than others in certain circumstances. Therefore, the investigation of longitudinal flying qualities divides nicely into the study of "Nonmaneuvering Tasks" and "Maneuvering Tasks."

Nonmaneuvering tasks are defined as those tasks during which the transition from one equilibrium flight condition to another is accomplished smoothly and gradually. Nonmaneuvering tasks result in essentially unaccelerated flight conditions. Tasks which can be classified as nonmaneuvering are:

1. Take-off
2. Climb
3. Cruise
4. Loiter
5. Glide
6. Descent
7. Approach
8. Wave-off

In general, the pilot's opinion of longitudinal flying qualities during nonmaneuvering tasks is most affected by the characteristics of the longitudinal control system, longitudinal control force and elevator position stability, and the frequency and damping of the long period or "phugoid" mode of motion. (The initial response of the airplane to a longitudinal control input is greatly dependent on the characteristics of the airplane short period motion. However, during the study of the nonmaneuvering tasks, the initial response characteristics may be temporarily ignored. The main areas of concern during nonmaneuvering tasks are the long term stability of the airplane and associated airspeed changes between equilibrium flight conditions.)

Maneuvering tasks are defined as those tasks which result in accelerated flight conditions; during maneuvering tasks, transitions from one equilibrium flight condition to another are made quickly, and possibly, somewhat roughly. Tasks which may be included in this category are:

1. Air-to-air combat

2. Ground attack

3. Reconnaissance

4. Low altitude terrain-following and avoidance

5. In-flight refueling

In general, the pilot's opinion of longitudinal flying qualities during maneuvering tasks is most affected by the characteristics of the longitudinal control system, longitudinal maneuvering stability, variation of normal acceleration with angle of attack, and frequency and damping of the airplane short period motion. The main areas of concern during maneuvering tasks are the initial response of the airplane to a longitudinal control input (short term characteristics) and associated normal acceleration changes between equilibrium flight conditions.

The total mission of any airplane will require the pilot to perform some combination of maneuvering and nonmaneuvering tasks. The various tasks required for mission accomplishment must be determined in order to establish the scope of the longitudinal flying qualities investigation. Since mission accomplishment for all airplanes requires numerous nonmaneuvering tasks, the investigation of the longitudinal flying qualities during these tasks will comprise a large part of any test program. Maneuvering tasks are not so universally required in all missions; therefore, the longitudinal flying qualities during these tasks will be rigorously investigated in some airplanes and less stringently investigated in others.

The provision of satisfactory longitudinal stability and control characteristic is probably the single most important duty of the stability and control design engineer. The pilot exerts a majority of his effort to controlling the longitudinal modes of motion. When an optimum blend of longitudinal stability and controllability is provided, the pilot finds the airplane easy and pleasant to fly. This allows the performance of mission tasks with simplicity and precision, thus enhancing overall mission effectiveness.

4.2 THEORY - NONMANEUVERING TASKS

4.2.1 Static Longitudinal Stability and Control in Unaccelerated Flight

For simplicity, the concepts of static longitudinal stability will first be presented for an airplane in gliding flight (power off) with no propeller and with the longitudinal control surface rigidly restrained in one position. Later, the effects of power and freedom of movement of the longitudinal control surface will be introduced.

4.2.1.1 STICK-FIXED OR ELEVATOR-FIXED LONGITUDINAL STABILITY

The variation of static pitching moments about the airplane's center of gravity with lift coefficient is the principal measure of the airplane's static longitudinal stability. The manner in which the total pitching moment varies with lift coefficient depends on contributions from the wing, fuselage, and nacelles, and the horizontal tail for a given configuration and flight condition. Generally, the contributions of the wing, fuselage, and nacelles is <u>destabilizing</u>; together these components generate nonrestoring pitching moments when changes in lift coefficient occur. If the airplane is to possess static longitudinal stability, the horizontal tail must be designed to overcome the destabilizing influence of the remainder of the airplane's components. The contribution of the horizontal tail to static longitudinal stability is powerful and almost always <u>strongly stabilizing</u>. The design and location of the horizontal tail determine the magnitude of the contribution which is normally expressed through the following nondimensional parameters:

$$\overline{V} = \text{tail volume coefficient} = \frac{S_t}{S_w}\,\frac{\ell_t}{\bar{c}} \qquad \textit{eq 4.1}$$

Where:

S_t = area of horizontal tail in square feet

S_w = area of wing in square feet

ℓ_t = distance from airplane center of gravity to the aerodynamic center of the tail, or “tail arm” length, in feet

$\bar{c}$ = average chord length of wing in feet

Tail Volume Coefficient is a measure of the size and location of the horizontal tail in relation to the size of the wing and center of gravity, respectively.

$$\eta_t = \text{tail efficiency factor} = \frac{q_t}{q} \qquad \textit{eq 4.2}$$

Where:

q_t = dynamic pressure at horizontal tail in pounds per square foot

q = dynamic pressure of free stream prior to encountering the wing and fuselage of the airplane in pounds per square foot

Tail Efficiency Factor is a measure of the change in energy level of the airflow. The dynamic pressure of the airflow at the horizontal tail is reduced because the airflow must first encounter the wing, fuselage, nacelles, and other protrusions prior to reaching the horizontal tail. Lift curve slope of the horizontal tail is denoted as follows:

$$\frac{dC_{L_t}}{d\alpha_t} = a_t \qquad \textit{eq 4. 3}$$

The angle of attack at the horizontal tail will not be the same as the wing angle of attack because of differences in wing and tail incidence, the downwash created by the wing lift production (Figure 4.2).

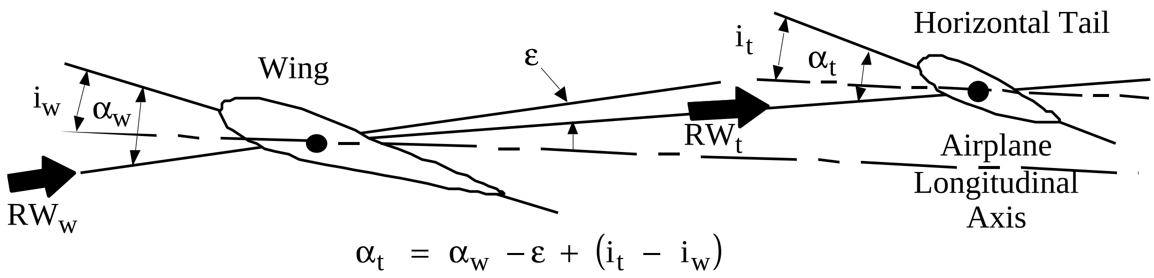


Figure 4.2
Relationship of Wing and Tail Angles of Attack

α_t = angle of attack at horizontal tail

α_w = angle of attack at wing

ε = downwash angle

i_t = incidence of horizontal tail

i_w = incidence of wing

RW_w = relative wind at wing

RW_t = relative wind at tail

Classical relationships of pitching moment coefficient with lift coefficient are shown in Figure 4.3. Several interesting observations may be drawn from a study of this figure. (The sign convention used here is arbitrary; i.e., nose-up pitching moments are assigned positive sign, nose-down pitching moments are negative. Static stability is thus indicated by a negative slope of the $C_{m_{CG}} - C_L$ relationship.)

As previously mentioned, the wing and fuselage contribution to static longitudinal stability is usually destabilizing, while the horizontal tail contribution is usually strongly stabilizing. As shown in Figure 4.3, the complete airplane possesses some degree of static longitudinal stability. The airplane is in trim, i.e., the pitching moments all add up to zero, at the point where the complete airplane curve crosses the horizontal axis. It can be seen that in order to exhibit both static longitudinal stability (negative slope) and a trim condition (cross the horizontal axis), the complete airplane curve must intersect the vertical axis at a positive value of $C_{m_{CG}}$. (It should perhaps be pointed out that although the intercept at $C_L = 0$ is a useful reference point, it does not correspond to a condition that can be achieved in equilibrium flight.)

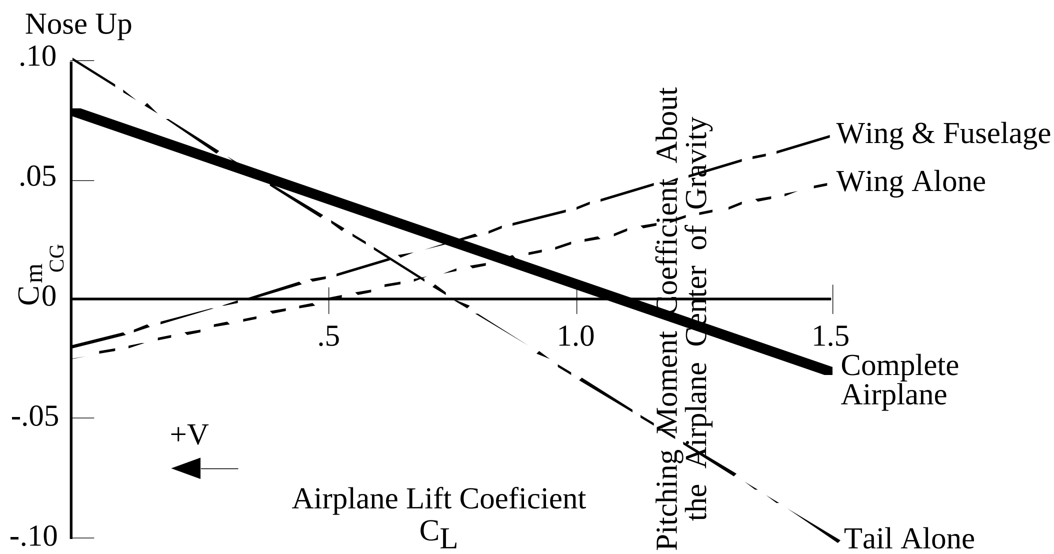


Figure 4.3
Classical Longitudinal Stability Relationships

The longitudinal stability equation which defines the slope of the pitching moment coefficient-lift coefficient relationship may be written as follows for the airplane in gliding flight with fixed controls and no propeller:

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{\text{Airplane}} = \frac{X_a}{\bar{c}} + \left.\frac{dC_m}{dC_L}\right|_{\substack{\text{Fuselage}\\\text{Nacelle}}} - \frac{a_t}{a_w}\,\bar{V}\eta_t\left(1 - \frac{d\varepsilon}{d\alpha}\right) \qquad \textit{eq 4.4}$$

Where:

$\frac{X_a}{\bar{c}}$ = Wing contribution, a measure of the location of the aerodynamic center of the wing in relation to the center of gravity of the airplane (Figure 4.4)

$\left.\frac{dC_m}{dC_L}\right|_{\substack{\text{Fuselage}\\\text{Nacelle}}}$ = Fuselage and nacelle contribution

$-\dfrac{a_t}{a_w}\bar{V}\eta_t\left(1-\dfrac{d\varepsilon}{d\alpha}\right)$	=	Horizontal tail contribution
a_t	=	Lift slope curve of horizontal tail
a_w	=	Lift curve slope of wing
$\bar{V}$	=	Tail volume coefficient
η_t	=	Tail efficiency factor
$\dfrac{d\varepsilon}{d\alpha}$	=	Change in downwash angle with change in wing angle of attack

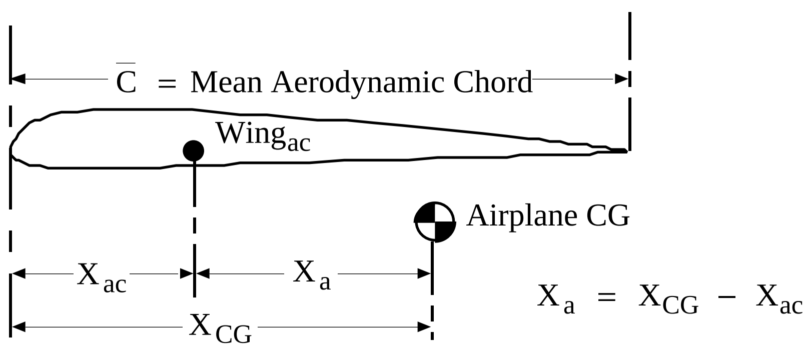


Figure 4.4
Relationship of Wing Aerodynamic Center to Airplane Center of Gravity

For any given airplane configuration, the longitudinal stability equation is fixed except for the wing contribution, which can be changed markedly by movement of the airplane center of gravity (CG). A shift of CG has a very small influence on the tail contribution (through the tail volume coefficient, $\bar{V}$) and negligible influence on the fuselage and nacelle contribution. However, for every percent of the mean aerodynamic chord that the CG is moved aft, $\left(\frac{dC_m}{dC_L}\right)_{Wing}$ increases positively (destabilizing) one percent. Center of gravity movement, therefore, has a powerful influence on the airplane's static longitudinal stability and is probably the single most important variable in static longitudinal stability. The effect of CG shift on the pitching moment coefficient-lift coefficient curve is shown in Figure 4.5. It should be noted that all the curves rotate about a constant pitching moment coefficient at a lift coefficient of zero.

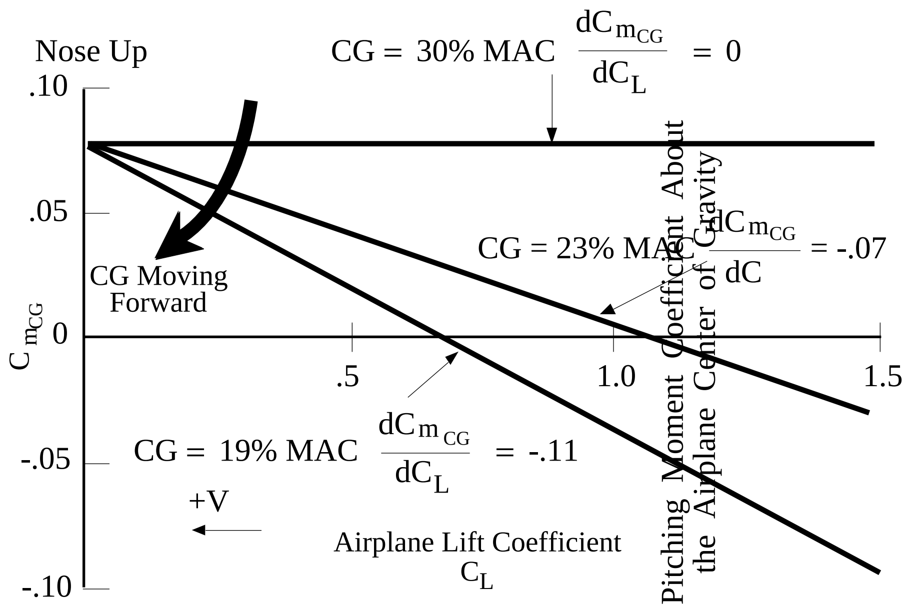


Figure 4.5
Center of Gravity Effects on Static Longitudinal Stability

An examination of Figure 4.5 shows that as the CG is moved aft, the slope of the pitching moment curve becomes more positive; i.e., less stable. A CG position will be reached at which the slope becomes zero. This CG position, at which the airplane exhibits neutral static longitudinal stability with the cockpit control stick or longitudinal control surface fixed, is called the stick-fixed or elevator-fixed neutral point and is denoted by the symbol (N_0). Once the neutral point is known, the slope of the pitching moment curve, i.e., the index of the longitudinal stability of the airplane, can be obtained for any airplane CG position with good accuracy from the following relationship:

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Airplane} = \frac{X_{CG}}{\bar{c}} - N_0 \qquad \textit{eq 4.5}$$

The distance between the actual CG and the neutral point of the airplane, expressed in percentage of mean aerodynamic chord, is called the static margin. (For the case just presented., i.e., the stick-fixed case, the distance would be called the stick-fixed static margin.)

4.2.1.2 LONGITUDINAL CONTROL

The static longitudinal stability presentation has not, to this point, included discussions of the effects for providing a means of longitudinal control nor the effects of power. The scope of the presentation will now be expanded to include longitudinal control; however, power effects will still be neglected for the time being.

For a typical airplane, the curve of pitching moment coefficient versus lift coefficient is shown in Figure 4.6. This airplane, in the condition shown, possesses static longitudinal stability.

However, it is in equilibrium at only one value of lift coefficient (point A). It the pilot wishes to decelerate and fly at a lift coefficient of 1.0 (point B), the airplane must be equipped with some means of overcoming the nose-down pitching $\left(C_{m_{CG}} = -.05\right)$. The problem then is to find a means of changing the lift coefficient for zero pitching moment from point A to point C. The best means of doing so is to merely shift the curve up without changing its slope. (If the slope is changed, the stability of the airplane is changed.)

Obviously, the more stability the airplane possesses, the more powerful must be the means of overcoming the restoring pitching moments. This situation may place a limit on the amount of stability permissible in any airplane.

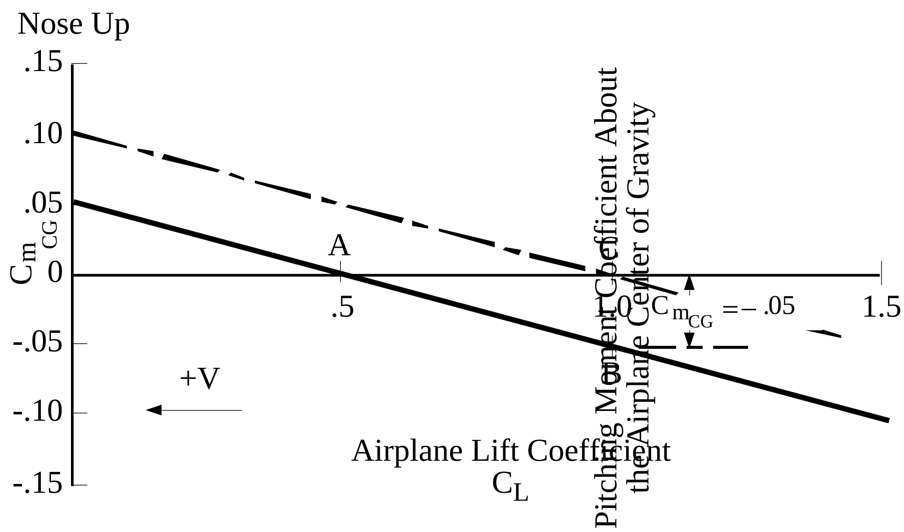


Figure 4.6
Typical Variations of $C_{m_{CG}}$ Versus C_L

To find a means for overcoming the restoring pitching moments, the equilibrium equation of static longitudinal stability is presented (propeller off and power off for simplicity).

$$C_{m_{CG}} = C_{m_{ac}} + \frac{X_a}{\bar{c}} C_L + C_{m_{CG}}{}_{\substack{Fus\\Nac}} - a_t \, \alpha_t \, \eta_t \, \bar{V} \qquad \textit{eq 4.6}$$

The three terms of this equation which might be used to change the lift coefficient for zero pitching moment at $C_{m_{ac}}$, $\frac{X_a}{\bar{c}}$, and α_t. The wing pitching moment about its aerodynamic center $\left(C_{m_{ac}}\right)$ is a function of wing camber and aerodynamic twist of the wing. This moment can be controlled by a flap at the wing trailing edge, a common control used by airplanes without horizontal tails. For several reasons, it is not a practical longitudinal control for airplanes with horizontal tails.

The term $\frac{X_a}{\bar{c}}$ is purely a function of CG position, and the mechanical complexity and <u>change of stability</u> associated with shifting CG positions rules out “CG shifting” as a means of longitudinal control.

The final term to consider is the tail angle of attack, α_t. Tail angle of attack can be changed by utilizing a moveable horizontal tail (sometime called a "slab tail") or by providing a moveable flap on the trailing edge of a fixed horizontal stabilizer. Changing the angle of attack of the horizontal tail can produce large changes in pitching moment without significant changes in longitudinal stability. It is the most powerful and most commonly used means of longitudinal control.

The magnitude of the pitching moment coefficient obtained per degree deflection of the longitudinal control surface is called the <u>longitudinal control power</u> and is written as follows for the "slab tail" and the conventional elevator:

<u>"Slab Tail"</u> $$C_{m_{i_t}} = \frac{dC_{m_{CG}}}{di_t} = -\frac{dC_L}{d\alpha_t}\,\eta_t \bar{V} = -a_t\;\eta_t\;\bar{V} \qquad \textit{eq 4.7}$$

<u>Elevator</u> $$C_{m_{\delta_e}} = \frac{dC_{m_{CG}}}{d\delta_e} = -\frac{dC_L}{d\alpha_t}\,\frac{d\alpha_{t\,EFF}}{d\delta_e}\,\bar{V}\eta_t = -a_t\;\tau\;\eta_t\;\bar{V} \qquad \textit{eq 4.8}$$

Where:

δ_e = elevator deflection from neutral, in degrees, trailing edge up considered negative

$\frac{d\alpha_{t_{EFF}}}{d\delta_e}$ = rate of change of effective tail angle of attack with elevator deflection, sometimes given the symbol τ. It is a function of the ratio of the area of the elevator to the area of the entire horizontal tail; for the "slab tail", τ = 1.0

Elevator control power, $C_{m\delta_e}$ will be used for the remainder of the discussion of static longitudinal stability. The change in equilibrium lift coefficient due to deflecting the elevator may be studied by again referring to the equilibrium equation of static longitudinal stability. The only term affected by the elevator deflection is the tail angle of attack, α_t, which can be broken down in terms of wing angle of attack, downwash angle, tail and wing incidence angle, and change in angle of attack due to elevator deflection:

$$\alpha_t = \alpha_w - \varepsilon - i_w + i_t + \tau\delta_e \qquad \textit{eq 4.9}$$

Thus, power off, propeller off equilibrium equation can be rewritten:

$$C_{m_{CG}} = C_{m_{ac}} + \frac{X_a}{\bar{c}} C_L + C_{m_{CG_{\substack{Fus\\Nac}}}} - a_t \left(\alpha_w - \varepsilon - i_w + i_t + \tau\delta_e\right) \bar{V}\eta_t \qquad \textit{eq 4.10}$$

The control of the equilibrium lift coefficient is obtained through the influence of the term $\tau\delta_e$ of the equilibrium equation. It is important to note that <u>a change in elevator deflection does not change the slope of the pitching moment curve</u> $\left(\frac{dC_{m_{CG}}}{dC_L}\right)$.

An example of the curves of pitching moment coefficient versus lift coefficient for several elevator angles is shown in Figure 4.7. The airplane can now be flown in equilibrium flight at any lift coefficient in the unstalled range by merely <u>changing the elevator position</u>. It should also be noted that, at least for the power off case, <u>elevator deflection has no effect on static longitudinal stability</u>.

The in-flight measurement of pitching moments about the airplane CG for different values of lift coefficient (or airspeed) would be a tedious, if not impossible, undertaking. This measurement can be made to some degree of accuracy in a wind tunnel. However, an in-flight method is needed to determine or estimate the static longitudinal stability of the real airplane. Using the principles already presented, a method can now be developed.

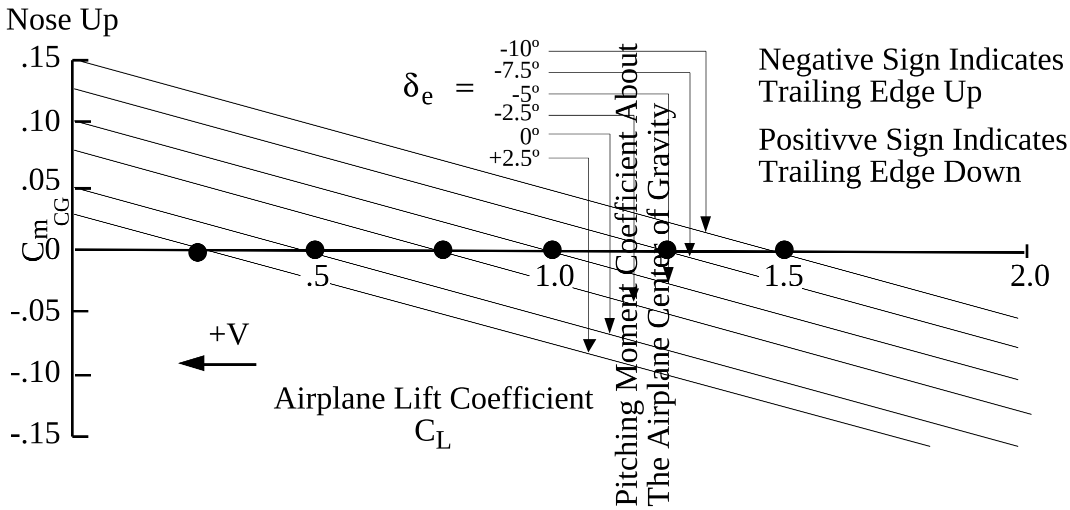


Figure 4.7
Classical Variation of $C_{m_{CG}}$ Versus C_L for Various Elevator Angles

4.2.1.3 ELEVATOR POSITION STABILITY OR LONGITUDINAL CONTROL POSITION STABILITY

First, a cross-plot is made of the elevator angle required for equilibrium versus equilibrium lift coefficient (from Figure 4.7). The cross-plot is presented as Figure 4.8. It must be emphasized that equilibrium conditions, i.e., zero pitching moments about the airplane CG, is represented by every point on the curve of Figure 4.8. (Equilibrium conditions do not necessarily imply, of course, that the airplane is "force trimmed" from the pilot's standpoint.) The elevator position versus lift coefficient curve can be analytically represented by:

$$\delta_e = \delta_{e_{C_L = 0}} - \frac{\left(\frac{dC_m}{dC_L}\right)_x}{C_{m_{\delta_e}}} C_L \qquad \textit{eq. 4.11}$$

where $\delta_{e_{C_L = 0}}$ is the elevator angle required for zero lift coefficient, and is a constant. Although zero airplane lift coefficient cannot be attained in equilibrium flight, the fact that $\delta_{e_{C_L = 0}}$ is a constant is an important aid in the analysis of flight test data, as will be seen later.

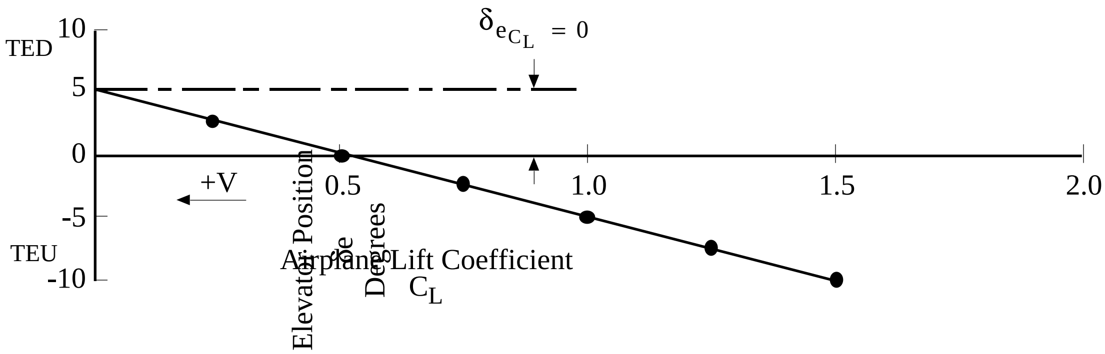


Figure 4.8
Elevator Position for Zero Pitching Moment Versus
Airplane Lift Coefficent

The slope of the curve of Figure 4.8 is obtained by differentiating the last equation with respect to lift coefficient:

$$\frac{d\delta_e}{dC_L} = \frac{-\left(\frac{dC_m}{dC_L}\right)_x}{C_{m_{\delta_e}}} \qquad \textit{eq. 4.12}$$

The elevator position required to vary the equilibrium lift coefficient (or equilibrium airspeed) varies directly with the stick-fixed (or elevator-fixed) static longitudinal stability, $\frac{dC_{m_{CG}}}{dC_L}$, and inversely with the elevator control power, $C_{m_{\delta_e}}$. This relationship of elevator position versus lift coefficient or airspeed in equilibrium flight is often termed elevator position stability or longitudinal control position stability. By measuring this relationship in equilibrium flight, a determination of the sign, but not the degree, of the stick-fixed static longitudinal stability may be made. The degree of stability cannot be determined unless the numerical value of the elevator control power is known. From the last equation, it is seen that, if $\frac{dC_{m_{CG}}}{dC_L} = 0$, i.e., the CG is at the stick-fixed neutral point, the slope of the elevator position versus lift coefficient curve will also be zero (Figure 4.9). This fact will be used later to estimate the airplane's stick-fixed or elevator-fixed neutral point. The neutral point determined from elevator position versus lift coefficient plots is often, more correctly, called the elevator position neutral point or longitudinal control position neutral point.

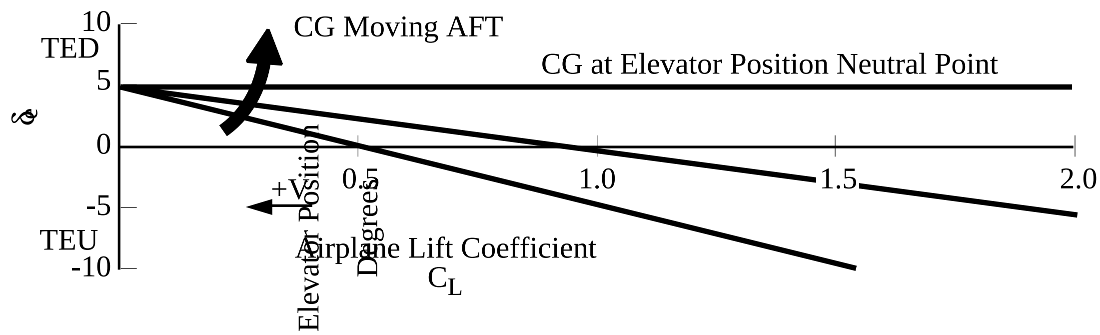


Figure 4.9
Classical CG Effects on Elevator Position Stability

4.2.1.4 STICK-FREE OR ELEVATOR-FREE LONGITUDINAL STABILITY

In the previous discussion, static longitudinal stability was related to the variation of elevator position or longitudinal control position with lift coefficient or airspeed. This variation was shown to be a function of the stability criterion, $\frac{dC_{m_{CG}}}{dC_L}$, <u>with the longitudinal control rigidly restrained in a fixed position</u>. The discussion will now be expanded to include the effects of allowing the longitudinal control surface to “float” in response to some variable in flight conditions. The classical definition of control surface float is “to ride freely in the airstream, changing position in response to pressure distribution over the surface.” The classical definition would apply only to a <u>reversible control system</u>, since the <u>irreversible control system</u> incorporates no control surface response to surface pressure distribution. However, many irreversible control systems incorporate features (stability augmentors) which move a control surface, independent of pilot action, in response to dynamic pressure, normal acceleration, angular rates, or various other flight variables. This movement of control surfaces in irreversible systems can also be thought of as control surface “float.” At any rate, freeing the longitudinal control surface, i.e., allowing it to respond to some flight variable, may have profound effects on the static longitudinal stability of airplanes equipped with either reversible or irreversible longitudinal control systems.

4.2.1.5 STICK-FREE STATIC LONGITUDINAL STABILITY - REVERSIBLE CONTROL SYSTEM

If the airplane is equipped with a reversible longitudinal control system, the longitudinal control surface may float with or against the relative wind at the horizontal tail. The direction and degree of float will depend upon the pressure distribution over the control surface and the hinge moments created at the control surface hinge line by the pressure distribution. The pressure distribution, and therefore the hinge moments, are governed by two major variables - the angle of attack of the horizontal tail and the deflection of the elevator with respect to the horizontal tail.

If the elevator is uncambered and hinged at its leading edge, the variation of hinge moment with horizontal tail angle of attack for zero elevator deflection will be as shown in Figure 4.10. As angle of attack is increased positively, the pressure distribution creates a hinge moment which tends to make the elevator float up.

Now, if horizontal tail angle of attack is maintained at zero and the elevator is deflected, hinge moments will be created as shown in Figure 4.11. As elevator deflection is changed from neutral, the pressure distribution generates a hinge moment which tends to restore the original elevator position.

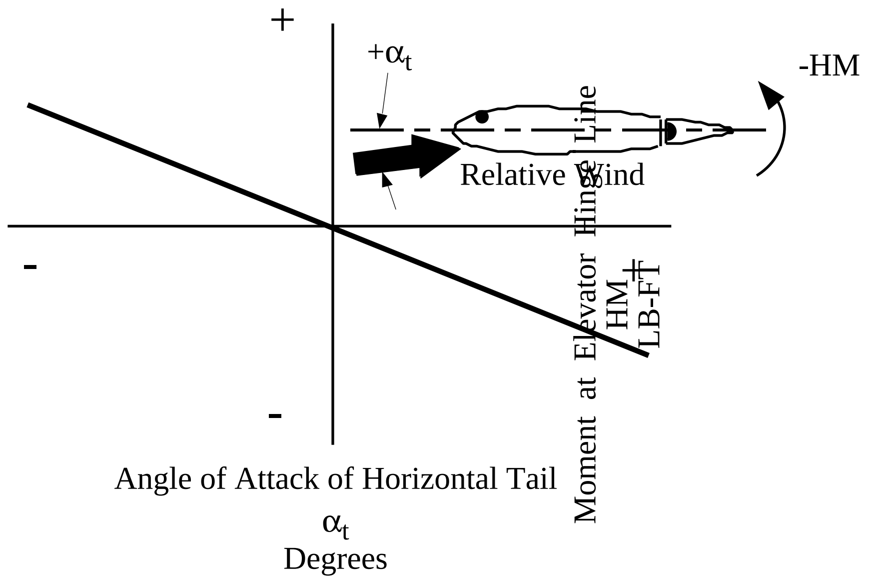


Figure 4.10
Hinge Moment Variation with Angle of Attack

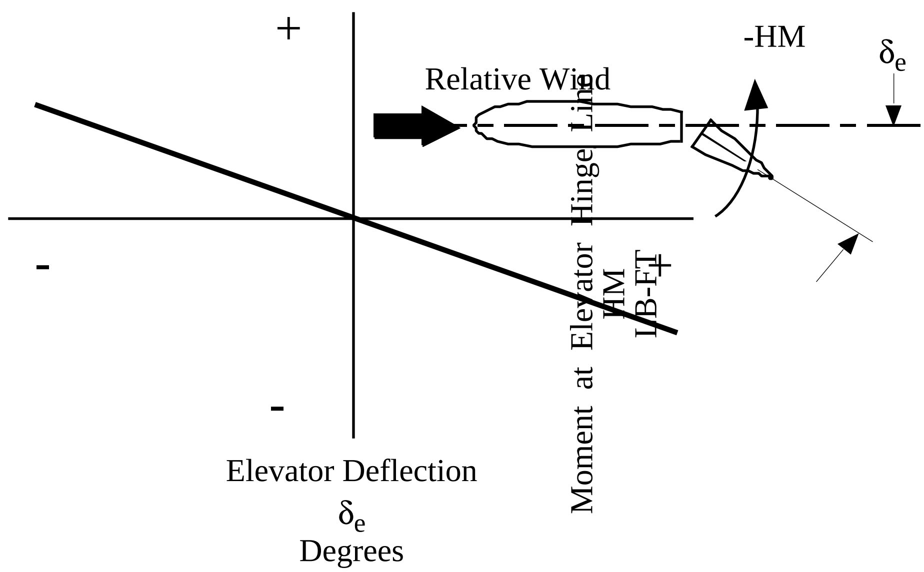


Figure 4.11
Hinge Moment Variation with Control Deflection

The total hinge moment, HM, is obtained by the addition of the two effects noted above. In coefficient form, this relationship may be expressed as follows:

$$C_{h_e} = C_{h_{\alpha_t}} \alpha_t + C_{h_{\delta_e}} \delta_e \qquad \textit{eq. 4.13}$$

Where:

C_{h_e} = total hinge moment coefficient, elevator

$C_{h_{\alpha_t}}$ = hinge moment coefficient variation with angle of attack at zero elevator deflection, normally carries a negative sign

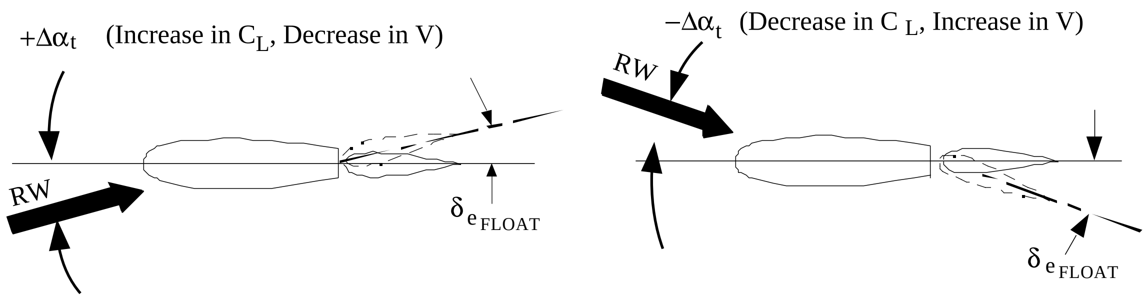
$C_{h_{\delta_e}}$ = hinge moment coefficient variation with elevator deflection at zero angle of attack, almost invariably carries a negative sign

When the total hinge moment coefficient is zero, an equilibrium condition is attained where the "floating tendency," $C_{h_{\alpha_t}}$, is just opposed by the "restoring tendency," $C_{h_{\delta_e}}$. When this equilibrium condition is attained, the elevator angle is called the "float angle" and may be expressed analytically as follows:

$$\delta_{e_{Float}} = -\frac{C_{h_{\alpha_t}}}{C_{h_{\delta_e}}} \alpha_t \qquad \textit{eq. 4.14}$$

If $C_{h_{\alpha_t}}$ and $C_{h_{\delta_e}}$ both have negative signs, the elevator will float up as angle of attack increases positively and float down as angle of attack increases negatively. <u>This effect reduces the static longitudinal stability of the airplane</u> (Figure 4.12). Analytically, the relationship between static longitudinal stability with stick- or elevator-free and stick- or elevator-fixed may be expressed as follows:

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Free} = \left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Fixed} + C_{m_{\delta_e}} \frac{d\delta_{e_{Float}}}{dC_L} \qquad \textit{eq. 4.15}$$



When the elevator is unrestrained, and floats so as to align itself with the relative wind, the float phenomenon reduces the stabilizing pitching moments generated by the horizontal tail. This effect reduces the static longitudinal stability of the airplane, i.e., stick-free stability is less than stick-fixed stability.

Figure 4.12
Effect of Elevator Float on Stabilizing Influence of Horizontal Tail

If the elevator control power coefficient $\left(C_{m_{\delta_e}}\right)$ and change in elevator float angle with change in lift coefficient $\left(\frac{d\,\delta_{e_{Float}}}{dC_L}\right)$ assume their normal sign (negative), it can readily be seen that stick-free static longitudinal stability will be less than stick-fixed static longitudinal stability (Figure 4.13).

Center-of-gravity movement has the same profound effect on stick-free stability as it had on stick-fixed stability. As the CG is moved aft, stick-free stability is reduced. If the CG is moved far enough aft, the slope of the pitching moment-lift coefficient curve becomes zero with the stick or elevator free. This CG position, at which the airplane exhibits neutral static longitudinal stability with the elevator free to float, is called the stick-free or elevator-free neutral point, and is denoted by the symbol (N_0'). Because the effect of elevator float on static longitudinal stability is generally destabilizing, the stick-free neutral point is usually forward of the stick-fixed neutral point. Once the stick-free neutral point is known, the stick-free static longitudinal stability can be obtained for any airplane CG position with good accuracy from the following relationship:

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Free} = X_{CG} - N_0' \qquad \textit{eq 4.16}$$

The distance between the actual CG of the airplane and the stick-free neutral point is called the stick-free static margin.

It is obvious that an in-flight method of measuring or estimating the stick-free static longitudinal stability is needed. A method can now be developed to estimate this important characteristic.

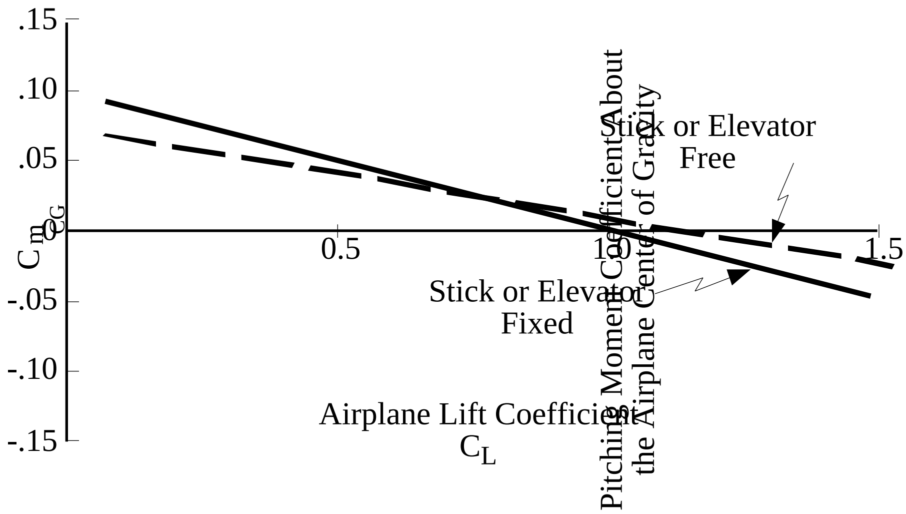


Figure 4.13
Typical Reduction of Static Longitudinal Stability Due to Freeing Elevator

4.2.1.6 STICK-FORCE OR LONGITUDINAL CONTROL FORCE STABILITY

It is convenient again to study curves of stick-fixed and stick-free static longitudinal stability (Figure 4.14).

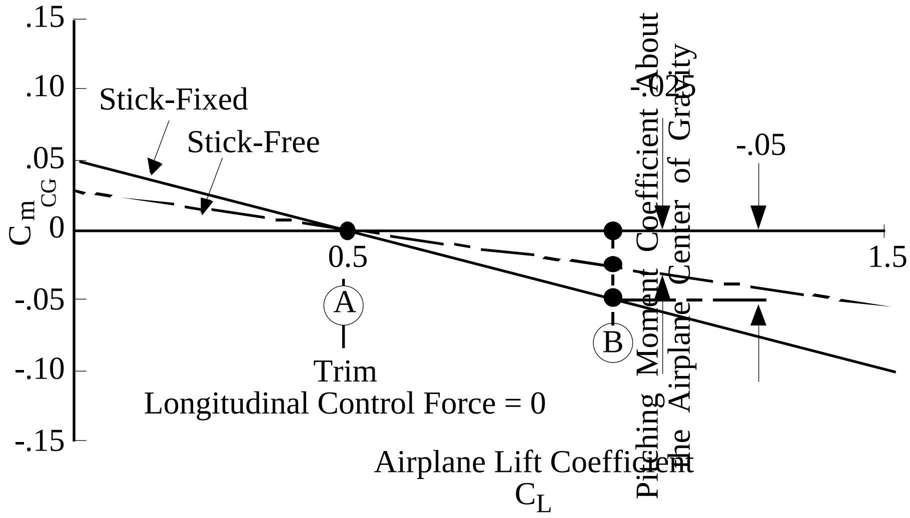
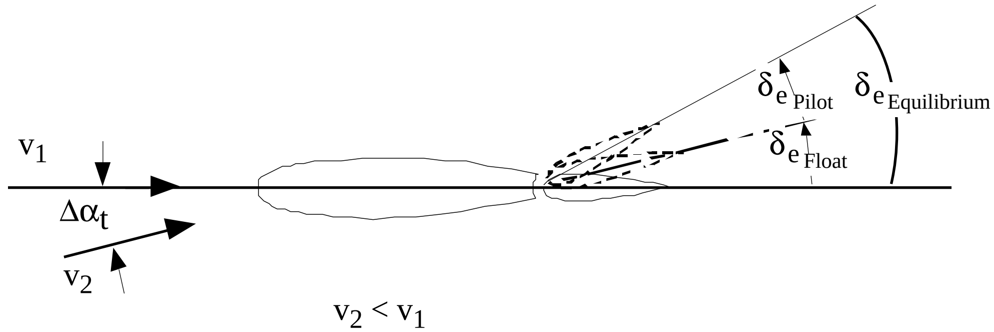


Figure 4.14
Stick-Fixed and Stick-Free Static Longitudinal Stability

In this case, assume that the airplane is trimmed for zero longitudinal control forces for both the stick-fixed and stick-free cases at the same lift coefficient or airspeed (point A). If the pilot now wishes to decelerate and fly at a lift coefficient of 1.0 (point B), he must overcome the stabilizing pitching moment represented by the distance between the stick-fixed curve at point B and the horizontal axis $\left(C_{m_{CG}} = -.05\right)$. Now, if the elevator is free to float, the pilot will only have to overcome the stabilizing pitching moment represented by the distance between the stick-free curve and the horizontal axis $\left(C_{m_{CG}} = -.025\right)$. The pilot must overcome the stick-free stability with a change in elevator position from the float position to the position for zero pitching moments $C_L = 1.0$. If he does not change the longitudinal trim setting, stick-free stability will be indicated by the <u>longitudinal control force required to move the elevator from its float position to the position for zero pitching moment</u> (Figure 4.15).



Longitudinal control forces are generated by the requirement to move the elevator from its float position to the position for zero pitching moments for V_2 *airspeed. The float angle,* $\delta_{e_{Float}}$ *, cannot be determined in flight. However, the longitudinal control force required to move the elevator from float to equilibrium is an index of elevator float, thus it is an index of stick-free stability.*

Figure 4.15
Generation of Longitudinal Control Forces - Reversible Control System

Thus, the variation of longitudinal control forces with airspeed about a force trim airspeed is indicative of the stick-free static longitudinal stability. This relationship can be expressed analytically as follows:

$$\frac{dF_s}{dV_e} = 2K \; \frac{W}{S} \; \frac{Ch_{\delta_e}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{Free} \; \frac{V_e}{V_{e_{Trim}}^2} \qquad \textit{eq. 4.17}$$

Where:

$\frac{dF_s}{dV_e}$ = longitudinal control force variation with equivalent airspeed about a force trim airspeed, $V_{e_{Trim}}$

K = a constant dependent on gearing ratio between the elevator and cockpit control stick, size of the elevator, and horizontal tail efficiency factor $\left(K = -GS_e \; \bar{c}_e \; \eta_t\right)$

$C_{h_{\delta_e}}$ = elevator hinge moment coefficient variation with elevator deflection

$\frac{W}{S}$ = wing loading, ratio of gross weight of the airplane to the total planform area of its wing

$\frac{dC_{m_{CG}}}{dC_{L_{Free}}}$ = stick-free stability

$C_{m_{\delta_e}}$ = elevator control power

From a study of this equation, it is obvious that the variation of longitudinal control force about a trim airspeed will at least indicate whether the static longitudinal stability of the airplane with the elevator free to float is positive, neutral, or negative. However, this variation will not indicate the degree of stability unless the numerical values of $C_{m_{\delta_e}}$, $\frac{W}{S}$, K, and $C_{h_{\delta_e}}$ are known. The relationship of longitudinal control force versus airspeed in equilibrium flight about a trim airspeed is often termed stick force stability or longitudinal control force stability. It is obvious that if $\left(\frac{dC_{m_{CG}}}{dC_L}\right)_{Free} = 0$,

i.e., the CG is at the stick-free neutral point, the slope of the longitudinal control force versus airspeed curve will also be zero (Figure 4.16). This fact will be used later to estimate the airplane's stick-free neutral point.

The neutral point determined from longitudinal control force versus airspeed plots will be the stick-free neutral point if, and only if, the longitudinal control system incorporates no force feel "gadgetry," such as springs or bob weights.

The neutral point determined from longitudinal control force versus airspeed plots is often, more correctly, called the stick force neutral point or the longitudinal control force neutral point.

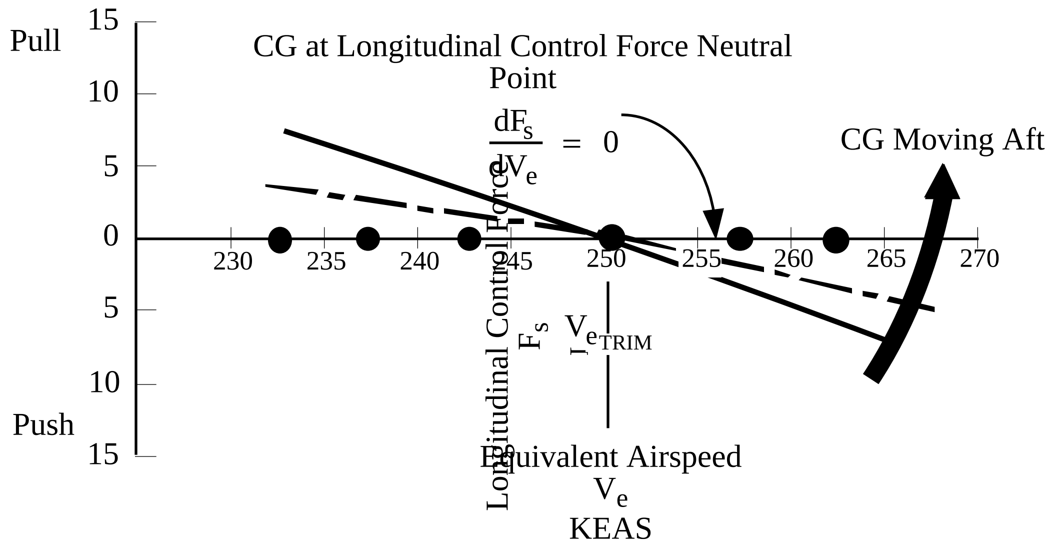


Figure 4.16
Classical CG Effects on Longitudinal Control Force Stability

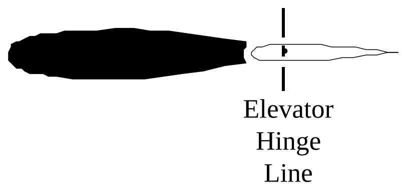
4.2.1.7 MINIMIZING LONGITUDINAL CONTROL SURFACE "FLOAT" IN THE REVERSIBLE CONTROL SYSTEM

The floating characteristics of the longitudinal control surface depend on the magnitudes of the parameters $C_{h_{\alpha_t}}$ and $C_{h_{\delta_e}}$. It is important to reduce the floating tendency of the control surface as much as possible in order to minimize the variation in static longitudinal stability between the stick-fixed and stick-free cases. This means the ratio of $\frac{C_{h_{\alpha_t}}}{C_{h_{\delta_e}}}$ should be as small as possible.

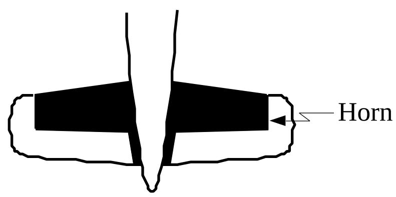
$$\delta_{e_{Float}} = \frac{C_{h_{\alpha_t}}}{C_{h_{\delta_e}}} \alpha_t \qquad \textit{eq 4.18}$$

Also, $C_{h_{\delta_e}}$ must not be too large or the longitudinal control forces will be excessive. Methods of controlling the magnitude of the hinge moment parameters are referred to as methods of aerodynamic balancing.

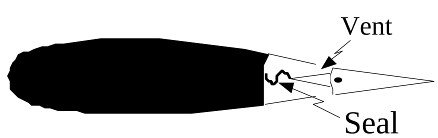
Common methods of aerodynamic balancing are shown in Figure 4.17. These methods all result in reducing the hinged moments created at the elevator hinge line when changes in angle of attack and elevator position occur. Aerodynamic balancing not only reduces the floating tendency but reduces longitudinal control forces required to deflect the surface.



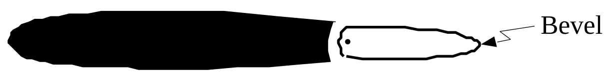
(A) Set-Back Hinge



(B) Horn Balance



(C) Internal Seal



(D) Beveled Trailing Edge

Figure 4.17
Methods of Aerodynamic Balance

4.2.1.8 STICK-FREE STATIC LONGITUDINAL STABILITY - IRREVERSIBLE CONTROL SYSTEM

For airplanes equipped with irreversible longitudinal control systems, freeing the longitudinal control surface, i.e., allowing it to respond to some flight variable, may have profound effects on static longitudinal stability. The nature of the irreversible control system, however, does not allow a control surface to move directly in response to a flight variable, in comparison to the reversible longitudinal control which may respond directly to surface pressure distribution by floating with or against the relative wind. The movement of the control surface in the irreversible system must be programmed within the control system. This is accomplished by providing sensors in the airplane to measure flight parameters, then feeding signals to the irreversible power control cylinders to move the control surface independent of the pilot's actions. This artificial "float" or static stability augmentation is almost always incorporated to attempt to correct stick-fixed static longitudinal instability, i.e.; an unstable elevator position - airspeed relationship. The acceptability of such a device must be determined in view of its reliability, the improvement in flying qualities which results, and the increase in mission effectiveness which can be realized. Generally, the use of such devices adds a marked degree of complexity to the control system.

For illustrative purposes, the following example is presented of an irreversible control system incorporating a device to provide artificial longitudinal control force stability. Assume that the airplane in some flight condition exhibits the unstable elevator position-airspeed relationship shown by the solid line of Figure 4.19. It is obvious that the elevator position instability would precipitate longitudinal control force instability for a "classical" irreversible control system in which longitudinal control force is a direct function of elevator position. However, devices can be incorporated in the longitudinal control system to change the elevator position, independent of pilot action, to an artificial "float" position as airspeed is varied about trim. One means of providing the artificial "float" is to incorporate an airspeed and altitude sensor with an "extendible link" in the longitudinal control system. These devices, then, can program "artificial elevator float" as airspeed is varied about trim - shown by the dashed line Figure 4.18. Since an extendible link in the control system is utilized, the "artificial elevator float" does not result in cockpit control stick motion. Longitudinal control forces are generated by the requirement for the pilot to move the elevator from the "artificial float" position to the position required for equilibrium. As shown in Figure 4.18, this movement is now in the direction which results in a positive or stable longitudinal control force-airspeed variation about trim (Figure 4.19). For a further discussion of the use of extendible links on irreversible control system, see pages 4.50 through 4.51.

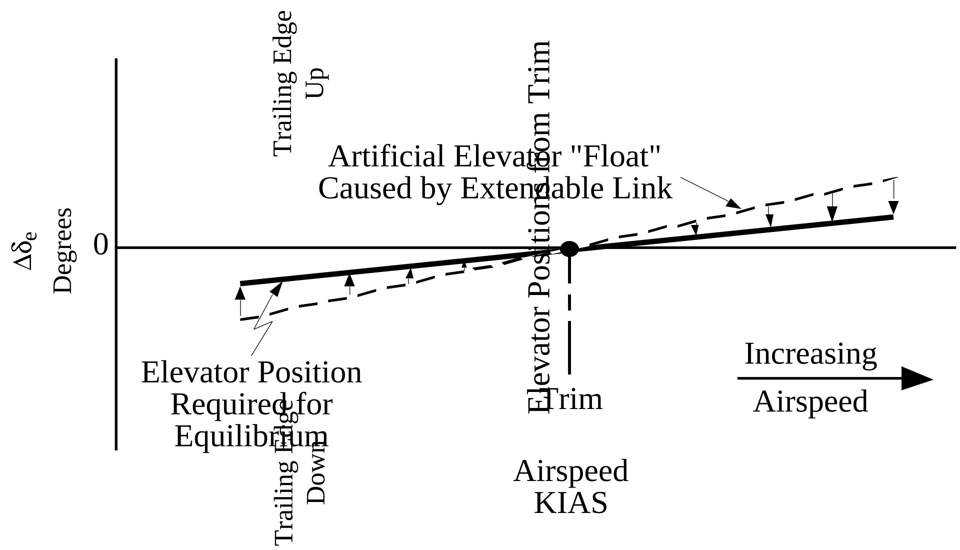


Figure 4.18
Typical Means of Providing Artificial Longitudinal Control Force Stability

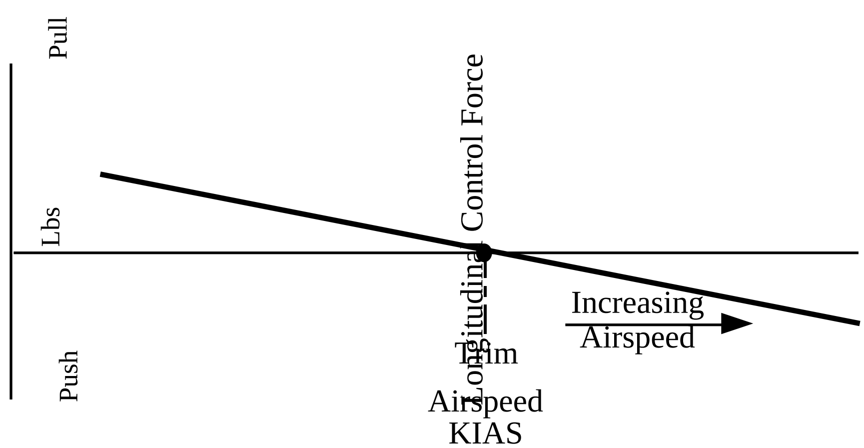


Figure 4.19
Devices in the Irreversible Control System Can Provide Longitudinal Control Force Stability Even When Elevator Position Instability Exists

4.2.1.9 POWER EFFECTS ON STATIC LONGITUDINAL STABILITY

The effects of engine operation on static longitudinal stability may be very significant. Propeller power effects will be considered first; these effects may be direct or indirect.

The direct propeller contribution arises as a result of the forces created by the propeller itself. The components of the force created by the running propeller at some angle to the relative wind are the thrust force, T_p, and the normal force, N_p (Figure 4.20). The generation of the thrust force is obvious. The normal force is generated as a result of the airflow being turned more perpendicular to the propeller disc as it passes through the disc. This effect is sometimes referred to as the "propeller fin effect."

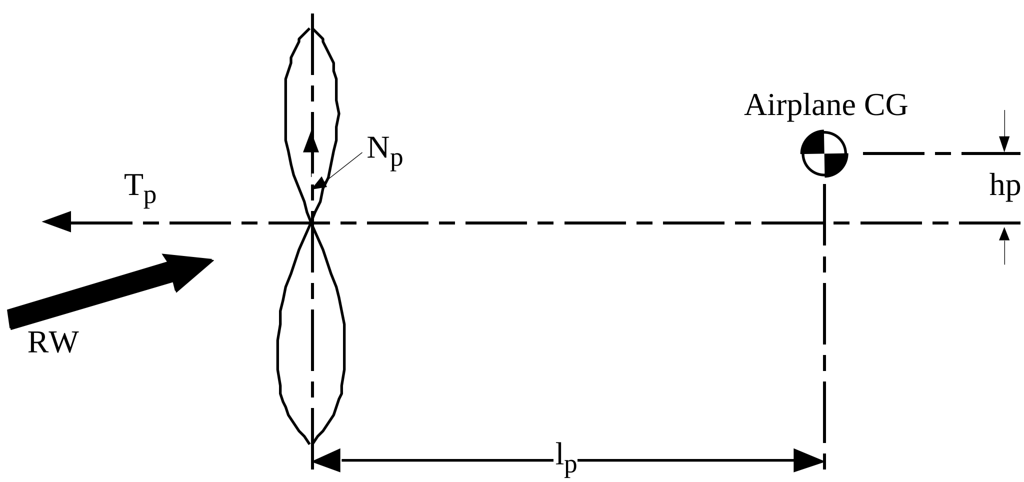


Figure 4.20
Forces Created by the Running Propeller

From a study of Figure 4.20, it may be rationalized that the effect of propeller power on static longitudinal stability depends on the location of the propeller with respect to the airplane center of gravity (CG). If the propeller is positioned ahead of and below the CG, direct propeller effects are destabilizing.

The indirect propeller effects are a result of slipstream interaction with the wing and horizontal tail and are composed of the following major contributions:

1. Effect of slipstream on wing-fuselage moments.

2. Effect of slipstream on wing lift coefficient.

3. Effect of slipstream downwash at the horizontal tail.

4. Effect of increased slipstream dynamic pressure on the tail.

Indirect propeller effects are difficult to predict. Their contribution to static longitudinal stability may be stabilizing or destabilizing, depending largely on whether the lift from the horizontal tail is acting up or down.

Direct and indirect propeller effects on static longitudinal stability are generally significant. For "conventional" propeller airplanes (propeller ahead of CG), the combined effects are usually destabilizing.

The effects of power on the static longitudinal stability of the jet propelled airplane are somewhat simpler to analyze. The turbojet unit generates three major contributions. These are the direct thrust effect, the direct normal force effect at the air inlet, and the effect of the induced flow at the horizontal tail due to inflow toward the jet exhaust.

The direct effects of thrust and normal force are the same as previously discussed for the propeller driven airplane (Figure 4.21). Whether the direct effects are stabilizing or destabilizing depends entirely on the vertical and horizontal position of the airplane center of gravity.

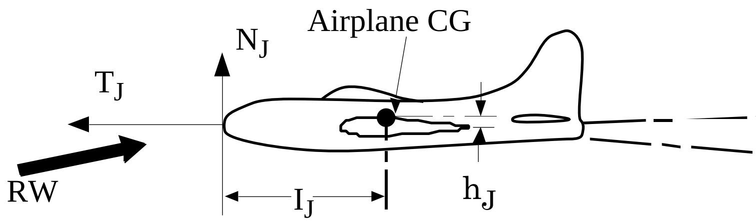


Figure 4.21
Forces Created by the Jet Engine

The indirect jet effect on static longitudinal stability is due to the jet exhaust creating an inclined flow pattern around itself. If the horizontal tail is located in this area of "exhaust inclined" flow, its angle of attack will be changed, thereby creating moments that influence the static longitudinal stability of the airplane. This indirect effect, sometimes called the "entrainment effect," is usually slightly destabilizing. The total influence of direct and indirect jet effects on static longitudinal stability is usually destabilizing.

4.2.2 Dynamic Longitudinal Stability and Control in Unaccelerated Flight

The previous discussion of longitudinal stability has been concerned only with equilibrium flight conditions. The discussion will not be expanded to study the means by which one equilibrium flight condition is changed to another equilibrium flight condition. This study of dynamic longitudinal stability and control characteristics will require the investigation of nonequilibrium longitudinal flight conditions.

The means by which the airplane may be stabilized at various lift coefficients and airspeeds has been previously developed. A typical response of the airplane in angle of attack and airspeed to a longitudinal control input through the two longitudinal modes of motion is shown in Figure 4.22. The control input (nose-up in this case) generates pitching moments which initially cause only changes in angle of attack. This is in response of the airplane through its short period mode of motion - airspeed is essentially constant for this response because the short time interval does not allow speed changes. This mode of motion affects both maneuvering and nonmaneuvering tasks because of its bearing on the initial response of the airplane. However, its characteristics are most critical for maneuvering tasks; therefore, the short period mode will be investigated in the subsequent section of longitudinal flying qualities during maneuvering tasks.

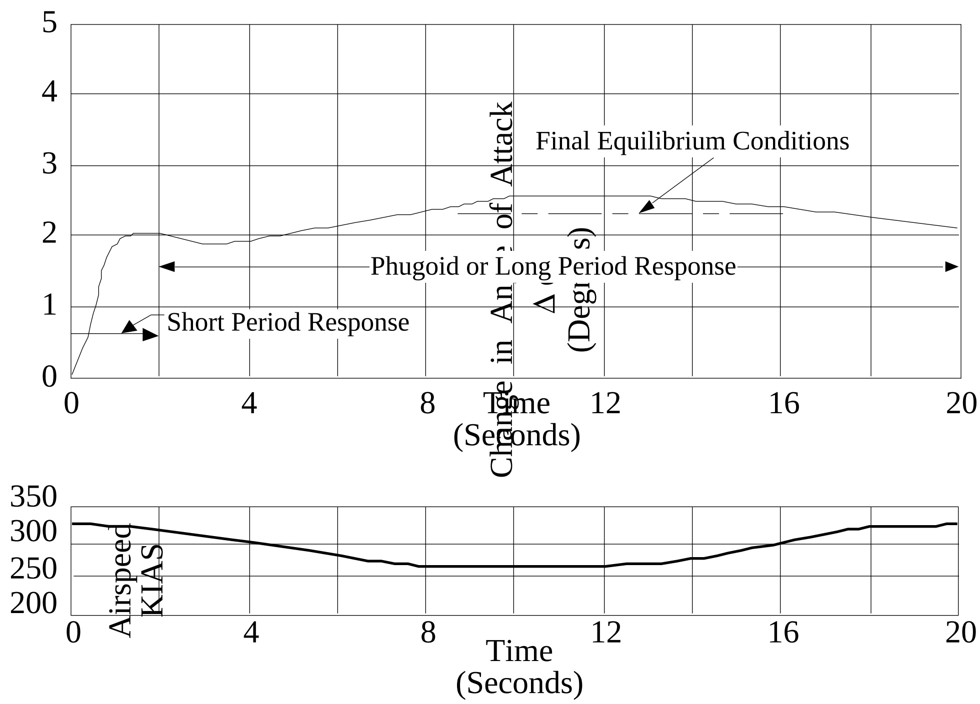


Figure 4.22
Typical Airplane Response to Longitudinal Control Input

The long period response of the airplane occurs after the short period motion has diminished to a near steady state condition. This typical long period motion is seen to be a second order response composed of airspeed variations at an essentially constant angle of attack. Of course, altitude and attitude will vary.

In normal flying of the airplane, the pilot would not allow the long period motion to cause the airspeed oscillation presented in Figure 4.22. If the pilot desired to restabilize at 260 KIAS, he would apply a small longitudinal control input to suppress the long period motion at about 10 seconds. However, it should now be apparent that the long period or "phugoid"[1] mode of motion is utilized by the pilot to make airspeed changes. Since a great

[1] The long period of motion was first described and named by F.W. Lanchester. Phugoid is from the Greek root for flee. It is believed Lanchester really wanted the Greek root for fly.

$$
\begin{array}{l|ccc|}
\text{Drag Characteristics} \rightarrow & S + D_u & D_\alpha - g & g \\
\text{Lift Characteristics} \rightarrow & \dfrac{L_u}{u_0} & S + \dfrac{L_\alpha}{u_0} & -S \\
\text{Pitching Moments Characteristics} \rightarrow & -M_u & -M_{\dot{\alpha}}S - M_\alpha & S^2 - M_{\dot{\theta}}S
\end{array} = 0
$$

Terms Generated by Changes in Horizontal Velocity

Terms Generated by Changes in Angle of Attack

Terms Generated by Changes in Pitch Attitude

S = Laplace Operator

g = acceleration due to gravity

u = horizontal velocity (u_0 = initial horizontal velocity)

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

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

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

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

$M_u = \dfrac{\partial M / \partial u}{I_{yy}}$ = change in pitching moment with 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 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.

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

Figure 4.23
The Longitudinal Determinant

deal of the pilot's effort during nonmaneuvering tasks will be directed toward making airspeed changes in level flight, the pilot will devote much of his attention during these tasks to controlling the long period mode of motion.

The remainder of this discussion will be directed toward describing the origin, characteristics, and parameters affecting the long period mode of motion.

4.2.2.1 ORIGIN OF THE PHUGOID MODE OF LONGITUDINAL MOTION

Without derivation, which can be found in appropriate literature, the determinant of the transformed longitudinal characteristic equation of motion for "small" disturbances may be written as shown in Figure 4.23.

Before proceeding, a few comments are in order concerning angle of attack stability and speed stability. These terms will be used extensively in discussions of longitudinal dynamics. Angle of attack stability may be expressed in coefficient form as $\frac{\partial c_m}{\partial \alpha}$ or C_{m_α}; i.e., the change in pitching moment coefficient with change in angle of attack at a constant speed. Angle of attack stability normally carries a negative sign; i.e., positive increase in (nose-up) generates a negative (nose-down) pitching moment. Under restricted conditions (power-off gliding flight at a low Mach number) where C_L is a unique function of α, angle of attack stability can be related directly to the familiar static stability criteria $\frac{dC_m}{dC_L}$ as follows:

$$\frac{dC_m}{dC_L} = \frac{\partial C_m / \partial \alpha}{\partial C_L / \partial \alpha} \qquad \textit{eq 4.19}$$

Thus, for these conditions, static longitudinal stability $\left(\frac{dC_m}{dC_L} < 0\right)$ guarantees angle of attack stability $\left(\frac{\partial C_m}{\partial \alpha} < 0\right)$.

Speed stability may be expressed in coefficient form as $\frac{\partial C_m}{\partial U}$ or C_{m_u}; i.e., the change in pitching moment coefficient with change in horizontal velocity at a constant angle of attack. The C_{m_u} term is normally very small for the moderate and low subsonic Mach numbers. In the high subsonic regime, C_{m_u} is normally negative and in many instances large enough to make the airplane statically unstable.

The solutions of the longitudinal determinant will provide useful information about the longitudinal modes of motion. The classic long period or "phugoid" approximation is of concern at present. In order to make this approximation, several assumptions must be made. These assumptions, based on flight experience and logical reasoning, are as follows:

1. The angle of attack stability, M_α, is large enough so that very small changes (i.e. near zero) of angle of attack are required to counter pitching moments generated by pitch rates, pitch accelerations, and velocity changes. This implies M_u is quite small and that the frequency of the phugoid is quite low.

2. The previous argument justifies the assumption that the angle of attack is a constant during the phugoid oscillation.

3. The assumption that little compressibility effects occur enhances the approximation.

If the above assumptions are valid, the lift and drag portions of the longitudinal determinant are the controlling factors for the long period motion. The classic long period approximation or "phugoid minor" may then be written as follows:

$$\begin{vmatrix} S + D_u & g \\ L_u/u_0 & -S \end{vmatrix} = 0 \qquad \textit{eq 4.20}$$

Solving the determinant yields the following second order characteristic equation:

$$S^2 + D_u S + g \frac{L_u}{u_0} = 0 \qquad \textit{eq 4.21}$$

Therefore, the undamped natural frequency and damping ratio of the phugoid mode of motion may be developed† as follows:

$$\omega_{n_p} = \text{undamped phugoid frequency} = \sqrt{2}\,\frac{g}{u_0} \qquad \textit{eq 4.22}$$

$$\zeta_p = \text{phugoid damping ratio} = \frac{1}{\sqrt{2}}\,\frac{C_D}{C_L} \qquad \textit{eq 4.23}$$

For a lightly damped oscillation, the damped natural frequency is approximately equal to the undamped natural frequency, so:

$$\omega_p = \text{damped natural frequency} \approx \sqrt{2}\,\frac{g}{u_0} \qquad \textit{eq 4.24}$$

The period of the long period motion is thus approximated by:

$$p_p\ (\text{sec}) = .138\, u_0\ (\text{where } u_0 \text{ is in feet per sec.}) \qquad \textit{eq 4.25}$$

Thus, the period of the phugoid is seen to be a function of horizontal velocity (u_0) or airspeed (V), about which the motion oscillates. This is a reasonable approximation for the phugoid period. It is readily seen that the period of the phugoid motion is very long.

The damping ratio of the phugoid motion may be approximated by the following relationship:

$$\zeta_p \approx \frac{.707}{L/D} \qquad \textit{eq 4.26}$$

† Several mathematical manipulations have been omitted.

This approximation for phugoid damping ratio is not as accurate as the approximation for the phugoid period. However, it does point out that phugoid damping varies inversely with the ratio L/D. The phugoid mode of motion, if allowed to persist, therefore exhibits a prolonged oscillation that damps very slowly.

4.2.2.2 CHARACTERISTICS OF THE PHUGOID MODE OF LONGITUDINAL MOTION

Additional insight into the long period or "phugoid" mode of motion may be gained by studying the flight path of an airplane during a phugoid motion which is allowed to persist. The actual motion involves alternate climbing and diving and airspeed variations between an excess at the bottom of a cycle and a deficiency at the top. During these oscillations, the airplane trades kinetic for potential energy and vice versa - corresponding to airspeed and altitude variations. To an observer with a stationary viewing point, the airplane motion during a longitudinal phugoid would appear as shown in Figure 4.24.

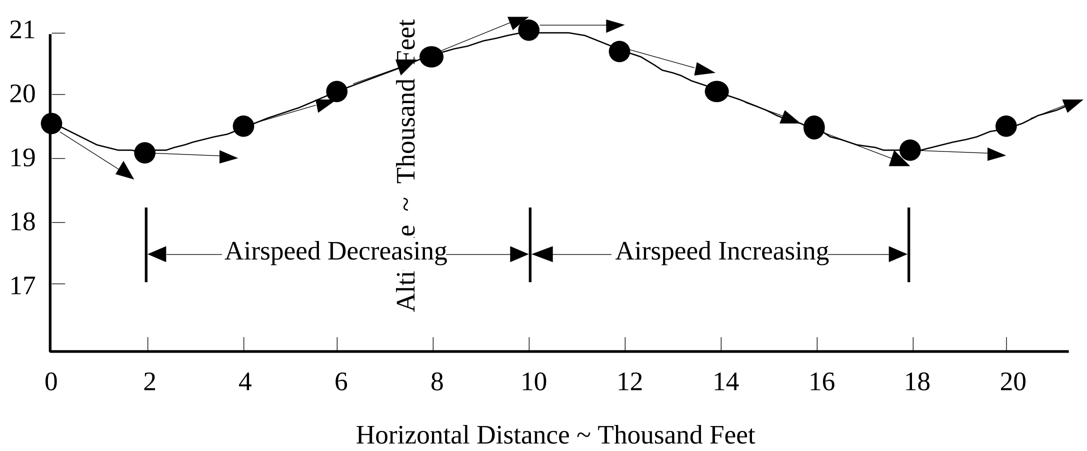


Figure 4.24
Typical Phugoid Flight Path (Stationary Viewing Point)

If the observer were flying alongside at constant airspeed, the airplane long period motion would appear as in Figure 4.25.

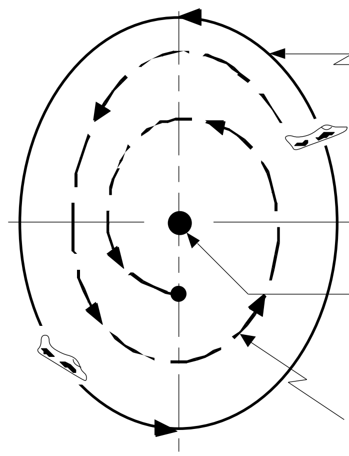


Figure 4.25
Typical Phugoid Motion (Moving Viewing Point)

From the moving viewing point, the airplane will appear to rise and fall like a mass suspended on a spring. For a constant angle of attack, the excess airspeed on the downswing produces excess lift; the deficiency of airspeed on the upswing results in less lift. These variations in lift result in net downward forces at the top of the oscillation and net upward forces at the bottom. These forces may be thought of as the effective spring constant in the system.

Drag forces vary also as the airplane oscillates in the long period motion. At the top of the cycle, where airspeed is reduced, drag is reduced. This results in a net forward force. Conversely, a net backward force is generated by the increase in drag at the bottom of the cycle. It is easily seen that these changes of drag would tend to damp the forward and backward components of motion. This would cause the elliptical path to degenerate into a spiral path toward the center opposite the observer’s position (Figure 4.25).

4.2.2.3 ADDITIONAL PARAMETERS AFFECTING THE PHUGOID MODE OF MOTION

The discussion of longitudinal long period motion to this point has included no consideration of varying CG, angle of attack stability (M_α), or speed stability (M_u) or the effects of power. The effect of varying these parameters will now be shown by utilization of the convenient root locus plots. The "classic" phugoid roots with the characteristics previously developed are shown in Figure 4.26. The "classic" short period roots are shown for completeness; however, the main concern here is the long period motion.

The long period mode of Figure 4.26 is typically stable, oscillatory, and lightly damped. The CG is somewhere forward of the stick-fixed neutral point. (Note: The stick-free case could be used; however, the influence of the free elevator on phugoid characteristics is usually negligible. The stick-fixed case was chosen arbitrarily for this discussion.)

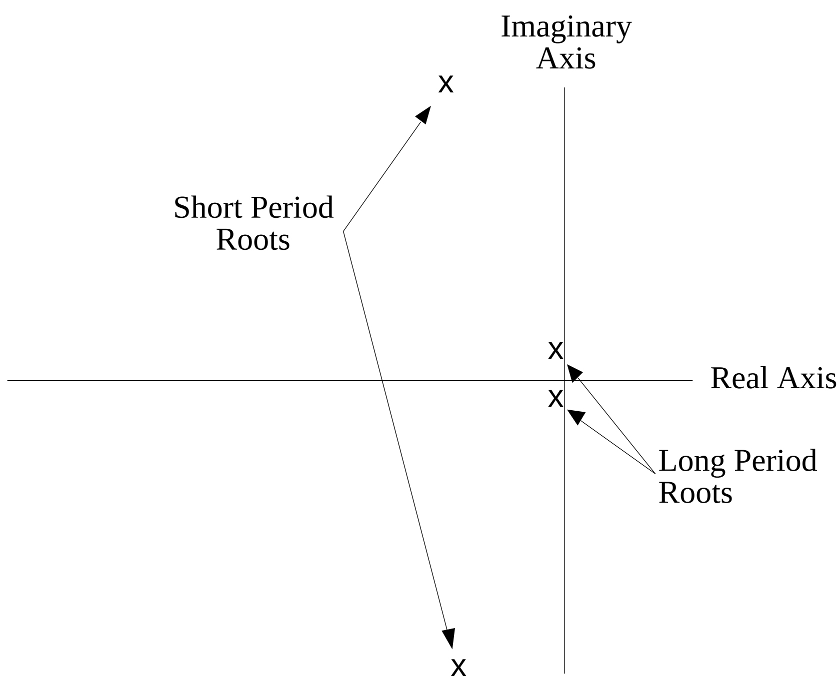


Figure 4.26
Complex Plane Representation of Classic
Phugoid and Short Period Modes of Motion

Now, if the CG is moved progressively further aft toward the neutral point, <u>the frequency of the phugoid mode decreases (period becomes longer) and the damping remains essentially constant</u> (Figure 4.27).

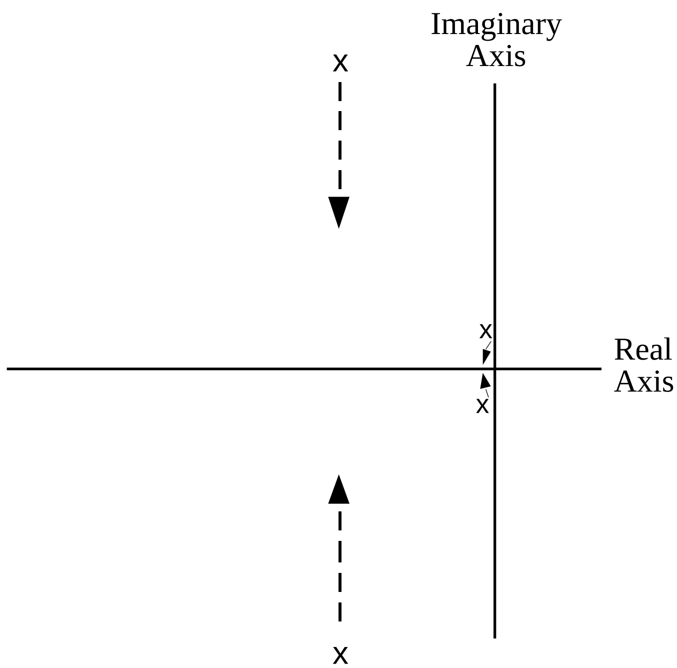


Figure 4.27
Effect of Aft CG Movement on Phugoid Mode

If the CG is moved far enough aft, the oscillatory phugoid mode degenerates into a pair of aperiodic modes represented by the branches AB and AC of Figure 4.28. The CG position at which the phugoid becomes aperiodic (Point A) is generally just slightly forward of the neutral point. When the CG is moved aft of the neutral point, the branch of the aperiodic mode AC crosses the imaginary axis representing a divergence mode - i.e., the airplane is statically unstable when the CG is aft of the neutral point. This situation is easy to visualize - any change of airspeed from a trimmed condition for a statically unstable airplane results in a further pure divergence in airspeed.

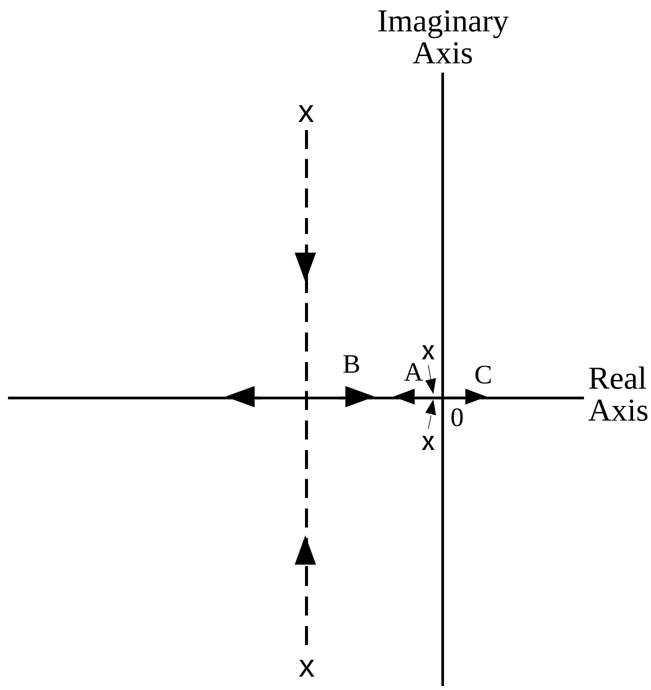


Figure 4.28
Degeneration of Phugoid Mode into Aperiodic Modes

If the CG is moved further aft past the neutral point, the branches of the phugoid mode and short period mode meet. At this point, a new oscillatory mode arises corresponding to the branches DE and DF of Figure 4.29. This is a stable oscillation whose damping and period both lie somewhere between those of the short period and phugoid. This mode of motion is of academic interest only since this far aft CG position is seldom encountered due to the strong static instability which would exist.

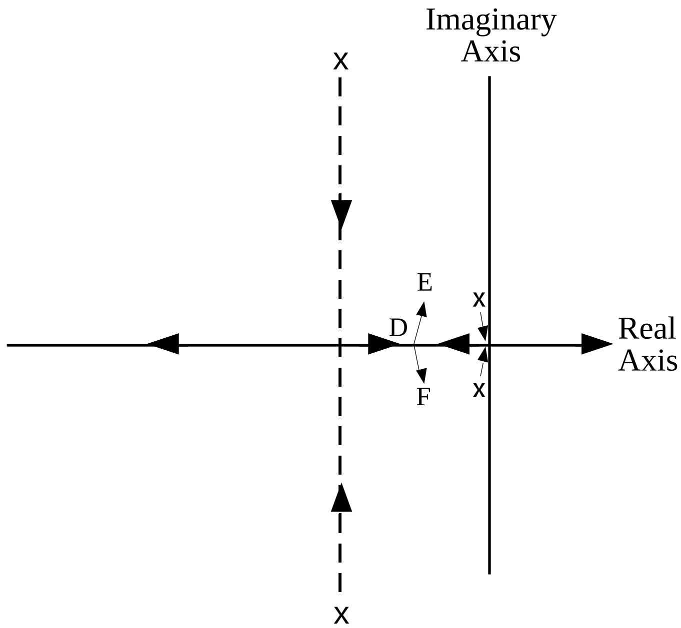


Figure 4.29
Generation of the "Third Longitudinal Oscillatory Mode"

The effect of varying angle of attack stability, M_α, can be studied by first assuming the M_u is zero, then allowing M_α to increase negatively from zero. (This is the normal sign of M_α, since, for stability, positive (nose-up) increases in angle of attack must generate negative (nose-down) pitching moments.) The effect of increasing angle of attack stability is shown in Figure 4.30, and is seen to be exactly the same effect as moving the CG forward from the stick-fixed neutral point.

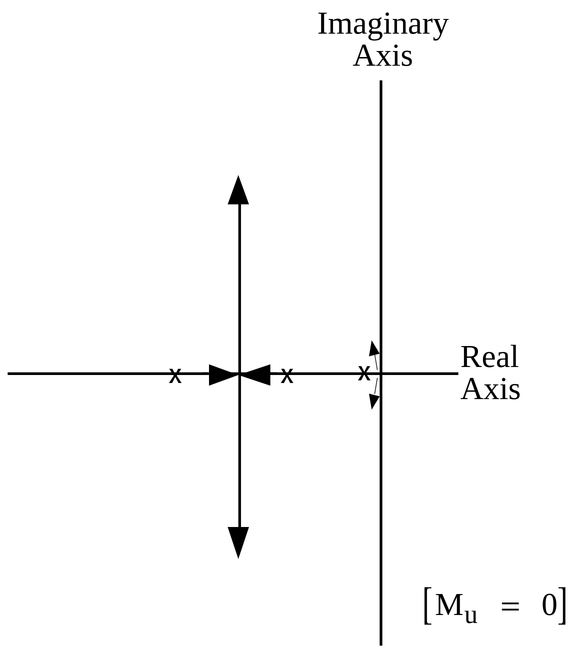


Figure 4.30
Effect of Increasing Angle of Attack Stability, M_α

The influence of changing speed stability, M_u on long period characteristics is mainly to change the frequency (or period) of the motion. If the roots are initially positioned according to the "classic approximation," and M_u is reduced (decreasing speed stability), the effect will be shown in Figure 4.31. If the product $M_u L_\alpha$ becomes greater than $M_\alpha L_u$, a branch of the long period mode crosses the imaginary axis and the motion becomes a nonoscillatory pure divergence. This phenomenon is easy to visualize - an increase in airspeed would generate nose-down pitching moments. The airplane possessing speed instability may be difficult to fly. This depends on the rate of divergence. Speed instability is quite often encountered in the transonic flight regime.

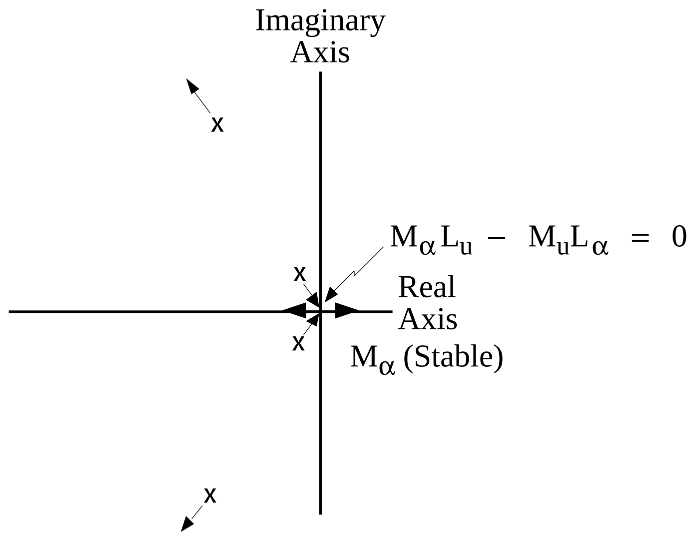


Figure 4.31
Influence of Reducing Speed Stability, M_u

If the roots are again initially positioned according to the “classic approximation” and M_u is increased positively, the effect may be as shown in Figure 4.32. If M_u is increased positively a sufficient amount, and if M_α is not too large, the long period motion may become an oscillatory divergence. If the motion is very divergent, flying qualities may be seriously degraded, although usually not as much as the condition of nonoscillatory pure divergence.

For the propeller driven airplane, engine operation may have a large effect on damping of the long period motion. For a constant brake horsepower, thrust increases at decreased airspeed adding a net forward force at the top of the cycle and vice versa. This phenomenon increases the damping of the phugoid oscillation. Jet engine operation has negligible influence on phugoid characteristics.

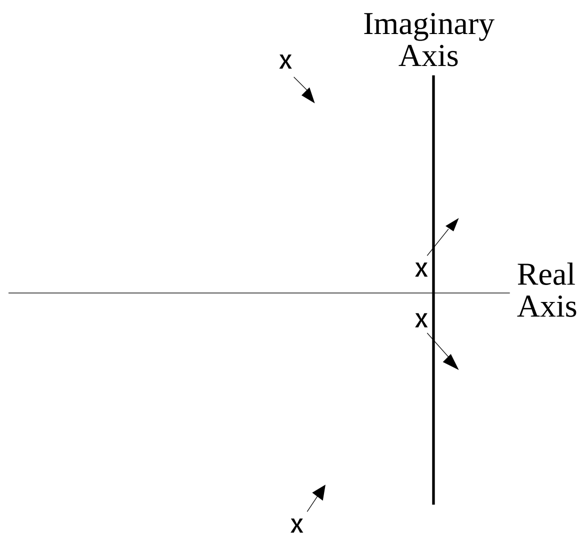


Figure 4.32
Possible Effect of Increasing Speed Stability, M_u

4.2.3 Longitudinal Control System Influence on Longitudinal Flying Qualities During Nonmaneuvering Tasks

Longitudinal control system design will have a profound effect on longitudinal flying qualities during nonmaneuvering tasks. A thorough understanding of the effects of control system "gadgetry" on longitudinal flying qualities is essential for the flight test engineer and test pilot.

4.2.3.1 GADGETRY USED IN BOTH REVERSIBLE AND IRREVERSIBLE LONGITUDINAL CONTROL SYSTEMS

The simple spring is often used to provide a steeper gradient of longitudinal control force versus airspeed and to provide good control stick centering. A simple spring arrangement and its effect on longitudinal control force stability is shown in Figure 4.33.

The preloaded downspring arrangement has a similar effect on longitudinal control forces as can be seen from Figure 4.34. When utilized in a reversible control system, the downspring has a tendency to "drive" the long period or phugoid motion divergent with controls free.

This is due to the steady unbalancing force which is always applied to the longitudinal control system. This unbalancing force is "trimmed out" by the pilot at the control force trim speed; however, the force becomes a factor as airspeed varies in the long period oscillation and may precipitate elevator inputs sufficient to destabilize the oscillation.

Although the bobweight is normally used to influence maneuvering characteristics, it has some effect on longitudinal control force stability as shown in Figure 4.35.

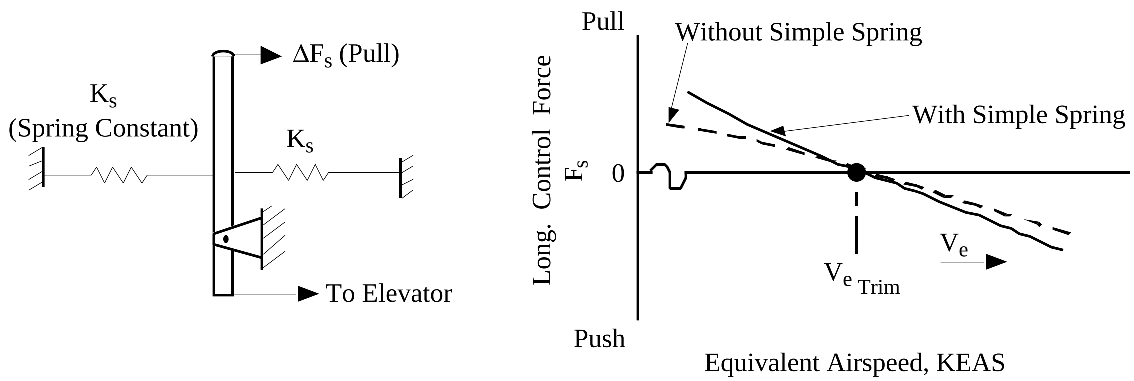


Figure 4.33
Simple Spring Arrangement and Influence on Static Longitudinal Stability Characteristics

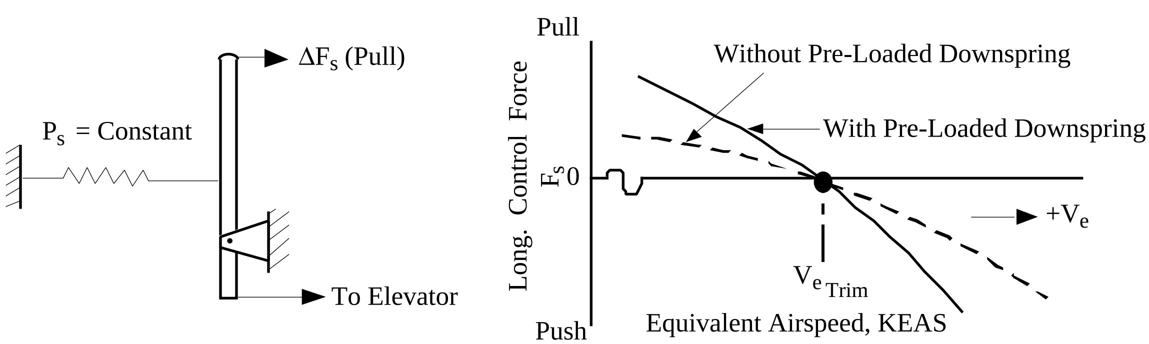


Figure 4.34
Pre-Loaded Downspring Arrangement and Influence on Static Longitudinal Stability Characteristics

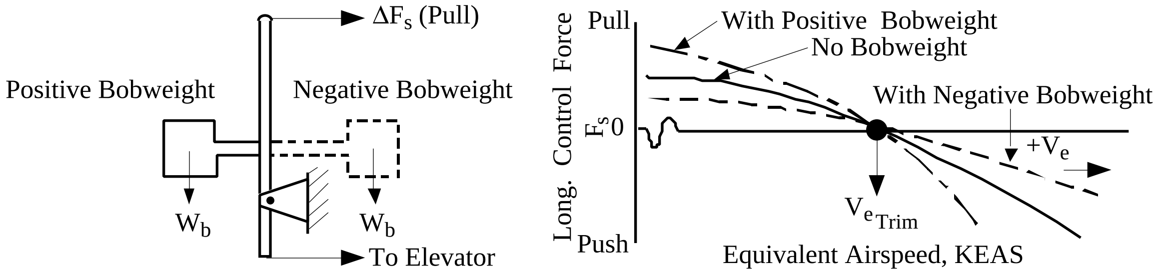


Figure 4.35
Bobweight Arrangement and Influence
on Static Longitudinal Stability Characteristics

4.2.3.2 GADGETRY USED ONLY IN REVERSIBLE LONGITUDINAL CONTROL SYSTEMS

The blow-down tab may be used to increase the gradient of longitudinal control force versus airspeed. As shown in Figure 4.36, the blow-down tab remains "on the stop" until the airplane has accelerated to a speed where the spring force is overcome. This speed must be slower than take-off speed or the pilot will be confronted with serious longitudinal control force nonlinearities at flying airspeeds.

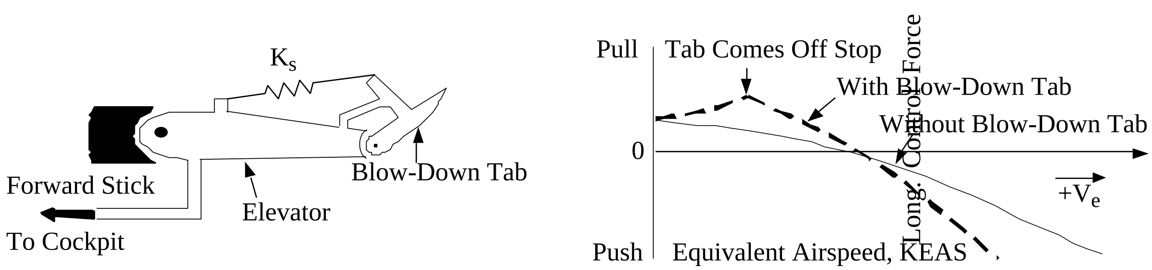
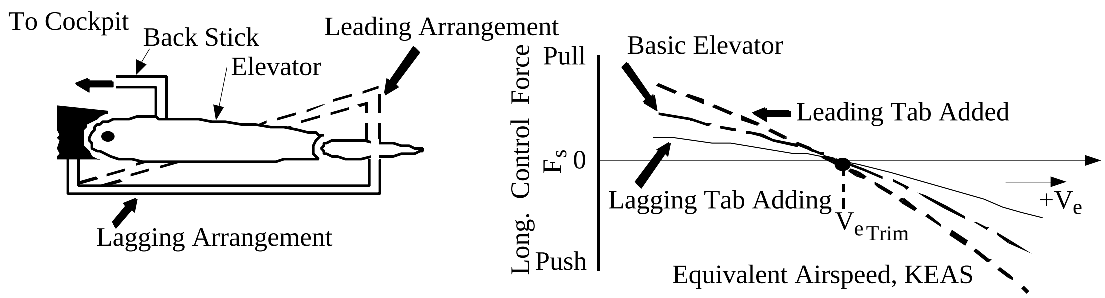


Figure 4.36
Blow-Down Tab Arrangement and Effect on Static
Longitudinal Stability Characteristics

Another means of modifying longitudinal stability and control characteristics is through the use of lagging and leading tabs. These arrangements have a dual effect in that they modify elevator float characteristics as well as changing longitudinal control force requirements (Figure 4.37).



Leading Tab Arrangement: Increases dF_s/dV_e, Reduces $\delta_{e_{FLOAT}}$

Lagging Tab Arrangement: Decreases dF_s/dV_e, Increases $\delta_{e_{FLOAT}}$

Figure 4.37
Leading and Lagging Tabs

The longitudinal servo tab is frequently used in very large airplanes or medium-sized airplanes capable of high subsonic airspeeds. With a servo tab arrangement, the pilot moves the servo tab through control stick motion. Movement of the servo tab generates forces and moments which cause the elevator to move (Figure 4.38). By use of the servo tab, longitudinal control forces required of the pilot are very greatly reduced.

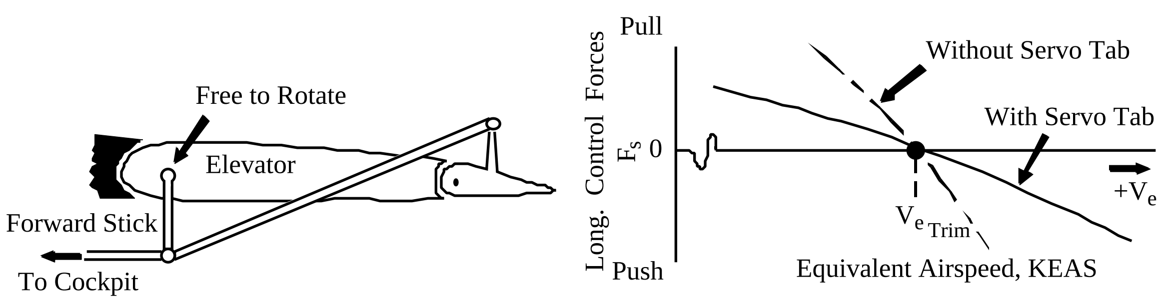


Figure 4.38
Longitudinal Servo Tab Arrangement

Another arrangement used to modify longitudinal stability and control characteristics is the preloaded spring tab, which is a modification of the servo tab described above. The preloaded spring tab does not modify longitudinal control forces about the trim airspeed until the preloaded force is exceeded (Figure 4.39). Once longitudinal control forces are greater than the preload of the spring, the servo action of the tab reduces the longitudinal control force-airspeed gradient.

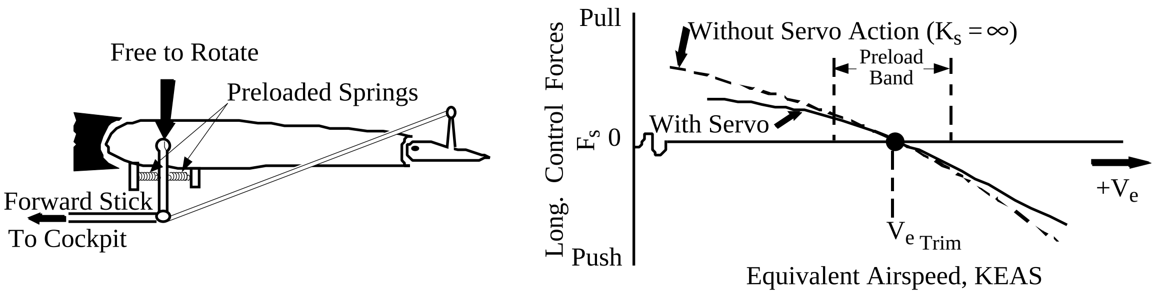


Figure 4.39
Pre-Loaded Spring Tab Arrangement

4.2.3.3 GADGETRY USED ONLY IN IRREVERSIBLE LONGITUDINAL CONTROL SYSTEMS

Fully irreversible longitudinal control systems generally incorporate simple springs, bob weights, and viscous dampers to provide the pilot with longitudinal control "feet." In addition, other control system gadgetry may be utilized to improve longitudinal stability and control characteristics. Some of these arrangements are discussed below.

The extendable link may be utilized to provide longitudinal control force stability even though the airplane exhibits elevator position instability. A typical extendable link arrangement is shown in Figure 4.40.

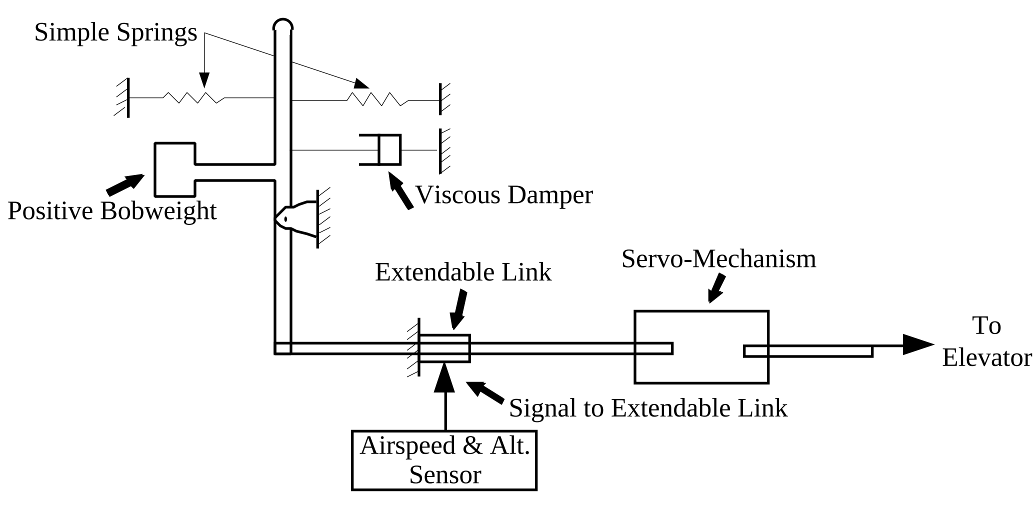


Figure 4.40
Irreversible Longitudinal Control System with Extendable Link

The extendable link mechanism may be used to program an artificial "float" into the longitudinal control system as airspeed is varied about trim (Figure 4.41). Since the pilot moves the elevator from the "float" position to the position required for equilibrium, longitudinal control forces are in the correct direction.

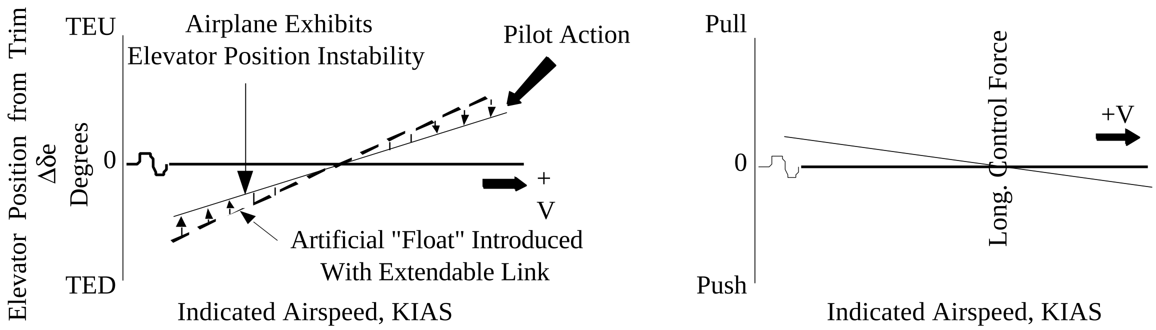


Figure 4.41
Extendable Link Can Provide Artificial Control Force Stability

Other devices sometimes used in irreversible longitudinal control systems are the mechanical advantage changer and the "q - bellows." Both these devices are used to cope with poor basic airplane characteristics, such as neutral or negative elevator position gradients or nonlinearities in relationship of elevator position with airspeed about trim. One of the characteristics usually generated by the action of both these devices is cockpit stick motion. However, this motion is usually so slight that it is not objectionable and usually is not noticeable. Certain types of "q - bellows" systems have demonstrated a propensity toward failure through atmospheric icing.

4.3 TEST PROCEDURES AND TECHNIQUES NONMANEUVERING TASKS

4.3.1 Preflight Procedures

A rigorous investigation of longitudinal flying qualities during nonmaneuvering tasks must begin with thorough preflight planning. The purpose and scope of the investigation must be clearly defined, then a plan of attack or method of test can be formulated.

Preflight planning must start with research. This includes a study of the airplane and a thorough study of the longitudinal control system - including stability and control augmentation if installed. All available information on longitudinal stability and control characteristics should be reviewed. Much useful information can be gained by conferences with pilots and engineers familiar with the airplane.

The particular nonmaneuvering tasks to be investigated must be determined and clearly understood by the flight test team. These tasks, of course, depend on the mission of the airplane. It is particularly important during the investigation of nonmaneuvering tasks to determine if these tasks will be performed in instrument flight (IFR) conditions or merely visual flight (VFR) conditions in operational use. Certain undesirable characteristics can be accepted for VFR missions, but are not acceptable for IRF missions.

The test conditions - configuration, altitude, center of gravity, trim airspeeds, and gross weight - must be determined. Test conditions should be commensurate with the mission environment of the airplane. Center of gravity position is extremely critical for longitudinal stability tests. If flight test time permits, tests at the most aft and most forward operational CG positions should be performed after adequate build up. If flight test time is limited, tests should be performed at the most aft operational CG position (aft critical loading). Note: If the test program is aimed at determining forward and aft CG limits for operational use, appropriate CG limits will be promulgated or recommended by the test activity or higher authority.

The amount and sophistication of instrumentation will depend on the purpose and scope of the evaluation. A good, meaningful qualitative investigation can be performed with only production cockpit instruments and portable instrumentation - hand-held force gauge, stopwatch, and tape measure. Automatic recording devices, such as oscillograph, magnetic tape, and telemetry, are very helpful in rapid data acquisition and may be essential in a long test program of quantitative nature. Special sensitive cockpit instruments are also very useful, not only aiding in data acquisition but also aiding in stabilization for equilibrium test points.

The final step in preflight planning is the preparation of pilot data cards. An example of a longitudinal stability and control data card for the investigation of nonmaneuvering tasks is shown in Figure 4.42. Many test pilots desire to modify data cards to their own requirements or construct data cards for each test. At any rate, the data cards should list all quantitative information desired and should be easy to interpret in flight. Blank cards should be utilized for appropriate qualitative pilot comments.

LONGITUDINAL STABILITY AND CONTROL RECORD NON-MANEUVERING TASKS		CARD NUMBER
AIRPLANE TYPE	PILOT	PTR-BIS
BUREAU NUMBER	T.O. GROSS WEIGHT	DATE
T.O. CG GEAR DOWN ___%MAC GEAR UP ___ %MAC	T.O. TIME _______	LAND TIME _______
EXTERNAL LOADING	CONFIGURATION	
TRIM AIRSPEED MACH ____ POWER ______	ALT. ________	LONG. TRIM ____________

BREAKOUT & FRICTION CN___________	CONTROL SYSTEM MECHANICAL CHARACTERISTICS	FREEPLAY
CONTROL SYSTEM OSCILLATIONS CN ___________		CENTERING

AIRSPEED Vo MACH NO. IMN	STATIC LONGITUDINAL STABILITY CN	LONG. CONTROL FORCE Fs	ELEVATOR POSITION δe STICK POSITION, δs	RATE OF CLIMB RATE OF DESCENT
TRIM FUEL ____				
TRIM FUEL ____		ACCELERATION - DECELERATION	MAX _____	MIN __________

DYNAMIC LONGITUDINAL STABILITY

SLOW START PHUGOID	FUEL ____________	FAST START PHUGOID	FUEL _____________
ELAPSED TIME, SEC	AIRSPEED, Vo	ELAPSED TIME, SEC	AIRSPEED, Vo
0		0	
	TRIM		TRIM
	TRIM		TRIM
	TRIM		TRIM
	TRIM		TRIM

ELEVATOR FLOAT DURING PHUGOID

EASE OF TRIM TO Fs = 0	TRIMMABILITY TRIM SENSITIVITY	TRIM RATE
LONG TERM TRIM HOLDING	TRIM SPEED BAND	LOCATION OF TRIM DEVICE

Figure 4.42
Longitudinal Stability and Control Record
for Non-Maneuvering Tasks

4.3.2 Flight Test Techniques

4.3.2.1 THE QUALITATIVE PHASE OF THE EVALUATION

Longitudinal stability and control characteristics must be evaluated in relation to their influence on various nonmaneuvering mission tasks. Therefore, the test pilot must devote a portion of the evaluation to performing or simulating the nonmaneuvering tasks which have been selected. While performing these tasks, the test pilot gains the essential qualitative opinion of the longitudinal flying qualities and should assign handling qualities ratings. Without recording a single item of data, the test pilot should be able to form a good opinion of the mission effectiveness of the airplane, at least for the particular task being evaluated. This opinion will be based on the amount of attention and effort the pilot must devote to “just flying the airplane.” Due consideration must be given during this phase of the test to the following factors:

1. Whether the mission task will be performed in VFR and IFR weather, or strictly in VFR conditions.

2. The availability of an autopilot or automatic flight control system for pilot relief.

3. If stability or control augmentation systems are installed, the consequences of their failure.

The test pilot's qualitative opinion of the airplane's longitudinal flying qualities in relation to the selected mission task is the most important information to be obtained. Therefore, this phase of the test must not be overlooked. The test pilot probably will have some ideas as to the particular characteristics which make the airplane easy or hard to fly even before proceeding to the quantitative phase of the testing. Use of the quantitative test techniques to be discussed below hopefully allows the test pilot to substantiate his qualitative opinion.

4.3.2.2 MEASUREMENT OF THE MECHANICAL CHARACTERISTICS OF THE LONGITUDINAL CONTROL SYSTEM

Mechanical characteristics of the longitudinal flight control systems have a major influence on longitudinal flying qualities. The mechanical characteristics to be evaluated are defined as follows:

1. Breakout forces including friction: The longitudinal cockpit control force from the trim position required to initiate movement of the longitudinal control surface.

2. Friction: Forces in the longitudinal control system resisting the pilot's effort to change the control position.

3. Freeplay: The longitudinal cockpit control motion from the trim position that does not initiate movement of the longitudinal control surface.

4. Centering: The ability of the longitudinal cockpit control and the longitudinal control surface to return to and maintain the original trimmed position when released from any other position.

5. Control System Oscillations: Oscillations in the longitudinal control system (elevator and cockpit control stick) resulting from external or internal disturbances.

4.3.2.2.1 Breakout Forces, Including Friction

Friction in the longitudinal control system is unavoidable, however, it should be kept as low as possible. The effect of friction (without breakout) on longitudinal flying qualities can be rationalized by a study of Figure 4.43. The true variation of longitudinal control force versus airspeed is represented by the solid line and the superimposed friction is represented by the dashed lines. In this case, the combination of a shallow control force gradient and significant longitudinal friction ((+) 1.5 pounds) create poor control characteristics about trim airspeed. These characteristics would be as follows:

1. Poor longitudinal control "feel" about trim in that the friction masks the longitudinal control force stability from 150 to 190 KIAS.

2. Poor trimmability in that the airplane will stabilize at any speed from 150 to 190 KIAS with the same longitudinal trim setting. This band of airspeed is called the "trim speed band."

By judiciously adding some breakout force to the longitudinal control system, the undesirable effects of friction in the control system may be eliminated. This effect may be rationalized by a study of Figure 4.44, which is the same plot as Figure 4.43 except for the addition of a breakout force and is a typical plot of longitudinal control force variation with airspeed for a real airplane. In this case, the addition of a breakout force equal to the friction force reduces the poor control "feel" about trim and reduces the trim speed band to zero. There are other advantages to having some breakout in the longitudinal control system. Breakout forces allow the pilot to rest his hand on the control stick without introducing inadvertent longitudinal control inputs.

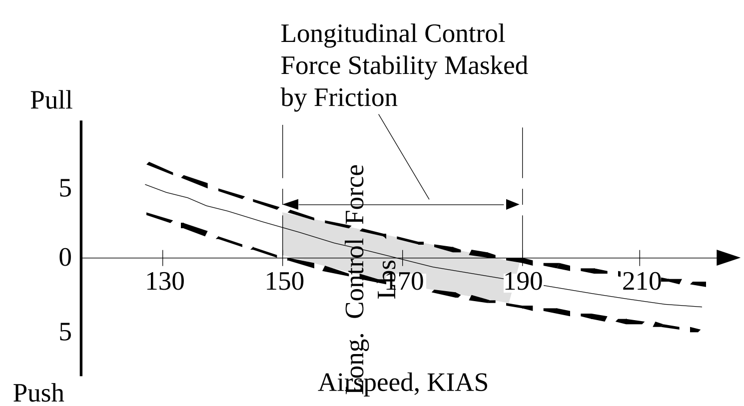


Figure 4.43
Longitudinal Control Friction Masking
Control Force Stability

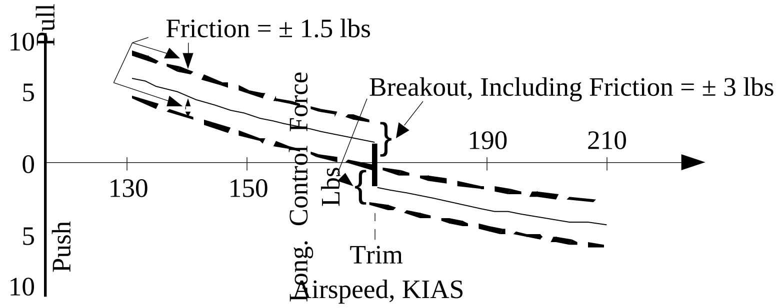


Figure 4.44
Addition of Breakout Force to the Longitudinal Control System

However, breakout forces must not be excessive or longitudinal flying qualities will again be degraded. For example, breakout forces must be suitably matched to the longitudinal control force stability. A combination of high breakout and very shallow longitudinal control force variation with airspeed (Figure 4.45) results in a noticeable control force nonlinearity about trim airspeed. This results in poor control "feel" about the trim airspeed. Since the pilot trims the airplane through stick force "feel", the high breakout force may result in poor trimmability. This is because the pilot has difficulty in determining when his applied stick force is equal to the breakout force (a criterion for trim) if the breakout is large.

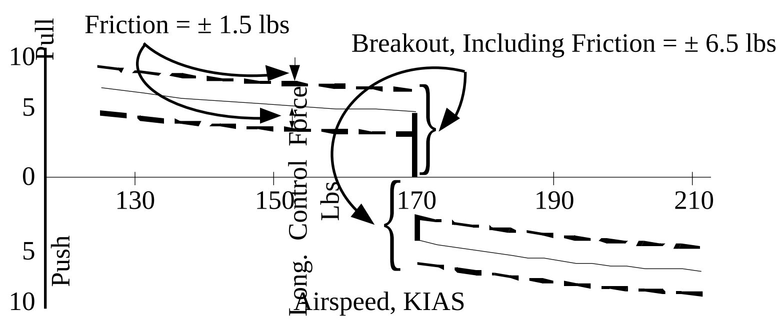


Figure 4.45
Poor Matching of Static Longitudinal Stability and Breakout Forces

In general, friction should be as small as possible in the longitudinal control system; some breakout is generally beneficial, but too much results in undesirable characteristics. Breakout forces, including friction may vary with atmospheric conditions, such as temperature and humidity, as well as trim conditions, such as longitudinal stick position. However, this variance is usually very small. For the irreversible longitudinal control system, breakout forces, including friction, usually do not vary from static condition (on the ground) through the airplane's entire flight envelope.

In should be obvious from studying Figures 4.44 and 4.45 that <u>breakout</u> force can never be measured alone, unless there is zero friction force. Therefore, <u>breakout forces, including friction</u>, are measured at the trim airspeed of the test, and <u>friction alone</u> is measured at stabilized airspeeds above or below trim airspeed. Breakout forces, including friction, are measured in flight with the hand-held force gauge by carefully stabilizing at the trim airspeed, then applying slow and smooth forward and aft longitudinal control forces in turn until movement of the elevator is detected. Movement of the elevator can be detected by visually observing elevator movement, use of an elevator position indicator, or by observing <u>airplane pitch attitude changes</u>. If the latter method must be used, extremely slow and smooth control force inputs should be made in small increments until pitch attitude response is noted. Caution must be exercised because the airplane will require a <u>finite time interval</u> to respond in pitch attitude to the elevator movement. This becomes particularly critical in large, slow responding, transport and patrol airplanes; therefore, for such airplanes, incremental force increases of approximately 1/2 pound are recommended. If automatic recording devices are utilized, breakout forces, including friction may be measured from the recording traces as shown in Figure 4.46. More rapid longitudinal force inputs will allow trace “breakaways” from the trimmed condition to be more easily identified but will aggravate the effect of time delays in the control system (for example, hydraulic actuator response lag) causing an increase in the apparent breakout (including friction) force.

Friction forces may be measured by stabilizing at airspeeds above or below trim airspeed (outside of the influence of breakout). After stabilizing at an airspeed above or below trim, the pilot <u>slowly</u> varies stick force until he observes elevator movement or airplane pitch attitude change. Longitudinal friction will be measured <u>as the difference between maximum and minimum longitudinal control forces required to maintain the stabilized airspeed</u> (see Figures 4.44 and 4.45).

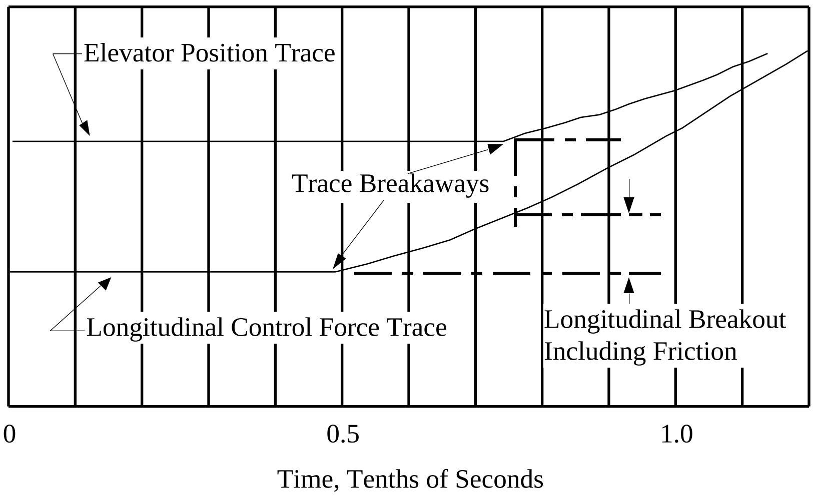


Figure 4.46
Use of Automatic Recording Trace for Determination of Longitudinal Breakout Forces, Including Friction

Breakout, including friction, may be measured on the ground for airplanes equipped with irreversible longitudinal control systems where longitudinal control force is merely a function of longitudinal control deflection. However, ground measurements should be checked with an in-flight measurement. It is obvious that in-flight measurement at various airspeeds is the only means of accurately determining these characteristics for the reversible control system.

4.3.2.2.2 Freeplay

Freeplay in the longitudinal control system should be as small as possible. Excessive freeplay will cause difficulty in performing precise maneuvers such as level accelerations or decelerations and tracking. The pilot will generally resort to flying the airplane “out of trim” during precise maneuvers to avoid the necessity to continually move the longitudinal control stick through excessive freeplay. Freeplay, expressed in inches or degrees of longitudinal cockpit control movement, is measured in flight at the trim airspeed much the same as breakout, including friction, was measured. Ground measurements may also be made for irreversible control systems.

4.3.2.2.3 Centering

Positive centering of the longitudinal control system is shown by an initial tendency of the cockpit control to return towards the trimmed position when released from a displaced position. If the control returns exactly to the trimmed position, then absolute centering is displayed. The longitudinal control system should exhibit positive centering in flight at any stabilized trim airspeed. Poor centering generally results in objectionable tracking characteristics or large departures in airspeed without constant pilot attention to the control of the airplane. Centering is qualitatively evaluated in flight at the trim airspeed by displacing the longitudinal cockpit control smoothly to various positions and observing its motion upon release. If poor centering is apparent, measurements of the difference between the trim position and the position attained after release may be made with automatic recording devices or hand-held cockpit instruments, such as a tape measure. Irreversible control system centering characteristics may be evaluated on the ground.

4.3.2.2.4 Control System Oscillations

Oscillations in the elevator control surface and the entire longitudinal control system, initiated by either external perturbations or pilot action, should be well-damped or deadbeat. Lightly damped or undamped motion can result in annoying and dangerous oscillations in normal acceleration, particularly during flight in turbulent air. Damping of the control system is measured in flight by abruptly deflecting and releasing the longitudinal cockpit control (sometime called a “rap” input) and observing the resulting motion in the control surface and/or the cockpit control stick. Use of automatic recording devices or a cockpit mounted elevator position indicator aides in data acquisition. If these are not available, the test pilot must resort to observing the motion of the cockpit control stick. It must be remembered, however, that motion of the control stick may or may not be the same as motion of the elevator, depending on the amount of freeplay in the longitudinal control system. Irreversible control system oscillation characteristics may be checked on the ground; however, these characteristics should be evaluated in flight to insure there is no coupling between airplane motion and control system dynamics.

4.3.2.3 MEASUREMENT OF LONGITUDINAL CONTROL FORCE STABILITY, ELEVATOR POSITION STABILITY, AND FLIGHT PATH STABILITY

It should be apparent that some degree of longitudinal control force and elevator position stability is desirable. These static longitudinal stability characteristics contribute to good longitudinal trimmability and maintenance of the trimmed condition. In addition, the longitudinal cockpit control forces and motions required in changing flight conditions are simple and natural if longitudinal control force and elevator position stability are present. However, if <u>too much stability</u> is present, the airplane may be <u>very difficult to control</u> in that large longitudinal control forces and position changes may be required to change airspeed in unaccelerated flight. The degree of longitudinal control force and elevator position stability which is desirable or acceptable in any airplane depends on <u>the mission of the airplane</u> and the multitude of pilot tasks required to accomplish that mission. However, it can be rationalized that for the nonmaneuvering tasks under evaluation, some degree of static longitudinal stability is desirable for pleasant longitudinal flying qualities. In addition, plots of longitudinal control force and elevator position versus airspeed should be smooth and their local gradients stable within a reasonable airspeed band about trim airspeed.

It is desirable that when the pilot changes airspeed by use of the elevator control alone, increases in airspeed are accompanied by decreases in flight path angle (γ) (i.e., less climb or more dive) and decreases in airspeed are accompanied by increases in flight path angle (i.e., more climb or less dive). This characteristic is referred to as <u>flight path stability</u> and is really a performance characteristic; i.e., it is dependent on whether the airplane is operating on the “front side,” “back side,” or “flat portion” of the power required curve. However, it may have a major influence on pilot <u>workload</u>, particularly in approach configuration where “back side” or “flat portion” operation may require continuous throttle and longitudinal control inputs to maintain desired airspeed and rate of descent. Flight path stability may be conveniently measured during static longitudinal stability tests by noting <u>rate of climb or rate of descent</u> at each test point. A plot of <u>change in rate of climb or descent</u> versus airspeed indicates flight path stability or instability (Figure 4.47).

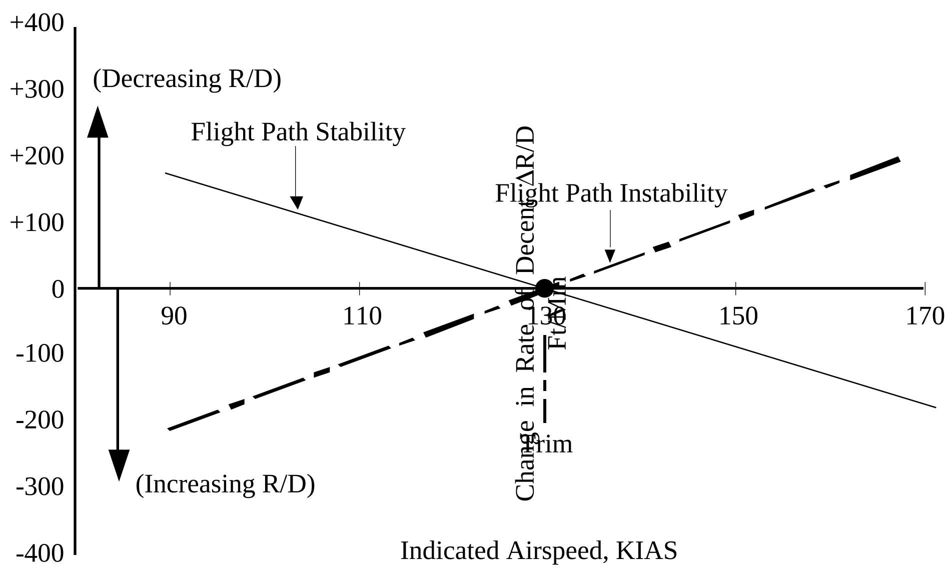


Figure 4.47
Means of Presenting Flight Path Stability Data

In analyzing flight path stability in the landing approach phase, a plot of flight-path angle versus true airspeed is used to determine specification compliance. The requirement is that the flight-path angle versus true-airspeed curve shall have a local slope at the minimum operational approach speed $\left(V_{0_{min}}\right)$ which is negative or less positive than:

a. Level 1 - 0.06 degrees/knot

b. Level 2 - 0.15 degrees/knot

c. Level 3 - 0.24 degrees/knot

The thrust setting shall be that required for the normal approach glide path at $V_{0_{min}}$. The slope of the flight-path angle versus airspeed curve at 5 knots slower than $V_{0_{min}}$ shall not be more than 0.05 degrees per knot more positive than the slope at $V_{0_{min}}$, as illustrated by Figure 4.48.

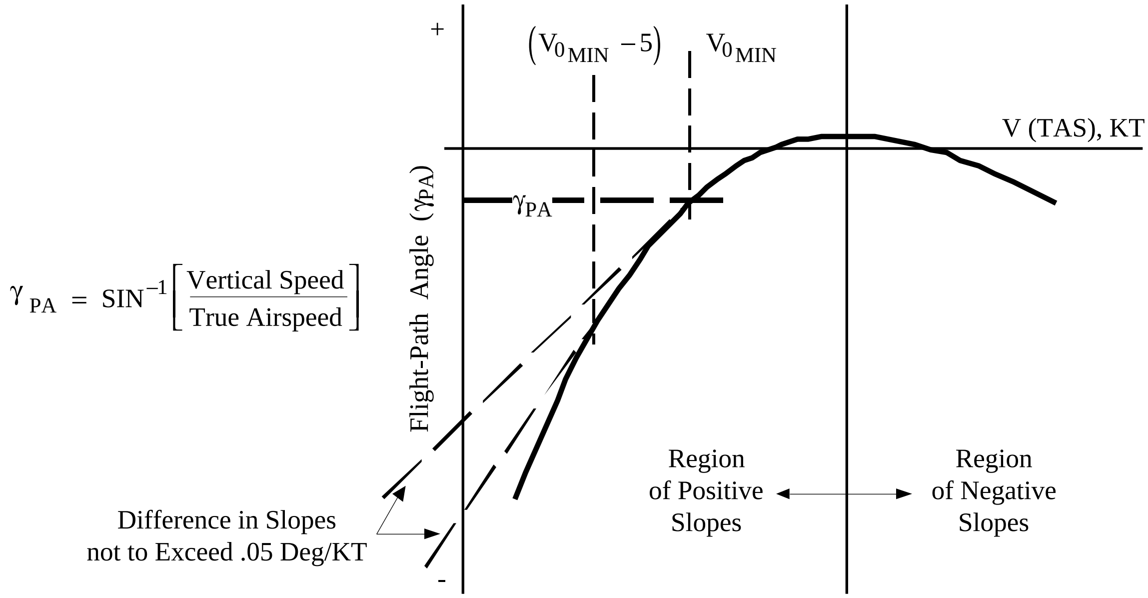


Figure 4.48
Flight Path Stability Data in Configuration PA

Several methods have been utilized for obtaining static longitudinal stability data in flight. Three will be presented here. The first method is called the stabilized point technique. It involves measuring data at a constant power setting and constant trim setting while varying airspeed about trim by varying altitude (or rate of climb or descent) with elevator position. The technique is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, a “trim shot” should be taken. Record appropriate data such as power, longitudinal trim setting, elevator or stick position, and fuel quantity. If a true airspeed indicator is not installed in the test airplane, OAT should be noted to determine true airspeed.

2. Without changing power or trim settings, vary airspeed to predetermined points about trim airspeed by varying altitude. The off-trim speeds used should cover a range of at least ± 15 percent of the trim speed or ± 50 KEAS whichever is less (except, of course, where limited by the service flight envelope). For power approach or land configuration, the range of airspeeds should extend to the stall speed. (A reversal or shallowing of the gradient of longitudinal control

force versus airspeed could precipitate a tendency toward inadvertent stall in these configurations.) It is recommended that at least three equally spaced points (preferably four) be selected at speeds both faster and slower than trim in order to adequately define the relationships to be plotted. At each selected point, airspeed must be carefully and precisely stabilized. After stabilization, a short automatic recording burst should be taken, and/or the following cockpit observations recorded on the data card:

a. Longitudinal control force (maximum and minimum, if friction is measurable, in order to aid in airing the data). Measurements may be made with the hand-held force gauge.

b. Elevator position or longitudinal cockpit control position. Longitudinal cockpit control position may be measured with a tape measure.

c. Rate of climb or rate of descent.

Altitude variance during these tests should not exceed $\pm$ 1000 feet from the base altitude. For configurations requiring power for level flight, acquiring data at first fast and then slow test airspeeds, etc., will facilitate remaining near the base altitude.

The second method is called the Slow Acceleration-Deceleration Technique and can be utilized only with automatic recording devices. However, it allows rapid data acquisition, particularly if the airspeed range is large. This method is not as accurate as the stabilized point method since true equilibrium conditions are never attained except at the trim airspeed. Nevertheless, if the acceleration and deceleration are performed slowly and smoothly (approximately 2 knots/second or less), the data obtained will be adequate for most flight test programs. The technique is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. Record appropriate cockpit data - power, longitudinal trim setting, and fuel quantity.

2. Start the automatic recording device and activate the event marker to denote a "trim shot." Leave the recording device running and initiate a slow acceleration or deceleration by applying a smooth longitudinal control force input. Power and trim settings should remain at the trim conditions. Adjust the acceleration or deceleration rate and actuate the event marker at predetermined airspeeds to simplify data reduction. The visual horizon should be used to maintain a constant acceleration or deceleration with frequent reference to the airspeed indicator. (If the process is done very slowly, or if the airspeed range is large, the pilot may desire to turn off the automatic recording devices between predetermined points.) Continue the acceleration or deceleration to one end of the airspeed range, then reverse longitudinal control force and proceed to the other extreme. The process should always terminate at the trim airspeed.

Data obtained by the Slow Acceleration-Deceleration Technique may be presented as shown in Figure 4.49. This particular method of data presentation, where every point on the plot is a data point, is sometime called "shot gunning" data. Plots like this are easy to derive if automatic data reduction facilities are available; it is apparent that obtaining the same plot by manual data reduction would be extremely laborious.

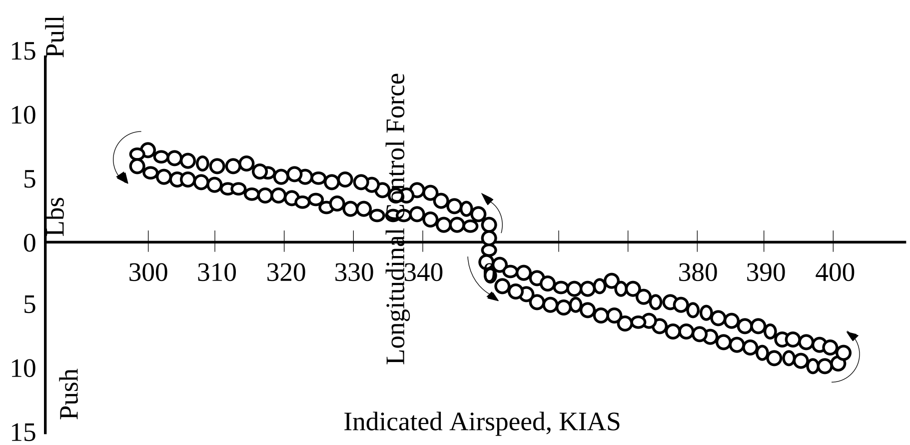


Figure 4.49
Typical Longitudinal Control Force Stability Data from a Slow Acceleration - Deceleration

The third method is called the Power Acceleration-Deceleration Technique and can be utilized only with automatic recording devices. This method is not as accurate as the stabilized point method since true equilibrium conditions are never obtained and power effects are not constant. Power effects can be observed in the data, however, since the data is obtained with maximum and minimum power. The technique is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. Record appropriate cockpit data - power, longitudinal trim setting, and fuel quantity. Start the automatic recording device and activate the event marker to denote a “trim shot”.

2. Retard the throttle slowly and smoothly to idle allowing the airplane to slow down. Leave the recording device running and actuate the event marker at predetermined airspeeds to simplify data reduction. A zero rate of climb is desired but a slight rate of climb or descent is acceptable if it is constant. If the airplane controls are moved back and forth between climbs and descents, additional errors will be introduced into the elevator position and stick force readings. A rapid crosscheck between the visual horizon and the rate of climb indicator will assist in maintaining the proper airplane rotation rate.

3. At predetermined minimum speed slowly and smoothly advance the throttles to full military power (maximum power if desired). As the airplane accelerates again activate the event marker at the predetermined airspeeds. When the maximum desired velocity is reached again slowly and smoothly retard the throttles to idle and transverse the speed range back to trim airspeed.

4. If the airspeed range is large and the acceleration/deceleration rates slow, the pilot may desire to turn off the automatic recording devices between the predetermined points.

4.3.2.4 MEASUREMENT OF LONG PERIOD OR PHUGOID CHARACTERISTICS

Damping and frequency (or period) of the long period longitudinal mode of motion have been shown to have little effect on longitudinal flying qualities in visual flight.[2] Under flight conditions where the pilot maintains close control over pitch attitude, he effectively damps the long period motion before it has a change to cause airspeed or altitude variations.

Unfortunately, pilot opinion of instrument flight characteristics is considerably affected by phugoid damping. Typical adverse effects on instrument flight of decreasing phugoid damping are:

1. Deterioration in the pilot's ability to trim at a precise desired airspeed.

2. Deviations from equilibrium trim conditions (altitude and airspeed) become more frequent and annoying.

3. The pilot's instrument flying technique may change. He may be required to monitor with increasing frequency the airspeed indicator, altimeter, and rate of climb indicator. With extreme "negative damping," the pilot may find it necessary to monitor very closely the horizon bar of the attitude gyro.

One study of the influence of phugoid damping on instrument flight characteristics revealed that the pilot utilized ten times more elevator inputs when the phugoid damping was negative (- 0.23) than when it was positive (+ 0.50). This reflects the increased pilot workload associated with poor long period characteristics.

The test technique for measuring phugoid characteristics (period and damping ratio) is very simple. The airplane is first stabilized and trimmed in the desired configuration at the desired flight condition. The elevator control alone is then used to stabilize at an airspeed approximately 15 KIAS slower or faster than the trim airspeed. Phugoids should

---

[2] However, if the period of the phugoid is low and the period of the airplane short period motion is high - such that $P_P < P_{SP}$, the pilot is likely to continually excite the phugoid in normal manuevering flight. This is generally not the case for most airplanes.

be initiated from airspeeds both slower and faster than trim for each trim airspeed being evaluated. Trim tab settings and power are maintained at the trim condition. The elevator control is then smoothly returned to the trim position and released to initiate the controls free oscillation; the 60-second sweep stopwatch is started simultaneously. The pilot then merely records a time history of airspeed (and altitude, if desired) until enough cycles are completed to define completely the characteristics of the oscillation (at least two cycles). Elapsed time and airspeed should be recorded at minimum airspeed, maximum airspeed, and trim airspeed points. Airspeed changes occur very slowly at minimum and maximum airspeed points; therefore, pitch attitude should be monitored to aid in recording the correct elapsed time at these points. Minimum and maximum airspeeds will occur at the point where pitch attitude is approximately the same as the initial trimmed pitch attitude (refer to Figure 4.24 in the Theory Section).

The pilot must keep the wings of the airplane completely level during the phugoid measurements without introducing any longitudinal control inputs. This may be accomplished by use of rudder inputs, lateral trim inputs, or side pressure on the cockpit control stick.

If the elevator is observed to "float" during the controls free oscillation, the phugoid characteristics should also be measured with the longitudinal cockpit control rigidly restrained in the trim position. (Instead of releasing the cockpit control stick after it is returned to the trim position, it is returned smoothly to the trim position and restrained there.) The effect of elevator float can be seen as the difference between the phugoids performed with controls free and controls fixed.

Automatic recording devices may be utilized to record phugoid characteristics. However, the long period and low damping of the phugoid require continuous operation of the automatic devices for a long time interval. Careful manual data acquisition yields results of useable accuracy.

4.3.2.5 MEASUREMENT OF LONGITUDINAL TRIMMABILITY

Longitudinal trimmability, as related to nonmaneuvering tasks, is indicated by the ease with which the pilot can reduce longitudinal control forces to zero at a precise airspeed and the ability of the airplane to maintain that trimmed condition without pilot attention.

Trimmability depends on all the characteristics previously discussed. In addition, it depends on the rate of operation[3] and sensitivity[4] of the longitudinal trim device as well as the physical location and ease of operation of the trim device in the cockpit.

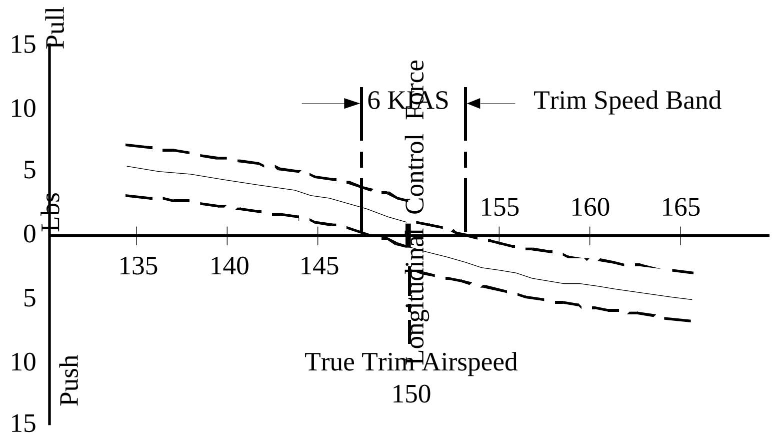
Trimmability determination is mainly a qualitative assessment by the pilot. However, measurement of the "trim speed band" is quantitative and requires some explanation.

The "trim speed band" is bounded by the maximum and minimum airspeeds at which the airplane will stabilize at a given trim setting without pilot applied forces. The technique for determining the "trim speed band" for a given trim airspeed, configuration, and altitude is rather difficult to explain on the round but easy to understand in flight. The airplane is first very carefully trimmed at the desired trim airspeed, configuration, and flight condition. Power and trim tab settings are maintained at the trimmed conditions. A very small longitudinal control force input is then applied in the nose-down direction and the airplane restabilized at an airspeed about 2 KIAS greater than trim speed. The longitudinal control is then released and the pitch attitude response and airspeed response of the airplane noted. If pitch attitude and airspeed remain at the new stabilized conditions, the limits of the trim speed band have not been exceeded. If pitch attitude and airspeed start to return to trim immediately upon releasing the longitudinal control, the limits of the trim speed band have been exceeded. The trim speed band is thus determined by both increasing and decreasing airspeed from trim until the limits are reached. The speed band below and above trim airspeed may not be the same since the trim airspeed may be anywhere in the trim speed band (Figure 4.50).

---

[3]Movement of the trim tab per unit time. This characteristic is independent of the flight conditions.

[4] Response of the trim device to a pilot trim input as indicated by the degree of longitudinal control force response to the trim input. This characteristic is a function of flight condition (dynamic pressure) as well as design of the trim system.



The Trim Airspeed at which the airplane stabilizes initially may be 148 KIAS. The Trim Speed Band measurement would then yield -1 KIAS and +5 KIAS as the limits. However, the same Trim Speed Band of 6 KIAS would be reported.

Figure 4.50
The Trim Speed Band

4.3.3 Postflight Procedures

As soon as possible after returning from the flight, the pilot should write a brief, rough qualitative report of the longitudinal flying qualities exhibited during the mission tasks under evaluation. This report should be written while the events of the flight are fresh in his mind. The qualitative opinion of the test pilot, appropriately related to the mission tasks under evaluation, will be the most important part of the final report of the longitudinal flying qualities.

Appropriate data should be selected to substantiate the pilot's opinion. Methods of data presentation are as numerous as flight test activities. No matter what method is used, the presentation should be clear, concise, and complete. The data presentation to be discussed here is only suggested and may be modified as desired by the test activity.

4.3.3.1 MECHANICAL CHARACTERISTICS OF THE LONGITUDINAL CONTROL SYSTEM

Mechanical characteristics are effectively presented in tabular form as shown in Figure 4.51. Longitudinal control surface damping is also effectively presented on a time history if automatic recording devices are utilized (Figure 4.52).

Breakout, Including Friction Lbs			Freeplay (Fwd & Aft) Inches	Damping Damping Ratio		Centering
Measured (Forward & Aft)	Specification Limits Mil. Spec.________					
	Min	Max		Elevator	Cockpit Control Stick	
$5\frac{1}{2}$	$\frac{1}{2}$	3	$\frac{1}{2}$	Deadbeat	.45	Positive

Figure 4.51
Suggested Table for Presenting Control Mechanical Characteristics

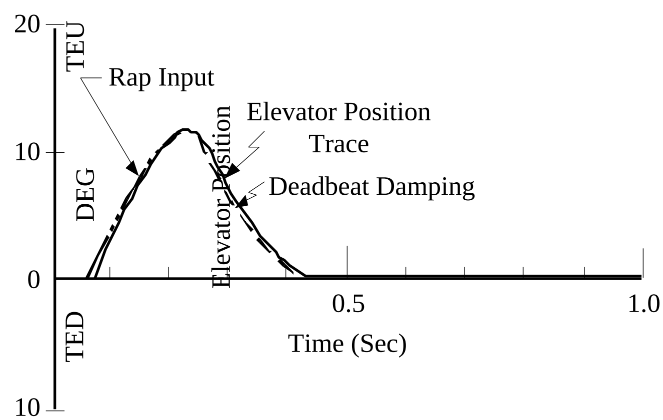
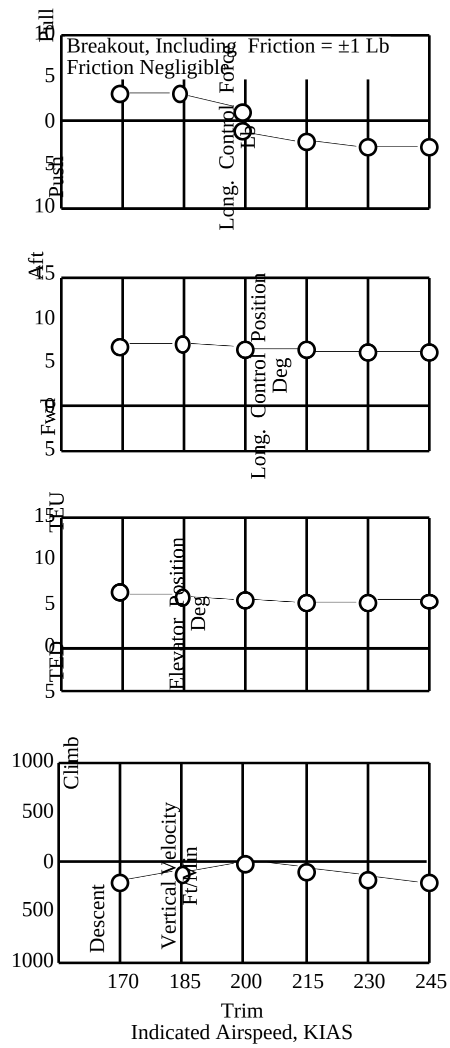


Figure 4.52
Longitudinal Control Surface Damping Presentation

4.3.3.2 STATIC LONGITUDINAL STABILITY CHARACTERISTICS

Static longitudinal stability characteristics may be presented as plots of longitudinal control force stability, elevator position stability, and flight path stability. (Flight path stability need not be presented in all cases. It should be presented for all Power approach configuration tests, however, as previously described in Figure 4.48.) Longitudinal cockpit control position variation with airspeed or Mach number about trim may also be presented. Typical plots are shown in Figure 4.53.



Configuration: Cruise
Loading: Normal Fighter
CG: 36% MAC

Gross Weight: 35,500 Lbs.
Stab Aug: On
Altitude: 3,000 Ft

Figure 4.53
Static Longitudinal Stability Characteristics

The effectiveness of longitudinal stability plots depends a great deal on the scales chosen. The gradients may be made to appear steep or shallow merely by changing the scale relationship of horizontal and vertical axes. Scales should be chosen so that the plots are compatible with the pilot's qualitative opinion; i.e., if the longitudinal control force variation with airspeed felt “light” or “shallow” to the pilot, scales should be chosen so that the relationship appears shallow.

Discussion of static longitudinal stability characteristics in the report of the test must be worded with care. The report must not imply that some characteristic was measured, where, in actuality, flight test data only indicated the characteristic. For example, in flight test work, the parameters recorded, such as longitudinal control force and elevator position variations with airspeed about trim, are only indications of stick-free and stick-fixed static longitudinal stability, respectively. In general, the use of the terms “stick-fixed,” “elevator-fixed,” “stick-free” and “elevator-free” is not recommended for the reporting of static longitudinal stability characteristics determined from flight tests. The terms are used extensively, of course, in text books, classroom work, and wind tunnel investigations. The language of the report should reflect the parameters which were actually measured. As an example, the following introductory sentence might be used in the report: “Static longitudinal stability, as indicated by the variation of longitudinal control force and elevator position about the trim airspeed, was slightly positive in all configurations tested.”

Caution must also be exercised when discussing the gradient changes which commonly occur at airspeeds below and above trim airspeed. As an example, the longitudinal control force stability plot of Figure 4.53 shows a reversal in the gradient at approximately 145 KIAS. This gradient reversal should most emphatically not be reported as static longitudinal instability, longitudinal control force instability, etc. If the author desires to discuss the reversal in gradient, it should be reported exactly as its exists; i.e., “The variation of longitudinal control force with airspeed exhibited a slightly stable gradient through trim airspeed; however, the gradient reversed smoothly at 15 KIAS faster than trim and push forces decreased to one pound at 30 KIAS faster than trim.”

Static longitudinal stability data is sometimes presented in tabular form when many loadings, configurations, altitudes, trim airspeeds, and CG positions have been utilized. An example is represented in Figure 4.54. Expression of longitudinal control force gradients in “pounds per knot” has particular merit for comparing the static stability characteristics for various configurations and CG positions.

Loading	Configuration	Altitude (ft)	Trim Airspeed (KCAS/M)	CG Position (% MAC)	Gradient (1) (lb/kt)
A	CR	40,000	.77	21.6$^{(3)}$	.18
A	P	40,000	.86	24.7	0
A	P	35,000	.91	18.1	-.15 >trim 0 < trim
C	CR	30,000	.67	23.3	.12
C	CR	30,000	.77	22.9	.09
A	CR	20,000	277/.60	25.2$^{(3)}$	0 > trim .20 < trim
A	G	20,000	211	22.9	0 > trim .19 < trim
A	D$^{(2)}$	20,000	212	24.2	.16
A	CR	10,000	261	15.5$^{(3)}$	.03 > trim .13 < trim
A	CR	10,000	289	25.5$^{(3)}$	.0 > trim .09 < trim
A	P	10,000	454/.83	21.0	0
A	P	10,000	461/.83	16.8	0 > trim .23 < trim
A	P	10,000	524/.94	24.1$^{(3)}$	-.24 > trim .07 < trim
C	CR	10,000	291	25.3	.04
C	CR	10,000	297	20.5	.13
C	P	10,000	445/.80	23.2	.09
A	TO	10,000	139	26.5$^{(3)}$	-.13 > trim .15 < trim
C	TO	10,000	166	25.4$^{(3)}$	.13
A	PA	10,000	147	21.7	.06
C	PA	10,000	135	21.5	.20
A	WO	10,000	116	22.2	.14
C	WO	10,000	124	24.1	.24

(1) Longitudinal Control Force Gradient Through Trim Airspeed. Stable Gradients Carry a Positive Sign; Unstable Gradients Carry a Negative Sign.

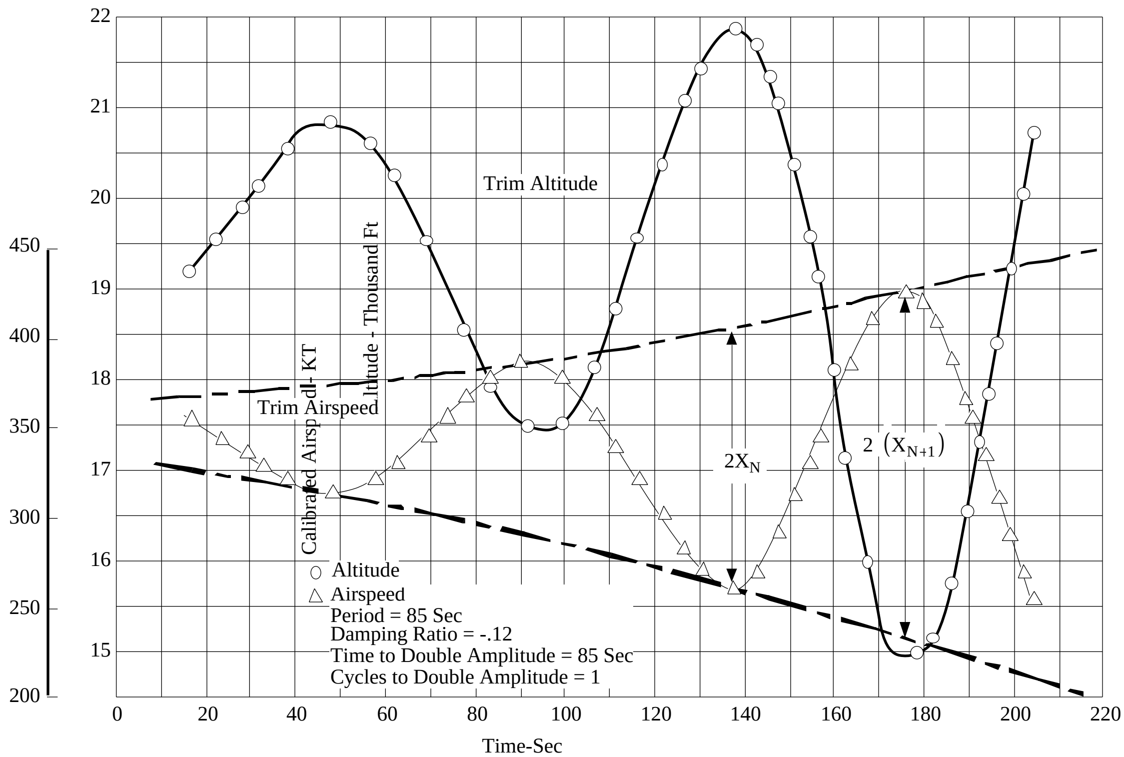
(2) IDLE Thrust.

(3) See CG-Gross Weight Relationship Shown in Appendix IV, Figure 1.

Figure 4.54
Static Longitudinal Stability Table

4.3.3.3 DYNAMIC LONGITUDINAL STABILITY CHARACTERISTICS - LONG PERIOD OSCILLATIONS

The phugoid or long period data are presented as time histories of airspeed for the controls free and controls fixed (if applicable) oscillations. Altitude may also be plotted on the time histories if desired. An example of long period data presentation is shown in Figure 4.55. Characteristics of the oscillation (period, damping ratio, time-to-half amplitude or time-to-double amplitude, cycles-to-half amplitude or cycles-to-double) may be presented on the time history if desired. The altitude or airspeed variation on the time history may be used to measure the characteristics; in Figure 4.55, the airspeed trace was arbitrarily chosen. Note that the phugoid motion oscillated about airspeed and altitude different than the assumed trim conditions. This could be caused by a large trim speed band or poor longitudinal control centering. However, the data is stall usable.



Model______Airplane
BuNo______

Configuration: CR	CG: 35.2% Mac
Loading: C	Gross Weight: 35,500 Lb
	Stab. Aug: On
	Method: Controls Free

Figure 4.55
Long Period Characteristics

4.3.3.4 LONGITUDINAL TRIMMABILITY

The determination of trimmability as presented herein is based on the test pilot's qualitative opinion. Therefore, a qualitative discussion of trimmability in the technical report is appropriate. The trim airspeed band may be shown on the plots of longitudinal control force stability, if the author desires (refer again to Figure 4.50).

4.3.3.5 DETERMINATION OF THE ELEVATOR POSITION AND LONGITUDINAL CONTROL FORCE NEUTRAL POINTS

If plots of longitudinal control force and elevator position stability have been obtained at more than one CG position for one configuration, altitude, trim airspeed, and power setting, neutral points may be computed. In order to obtain the most accurate neutral points, the following points should be remembered in the conduct of the static longitudinal stability tests:

1. Although tests at only two CG's are theoretically sufficient to obtain neutral points, the flight test engineer should insist that the airplane is tested with the CG in at least three widely separated locations. if it is feasible to place the most aft CG behind the neutral point, the neutral point can be determined by interpolation vice extrapolation, which should improve the overall accuracy.

2. The CG positions chosen should be evenly spaced (if possible) since even increments of CG travel result in even increments of elevator position at the same lift coefficient (see Figure 4.56). This fact aids in fairing plots of elevator position versus lift coefficient.

3. Airplane gross weight should be maintained near constant for the various CG positions, since the neutral point may vary with angle of attack and power setting. This is not too critical, however, since plotting the variables (elevator position and longitudinal control force) versus lift coefficient tends to eliminate the effect of variation in gross weight.

4. The in-flight tests are performed exactly as previously described for longitudinal control force and elevator position stability. Power and trim setting should not be altered as airspeed is changed about the trim conditions. In addition, power

should be the same at trim for all CG positions tested to eliminate power effects. The small differences in climb or descent angles which result should not have a significant influence on the accuracy of the data.

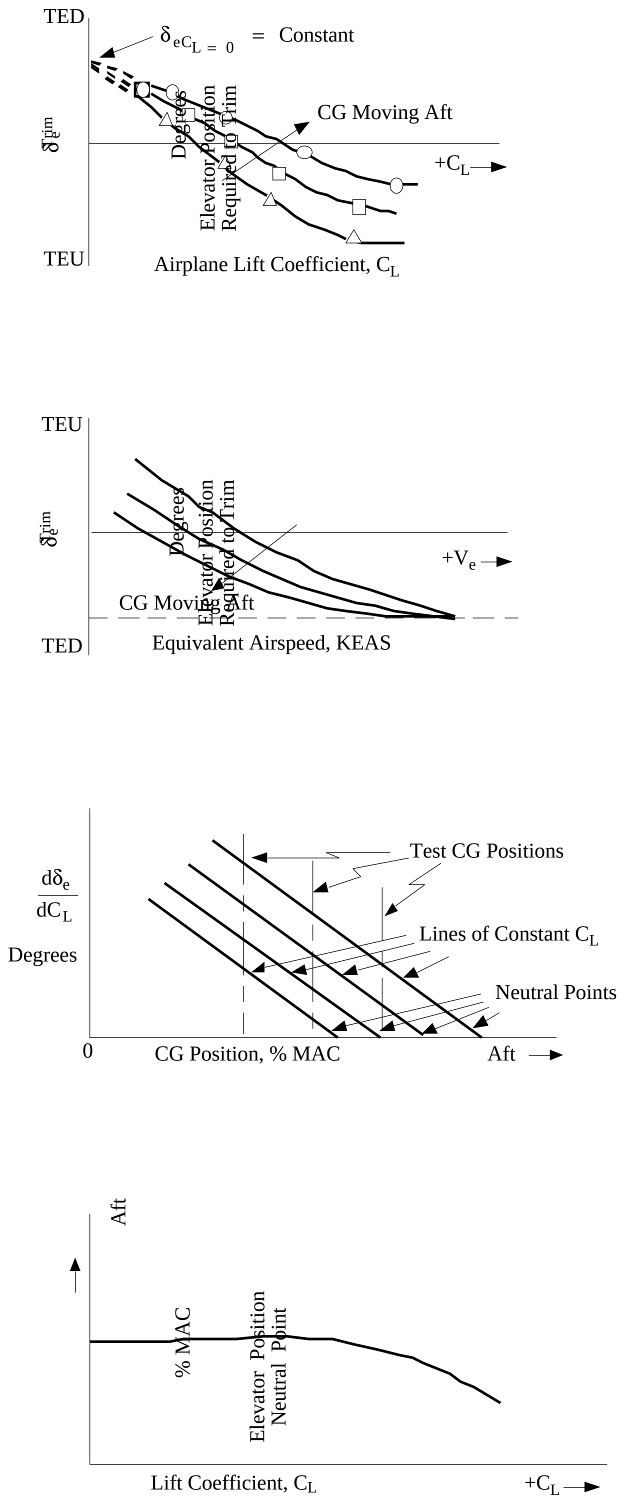
5. The pilot may have difficulty in attaining the same trim airspeed for the various CG positions tested if the trim speed band of the airplane is large. This will be evident in the longitudinal control force versus airspeed data. However, this inconsistency is eliminated when the curves of $\frac{F_s}{q}$ are replotted versus lift coefficient. The slopes of such curves are independent of the initial trim condition.

6. Accurate elevator position data is easy to obtain. However, accurate longitudinal control force data is extremely difficult to obtain because of friction in the control system. The test pilot must exercise care to insure that the forces measured are correct. If friction is large, maximum and minimum forces for equilibrium conditions must be measured at each stabilized point faster and slower than trim airspeed.

7. Data accuracy will be enhanced if <u>sideslip</u> is maintained constant as airspeed is varied about trim. Changes in pitching moments are generated by the horizontal tail moving in relation to the slipstream of propeller driven airplanes. This is not a particularly important point in testing pure jet airplanes.

8. Plotting the elevator position data versus lift coefficient vice airspeed tends to linearize the relationships. Also, the elevator position at zero lift coefficient is a constant. This is a useful fact in fairing the curves.

The graphical determinations of the elevator position and longitudinal control force neutral points are presented in Figure 4.56 and 4.57.



(1) Determine C_L from various values of V_e.

$$\left(C_L = \frac{W/S}{\frac{1}{2}\rho_{ssl} V_e^2} \right)$$

(2) Plot δ_e versus C_L for all CG positions tested.

(3) Fair lines using rules shown on figure.

(4) Using selected C_L values from the faired δ_e versus C_L curve, determine V_e at each CG position.

(5) Plot δ_e versus V_e for all CG positions tested.

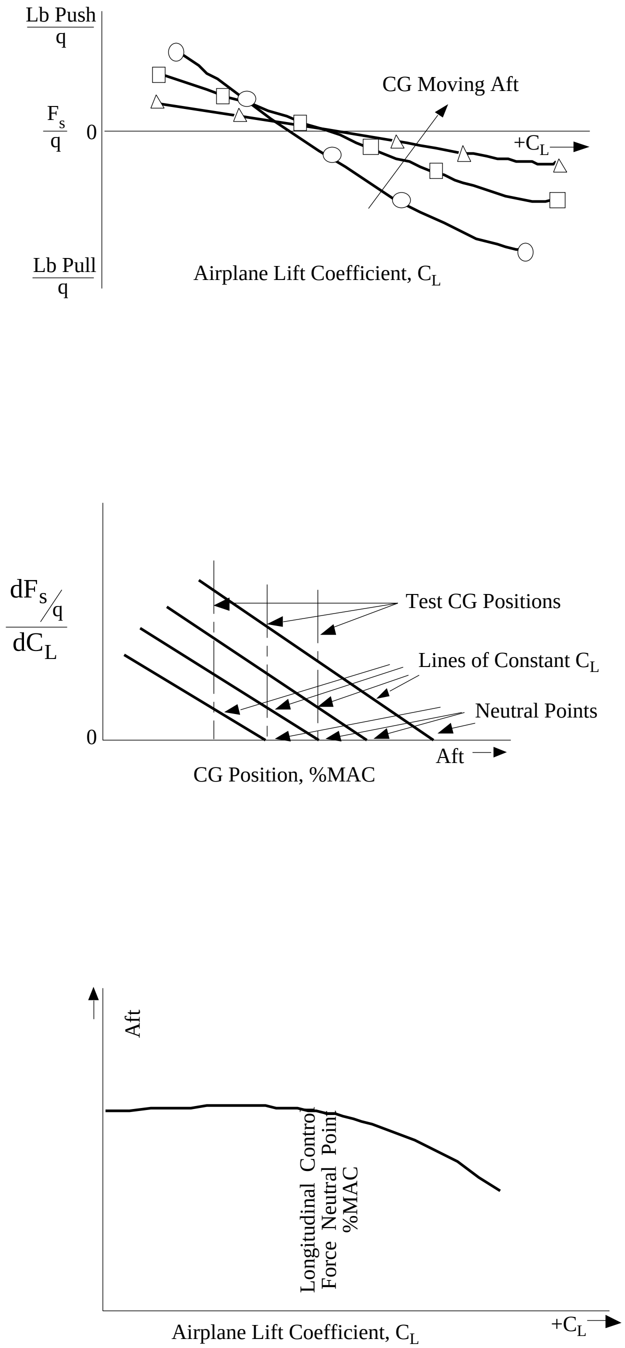
(6) From the δ_e versus C_L curves, at selected C_L and CG values, determine $\frac{d\delta e}{dC_L}$. If relationships are linear, $\frac{d\delta e}{dC_L}$ is constant for given CG position. Therefore, the neutral point is independent of C_L.

(7) Plot $\frac{d\delta e}{dC_L}$ versus CG for various lines of constant C_L.

(8) CG positions where $\frac{d\delta e}{dC_L}$ is zero for each CL is the neutral point for that C_L. Determine neutral points for several C_L values.

(9) Plot neutral points versus C_L.

Figure 4.56
Graphical Determination of the Elevator Position Neutral Point



(1) Compute $q = \frac{1}{2} \rho_{ssl} V_e^2$

(2) For each CG value, compute $C_L = \frac{W/S}{q}$

(3) Compute $\frac{F_s}{q}$

(4) Plot $\frac{F_s}{q}$ versus C_L for each CG position (include breakout including friction for irreversible control systems).

(5) Measure the slopes, $\frac{dF_s/q}{dC_L}$, and plot as a function of CG position for various C_L values.

(6) The CG positions where $\frac{dF_s/q}{dC_L}$ is zero for each C_L is the control force neutral point for that C_L. Determine neutral points for several C_L values.

(7) Plot control force neutral points versus C_L.

(8) Also plot F_s versus V_e for each CG position (include breakout including friction for irreversible control systems). This plot will aid the reader's understanding of the data.

Figure 4.57
Graphical Determination of Longitudinal Control Force Neutral Point

The elevator position neutral point is exactly the same as the stick- or elevator-fixed neutral point if these neutral points are defined by a neutral gradient of airplane pitching moment coefficient versus airplane lift coefficient. The longitudinal control force neutral point is almost always never exactly the same as the stick- or elevator-free neutral point because of gadgetry in the longitudinal control system. Adding various gadgetry results in an additional term being added to the longitudinal control force equation and an additional value being added to the relationship $\frac{dF_s/q}{dC_L}$.

4.4 SPECIFICATION REQUIREMENTS

Requirements for static and dynamic longitudinal flying qualities during nonmaneuvering tasks are contained in the following applicable paragraphs of Military Specification MIL-F-8785B(ASG), of 7 August 1969, hereafter referred to as the Specification.

3.2 Longitudinal Flying Qualities (except 3.2.1.1.1)

- 3.2.3.1 Longitudinal control in unaccelerated flight
- 3.5.2 Mechanical Characteristics (control system) (as applicable)
- 3.5.3 Dynamic Characteristics (as applicable)
- 3.5.4 Augmentation Systems (as applicable)
- 3.5.5 Failure of augmentation systems
- 3.6.1 Trim System (as applicable)

6.2 Definitions

6.5 Engine Considerations

6.6 Effects of aeroelasticity, control equipment, and structural dynamics.

The requirements of the Specification may be modified by the applicable airplane Detail Specification. Comments concerning individual requirements of the Specification are presented below.

3.2.1.1.1 Exception in Transonic Flight This paragraph is self-explanatory. It will be discussed more thoroughly ina subsequent section on Transonic and Supersonic Flying Qualities.

3.2.1.3 Flight-Path Stability The intent of this paragraph is to prohibit rapidly increasing descent rates at airspeeds below normal approach speed that might result in dangerous flight conditions or require excessive pilot workload to maintain glide path. From previous investigations, it has been determined that if the slope requirements of this paragraph are met, the pilot will be able to effectively use the elevator control alone to make small glide path adjustments.

4.5 NONMANEUVERING TASKS - GLOSSARY

4.5.1 Terms

Mode of Motion — Manner of doing, method. In this case, a method of changing flight conditions in the airplane's plane of symmetry.

Frequency — Number of cycles per unit time. A measure of the “quickness” of the motion.

Period — Time required per cycle. Inversely proportional to frequency.

Damping — Progressive diminishing in amplitude. A measure of the subsidence of the motion when excited.

Nonmaneuvering Tasks — Those tasks during which the transition from one equilibrium flight condition to another is accomplished smoothly and gradually; results in essentially unaccelerated flight conditions.

Incidence — The acute angle between a chord of an airfoil and the longitudinal axis of the airplane.

Tail Volume Coefficient — A measure of the size and location of the horizontal tail in relation to the size of the wing and the airplane center of gravity, respectively.

Tail Efficiency Factor — A measure of the modification in energy level of the airflow between the point where the airflow first encounters the airplane until it reaches the horizontal tail.

Neutral Point	The location of the center of gravity of an aircraft for which static longitudinal stability would be neutral. The neutral point may be described as "stick-fixed," "stick-free," "elevator-fixed," "elevator-free," "elevator position," or "longitudinal control force" depending on the manner in which it was determined.
Static Margin	The distance between the actual center of gravity and the neutral point of the airplane usually expressed as a percentage of the mean aerodynamic chord.
Longitudinal Control Power	A measure of the pitching moment coefficient change per degree deflection of the longitudinal control surface.
Float	As applied to the control surface of a reversible control system: to ride in the airstream.
Aerodynamic Balancing	Methods of controlling the magnitude of the hinge moment parameters.
Undamped Natural Frequency	The frequency of a dynamic system if zero damping is exhibited.
Damping Ratio	Ratio of the damping exhibited to the critical damping.
Spring Constant	As applied to a dynamic system, a measure of the static restoring tendency.
Aperiodic; Deadbeat	A motion which does not exhibit periodic oscillations.
Oscillatory	Characterized by periodic motion.

4.6 NONMANEUVERING TASKS - REFERENCES

1. *Advisory Group for Aeronautical Research and Development,* Flight Test Manual, Volume II, Stability and Control, by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control,* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training,* by F.D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F-1, September, 1962.

4. *Dynamics of Flight, Stability, and Control,* by Bernard Etkin, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

5. *Flight Evaluations of the Effect of Variable Phugoid Damping in a KTB-26B Airplane,* by Fred Newell and Donald W. Rhoads, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, WADC TR 56-223, Report No. TB-969-F-1, October 1956.

6. *Flight Evaluation of Various Phugoid Dynamics and* $1/T_{h_1}$ *Values for the Landing Approach Tasks;* by Charles R. Chalk, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. TC-1921-F-4, February, 1966.

7. *Flight Testing of Aircraft,* by M.G. Kotik, et al., NASA Technical Translation F-442, April, 1967.

8. *Handling and Performance Exercises Test Techniques,* Empire Test Pilots' School, Flying Department Notes.

9. *Investigation of Phugoid Effects on GCA Landings,* by Fred Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, WADC TR 57-650, Report No. TB-1142-F-1, December, 1957.

10. *Military Specification, Flying Qualities of Piloted Airplanes,* MIL-F-8785B(ASG), 7 August 1969.

11. *Minimum Flyable Longitudinal Handling Qualities of Airplanes,* by Gifford Bull, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Report No. TB-1313-F-1, December, 1959.

12. *Modern Flight Dynamics,* by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

13. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, U.S. Government Printing Office, Washington, 1959.

14. *Recommendations for Revision of MIL-F-8785(ASG),* "Military Specification - Flying Qualities of Piloted Airplanes," by C.R. Chalk, T.P. Neal, T.M. Harris, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Technical Report AFFDL-TR-67, March, 1967.

15. *Stability and Control,* USAF Aerospace Research Pilot School, Edwards Air Force Base, California, FTC-TIH-64-2004, revised November, 1964.

16. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

17. *Stability and Control*, Empire Test Pilots' School Ground School Notes, Book "C".

18. *Substantiation Information for Selected Longitudinal Requirements,* Cornell Aeronautical Laboratory Report No. BM-2238-F-1, May, 1967.

4.7 THEORY - MANEUVERING TASKS

4.7.1 Static Longitudinal Stability and Control in Accelerated Flight

The previous discussion on static longitudinal stability and control considered the airplane flying on equilibrium, unaccelerated flight paths. It is now necessary to study static longitudinal stability and control along curved flight paths. Obviously, every airplane must be capable of turning, at least to some degree. The subject of turning performance will not be considered here, although it is a subject of major interest in the performance testing of many airplanes. The assumption is made that the turning performance of the airplane is not limited by stability and control characteristics, but by engine or airframe characteristics. The areas of interest in this discussion are the static longitudinal stability and control characteristics exhibited by the airplane when it is subjected to accelerated flight conditions with the lift greater than or less than the weight; i.e., in maneuvering flight. It is essential that airplanes exhibit stability and controllability in maneuvering flight along curved flight paths. Obviously, if the mission of the airplane involves a great deal of maneuvering, the investigation of longitudinal maneuvering stability and control will consume a considerable amount of flight test time. Since all airplanes are required to perform some maneuvering, it is necessary to investigate these characteristics to some extent in every airplane.

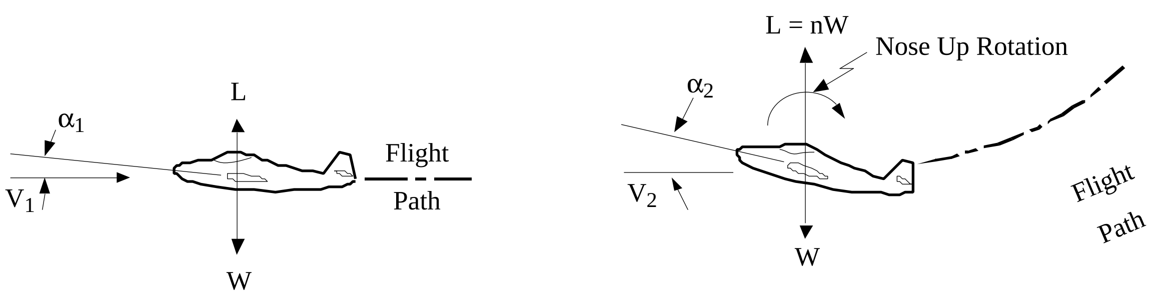
The flight path of the airplane may be curved by the pilot by performing wings level pull-ups or push-overs or by banking the airplane or by performing a combination of these maneuvers. For this study of static longitudinal stability and control in accelerated flight, consideration will be given to steady pull-ups and steady turns at a constant airspeed. Several relationships will be developed first for the steady pull-up maneuver; these relationships will then be expanded to the steady turn maneuver. The relationships for steady pull-ups are applicable also to steady push-overs.

4.7.1.1 STEADY PULL-UPS

Consider that the airplane is initially trimmed in straight and level flight. If a climb, then a dive with a wings level pull-out at the bottom are performed such that (at least for an instant) the original trimmed values of altitude and airspeed are regained, the airplane's original equilibrium conditions will have been modified in two ways (Figure 4.58).

1. The angle of attack and lift coefficient will be greater since extra lift is required to sustain the curved flight path.

2. The airplane will exhibit a steady nose-up rate of rotation (pitch rate) about its center of gravity. This pitch rate will be equal in magnitude to the rate of rotation of the airplane about the center of the pull-out.



(a) Level Flight
L = W
n = 1

(b) Manuevering Flight
$V_2 = V_1$
$\alpha_2 = \alpha_1$
L > W
n > 1

n = Normal Acceleration

Figure 4.58
Relationships of Unaccelerated and Accelerated Flight

Both these modifications generate changes in pitching moments about the airplane center of gravity which hopefully act so as to tend to restore the original unaccelerated flight condition. Of course, in order to stabilize the airplane in the accelerated maneuver, the pilot applies and holds the necessary amount of elevator deflection and longitudinal control force. Thus, the amount of elevator deflection and longitudinal control force required are indications of static longitudinal stability of the airplane in maneuvering flight. The increase in lift coefficient generates changes in pitching moments according to the familiar airplane static stability criterion in unaccelerated flight, $\frac{dC_m}{dC_L}$. The fact that the airplane is rotating generates an additional contribution to the total pitching moment. This contribution, usually quite powerful, is a result of the increase in effective angle of attack at the horizontal tail, due to the horizontal tail moving downward relative to the air (Figure 4.59). (The air may be considered to be moving upward relative to the tail.) The change in effective angle of attack at the horizontal tail during maneuvering generally contributes greatly to the stability of the airplane in accelerated flight. It can readily be seen that this contribution is directly dependent on the airplane pitch rate, $\dot{\theta}$, if airspeed is held constant. For the pull-up maneuver just described, a constant angle of attack pull-up, the magnitude of the pitch rate is a function only of normal acceleration if airspeed is held constant:

$$\dot{\theta}_{\text{pull-up}} = \frac{g\ (n-1)}{V} \qquad \textit{eq 4.27}$$

Where:

$\dot{\theta}$ = pitch rate, radians per second.

g = acceleration due to gravity, $\frac{\text{ft}}{\text{sec}^2}$.

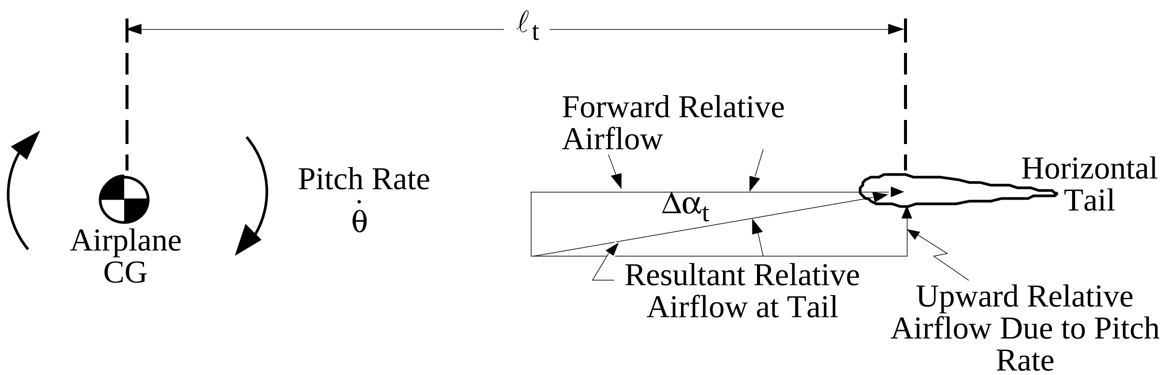
n = normal acceleration, g.

V = True airspeed, $\frac{\text{ft}}{\text{sec}}$.

Similarly, for the steady level turn, the magnitude of the pitch rate is expressed as follows:

$$\dot{\theta}_{\text{steady level turn}} = \frac{g}{V}\left(n - \frac{1}{n}\right) \qquad \textit{eq 4.28}$$

Because it is the pitch rate which causes the pilot to use more or less elevator deflection and longitudinal control force during maneuvering flight than was required during nonmaneuvering flight, the airplane normal acceleration, n, is generally used as the independent variable in maneuvering flight. This is a direct result of the last two equations, which show the fundamental relationships between pitch rate and normal acceleration in maneuvering flight. Thus, in flight test work, the parameters "elevator position per g" and "longitudinal control force per g" are measured at a constant airspeed as classical indexes of the longitudinal maneuvering stability of the airplane.



$\Delta\alpha_t$ = Effective Increase in Angle of Attack of Horizontal Tail

Figure 4.59
The Horizontal Tail Angle of Attack Changes with Pitch Rate

4.7.1.2 ELEVATOR POSITION MANEUVERING LONGITUDINAL STABILITY

The elevator positions required to stabilize the airplane at various values of lift coefficient in accelerated flight at a constant airspeed are generally not the same as the elevator positions required at various values of lift coefficient in unaccelerated flight. As stated previously, the angular rotation of the airplane in pitch during curvilinear flight creates an additional increment in effective tail angle of attack, which in turn generates an additional pitching moment about the center of gravity. This pitching moment may be expressed as:

$$M_{CG_{\text{Due to }\dot{\theta}}} = -a_t \frac{\ell_t^{2}\ \dot{\theta}}{V} q_t S_t \qquad \textit{eq 4.29}$$

Where:

a_t = lift curve slope of horizontal tail.

ℓ_t = "tail arm" length, in feet.

q_t = dynamic pressure at horizontal tail, in pounds per square feet.

S_t = area of horizontal tail, in square feet.

(Note that tail arm length has a powerful influence on the magnitude of the pitching moment).

In nondimensional coefficient form, "pitch rate damping," "damping in pitch," or "viscous damping in pitch" may be defined as:

$$C_{m_{\dot{\theta}}} = \frac{\partial C_{m_{CG}}}{\partial\left(\frac{\dot{\theta}\bar{c}}{2V}\right)} = -2a_t \; \eta_t \; \bar{V} \; \frac{\ell_t}{\bar{c}} \qquad \text{eq 4.30}$$

Where:

$\frac{\dot{\theta}\bar{c}}{2V}$ = the nondimensional pitch rate.

η_t = tail efficiency factor, nondimensional.

$\bar{V}$ = tail volume coefficient, nondimensional.

$\bar{c}$ = average chord length of wing in feet.

Without derivation, the elevator position required in steady, wings level pull-ups at a constant airspeed may be expressed as:

$$\delta_{e_{\text{Pull-Ups}}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}} \; \frac{W/S}{\frac{1}{2}\rho_{SSL} V_e^{\,2}} \left\{ \left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} n + \frac{C_{m_{\dot{\theta}}}\rho g \bar{c}}{4 \; W/S} (n - 1) \right\}$$

eq 4.31

Where:

δ_{e_0}	=	elevator angle required for zero lift coefficient and zero pitch rate; a constant.
$C_{m_{\delta_e}}$	=	elevator control power.
$\left(\frac{dC_m}{dC_L}\right)_{Fixed}$	=	stick-fixed static longitudinal stability.

The derivative of the last equation with respect to normal acceleration yields a classical index of longitudinal maneuvering stability for steady, wings level pull-ups at a constant airspeed:

$$\left(\frac{d\delta_e}{dn}\right)_{Pull-Ups} = -\frac{1}{C_{m_{\delta_e}}}\,\frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2}\left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} + \frac{\rho g\bar{c}}{4\,W/S}\,C_{m_{\dot{\theta}}}\right\} \qquad \textit{eq 4.32}$$

Similarly, the elevator position required in <u>steady turns</u> at constant airspeed may be expressed as:

$$\delta_{e_{Steady\ Turns}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}}\,\frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2}\left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} + \frac{C_{m_{\dot{\theta}}}\rho g\bar{c}}{4\,W/S}\left(n - \frac{1}{n}\right)\right\}$$

eq 4.33

Note that the only difference between the expressions for elevator angle required in pull-ups and steady turns arises from the difference in expressions for pitch rate, $\dot{\theta}$, presented earlier. Taking the derivative of the last equation with respect to normal acceleration yields the classical index of longitudinal maneuvering stability for steady turns at a constant airspeed:

$$\left(\frac{d\delta_e}{dn}\right)_{Steady\ Turns} = -\frac{1}{C_{m_{\delta_e}}}\,\frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2}\left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} + \frac{C_{m_{\dot{\theta}}}\rho g\bar{c}}{4\,W/S}\left(n + \frac{1}{n^2}\right)\right\}$$

eq 4.34

Several important observations may be derived from a study of the equations for $\left(\frac{d\delta_e}{dn}\right)_{Pull-Ups}$ and $\left(\frac{d\delta_e}{dn}\right)_{Steady\ Turns}$:

1. For longitudinal stability in maneuvering flight, $\left(\frac{d\delta_e}{dn}\right)$ must carry a negative sign; i.e., to stabilize the airplane at a higher value of positive normal acceleration, more trailing edge up (TEU - negative direction) elevator must be applied.

2. Both equations contain two parts (See Figure 4.60). The first part is a "stability term" due to the stick-fixed static longitudinal stability; the second part is a "damping term" arising from the change in effective angle of attack at the horizontal tail due to the pitch rate.

3. A little more elevator is required to pull the same normal acceleration increment in steady turns than in pull-ups. The difference in the gradient of elevator position versus normal acceleration is directly proportional to $\frac{1}{n^2}$; therefore the difference becomes very small at high levels of normal acceleration. $\left(\text{If } n = 5, \frac{1}{n^2} = .04\right)$. (See Figure 4.61).

Center of gravity movement naturally has a profound effect on maneuvering longitudinal stability through both the stability term and the damping term (Figure 4.62). As the CG is moved aft, $\left(\frac{dC_m}{dC_L}\right)_{Fixed}$ becomes smaller in magnitude. When the CG is at the stick-fixed neutral pint, the gradient $\left(\frac{d\delta_e}{dn}\right)$ is only a function of the damping term. (The damping term decreases slightly in magnitude as the CG is moved aft because the tail arm length is decreased.) If the CG is moved far enough aft, the gradient $\left(\frac{d\delta_e}{dn}\right)$ becomes zero; this CG position is called the stick-fixed maneuvering neutral point, N_M. Thus, the stick-fixed maneuvering neutral point N_M, should always be aft of the stick-fixed neutral point N_0, if $\frac{dC_m}{dC_L}$ in level flight is the same in maneuvering flight.

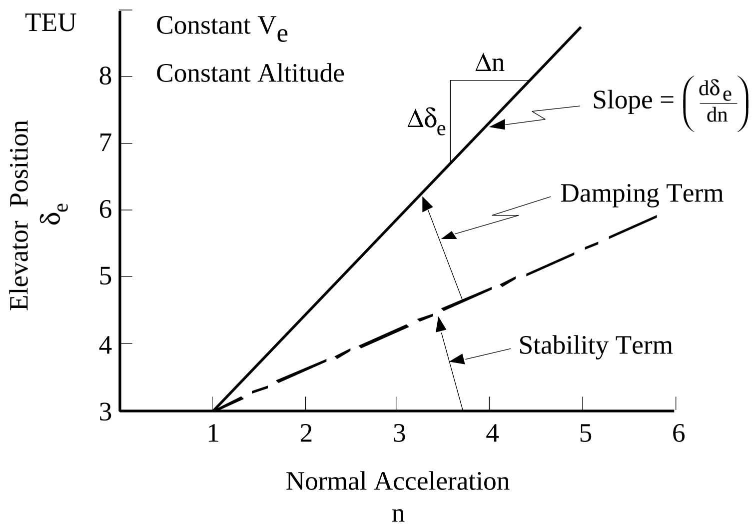


Figure 4.60
Classical Variation of Elevator Position in Maneuvering Flight

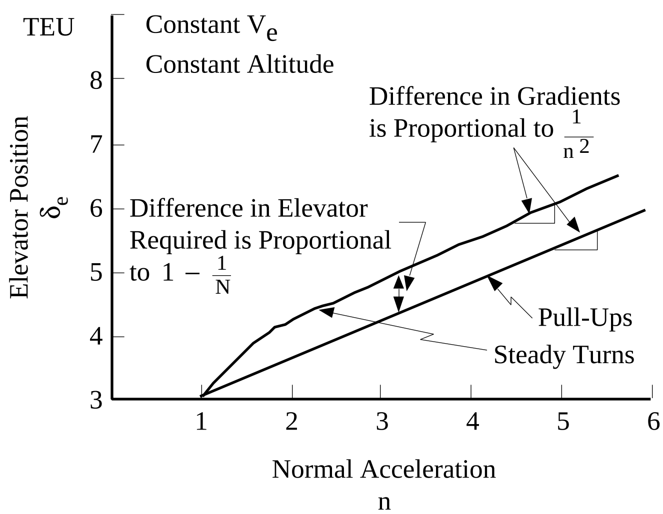


Figure 4.61
Relationships of Maneuvering Stability Characteristics in Steady Turns and Pull-Ups

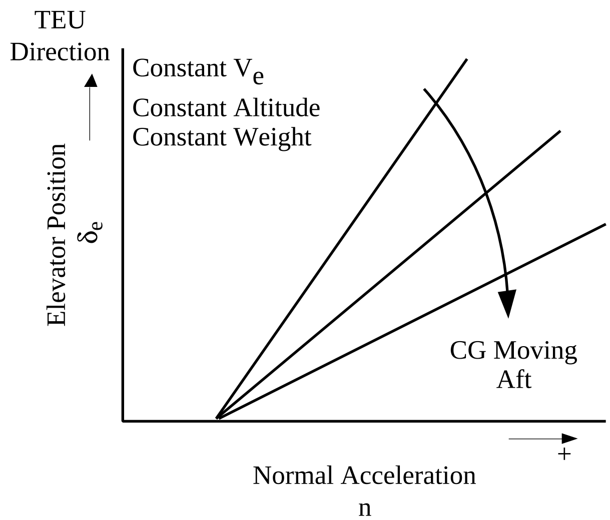


Figure 4.62

CG Movement Effects on $\left(\frac{d\delta_e}{dn}\right)$

The effects of altitude on the elevator position gradient in maneuvering flight may be studied by considering a constant CG position and constant equivalent airspeed while varying altitude. Altitude variation, for these conditions, has no effect on the stability term. However, as altitude is increased, the damping term decreases because of the reduction in density; therefore the elevator position gradient in maneuvering flight decreases with altitude increase at a constant equivalent airspeed (Figure 4.63).

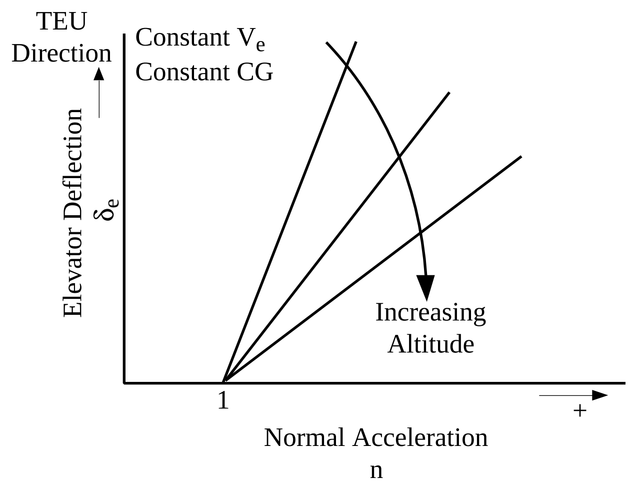


Figure 4.63
Altitude Effects on $\left(\frac{d\delta_e}{dn}\right)$ **at Constant** V_e

(NOTE: The reader should not receive an erroneous impression from this discussion. Pilots do not normally fly at the same equivalent airspeeds at high altitudes as they fly at low altitudes. Generally, Ve is less at high altitude. Therefore, the pilot's natural impression of the elevator position variation with normal acceleration at high altitudes may be that it is greater than at low altitudes. This is due to the fact that more elevator deflection is required to produce a unit change in normal acceleration at the lower dynamic pressure (lower equivalent airspeed) existing at the higher altitude.) If altitude is varied at a constant Mach number, the elevator position gradient in maneuvering flight increases with increase in altitude as shown in Figure 4.64.

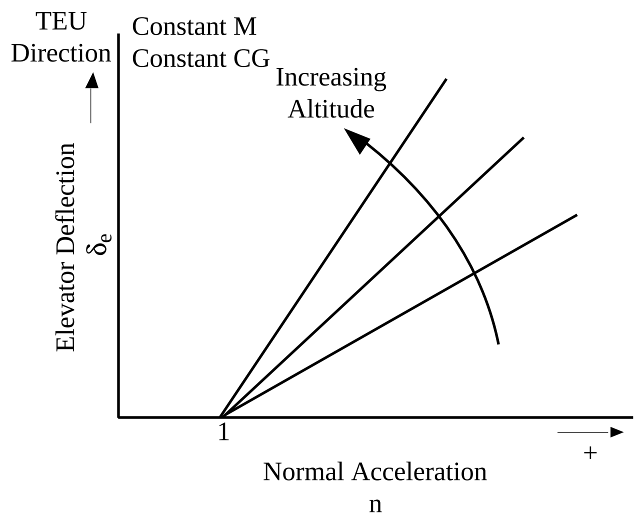


Figure 4.64

Altitude Effects on $\left(\frac{d\delta_e}{dn}\right)$ **at Constant Mach**

This is due to a slight increase in the damping term and a considerable increase in the stability term with increase in altitude at a constant Mach number.

Airspeed variation has a very large influence on the elevator position gradient in maneuvering flight since V_e^2 appears in the equations for $\frac{d\delta e}{dn}$. An increase in equivalent airspeed decreases the gradient of elevator position with normal.

Because of the large effect of airspeed variation on the elevator position gradient, it is extremely important that the pilot maintain close control over airspeed during the flight test measurement of maneuvering stability characteristics. Small errors in airspeed can generate erroneous data as shown in Figure 4.66.

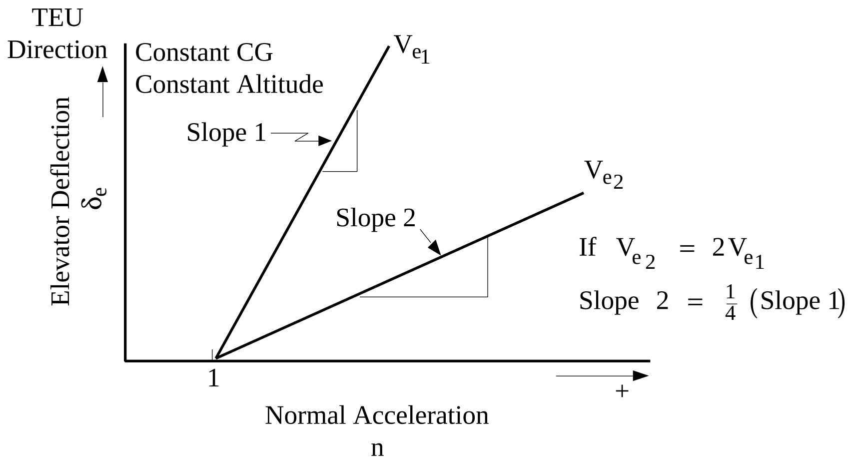
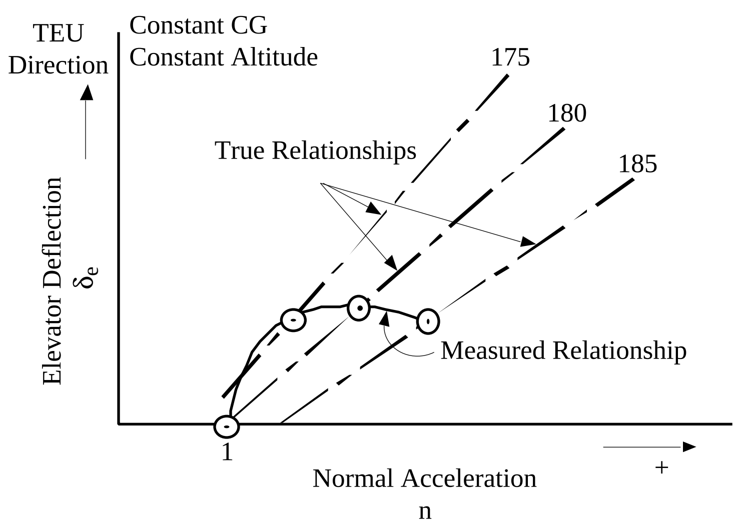


Figure 4.65

Effect of Varying Equivalent Airspeed on $\frac{d\delta_e}{dn}$



Assume the test pilot desired to measure the elevator position gradient in manuevering flight at 180 KEAS. If he measured points 5 KEAS slow, on speed, and 5 KEAS fast (in turn), the erroneous relationship shown above is a possible result.

Figure 4.66
Effect of Poor Airspeed Control on Manuevering Stability Data

4.7.1.3 LONGITUDINAL CONTROL FORCE MANEUVERING STABILITY

The second criterion for longitudinal stability in maneuvering flight is the longitudinal control force variation with normal acceleration at a constant airspeed. This parameter, commonly called “stick force per g” has a tremendous effect on the overall flying qualities of all airplanes. If the mission of the airplane requires extensive maneuvering, the stick force gradient in maneuvering flight is perhaps the most important single characteristic of the airplane.

Longitudinal control forces in maneuvering flight are generated by the requirement for the pilot to move the elevator control to the position required for maintenance of the accelerated condition. If the control system is reversible, elevator “float” may modify the angle through which the pilot must move the elevator. For the irreversible control system, classical elevator “float” is not a factor, although artificial elevator float may be introduced by extendible link devices, mechanical advantage changers, etc. Of course, longitudinal control forces in maneuvering flight may also be modified by various other control system “gadgetry” in reversible or irreversible control systems. In this manual, maneuvering control forces will be discussed for the reversible control system, then the irreversible control system. The effects of various devices and “gadgetry” on longitudinal maneuvering forces will then be presented.

4.7.1.4 STICK FORCES IN MANEUVERING FLIGHT - REVERSIBLE CONTROL SYSTEM

For the reversible control system, the longitudinal control forces required in steady wings-level pull-ups and in steady turns may be expressed as follows:

$$F_{s_{\text{Pull-Up}}} = K\frac{W}{S}\,\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{\text{Free}}\left\{\frac{V_e^{\,2}}{V_{e_{\text{Trim}}}^2} - n\right\} + K\,\tfrac{1}{2}\,\rho\,\ell_t\,g\,(n-1)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.35

$$F_{s_{\text{Steady Turns}}} = K \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} \left\{\frac{V_e^2}{V_{e_{\text{Trim}}}^2} - n\right\} + K \tfrac{1}{2}\rho \ell_t g \left(n - \frac{1}{n}\right)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.36

Where:

K	=	a constant dependent on gearing ratio between the elevator and cockpit control stick, size of the elevator, and horizontal tail efficiency factor.
$C_{h_{\delta_e}}$	=	elevator hinge moment coefficient variation with elevator deflection.
$C_{m_{\delta_e}}$	=	elevator control power.
$\left(\frac{dC_m}{dC_L}\right)_{\text{Free}}$	=	stick-free static longitudinal stability.
$C_{h_{\alpha_t}}$	=	elevator hinge moment variation with change in angle of attack of the horizontal tail.
τ	=	rate of change of effective angle of attack with change of elevator deflection.

Again, note that the only difference in the two equations arises from the difference in expressions for pitch rate in steady wings level pull-ups and in steady turns. The derivative of these equations with respect to normal acceleration (at a constant airspeed) yields the following classical indices of longitudinal maneuvering stability for the reversible control system.

$$\left(\frac{dF_s}{dn}\right)_{\text{Pull-Up}} = -K \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} + K \tfrac{1}{2} \rho \ell_t g \left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$ *eq 4.37*

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turn}} = -K \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} + K \tfrac{1}{2} \rho \ell_t g \left(1 + \frac{1}{n^2}\right)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.38

Several important observations may be drawn from a study of the last two equations:

1. For longitudinal stability in maneuvering flight, increases in longitudinal control pull force must be used to stabilize the airplane at higher values of positive normal acceleration.

2. Both equations contain two parts (see Figure 4.67). The first part is a "stability term" due to the stick-free static longitudinal stability. The second part is a "damping term" arising from the change in effective angle of attack at the horizontal tail due to the pitch rate.

3. A little more longitudinal control force is required to pull the same normal acceleration increment in steady turns than in pull-ups. The difference in the gradient $\left(\frac{dFs}{dn}\right)$ is directly proportional to $\frac{1}{n^2}$; therefore, the difference becomes very small at high levels of normal acceleration. (See Figure 4.68.)

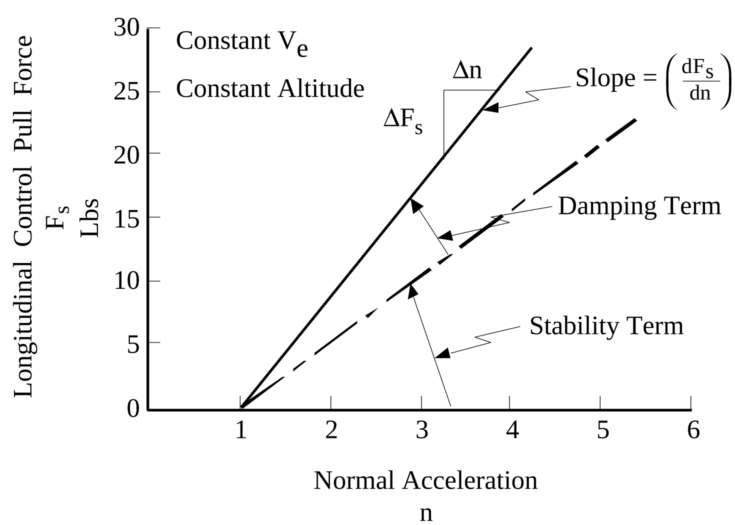


Figure 4.67
Classical Variation of Longitudinal Control Force in Maneuvering Flight

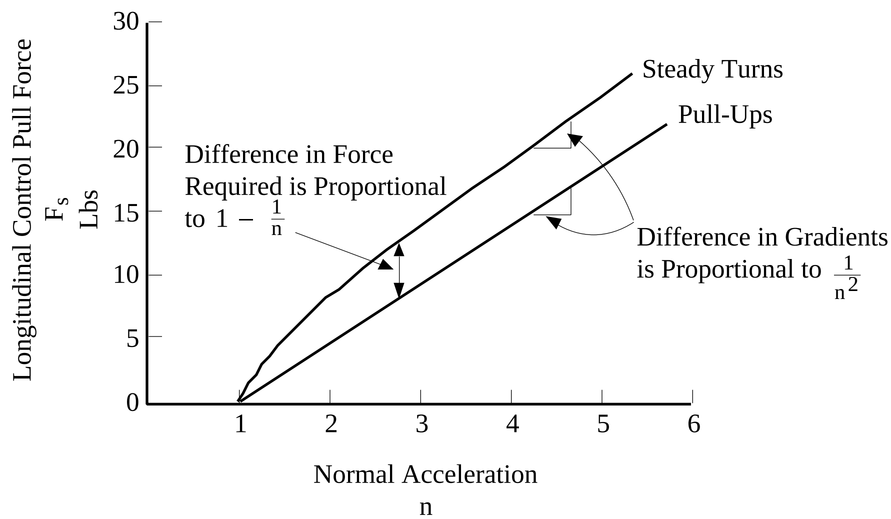


Figure 4.68
Relationship of Maneuvering Stability Characteristics in Steady Turns and Pull-Ups

Center of gravity movement naturally has a profound effect on longitudinal control force requirements in accelerated flight (Figure 4.69).

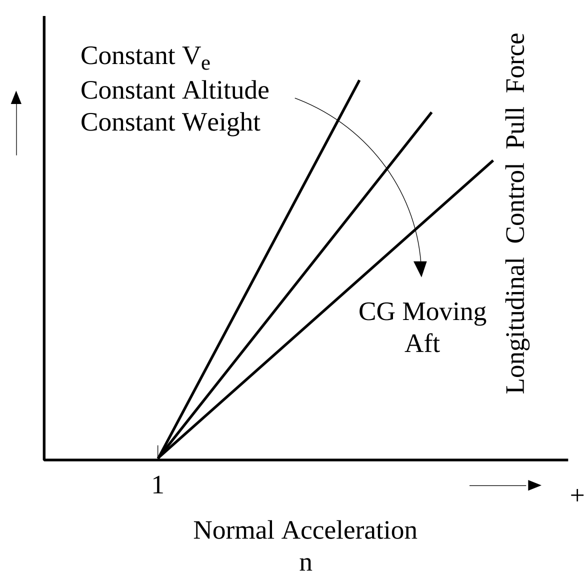


Figure 4.69
CG Movement Effects on $\left(\frac{dF_s}{dn}\right)$ **or** $\left(\frac{d\delta_e}{dn}\right)$

As the CG is moved aft, $\left(\frac{DC_m}{dC_L}\right)_{Free}$ becomes smaller in magnitude. When the CG is at the stick-free neutral point, the gradient $\left(\frac{dF_s}{dn}\right)$ is only a function of the damping term. (The damping term decreases slightly in magnitude as the CG is moved aft because the tail arm length is decreased.) If the CG is moved far enough aft, the gradient $\left(\frac{dF_s}{dn}\right)$ becomes zero; this CG position is called the stick-free maneuvering neutral point, N_m'. The stick-free maneuvering neutral point generally is aft of the stick-free neutral point, N_o'. In certain instances power or Mach effects may cause this relationship to be reversed.

The effects of altitude variation on the longitudinal control force gradient in maneuvering flight at a constant CG and constant equivalent airspeed is shown in Figure 4.70. For these conditions, altitude variation has no effect on the stability term of the equations; however, the damping term decreases because of the reduction in density. Therefore, for the reversible control system, the longitudinal control force gradient in maneuvering flight decreases with increase in altitude at a constant equivalent airspeed.

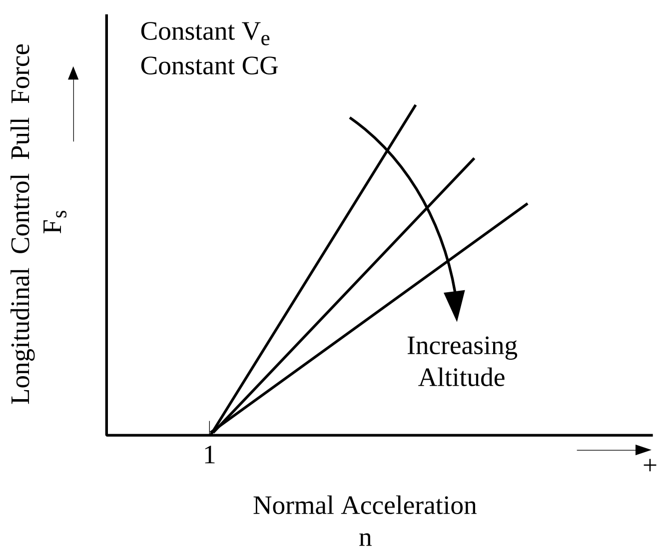


Figure 4.70
Altitude Effects on $\left(\frac{dF_s}{dn}\right)$ **at Constant** V_e
for the Reversible Control System

If altitude is varied at a constant Mach number, the longitudinal control force gradient in maneuvering flight again decreases in altitude for the reversible control system. This due to the decrease in the damping term because of the density decrease (Figure 4.71).

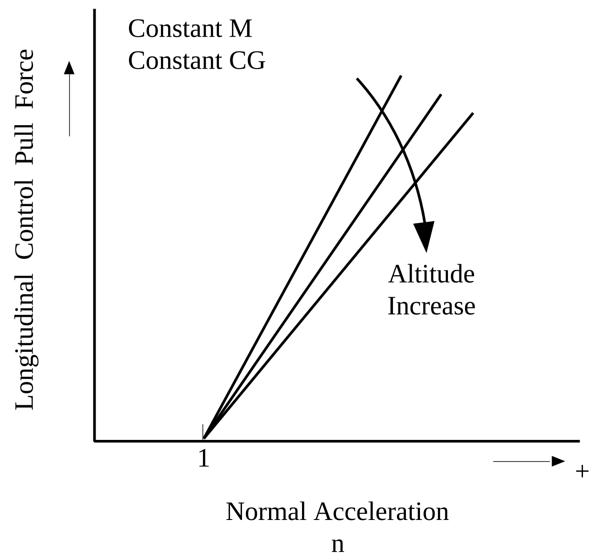


Figure 4.71
Altitude Effects on $\left(\frac{dF_s}{dn}\right)$ at Constant Mach
for the Reversible Control System

The effects of airspeed variation on longitudinal control forces in maneuvering flight for the reversible control system are interesting to study. First of all, the classical equations were developed by assuming that the airplane was initially trimmed in unaccelerated flight at a force trim speed, $V_{e_{Trim}}$. As long as $V_e = V_{e_{Trim}}$, the longitudinal control forces required in maneuvering flight do not vary as trim airspeed is varied if other factors remain constant (Figure 4.72). However is Ve is allowed to vary from $V_{e_{Trim}}$, the control forces vary considerably (Figure 4.73) (V_e^2 appears in the equations for longitudinal control force.)

It should be noted that $\left(\frac{dF_s}{dn}\right)$ does not vary even if airspeed varies from trim airspeed. Because of the situation shown in Figure 4.73, it is extremely important that the test pilot maintain precise control over airspeed during the flight test measurement of "stick force per g." If airspeed is allowed to vary form trim airspeed, erroneous impressions of maneuvering stability characteristics can be the result (Figure 4.74).

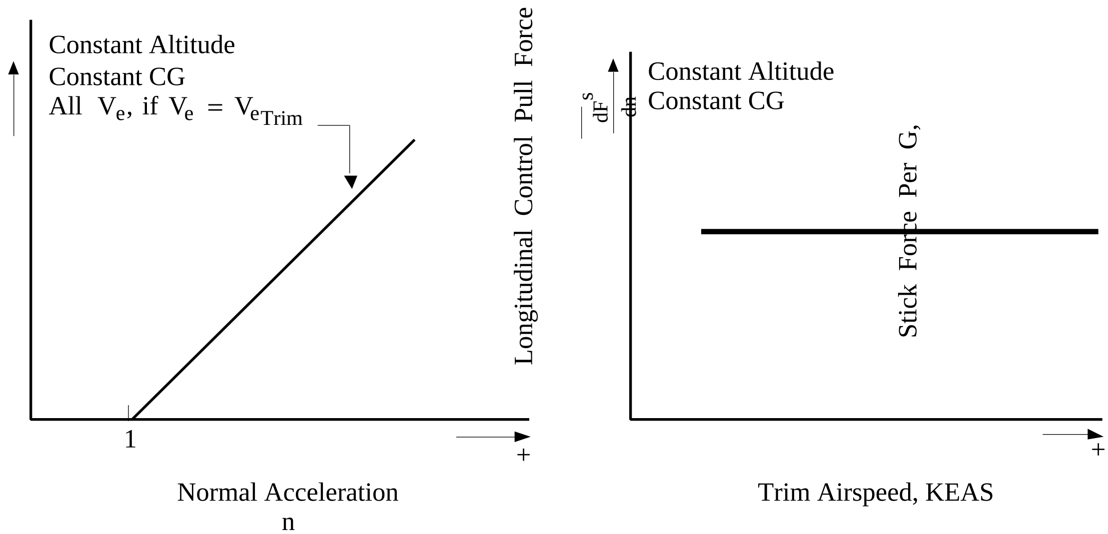


Figure 4.72
For the Reversible Control System "Stick Force per g" is not Affected by Changes in Trim Airspeed

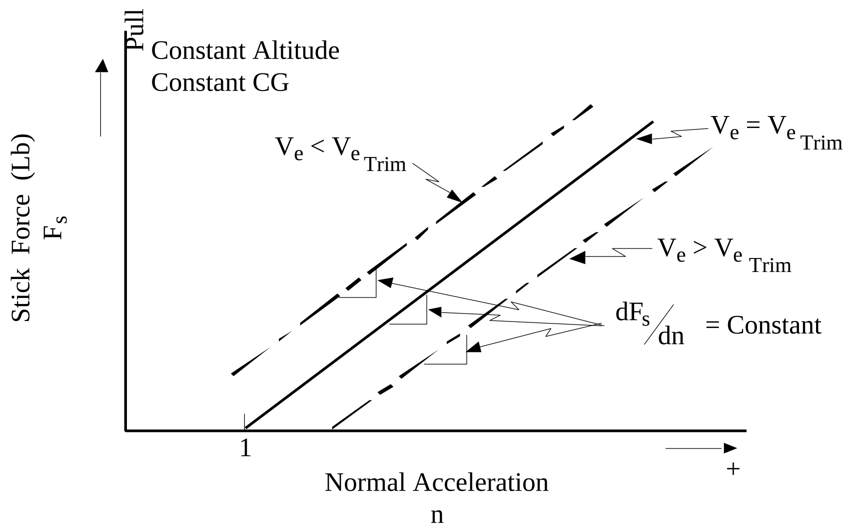
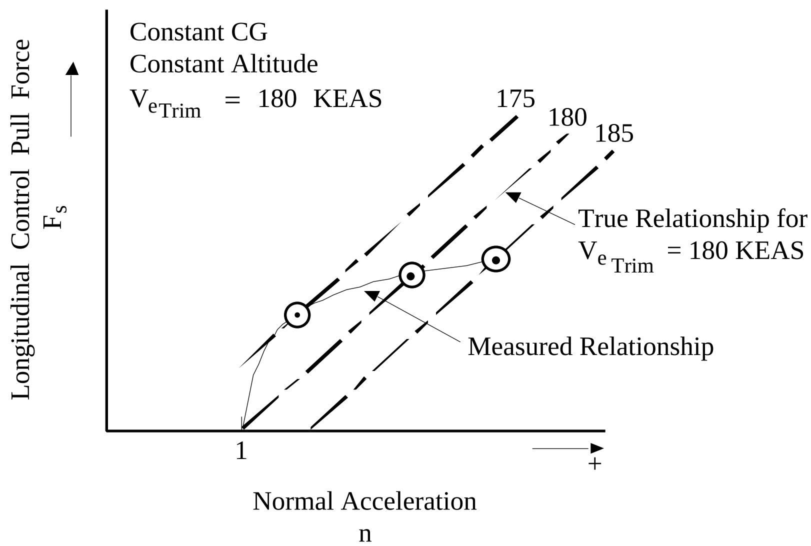


Figure 4.73
Affect of Speed Variation from Trim Airspeed on "Stick Force per g"



Assume the test pilot desired to measure the longitudinal control force variation with normal acceleration at a force trim speed of 180 KEAS. If he measured points slow, on speed, and fast (in turn), the erroneous relationship shown above is a possible result.

Figure 4.74
Effect of Poor Airspeed Control on Maneuvering Stability Data

4.7.1.5 STICK FORCES IN MANEUVERING FLIGHT-IRREVERSIBLE CONTROL SYSTEM

Some of the characteristics of longitudinal control force variation in maneuvering flight for the reversible control system are the same for the irreversible control system. These are:

1. More longitudinal control force is required to pull the same normal acceleration increment in steady turns than in pull-ups. However, the difference in $\left({}^{dF_s}/_{dn}\right)$ between the two cases is very small at high normal acceleration.

2. Aft CG movement decreases “stick force per g,” if other factors remain constant.

3. Poor airspeed control during the measurement of “stick force per g” can result in erroneous impressions of longitudinal maneuvering stability.

Equations for longitudinal control force variation in maneuvering flight will now be presented for two types of irreversible control system. For simplicity, the equations for steady turns only will be presented.

Assume the irreversible control system is designed such that longitudinal control force is directly proportional to elevator deflection; i.e.:

$$F_s = K_1 \Delta\delta e \qquad \textit{eq 4.39}$$

where K_1 = a constant describing the characteristics of the system such as strength of the feel spring, gearing ratio, etc. (This is one of the simplest and most widely used longitudinal control systems, generally containing a linear feel spring.)

For this type of irreversible longitudinal control system, longitudinal control force variation with normal acceleration in steady turns at a constant trim airspeed may be written as follows:

$$\left(\frac{dFs}{dn}\right)_{\text{Steady Turns}} = -\frac{K_1}{C_{m_{\delta_e}}}\,\frac{W/S}{\frac{1}{2}\rho_{SSL}\,V_{e_{Trim}}^2}\left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot\theta}}\rho g\bar{c}}{4\,W/S}\left(1-\frac{1}{n^2}\right)\right\}$$

eq 4.40

The effects of trim airspeed variation of "stick force per g" for this type control system are shown in Figure 4.75, for <u>no compressibility effects</u>. Note the difference between Figures 4.75 and 4.72. Also note that for the irreversible control system, longitudinal forces are dependent on stick-fixed stability vice stick-free stability.

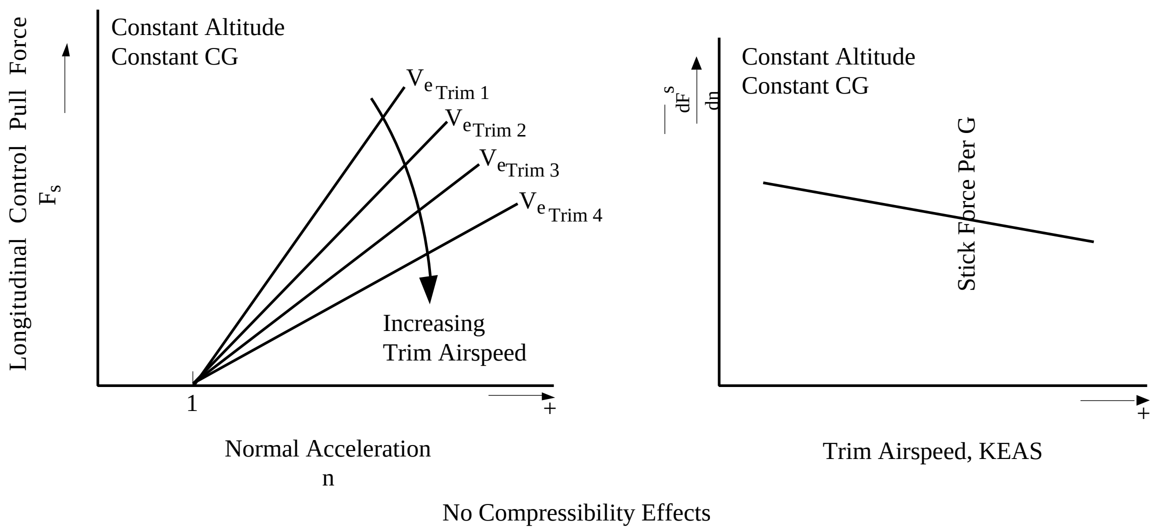


Figure 4.75
Effect of Varying Trim Airspeed on "Stick Force per g"
for the Irreversible Control System Where $F_s = K_1\Delta\delta_e$

Consider now a different irreversible control system which incorporates a dynamic pressure (q) sensor such that:

$$F_s = K_2 q \Delta\delta_e \qquad \textit{eq 4.41}$$

where K_2 = a constant describing the characteristics of the system, such as strength of the feel spring, gearing ratio, etc. (This type longitudinal control system is commonly called a "q-feel" system.)

For this type of irreversible longitudinal control system, longitudinal control force variation with normal acceleration in steady turns at a constant trim airspeed may be written as follows:

$$\left(\frac{dF_s}{dn}\right)_{Steady\ Turns} = -\frac{K_2 \ W/S}{C_{m_{\delta_e}}} \left\{\left(\frac{dC_m}{dC_L}\right)_{Fixed} + \frac{C_{m_{\dot{\theta}}} \rho g \bar{c}}{4\ W/S}\left(1 - \frac{1}{n^2}\right)\right\} \qquad \textit{eq 4.42}$$

The influence of trim airspeed variation on "stick force per g" for this type of control system is the same as for the reversible control system (see Figure 4.72) if no compressibility effects are present.

4.7.1.6 EFFECTS OF COMPRESSIBILITY ON MANEUVERING STABILITY

The previous discussions have neglected compressibility effects (high Mach number flight) which may have a profound influence on maneuvering control forces. Without proceeding deeply into transonic and supersonic flight testing, which will be discussed in a subsequent section, compressibility generates the following phenomenon which influence the maneuvering force gradient:

1. The wing aerodynamic center shifts aft in the transonic flight regime, which increase $\left(\frac{dC_m}{dC_L}\right)$. (This is analogous to a forward shift in airplane center of gravity.)

2. Shock wave formation and change in pressure distribution reduce the effectiveness of the longitudinal control surface, particularly if the surface is an elevator vice a stabilator.

Both the effects listed above tend to increase "stick force per g" and "elevator position per g" for both the reversible and irreversible control system. Typical influence on maneuvering stability is shown in Figure 4.76.

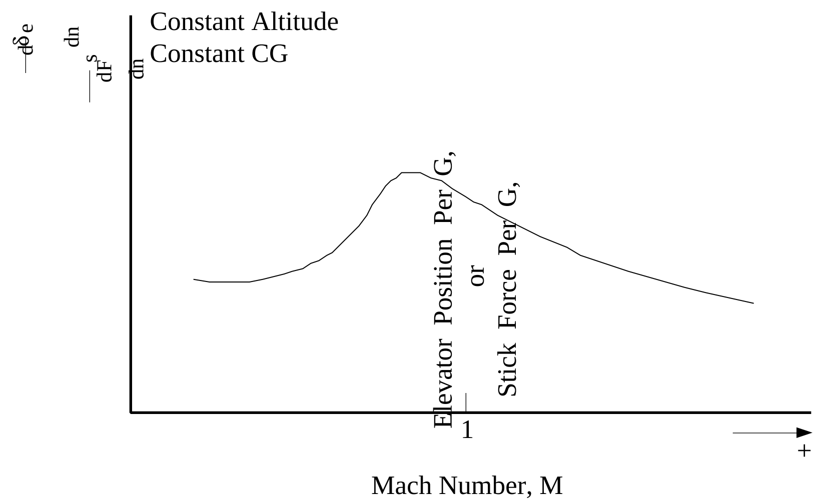


Figure 4.76
Compressibility Influence on Maneuvering Stability

4.7.1.7 EFFECTS OF LONGITUDINAL CONTROL SYSTEM "GADGETRY" ON CONTROL FORCES IN MANEUVERING FLIGHT

Longitudinal control system "gadgetry" has been introduced earlier in this section and its effect on longitudinal flying qualities during nonmaneuvering tasks discussed. Schematics of these devices were presented in that part; therefore, many of the schematics will not be reproduced here.

The most commonly used means of alerting longitudinal control forces in maneuvering flight is through the use of bobweights. The addition of a "positive bobweight" - a bobweight mounted so as to oppose movement of the longitudinal control during accelerated flight - increases the "stick force per g" in maneuvering flight (Figure 4.77). Conversely, the negative bobweight decreases "stick force per g".

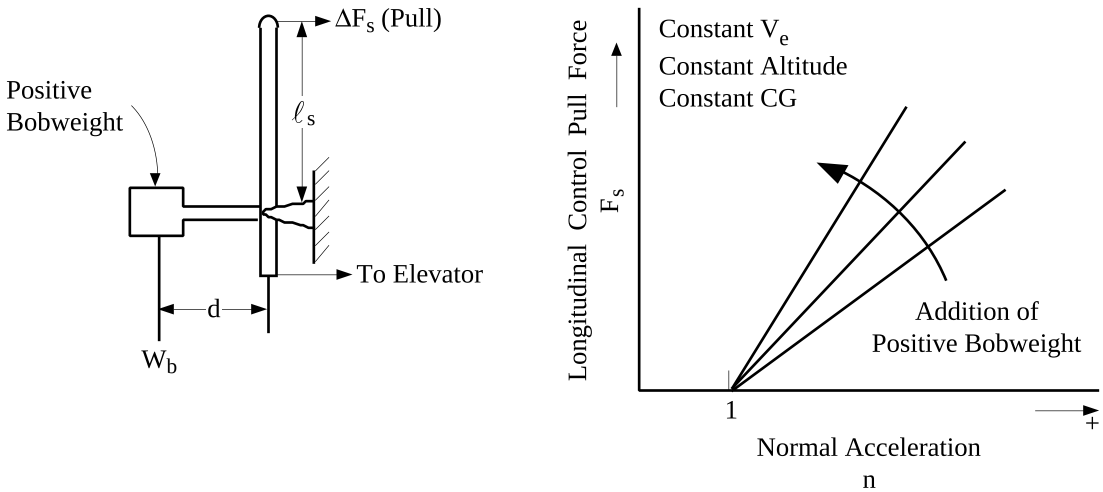


Figure 4.77
Bobweight Arrangement and Influence on Longitudinal Control Forces During Maneuvering Flight

The following devices, generally used to correct shallow longitudinal control force versus airspeed relationships in unaccelerated flight, usually increase "stick force per g" in accelerated flight:

1. simple spring[5]
2. downspring[5]
3. Leading tab

---

[5] The constant load downspring has no effect of "stick force per g" if it merely adds a preload force to the longitudinal control system. This is generally not the case since simple springs and downsprings normally add forces as a function of stick displacement. Of course, this arrangement does increase "stick force per g."

Lagging tabs and Servo tabs, generally used to reduce longitudinal control force in unaccelerated flight, also reduce longitudinal control forces in accelerated flight.

The blow-down tab does not affect "stick force per g" as long as the pilot maintains trim airspeed precisely during the in-flight measurement.

The preloaded spring tab, has an interesting influence on maneuvering control forces in that it introduces artificial nonlinearity into the "stick force per g" plots (Figure 4.78).

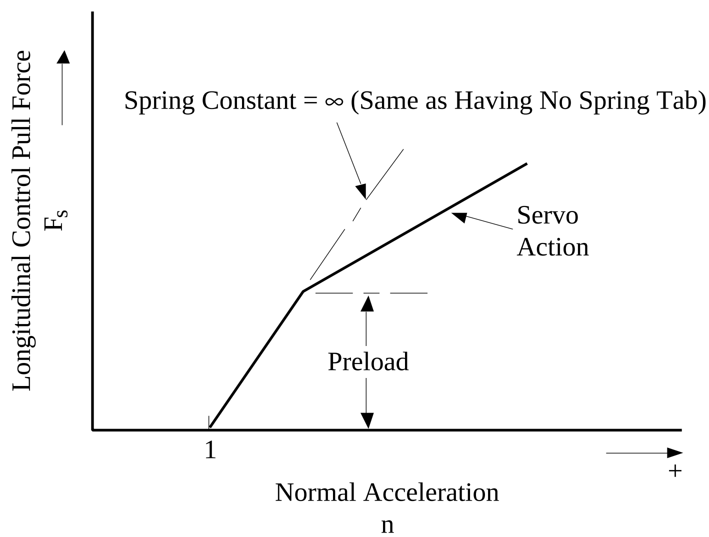


Figure 4.78
Influence of Pre-Loaded Spring Tab on Maneuvering Control Forces

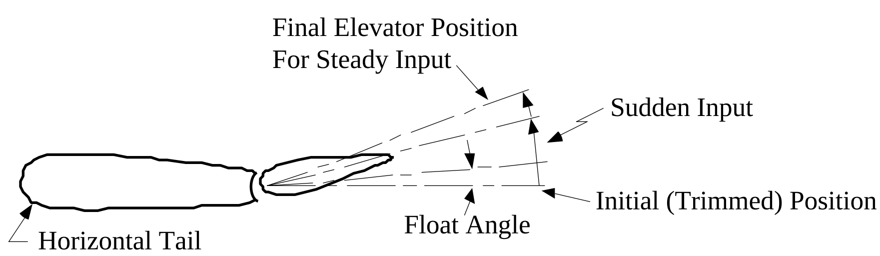
4.7.1.8 EFFECTS OF "RAPID MANEUVERS" ON MANEUVERING STABILITY

The discussion of maneuvering stability has, to this point, considered only steady-state conditions where dynamic equilibrium has been achieved. During transient maneuvers with rapid inputs of stick force and elevator position (sometimes called sudden pull-ups), the simple relationships previously presented no longer apply. It is extremely important, of course, that the maneuvering stability characteristics (particularly "stick force per g") during sudden maneuvers be such that the airplane is not easily overstressed. In addition, the dynamic characteristics of the airplane (short period damping, in particular) and the phasing between the pilot's force inputs to the control stick and the resulting stick motion and normal acceleration response must be such that the airplane is not prone to pilot-induced-oscillations in rapid maneuvering. Several factors affecting maneuvering stability during abrupt maneuvering will now be presented.

Consider an airplane equipped with a reversible longitudinal control system with no bobweight. If the pilot applies and holds a rapid input of longitudinal control pull force to this system, the airplane response in normal acceleration will generally be <u>less</u> for the <u>sudden</u> force input than for the equivalent <u>steady</u> force input. This is due to the fact that the elevator does not have sufficient time to reach its "float" position in the rapid maneuver. Therefore, the longitudinal control force are higher per unit change in normal acceleration in the sudden maneuver as compared to the steady maneuver if the elevator is not over-balanced (see Figure 4.79). (This is the same as saying the response in normal acceleration per unit input of longitudinal control force is less in the sudden maneuver.) The difference in control force variation with normal acceleration between steady maneuvering flight and sudden maneuvering is dependent on the <u>rapidity</u> of the sudden input. This difference is largest at low values of normal acceleration and smallest at high values. (At high levels of normal acceleration, the steady pull-up or steady turn maneuver must be fairly <u>rapid</u> to attain the high normal acceleration at a constant airspeed.)

For the irreversible control system which exhibits no classic elevator float, viscous dampers or other devices may be used to discourage rapid longitudinal control inputs. These device tend to effectively increase maneuvering control forces during rapid, abrupt maneuvering exactly the same as shown in Figure 4.79. If the rapidity or the suddenness of the input is increased, the difference between sudden and steady control forces in maneuvering flight is increased.

However, consider the reversible control system again where the elevator is very closely balanced $\left(C_{h_{\delta_e}}\right.$ is very small). Satisfactory control forces in maneuvering flight can be achieved for this situation by making $C_{h_{\alpha_t}}$ <u>slightly positive</u>. This causes the elevator to "float" opposite to the direction shown in Figure 4.79. However, in rapid maneuvers, large elevator deflections may be obtained before the airplane's response builds up the longitudinal control force through the floating tendency. This will generate large normal acceleration changes for undesirably small control forces in sudden maneuvers, while in the steady state maneuver, longitudinal control force variation with normal acceleration may be satisfactory (see Figure 4.80).



Sudden Longitudinal Force Input is the same as the Steady Longitudinal Force Input. For the Steady Maneuver, Final Elevator Deflection is Greater because of Float Angle, which develops after a Finite Time Interval.

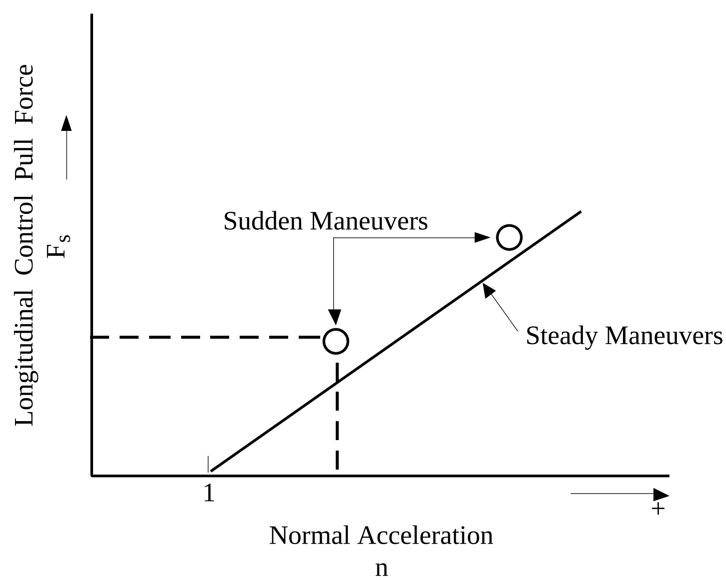


Figure 4.79
Relationship Between Maneuvering Control Forces for Steady and Sudden Maneuvering for the Reversible Control System

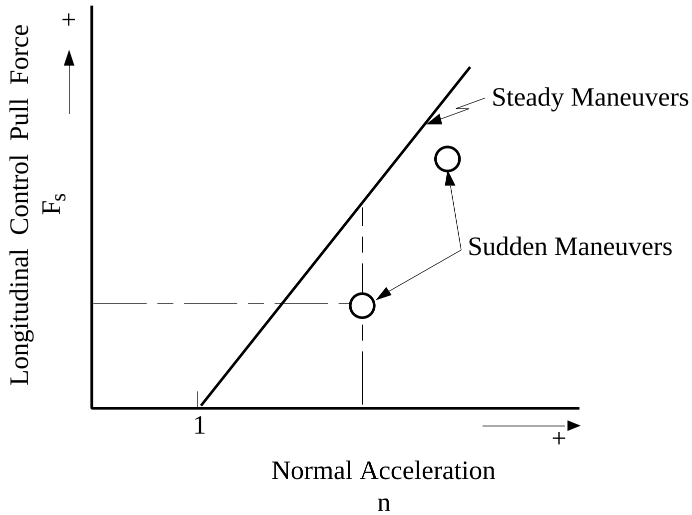


Figure 4.80
Typical "Stick Force per g" Characteristics for the Closely Balanced or Overbalanced Elevator-Reversible Control System

The bobweight, previously introduced as a gadget used to tailor maneuvering control forces in steady maneuvers, can have a serious degrading influence on longitudinal flying qualities during rapid or sudden maneuvers. In any type of control system the bobweight tends to alter the phasing between the pilot's force inputs and the resulting stick motion and normal acceleration response. Consider the case of an airplane which obtains all or nearly all its maneuvering force gradient (stick force per g) in steady maneuvering flight from a positive bobweight. In rapid maneuvering of this airplane, <u>the cockpit control stick can be moved with very small force inputs</u> to initiate the sudden maneuver. As normal acceleration develops, the bobweight, responding to the normal acceleration, attempts to pull the control stick back to neutral. This requires the pilot to add increasing longitudinal pull forces to maintain the control input. The same relationship between sudden and steady maneuvers shown in Figure 4.80 again apply for this situation. In addition, the pilot may induce objectionable high-frequency oscillations in normal acceleration in attempting to perform rapid maneuvering tasks under these conditions. In extreme cases, if the damping of the longitudinal control system is poor, the pilot feels the control stick constantly slapping against his hand during rapid maneuvering.

In an attempt to alleviate poor control characteristics in sudden maneuvering for control systems utilizing bobweights, the arrangement shown in Figure 4.81 is sometimes used. The bobweight is not only sensitive to normal acceleration, it is sensitive to rate of change of normal acceleration, or pitch acceleration, $\ddot{\theta}$.

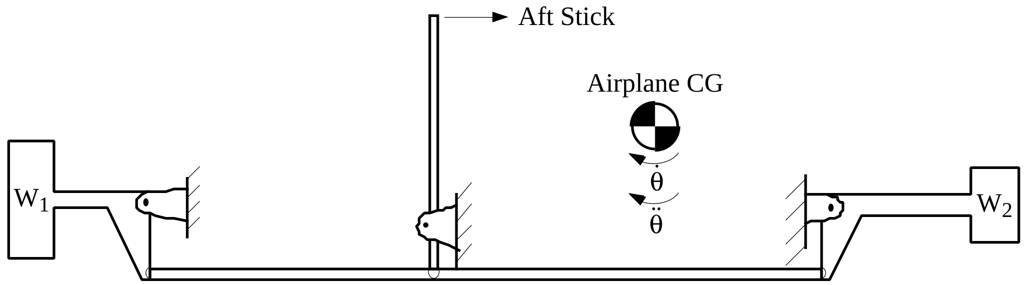


Figure 4.81
Bobweight Arrangement Utilizing Bobweights
Fore and Aft of the Airplane Center of Gravity

From a study of Figure 4.81, the rationalization may be made that during rapid maneuvering (during which normal acceleration and pitch rate are changing), the fore and aft bobweight arrangement applies additive forces to the control system which <u>oppose the pilot's control input</u>. This tends to increase maneuvering control forces in sudden pull-ups, etc. During steady maneuvering flight (during which normal acceleration and pitch rate are constant), the fore and aft bobweights apply individual forces to the control system which tend to cancel each other. (Bobweight W_1 would increase "stick force per g" and bobweight W_2 would decrease "stick force per g".) The overall effect depends, of course, upon the relative size of the bobweights, as well as their placement with respect to the airplane CG and the cockpit control stick. However, the overall effect would generally be that shown in Figure 4.79.

4.7.2 Dynamic Longitudinal Stability and Controls Related to Maneuvering Tasks

The previous discussion of longitudinal maneuvering stability has been concerned mainly with equilibrium flight conditions. The discussion will now be expanded to study the means by which one equilibrium flight condition is changed to another equilibrium flight condition.

The means by which the airplane may be brought into a condition of equilibrium during maneuvering tasks has been previously developed. Further, the typical response of the airplane to a longitudinal control input thought the two longitudinal modes of motion was presented earlier in the discussion of nonmaneuvering tasks. It is convenient to again refer to this typical response (Figure 4.82). Note that the control input generates pitching moments which initially cause only changes in angle of attack (and normal acceleration) at a constant airspeed. This is the response of the airplane through its short period mode of motion. The characteristics of this mode of motion greatly influence the pilot's ability to perform both maneuvering and nonmaneuvering tasks. It's characteristics are particularly critical for maneuvering tasks. Characteristics of the phugoid or long period mode have little influence during maneuvering tasks because:

1. The pilot generally has close control over pitch attitude during maneuvering tasks, which effectively damps the phugoid motion.

2. The pilot is continually changing the airplane's flight path during maneuvering tasks. The short time interval between changes in the airplane's flight path does not allow the phugoid motion to develop.

It should be apparent from a study of Figure 4.82 that the short period mode of motion is a second order response composed of angle of attack (and normal acceleration) variations at an essentially constant airspeed. Thus the pilot utilizes the short period mode to make angle of attack and normal acceleration changes; therefore, during maneuvering tasks, the pilot will devote much of his attention to controlling the short period mode of motion.

The remainder of this discussion will be direct toward describing the origin, characteristics, and parameters affecting the short period mode of motion.

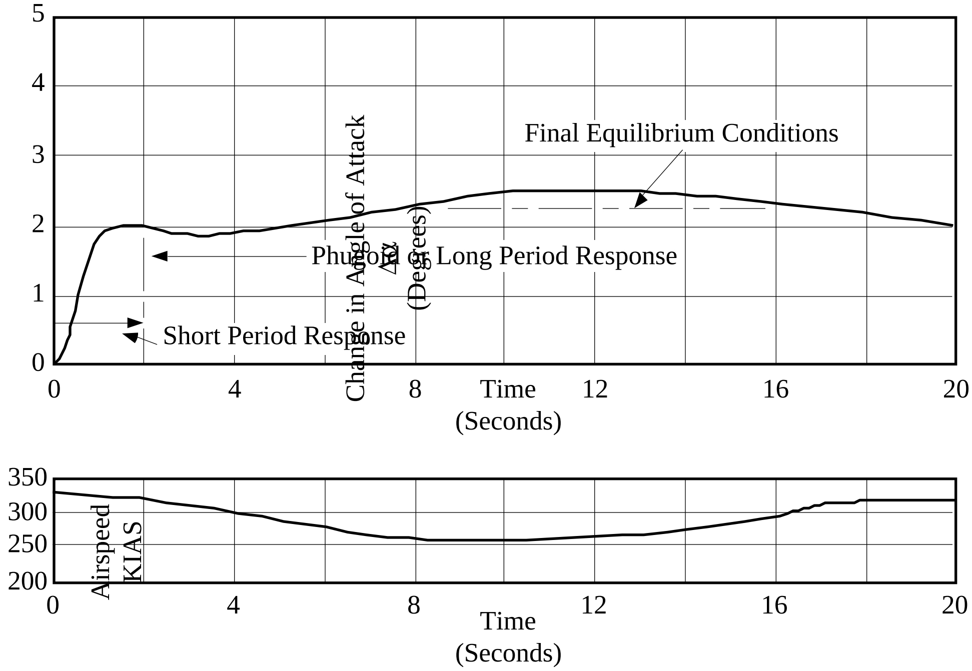


Figure 4.82
Typical Airplane Response to Longitudinal Control Input

4.7.2.1 ORIGIN OF THE SHORT PERIOD MODE OF LONGITUDINAL MOTION

Without derivation, the determinant of the transformed longitudinal equation of motion for “small” disturbances may be written as shown in Figure 4.83.

133

$$\begin{array}{l} \text{Drag Characteristics} \rightarrow \\[2ex] \text{Lift Characteristics} \rightarrow \\[2ex] \text{Pitching Moments Characteristics} \rightarrow \end{array} \begin{vmatrix} S + D_u & D_\alpha - g & g \\ \dfrac{L_u}{u_0} & S + \dfrac{L_\alpha}{u_0} & -S \\ -M_u & -M_{\dot\alpha}S - M_\alpha & S^2 - M_{\dot\theta}S \end{vmatrix} = 0$$

Terms Generated by Changes in Horizontal Velocity

Terms Generated by Changes in Angle of Attack

Terms Generated by Changes in Pitch Attitude

S = Laplace Operator

g = acceleration due to gravity

u = horizontal velocity (u_0 = initial horizontal velocity)

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

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

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

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

$M_u = \dfrac{\partial M/\partial u}{I_{yy}}$ = change in pitching moment with 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 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.

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

Figure 4.83
The Longitudinal Determinant

The solutions of the longitudinal determinant will provide useful information about the longitudinal modes of motion. The classic short period approximation is now of concern. In order to make this approximation several assumptions must be made. These assumptions, based on flight experience and logical reasoning, are as follows:

1. Airspeed remains constant during the motion.
2. Short period motion is not affected by pitch attitude; however, the short period mode is sensitive to pitch rate.
3. Drag characteristics have no influence on the short period mode.
4. Low Mach number (no compressibility effects).

If the above assumptions are valid, the lift and moment portions of the longitudinal determinate (with airspeed terms set equal to zero) are the controlling factors for the short period motion. The "classic" short period approximation may then be written as follows:

$$\begin{vmatrix} S + \frac{L_\alpha}{u_0} & -1 \\ -M_{\dot{\alpha}}S - M_\alpha & S - M_{\dot{\theta}} \end{vmatrix} = 0 \qquad \textit{eq 4.43}$$

Solving the determinant yields the following second order characteristic equation:

$$S^2 + \left(\frac{L_\alpha}{u_0} - M_{\dot{\theta}} - M_{\dot{\alpha}}\right)S - \left(M_\alpha + \frac{L_\alpha}{u_0}M_{\dot{\theta}}\right) = 0 \qquad \textit{eq 4.44}$$

The undamped natural frequency of the short period mode of motion may be developed† as follows:

$\omega_{n_{sp}}$ = undamped short period frequency

$$= \sqrt{\frac{\frac{1}{2}\ P_a\ M^2}{I_{yy}}\ S\bar{c}\ C_{L_\alpha}\left(\frac{X_{CG}}{\bar{c}} - N_M\right)} \qquad \textit{eq 4.45}$$

Where:

γ	=	The ratio of the specific heat of a gas at constant volume to that at constant pressure (γ is a constant, generally taken as 1.4).
P_a	=	absolute pressure, pounds per square foot.
M	=	Mach number
C_{L_α}	=	change in lift coefficient per unit change in angle of attack (life curve slope).
$\frac{X_{CG}}{\bar{c}} - N_M$	=	nondimensional distance between the airplane CG and stick-fixed maneuvering neutral D (sometimes called maneuver margin or maneuvering margin.

A simple expression for short period damped natural frequency is derived if the following assumption are made:

$$M_{\dot{\alpha}} \doteq 0$$
$$L_{\alpha/u_0} \doteq -M_{\dot{\theta}} \qquad \textit{eq 4.46}$$

† Several mathematical manipulations have been omitted.

The undamped natural frequency of the short period mode of motion may then be developed as follows† :

$$\omega_{sp} = \sqrt{-M_\alpha} \qquad \textit{eq 4.47}$$

Several important relationships can be gathered from a study of the equation for $\omega_{n_{sp}}$:

1. The undamped natural frequency of the short period motion increases as Mach number <u>increases</u>; thus the period decreases with increase in Mach number. (The "quickness" of the motion increases.)

2. The undamped natural frequency of the short period motion decreases with increase in pressure altitude at a constant Mach number.

3. The undamped natural frequency of the short period motion decreases as the airplane CG is moved aft toward the stick-fixed maneuvering neutral point. This is analogous to weakening the spring in the spring-mass-damper system. When the CG is at the stick-fixed maneuvering neutral point, the undamped natural frequency is zero; i.e., the motion is nonoscillatory.

4. The undamped natural frequency of the short period motion decreases with an increase in moment of inertia in pitch. This is analogous to increasing the mass in the spring-mass-damper system.

5. The damped natural frequency of the short period motion is only dependent on angle of attack stability, M_α , if certain simplifying assumptions are valid.

† Several mathematical manipulations have been omitted.

The damping ratio of the short period mode of motion may be developed† as follows:

$$\zeta_{sp} = \frac{\sqrt{\frac{\rho S}{2}}}{2\sqrt{-\frac{\bar{c}}{I_{yy}} C_{L_\alpha}\left(\frac{X_{CG}}{\bar{c}} - N_M\right)}} \left\{ \frac{C_{L_\alpha}}{w/g} - \frac{C_{m_{\dot{\theta}}}\bar{c}^2}{2I_{yy}} - \frac{C_{m_{\dot{\alpha}}}\bar{c}^2}{2I_{yy}} \right\} \qquad \textit{eq 4.48}$$

Where:

C_{L_α} = lift curve slope coefficient

$C_{m_{\dot{\theta}}}$ = pitch rate damping coefficient

$C_{m_{\dot{\alpha}}}$ = “downwash lag” term coefficient

Certain important effects are visible from this relationship:

1. Increasing lift curve slope, increasing pitch rate damping, and increasing the “downwash lag” term increases damping of the short period mode of motion.

2. Increasing angle of attack stability decreases short period damping.

3. Moving the CG forward decreases short period damping.

4. Damping of the short period mode of motion is not a direct function of airspeed or Mach number.

4.7.2.2 CHARACTERISTICS OF THE SHORT PERIOD MODE

Additional insight into the short period mode of longitudinal motion may be gained by studying the flight path of an airplane during a short period oscillation. Figure 4.84 shows a typical short period motion. It is so rapidly damped out that the transient has

† Several mathematical manipulations have been omitted.

virtually disappeared in a very short horizontal distance. The deviation of the flight path from the original flight path is generally small, the principal feature of the motion being the rapid rotation of the airplane in pitch. (Compare the short period flight path with the phugoid flight path presented earlier.)

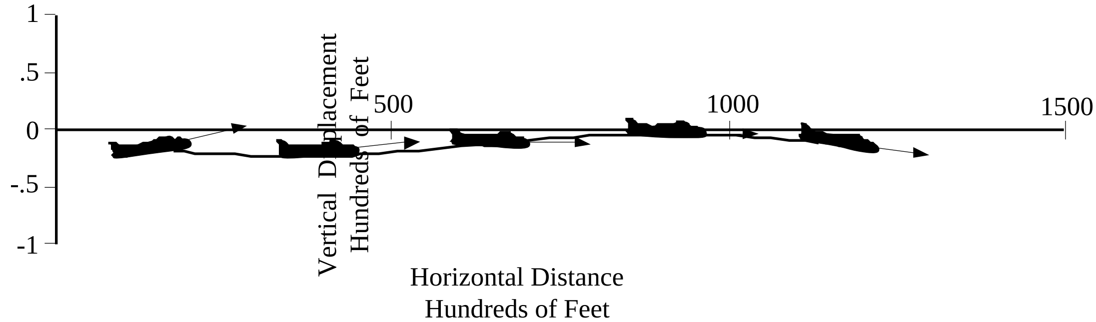


Figure 4.84
Typical Short Period Flight Path

4.7.2.3 EFFECTS OF VARIOUS PARAMETERS ON SHORT PERIOD MODE OF MOTION

The influence of varying several parameters on the short period motion will now be shown using the convenient root locus plots. The "classic" short period roots, as well as the "classic" phugoid roots are shown in Figure 4.85.

The short period mode shown in Figure 4.85 is typically stable, oscillatory, and well damped. It is assumed that the CG is somewhere forward of the stick-fixed neutral point.

The effect of varying angle of attack stability, M_α, can be studied by first assuming the M_u is zero, then allowing M_α to increase negatively from zero. (This is the normal sign for M_α, since, for stability, positive (nose up) increases in angle of attack must generate negative (nose down) pitching moments.) The effect of increasing angle of attack stability is shown in Figure 4.86. (This is exactly the same effect as moving the CG forward.)

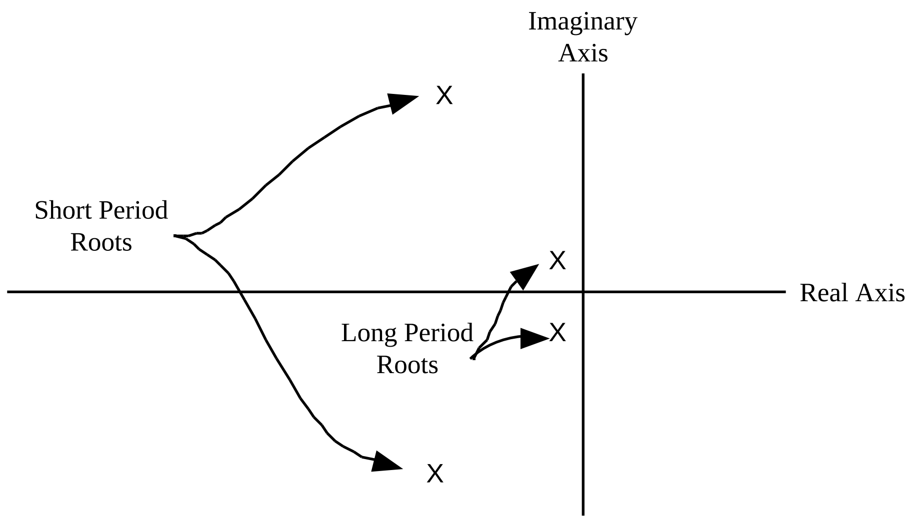


Figure 4.85
Complex Plane Representation of Classic Short Period and Phugoid Modes of Motion

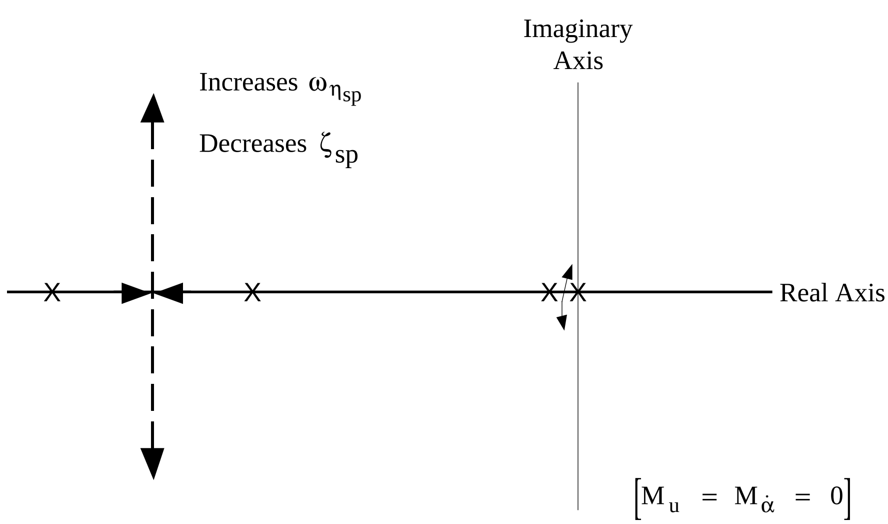


Figure 4.86
Effect of Increasing Angle of Attack Stability, M_α

A typical effect of airspeed variation in the subsonic flight regime on the short period motion is shown in Figure 4.87. As stated previously, short period damping is independent of airspeed, although undamped natural frequency increases with increasing airspeed.

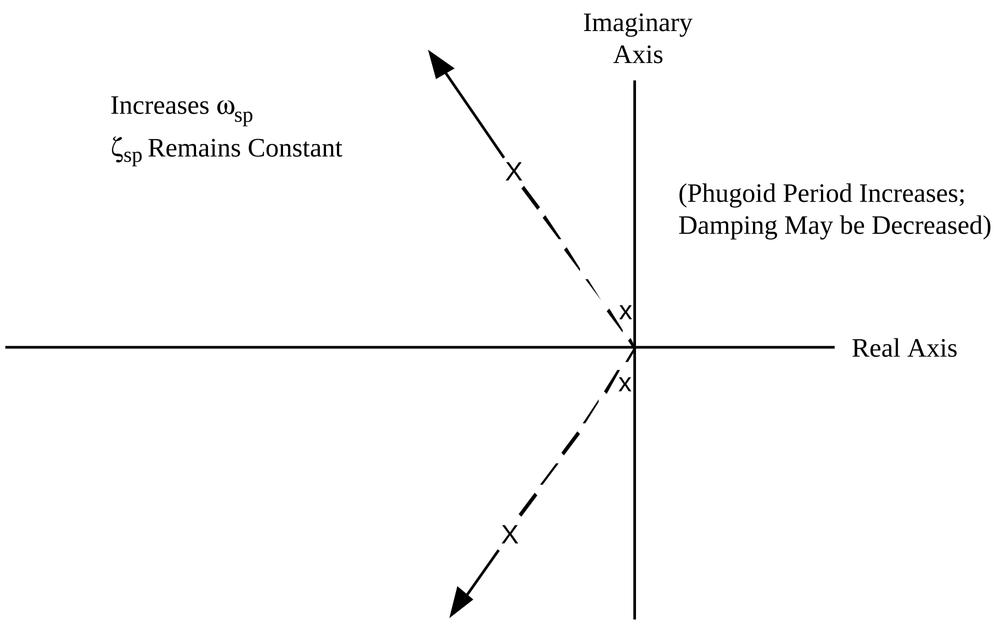


Figure 4.87
Influence of Increasing Airspeed on Longitudinal Short Period and Phugoid Characteristics

The influence of changing speed stability, M_u, on short period characteristics is shown in Figure 4.88. The most apparent phenomenon to the pilot will be the divergent, nonoscillatory phugoid tendency if M_u is less than zero. Only in the transonic flight regime is speed instability (negative M_u) generally encountered. This situation might be characterized by a well-damped, high frequency short period motion; yet a pure divergence in airspeed if speed is altered from trim, at least for small disturbances.

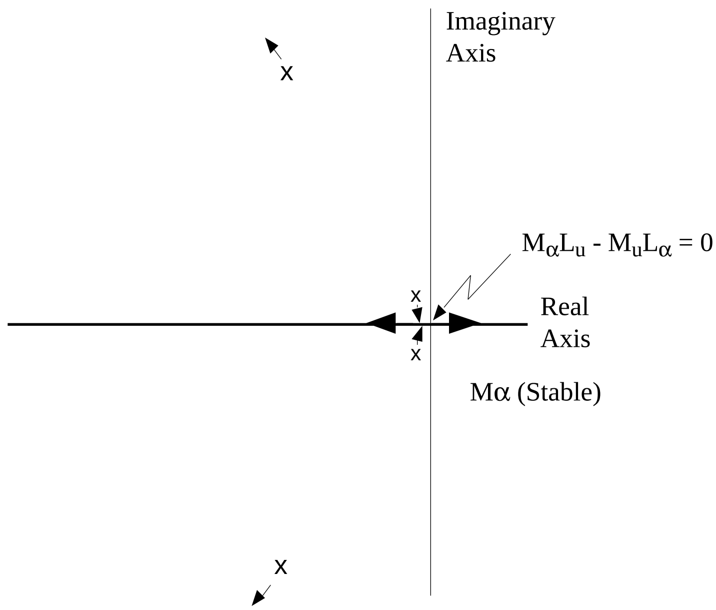
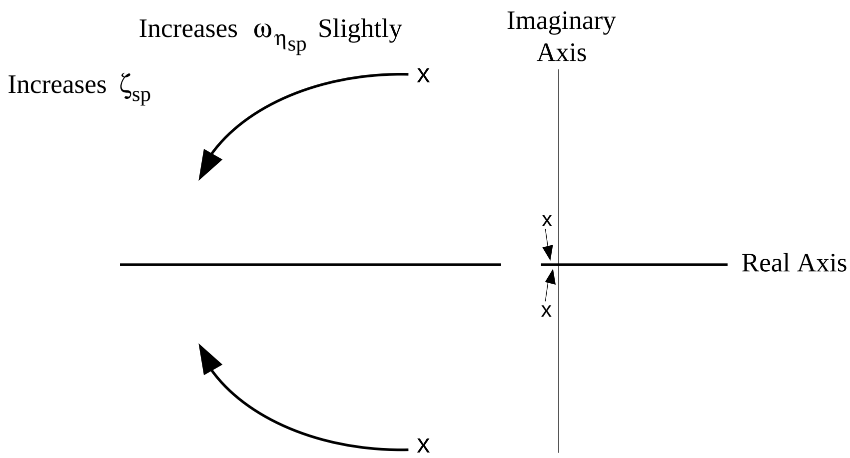


Figure 4.88
Influence of Reducing Speed Stability, M_u

One of the means of artificially augmenting short period damping is through the use of a pitch rate damper. This device senses pitch rate and applies proportional longitudinal control inputs which artificially increases $M_{\dot{\theta}}$. The effect is shown in Figure 4.89.

Another means of artificially adding short period damping is via utilization of a pure pitch attitude sensor (Figure 4.89). This device is not generally very good because it also increases the frequency of the short period motion considerably. This results in a very "rough ride" in turbulent air, particularly at high dynamic pressure.



Classical Influence of Increasing Pitch Rate Damping, $M_{\dot{\theta}}$, on Longitudinal Motion

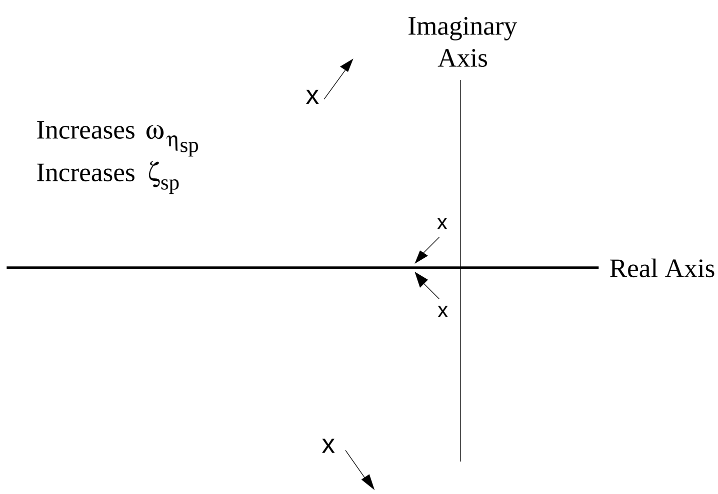


Figure 4.89
Effect of Increasing $M_{\dot{\theta}}$

4.8 TEST PROCEDURES AND TECHNIQUES MANEUVERING TASKS

4.8.1 Preflight Procedures

A thorough investigation of longitudinal flying qualities during maneuvering tasks must begin with thorough preflight planning. The purpose and scope of the investigation must be clearly defined, then a plan of attack or method of test can be formulated.

Preflight planning must start with research. This includes a study of the airplane and a thorough study of the longitudinal control system - including stability and control augmentation if installed. The design of the longitudinal control system should have a major influence on both the scope of the investigation and the emphasis during the investigation. The theory presented earlier for longitudinal maneuvering stability should provide excellent direction to the test pilot and engineer in formulating a test program for the investigation of longitudinal flying qualities during maneuvering tasks. For example, major emphasis during maneuvering stability tests on airplanes with reversible control systems should be on the linearity of the longitudinal control force variation with normal acceleration at several selected trim airspeeds. Conversely, for airplanes with irreversible control systems, major emphasis should be placed on the variation of the longitudinal control force - normal acceleration gradient (stick force per g) with altitude and airspeed or Mach number. Theory, although not always complete and not always classically applicable to the practical tests, generally leads to the emphasis presented above because of the following:

1. Nonlinear hinge moment characteristics at large elevator deflections and high Mach numbers can generate serious nonlinearities in maneuvering control forces for the airplane equipped with a reversible longitudinal control system. The nature of the irreversible control system results in no aerodynamic force feedback to the pilot from nonlinear hinge moments.

2. Reversible control system are usually utilized in airplanes with relatively restricted flight envelopes. This fact, in conjunction with knowledge of the characteristics of the reversible control system, leads to the rationalization that the gradient of longitudinal control force with normal acceleration should not vary greatly throughout the operational flight envelope for these airplanes. This is generally not so for the irreversible control system. Because of the large flight envelopes usually associated with airplanes possessing irreversible control systems and the characteristics of irreversible control systems the gradient of longitudinal control force with normal acceleration can vary drastically within the operational flight envelope.

Preflight research also involves reviewing all available information on longitudinal stability and control characteristics. Much useful knowledge may be gained from conferences with pilots and engineers familiar with the airplane.

The particular maneuvering tasks to be investigated must be determined and clearly understood by the flight test team. These tasks, of course, depend on the mission of the airplane. Knowledge of the mission and the maneuvering tasks allows determination of appropriate test conditions - configurations, altitudes, centers of gravity, trim airspeeds, and gross weights. Test conditions should be commensurate with the mission environment of the airplane. Center of gravity position is, of course, extremely critical for these tests. If flight test time permits, tests at the most aft and most forward operational CG positions should be performed after an adequate build-up program. If flight time is limited, tests should be performed (with care) at the most aft operational CG position (aft critical loading). Note: Maneuvering longitudinal control force gradients (stick force per g) may actually limit forward and/or aft CG positions for operational use. If the test program is aimed at determining these limits, appropriate CG restrictions will be promulgated or recommended by the test activity or higher authority.

The amount and sophistication of instrumentation will depend on the purpose and scope of the evaluation. A good, meaningful qualitative investigation can be performed with only production cockpit instruments and portable instrumentation-hand-held force gauge and stopwatch. Automatic recording devices, such as oscillograph, magnetic tape, and telemetry, are very helpful in rapid data acquisition and may be essential in a long test program of quantitative nature. Special sensitive cockpit instruments are also very useful, not only aiding in data acquisition but also aiding in stabilization for equilibrium tests points.

The final step in preflight planning is the preparation of pilot data cards. An example of a longitudinal stability and control data card for the investigation of maneuvering tasks is shown in Figure 4.90. Many test pilots desire to modify data cards to their own requirements or construct data cards for each tests. At any rate, the data cards should list all quantitative information desired and should be easy to interpret in flight. Blank cards should be utilized for appropriate qualitative pilot comments.

LONGITUDINAL STABILITY AND CONTROL RECORD MANUEVERING TASK	CARD NUMBER

AIRPLANE TYPE	PILOT	PTR-BIS
BUREAU NUMBER	T.O. GROSS WEIGHT	DATE

T.O. CG GEAR DOWN ____ %MAC GEAR UP____ %MAC	T.O. TIME ________ LAND TIME ________
EXTERNAL LOADING	CONFIGURATION

TRIM AIRSPEED MACH ________	POWER ________	ALT. ________	LONG. TRIM ____________

BREAKOUT & FRICTION CN ________	CONTROL SYSTEM MECHANICAL CHARACTERISTICS	FREEPLAY
CONTROL SYSTEM OSCILLATIONS CN ____________		CENTERING

METHOD	MANUEVERING LONGITUDINAL STABILITY					AIRSPEED DEVIATION
	CN	TARGET N	NORMAL ACCELERATION N	Fs	δ_e OR δ_s	
TRIM SHOT FUEL ________		1.0		0		0
FUEL ________						

WIND-UP TURN FUEL ________ CN ________	REMARKS:

METHOD	SHORT PERIOD CHARACTERISTICS		HALF-CYCLE AMPLITUDE RATIO
	CN	PERIOD	
TRIM SHOT FUEL ________			

PIO INVESTIGATION

EASE OF TRIM TO $F_s = 0$	TRIMMABILITY TRIM SENSITIVITY	TRIM RATE

Figure 4.90
Longitudinal Stability and Control Record for Maneuvering Tasks

4.8.2 Flight Test Techniques

4.8.2.1 THE QUALITATIVE PHASE OF THE EVALUATION

Longitudinal stability and control characteristics must be evaluated in relation to their influence on various maneuvering mission tasks. Therefore, the test pilot must devote a portion of the evaluation to performing or simulating the maneuvering tasks which have been selected. While performing these tasks, the test pilot gains the essential qualitative opinion of the longitudinal flying qualities and should assign handling qualities ratings. Without recording a single item of data, the test pilot should be able to form a good opinion of the mission effectiveness of the airplane, at least for the particular tasks being evaluated. This opinion will be based on the amount of attention and effort the pilot must devote to "just flying the airplane". Due consideration should be given during this phase of the test to the following considerations:

1. Whether the mission tasks will be performed in VFR and IFR weather, or strictly VFR conditions.

2. The amount of time and effort the pilot must devote to duties other than "just flying the airplane" - duties such as setting up a weapons system, coordinating multiplane tactics, communicating with other aircraft or a controlling station, etc.

3. If stability or control augmentation systems are installed, the consequences of their failure.

The test pilot's qualitative opinion of the airplane's longitudinal flying qualities in relation to the selected mission task is the most important information to be obtained. Therefore, this phase of the test must not be overlooked. The test pilot probably will have some ideas as to the particular characteristics which make the airplane easy or hard to fly even before proceeding to the quantitative phase of the testing. Use of the quantitative test techniques described below hopefully allows the test pilot to substantiate his qualitative opinion.

4.8.2.2 MEASUREMENT OF THE MECHANICAL CHARACTERISTICS OF THE LONGITUDINAL CONTROL SYSTEM

Mechanical characteristics of the longitudinal flight control system have been previously introduced earlier in this section on Longitudinal Flying Qualities. Therefore, test techniques for measuring mechanical characteristics will not be restated. This discussion is mainly concerned with the direct effects of mechanical characteristics on longitudinal flying qualities during maneuvering tasks.

4.8.2.2.1 Breakout Forces, Including Friction

Friction in the longitudinal control system, since it is usually small, generally has little or no effect on maneuvering handling qualities. However, if friction (without breakout) is very large, longitudinal flying qualities during maneuvering tasks may be seriously degraded. A large amount of friction would introduce poor control “feel” in maneuvering flight in that the friction would necessitate significant longitudinal control force inputs before an airplane response would be apparent (Figure 4.91). This would be particularly true while maneuvering at low values of normal acceleration since the friction would effectively “mask” the airplane’s true “stick force per g” gradient, particularly if the gradient were rather shallow.

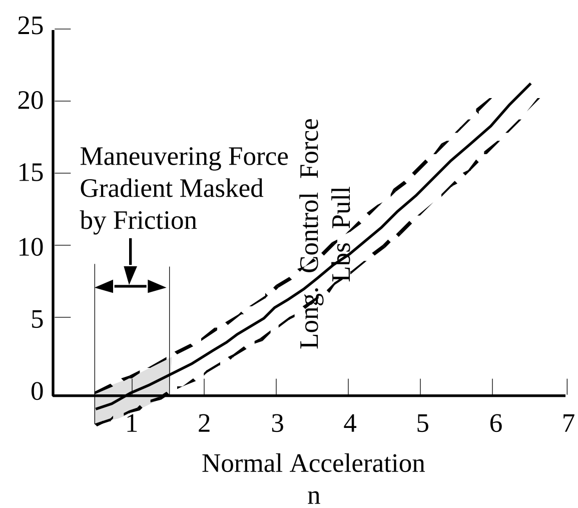


Figure 4.91
Longitudinal Control Friction Effects on Maneuvering Force Gradient

A judicious amount of longitudinal control breakout force generally is beneficial to longitudinal flying qualities during maneuvering tasks. It may reduce excessive sensitivity in longitudinal control feel about trim for certain airplane flight conditions (high natural frequency and low damping of the airplane short period mode, low "stick force per g" gradient). Addition of some breakout force may reduce otherwise severe pilot-induced-oscillation (PIO) tendencies for these flight conditions. However, if too much breakout force is added, the pilot feels a "lag" in the control system which may cause him to overcontrol (attempt to drive the airplane to the response he desires) and generate pilot-induced-oscillations.

Breakout forces must be suitably matched to the longitudinal maneuvering stability characteristics of the airplane. A combination of large breakout and shallow gradient of longitudinal control forces in maneuvering flight (Figure 4.92) results in artificial maneuvering force nonlinearity about trim. This generates poor longitudinal control feel when the pilot attempts to track precisely at low values of normal acceleration.

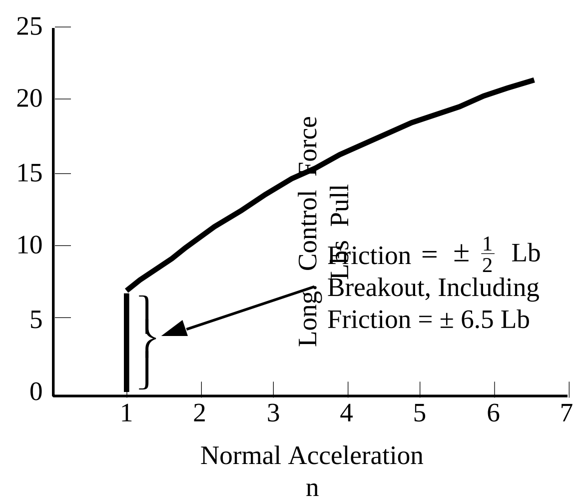


Figure 4.92
Poor Matching of Longitudinal Maneuvering Force Gradient and Breakout Forces

In summary, some longitudinal breakout force is usually beneficial to longitudinal flying qualities during maneuvering tasks, however, too much results in undesirable characteristics. Friction generally should be as small as possible in the longitudinal control system.

4.8.2.2.2 Freeplay

Freeplay in the longitudinal control system should be as small as possible. Excessive freeplay results in difficulty in performing precise tracking tasks at low values of normal acceleration about trim. The pilot will generally resort to tracking slightly "out of trim" during precise maneuvering to avoid continually moving the longitudinal control stick through a large "dead band" of freeplay.

4.8.2.2.3 Centering

Positive centering of the longitudinal cockpit control stick contributes to good longitudinal flying qualities during maneuvering tasks; positive centering allows the pilot to change normal acceleration, angle of attack, and pitch attitude toward the trim (one g) condition merely by relaxing forward or aft force on the control stick.

4.8.2.2.4 Control System Oscillations

Oscillations in the elevator control surface and the entire longitudinal control system, initiated by either external perturbations or pilot inputs, should be essentially deadbeat. Lightly damped or undamped motion can result in annoying motion in the cockpit control stick during rapid maneuvering, as well as objectionable oscillations in normal acceleration.

4.8.2.2.5 Measurement of Longitudinal Maneuvering Stability

Longitudinal maneuvering stability characteristics have been shown to have a major influence on the pilot's opinion of the airplane during maneuvering tasks. In particular, the longitudinal control force variation with normal acceleration, or "stick force per g", is a primary "control feel" parameter. This parameter is of tremendous importance for airplanes which will be maneuvered extensively in operational use; however, it must be investigated to some degree in all airplanes, irregardless of their missions.

4.8.2.2.6 Stick Force per g

The pilot's opinion of the maneuvering capabilities of the airplane are directly related to the "stick force per g" gradient; therefore, it is necessary to design the airplane very carefully to maintain this gradient within acceptable limits. The acceptability of a particular airplane's "stick force per g" gradient will generally depend on at least the following considerations:

1. The amount of maneuvering and the nature of the maneuvering tasks required for mission accomplishment. If the airplane is designed to be maneuvered extensively, the "stick force per g" gradient must be low enough so that the pilot is not fatigued excessively. However, the "stick force per g" gradient must not be too low or the control feel may be too light and sensitive. Additionally, there may also be inadequate protection against inadvertent overstress with a low force gradient.

2. The limit load factor, or "g tolerance" of the airplane. Obviously, the "stick force per g" gradient must be high enough to discourage inadvertent overstress. "Stick force per g" gradients must be higher for airplanes with low g - tolerances than for airplanes with high g- tolerances. The pilot rightly expects untrimmed stick forces to be high when the airplane is maneuvered near its limit load factor.

3. The type of cockpit longitudinal controller; i.e., whether the airplane is equipped with a wheel or center-stick controller. A wheel or yoke grip is usually located higher with respect to the pilot's seat than a center-stick, therefore, the pilot is able to exert larger forces, even with one hand. Considering also that the pilot is able to comfortably use both hands with a wheel controller leads to the rationalization that the <u>maximum</u> acceptable "stick force per g" gradients can be higher with a wheel controller than with a center-stick controller. Similarly, the <u>minimum</u> acceptable "stick force per g" gradient must generally be higher with a wheel controller because the pilot's arm is usually unsupported. The pilot has very good "vernier" control with a center-stick even with a low "stick force per g" gradient because the forearm is usually supported on the thigh.

4. There is some evidence from flying qualities investigations to indicate that "stick force per g" should be higher at low speeds than at high speeds. This is probably due to the fact that the pilot maintains tight control over normal acceleration at high speeds, then gradually switches to tight control of pitch attitude at low speeds. Thus the pilot tends to use "stick force per g" as a primary control feel parameter at high speeds, then switches to longitudinal control force per unit change in angle of attack $\left(\frac{F_s}{\alpha}\right)$ as a primary control feel parameter at low speeds. In order to utilize the same criteria $\left(\frac{F_s}{g}\right)$ for both slow and fast speeds, criteria for "stick force per g" at low speeds can be made inversely proportional to the parameter $\frac{n}{\alpha}$[6] (change in normal acceleration per unit change in angle of attack, a direct measure of how much rotation of the airplane is required to obtain the normal acceleration response). Use of this type of requirement can be justified by study of the following constant speed approximation:

$$\frac{F_s}{\alpha} = \left(\frac{F_s}{n}\right)\left(\frac{n}{\alpha}\right) \qquad \textit{eq 4.49}$$

It is very desirable that the plots of longitudinal control force versus normal acceleration be linear within the range of normal accelerations which would normally be attained during maneuvering tasks in operational use. Some nonlinearity must be expected in all airplanes; however, the departure from linearity should not cause excessive differences between the local "stick force per g" gradient and the average "stick force per g" gradient. The local gradient is defined by the slope of a tangent to the curve at any point. The average gradient is defined by the slope of a line drawn from the lg point where breakout including friction is overcome to the point on the curve under consideration (see Figure 4.93). In general, a departure from linearity which results in the local gradient differing from the average gradient by more than 50 percent is considered excessive.

---

6 $\frac{n}{\alpha}$ is directly proportional to the slope of the airplane lift curve and the square of the airplane velocity.

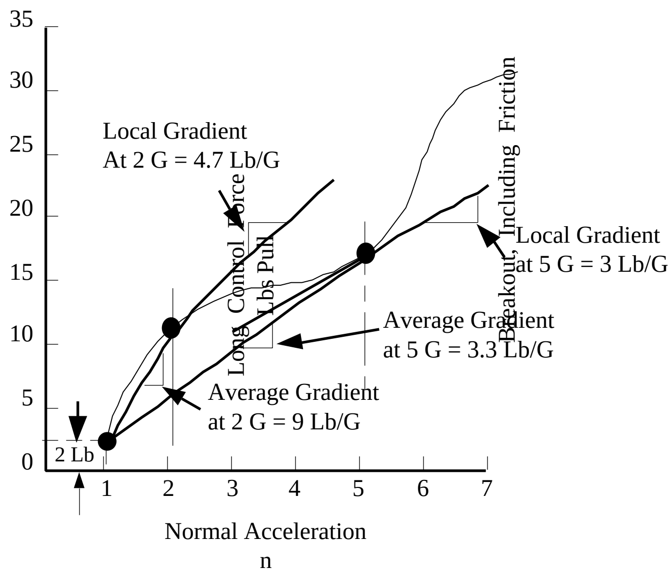


Figure 4.93
Graphical Explanation of Local Gradients and Average Gradients of Stick Force per g

4.8.2.2.7 Transient Control Forces

During abrupt maneuvers, the longitudinal control forces must not be too light, or the pilot may inadvertently overstress the airplane while attempting to maneuver rapidly. A satisfactory "stick force per g" gradient in steady, smoothly controlled flight is not absolute assurance that transient control forces will not be too low. Essentially, it should be more difficult to overstress the airplane during abrupt, sudden maneuvers than during steady maneuvers. Thus, one requirement on transient longitudinal control forces is that the control force required to attain a certain normal acceleration in a sudden or abrupt maneuver should not be objectionably light and the buildup of control force during the maneuver entry shall lead the buildup of normal acceleration. Another criterion which has been developed is a requirement on the ratio of longitudinal control force to normal acceleration during maneuvers in which the pilot sinusoidally pumps the longitudinal control at various frequencies. This criterion states that these ratios should always be greater than 3.0 pounds per g for a center-stick controller and 6.0 pounds per g for wheel controller.

4.8.2.2.8 Elevator Position per g

Of lesser influence on the pilot's opinion of the airplane during maneuvering tasks is the variation of elevator position with normal acceleration in maneuvering flight, or "elevator position per g". However, a positive elevator position gradient in maneuvering flight; i.e., increasing trailing edge up elevator deflection for increasing positive normal acceleration is essential for satisfactory unaugmented longitudinal flying qualities; it is also indicative of good basic airplane design. No maximum or minimum limits are placed on the elevator position variation in maneuvering flight. The only criterion is that increases in trailing edge up elevator deflection shall be required to maintain increases in positive normal acceleration throughout the range of attainable acceleration.

4.8.2.2.9 Stick Position per g

The longitudinal cockpit control motion required in maneuvering flight has some effect on the pilot's opinion of the airplane during maneuvering tasks. Qualitative and quantitative criteria have been developed for the variation of cockpit control position with normal acceleration in maneuvering flight. "Stick position per g" should at least be positive-increasing aft cockpit control position required to maintain increases in positive normal acceleration - and the cockpit control motions required should not be so large or small as to be objectionable. A quantitative criterion that has been developed for Category A Flight Phases is as follows: the average gradient of longitudinal control force per inch of cockpit control motion during maneuvering flight should not be less than 5.0 pounds per inch for Levels 1 and 2 (this is actually designed to discourage excessive control motion). However, flying qualities investigations have shown fairly conclusively that <u>some</u> finite "stick position per g" level is desirable during maneuvering tasks. The main benefit of the stick motion during maneuvering is the "filtering action" which the stick motion has on the pilot's control inputs. For instance, if an airplane exhibits a shallow longitudinal control force gradient in maneuvering flight and little or no "stick force per g" gradient, the pilot has little longitudinal control "feel" in terms of either force or motion and he may tend to overcontrol during precise maneuvering tasks. The "stick force per g" gradient for this case may be optimum for the mission and characteristics of the airplane; if so, increasing the stick motion during maneuvering flight may be the solution for the overcontrolling tendency.

4.8.2.2.10Steady Pull-Ups

Test techniques which may be used to measure maneuvering stability characteristics - "stick force per g" "elevator position per g," "stick position per g," and $\frac{n}{\alpha}$ will now be introduced. The first technique to be presented is the steady pull-up method.

The steady pull-up method involves obtaining data at a constant power setting, a constant longitudinal trim setting, and a constant airspeed (trim airspeed) while varying normal acceleration by varying pitch rate during stabilized wings-level pull-ups. It is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, a "trim shot" should be taken. Record appropriate data such as power, longitudinal trim setting, trim elevator and/or stick position, and <u>fuel quantity</u>. Note any correction to be applied to cockpit sensitive accelerometer readings ("tare" correction) and set the floating pointers of the accelerometer to 1 g.

2. Without changing power or trim settings, decelerate in climbing attitude (zoom climb) then push over to enter a shallow dive toward the original trim altitude. As the airspeed increases toward the trim airspeed, steadily apply a pull longitudinal control force to establish a nose-up pitch rate and increase normal acceleration to approximately that selected for the test point.

3. Is using the hand-held force gauge on a center-stick controller, the force input must be made <u>through</u> the force gauge; i.e., with the force gauge already applied to the control stick. The <u>transient force input</u> necessary to initiate the pitch rate may be different from the <u>steady force input</u> required to maintain the established normal acceleration. The test pilot must keep in mind that the floating pointer of the force gauge will remain at the maximum force applied, which may not be the <u>steady force</u> which he desires to measure.

4. For a short period of time during the steady wings-level pull-up, airspeed, longitudinal control force, and normal acceleration will be stabilized. During this period of time, the pilot should activate the automatic recording devices and mentally note stick force, stick or elevator position, and normal acceleration. If using the hand-held force gauge, look at it quickly during this period; do not rely solely on the floating pointer. For wheel-or yoke-control airplanes, it is possible to establish the normal acceleration with one hand while holding the force gauge in readiness with the other. Then when stabilization is attained, the force gauge can be applied quickly and the steady control forces measured.

5. Airspeed control is critical for this test. Deviations in airspeed from trim airspeed of more than ± 5 KIAS during data gathering is considered unacceptable.

6. Altitude would be within ± 2000 feet of the base altitude during the stabilized portion of the steady pull-up. Pitch attitude during the gathering of data should be within ± 15 degrees of the original trim pitch attitude.

7. The technique of arriving at the desired airspeed, altitude, and attitude with approximately the desired g is difficult, but can be mastered with practice. (Do not discard an otherwise perfect data point if the exact target g is not attained. A reasonable spread of normal acceleration is all that is required.)

8. After the run, the pilot should decelerate in a zoom climb in preparation for the next data point while recording appropriate information on the pilot's data card: counter number (if applicable), g attained, stick force, stick and/or elevator position, and deviation from trim airspeed (if any) during measurement.

9. Normal acceleration should be increased in steps from near lg toward the maximum useable in operational use. The maximum useable may be limited by structural considerations, severe buffeting, or stall. If any of these limiting cases are reached, no further efforts should be made to increase the applied normal acceleration. (Buffet onset normal acceleration should be noted, if

reached prior to maximum useable normal acceleration, since it is indicated on the data plots. (Nonlinearities in "stick force per g" and $\frac{n}{\alpha}$ usually occur at normal acceleration levels past buffet onset.)

10. As applied normal acceleration increase, the deceleration prior to entering the dive, the steepness of the dive, and the rapidity of the control force input to initiate the pitch rate must be increased. As a matter of fact, for some airplanes, the pull-up for high-g points may have to be initiated at airspeeds faster than trim airspeed because it may be impossible to keep the airplane from decelerating as the normal acceleration is applied.

4.8.2.2.11Steady Pushovers

The steady pushover is probably the optimum method of obtaining maneuvering stability characteristics at less than 1g. This method is simply a "reverse steady pull-up." It is performed exactly as the steady pull-up except:

1. A dive is entered initially to increase airspeed from trim, then a climb is initiated toward the original trim altitude. As the airspeed decelerates toward the trim airspeed, steadily apply a push longitudinal control force to establish a nose-down pitch rate and decrease normal acceleration.

2. For a short period of time during the steady, wings-level, pushover, airspeed, longitudinal control force, and normal acceleration will be stabilized. Record or note pertinent parameters at this time.

3. Minimum normal acceleration attainable during these tests will probably be limited by the trailing edge down elevator deflection stops or the airplane structural units.

4.8.2.2.12Steady Turns

This method involves obtaining data at a constant power setting, constant longitudinal trim settings, and a constant airspeed (trim airspeed) while varying normal acceleration by varying pitch rate during stabilized turns in both directions. This method is somewhat easier than wings-level steady pull-ups because the test pilot has a better opportunity to stabilize exactly on trim airspeed and normal acceleration. Additionally, because of the nature of the technique, the stabilized condition can be maintained for a longer time period, which facilitates obtaining all the required data. Steady turns are performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, a "trim shot" should be taken. Record appropriate cockpit data such as power, longitudinal trim setting, trim elevator and/or stick position, and <u>fuel quantity</u>. Note any correction to be applied to cockpit sensitive accelerometer readings ("tare" correction) and set the floating pointers of the accelerometer to 1g.

2. Without changing power or trim settings, roll the airplane slowly and smoothly into a turn while simultaneously lowering the nose slightly to <u>maintain trim airspeed</u>. If using the hand-held force gauge on a center-stick controller, the longitudinal force input will have to be made <u>through</u> the force gauge.

3. When well stabilized on trim airspeed, bank angle, and normal acceleration, mentally note longitudinal control force, stick or elevator position, and normal acceleration. If using the hand-held force gauge, <u>look at it</u> if possible during this period. If the airplane is equipped with a wheel controller, stabilize with one hand while holding the force gauge in readiness with the other, then apply the force gauge and measure the force. If automatic recording devices are utilized, "take a picture" of the stabilized condition.

4. After the run, roll wings level and climb in preparation for the next test point while recording appropriate cockpit data: counter number (if applicable), g attained, stick force, stick and/or elevator position, and deviation from trim airspeed (if any) during measurement.

5. Airspeed again is the critical parameter for this test. The test pilot must note or record data only when stabilized precisely on trim airspeed. Deviation from trim by more than $\pm$ 5 KIAS is considered unacceptable.

6. Stabilized data points should be obtained at 30, 45, and 60 degrees of bank angle, then in approximately one-half g increments to the maximum useable normal acceleration. Again, do not discard a perfectly good data point if the exact value of normal acceleration is not attained. A reasonable spread of normal acceleration is all that is necessary. Only slight increases in bank angle are necessary above 2 g in order to increase substantially the g increment.

7. Little altitude is generally lost for the stabilized points at 60 degrees of bank or less; therefore, a considerable time interval can be spent attaining good stabilization without exceeding the allowable altitude band (base altitude ± 2000 feet). At greater bank angles (higher normal acceleration), the test pilot should start above the test altitude prior to entering the steady turn. Obviously, at these higher levels of normal acceleration, stabilization must be quicker because altitude is being lost rapidly.

8. At the higher normal acceleration levels (60-90 degrees of bank), top or bottom rudder should be utilized as an aid in precise airspeed control. A little bottom rudder can salvage a run if the airspeed starts to decrease. Usually, if airspeed increases sharply, top rudder will not be effective in stopping the increase, thus the run must be aborted.

9. Both left and right steady turns should be performed. For jet airplanes, little variation in maneuvering stability characteristics is generally attributable to the direction of turn. For propeller driven types, large differences may be noted due to direction of the turn; these differences are usually caused largely by "gyroscopic effects."

10. The time and effort required to obtain maneuvering stability characteristics at less than 1 g in steady turns is excessive. Therefore, these characteristics should be obtained during steady wings-level pushovers previously described.

4.8.2.2.13Wind-Up Turns

The third method which may be used to obtain maneuvering stability data is the "wind-up turn". This technique is exactly the same as the "alternate technique for accelerated stall investigations" presented previously. It merely involves gradually increasing normal acceleration from 1 g to maximum useable in a wind-up turn (left or right) at constant airspeed. The wind-up turn is a convenient method to utilize for obtaining a large amount of data in a short period of time if automatic recording devices are utilized; it is also a good "quick look" qualitative technique even without automatic recording devices. The wind-up turn should be performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. Record a "trim shot" with the automatic recording devices. Record appropriate cockpit data such as power, longitudinal trim setting, trim elevator and/or stick position, and <u>fuel quantity</u>.

2. Actuate the automatic recording devices and smoothly and slowly roll into the wind-up turn. Increase normal acceleration smoothly and slowly by gradually increasing bank angle and aft stick position while <u>maintaining airspeed constant</u>. At high levels of normal acceleration (bank angles greater than 60 degrees), use rudder inputs to aid in airspeed control. Actuate the event marker at predetermined g increments, at buffet onset, and at maximum useable normal acceleration. Deactivate the instrumentation while recovering to 1 g flight conditions. Record counter number and set up for the next run.

3. Some flight test activities have advocated the "wind-down turn" as a means of obtaining maneuvering stability data at less than 1 g normal acceleration. However, this method requires extreme pilot skill and is hardly worth the time and effort involved.

4.8.2.2.14Sinusoidal Stick Pumping, Out-of-Trim Releases, and Sudden Pull-Ups

Three methods will be introduced through which transient control force requirements in abrupt maneuvers may be determined. The technique to be utilized in a particular test program will depend on the amount of instrumentation available and the quantitative requirement being used as a criteria.

Sinusoidal stick pumping at various frequencies can be used to determine the minimum transient stick force per g ratio. (The minimum stick force per g ratio results when the control system is pumped sinusoidally at a frequency close to (the closeness depending on short period damping ratio) the stick free airplane short period natural frequency.) This technique requires automatic recording devices in the test airplane. The procedure is merely to trim the airplane in the desired configuration at the desired flight condition, then merely pump the cockpit control stick fore and aft sinusoidally at various frequencies. The test pilot should attempt to include the frequency at which maximum normal acceleration response is obtained for the lowest control force inputs. The amplitudes of fore and aft stick motion, push and pull stick forces, and positive and negative load factor excursions should be as nearly equal as possible. The sinusoidal stick pumping is recorded on the automatic recording traces. Typical results are shown in Figure 4.94. The minimum ratio of stick force per g in the transient maneuver should be greater than 3.0 pounds per g for center-stick controllers and 6.0 pounds per g for wheel controllers.

Out-of-trim stick releases is a method of "artificially" introducing a rapid pull-up. The airplane is trimmed in the desired configuration and flight condition. It is then rolled into a steady turn and stabilized at a desired normal acceleration. Maintaining the steady turn, the test pilot notes the stick force required, then reduces it to zero by retrimming. The airplane is then rolled out of the turn, and without retrimming, returned to the test altitude and trim airspeed. (The trim airspeed is stabilized with wings level by maintaining a push force on the control stick.) The pilot then merely releases the stick and notes the peak normal acceleration response. The stick force required in the steady turn and the peak normal acceleration response provide a point which may be compared to the "stick force per g" gradient for steady pull-ups. Caution should be exercised in performing this test; start with low g points and build up to higher values of normal acceleration. This test is not rigorous although valuable qualitative information can be obtained. The validity of this test is particularly questionable if the longitudinal control system has appreciable friction or if control system centering is poor.

The sudden pull-up merely involves measuring the ratio of longitudinal control force to normal acceleration change with various rates of cockpit control motion. The airplane is trimmed in the desired configuration and flight condition. If a hand-held gauge

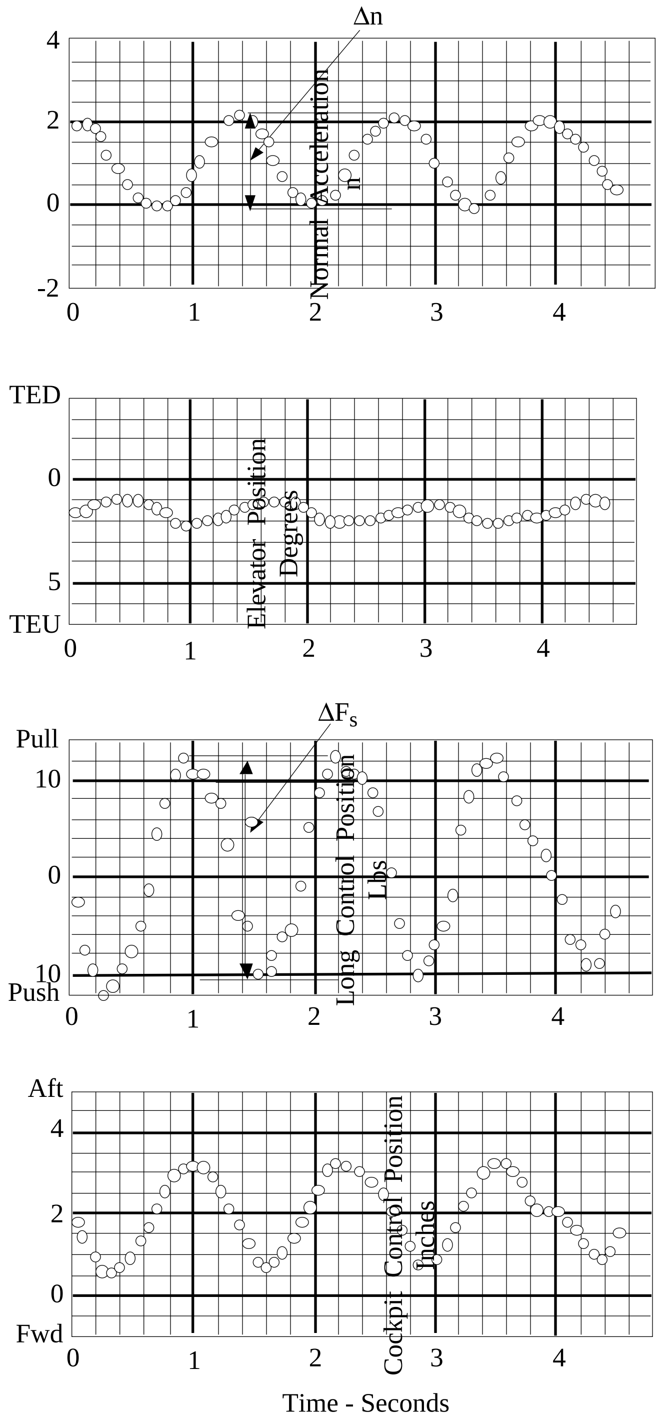


Figure 4.94
Typical Data from Sinusoidal Stick Pumping

is being used it is applied to the cockpit control stick so that the abrupt force input can be made through the force gauge and recorded with the floating pointer. The cockpit control stick is then smartly and rapidly deflected to the rear a predetermined, safe amount, then returned to the trim position. The peak longitudinal control force and peak normal acceleration during the abrupt maneuver are noted. This ratio is then compared to “stick force per g” gradients in steady pull-ups. The sudden pull-ups should be performed with various rates of cockpit control motion; the total elapsed time for the cockpit control input (from start to return to the trim position) should be varied from approximately one-half to 6 seconds. If the airplane is instrumented for automatic recording of stick force, normal acceleration, and elevator position, a continuous record of the entire maneuver will yield the necessary quantitative information. This test should be performed with due caution; the test pilot should make initial elevator inputs rather small until a good feel for the “g-response” in abrupt maneuvers is obtained.

4.8.2.3 MEASUREMENT OF LONGITUDINAL SHORT PERIOD CHARACTERISTICS

Damping and frequency (or period) of the airplane short period mode of motion have been shown to have a profound effect on overall longitudinal flying qualities. However, it is most appropriate to investigate the characteristics of this motion during maneuvering tasks because of the effect of these characteristics on the <u>response</u> of the airplane to external perturbations or longitudinal control inputs. It is necessary to discuss the effect of varying short period characteristics by varying only one parameter at a time. For the discussion of the effect of varying short period frequency it is assumed that the damping ratio of the short period motion is fixed at an acceptable level.

4.8.2.3.1 Short Period Frequency

The parameter, $\omega_{d_{sp}}$ is the damped frequency of the second-order, short period mode of motion. If it is a real, positive number, it is directly related to the physical frequency (or quickness) with which the airplane responds to an elevator input or an external disturbance. In visual flight, the pilot notes this frequency of response by reference to the pitch attitude of the airplane, the normal accelerometer, or angle of attack indicator. The pilot is also sensitive to this frequency of response through the normal acceleration he feels. When flying by reference to instruments, the normal accelerometer, angle of attack indicator, and "normal acceleration feel" provide cues of the response frequency of the airplane. Obviously, the damped frequency of the short period mode of motion has a very large influence on the pilot's opinion of the longitudinal flying qualities of the airplane. However, the damped frequency is dependent on damping ratio as well as the undamped natural frequency. Therefore, airplane short period flying qualities requirements and data are usually presented in terms of the undamped natural frequency, $\omega_{n_{sp}}$ and damping ration ζ_{sp}. Although the undamped natural frequency might seem to be of academic interest only (at first glance) it will now be shown that it is actually a useful means of describing the longitudinal maneuvering behavior of the airplane as the pilot sees it.

With satisfactory damping of the short period mode, the following rationalizations may be made concerning the effect of various short period natural frequencies on longitudinal flying qualities:

1. For "low" $\omega_{n_{sp}}$ values - the pilot finds that the airplane tends to "dig-in" during maneuvering. This characteristic is explained by the fact that the airplane does not respond quickly enough initially to the pilot's control input. The pilot therefore tends to put in too large an input when attempting to make a rapid flight path change, such as a sharp pull-up or rapid turn entry. The large input yields the desired initial response; however, the pilot soon finds that the final response, once it develops, is more than he wanted. Thus it is the initial response which the pilot finds lacking when attempting vigorous maneuvering tasks at low short period natural frequencies. If the airplane is always maneuvered slowly and smoothly, the pilot probably does not object to the slow initial response. (The large transport or passenger airplane, with large moments of inertia in pitch, are characterized by low short period natural frequencies. Since these airplanes do not have to be maneuvered extensively in their missions, however, the pilot may feel the response characteristics are perfectly satisfactory.) Trimmability may be impaired somewhat if $\omega_{n_{sp}}$ is too low. This is due to the fact that every trim input the pilot makes requires a relatively long time interval to take effect. Thus the pilot thinks he is in trim initially, but finds later that a little further trim correction is necessary. The pilot does not have a good, firm knowledge of when the trim setting is exactly correct.

2. If the short period undamped natural frequency is "medium" to "high", the response of the airplane to longitudinal control inputs is generally satisfactory for maneuvering tasks. The airplane is quick responding longitudinally and the pilot will generally feel very confident during gunsight tracking or bombing deliveries. During vigorous maneuvering, the pilot has a strong, positive feeling that the normal acceleration response will be exactly what was desired when the elevator input was made. This "predictability factor" is important to the pilot. Additionally, the medium to high short period frequency enhances longitudinal trimmability. With the medium to high frequency, every correction made during the task of trimming takes less time and comes to a completion quicker. This gives the pilot the feeling that he knows exactly what trim correction is necessary. In other words, the airplane's longitudinal trim point is well defined and corrections to the trim point are made quickly.

3. For "very high" $\omega_{n_{sp}}$ values - the pilot may complain that the initial response of the airplane is too fast or too quick. This is due to the fact that the high natural frequency makes the airplane too sensitive and responsive to very small longitudinal control inputs. During precise tracking maneuvers, the pilot tends to "bobble" the nose position of the airplane. This may impair precise placement of ordnance during certain maneuvering tasks required in mission accomplishment. If the airplane is flown in turbulence, it may respond so abruptly through angle of attack and normal acceleration changes that the pilot is subjected to an uncomfortable, teeth-rattling ride. Flying qualities investigations have shown that increasing "stick force per g" gradients tend to attenuate the sensitivity and "bobbling" tendencies associated with high short period natural frequencies. The higher $\left(\frac{F_s}{n}\right)$ gradients merely require the pilot to use <u>larger</u> force inputs during any maneuver, which tends to decrease the initial abruptness and sensitivity experienced with lighter $\left(\frac{F_s}{n}\right)$ gradients. However, this type of compromise is never completely satisfactory since <u>steady</u> longitudinal control forces in pull-ups and turns may become excessive. If the short period natural frequency is <u>very high</u>, even the best compromise value of $\left(\frac{F_s}{n}\right)$ cannot make the maneuvering characteristics acceptable.

4.8.2.3.2 Short Period Damping

The parameter, ζ_{sp}, is the damping ratio of the short period mode of motion. Its value strongly affects the time or dynamic response of the airplane to a longitudinal control input or an external disturbance. Pilots are very sensitive to this parameter. It may be detected in visual flight by observing the pitch attitude of the airplane as the airplane responds to an elevator input. The pilot notes the peak value and oscillatory nature of the response. The damping ratio can be detected in instrument flight by reference to the normal accelerometer or angle of attack indicator.

Short period damping ratio has a direct effect on piloting technique and the pilot's opinion of the longitudinal flying qualities of the airplane, particularly during maneuvering tasks. At a constant short period undamped natural frequency of reasonable value, the pilot's description of the airplane can be varied from "over-responsive" to "sluggish" merely by changing the damping ratio. Assuming a satisfactory $\omega_{n_{sp}}$, the following rationalizations may be made concerning the effect of various short period damping ratios on longitudinal flying qualities.

1. For <u>very low</u> damping ratios - the airplane short period motion is very easily excited by pilot inputs or external disturbances. Once excited, the motion (pitch attitude, normal acceleration, and angle of attack oscillations) tends to persist for a relatively long period of time. When the pilot attempts to maneuver the airplane vigorously, he finds the longitudinal response is oscillatory and the resulting oscillations in angle of attack and normal acceleration disconcerning and uncomfortable. Thus, the pilot will probably switch to cautious longitudinal control inputs in an attempt to keep from exciting the short period motion. Longitudinal control forces required in maneuvering will probably <u>feel</u> lighter to the pilot than the actual force gradient. This is because the initial response of the airplane is quicker than the pilot <u>thinks</u> it should be, therefore, the pilot <u>thinks</u> he applied more force than he should have applied.

2. For <u>low</u> short period damping ratios - the airplane short period motion is still quite apparent to the pilot, however, it is very noticeably damped. The pilot may still use somewhat cautious control inputs because a noticeable overshoot in desired angle of attack and normal acceleration occurs when large, abrupt

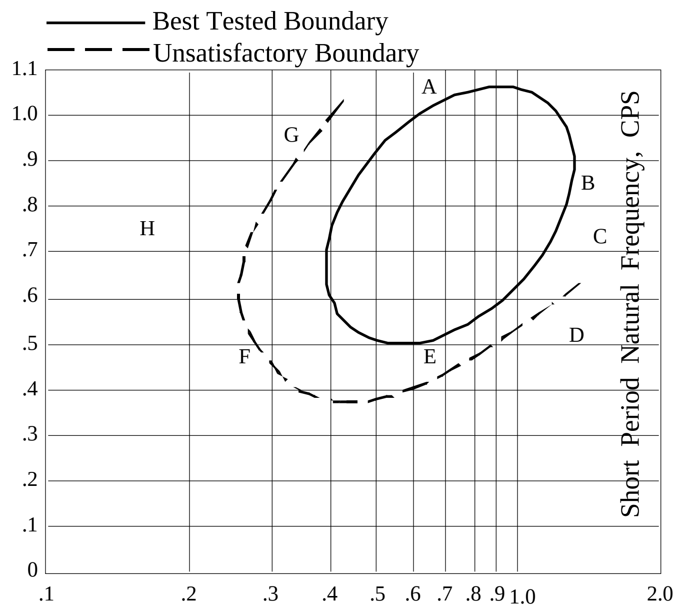
inputs are made. However, the pilot will feel more comfortable in maneuvering vigorously than he would with the very low short period damping. Longitudinal control forces in maneuvering flight may still seem a bit light.

3. For moderate short period damping ratios - the airplane short period motion is natural and predictable. The response of the airplane to a longitudinal control input is such that the pilot feels that he can change angle of attack, pitch attitude, and normal acceleration to whatever values he desires. In addition, the pilot feels that he can make these changes precisely without any overshoot or undershoot in amplitude. Longitudinal control forces during maneuvering flight feel normal. The pilot thus feels very secure in maneuvering the airplane vigorously; maneuvering tasks required in mission accomplishment are performed without undue pilot effort.

4. The fairly heavy and heavy short period damping, the airplane short period motion is not evident to the pilot. The response of the airplane to longitudinal control input approaches a steady state value with a minute overshort or it approaches the steady state purely asymptotically. As the short period damping ratio increases, the airplane response becomes slower and slower; the pilot resorts to "forcing" the initial response by applying large elevator inputs to get the response started. For this situation, the pilot describes the airplane as "sluggish" during maneuvering, and because he resorted to using large initial elevator inputs, longitudinal maneuvering control forces feel higher than normal.

4.8.2.3.3 The Short Period "Thumbprint."

The results of a pilot opinion study have indicated that there are combinations of short period undamped natural frequencies and damping ratios which provide satisfactory longitudinal flying qualities for tracking maneuvering tasks. A typical display of these combinations, or "thumbprint" as it is commonly called, is presented in Figure 4.95 with various pilot comments describing the airplane's longitudinal characteristics at appropriate points outside the thumbprint. (The "thumbprint" discussion (Figure 4.95) is valid for only one particular airplane; however, the trends are the same for all airplanes.)



Short Period Damping Ratio ζ
(Applicable for a Tracking Manuevering Task)

A. Moves in steps. Response initially a bit abrupt.

B. Response erratic or step-like. Forces too heavy. Not manueverable. Stick motion too great.

C. Not very manueverable. Stiff and sluggish. Force too heavy Good flying but not a fighter.

D. Bomber or heavy fighter. Not manueverable. Forces heavy and stick motion too great. Trims well.

E. Light bomber Not fighter type. Forces heavy. Too much stick motion. Not manueverable. Sluggish.

F. Response fast. Oscillatory. Difficult to track. Force initially light then stiffens.

G. Oscillatory, too closely coupled. Too responsive.

H. Dangerous - could exceed load factor. Highly oscillatory. Pilot reluctant to manuever. Very difficult to track.

Figure 4.95
The Short Period "Thumbprint"

4.8.2.3.4 Residual Oscillation

Any sustained residual oscillations in pitch should not interfere with the pilot's ability to perform the tasks required in the mission of the airplane. For levels 1 and 2, oscillations in cockpit normal acceleration of greater than $\pm$ .05g or pitch attitude oscillations greater than $\pm$ 3 mils (Category A Flight Phases) are considered excessive.

4.8.2.3.5 Additional Short Period Criteria

It is generally agreed that short period frequency and damping alone are not adequate to completely describe the acceptability or unacceptability of the short period response. An additional parameter has been utilized in an attempt to resolve discrepancies existing between the results of various pilot opinion studies in which only $\omega_{n_{sp}}$ and ζ_{sp} were considered. This parameter, $\frac{n}{\alpha}$, has been previously introduced in the discussion of maneuvering stability characteristics. The ratio of maximum pitching acceleration to steady state normal acceleration during maneuvering is approximately[7] equal to $\frac{\left(\omega_{n_{sp}}\right)^2}{n/\alpha}$. This pitching acceleration is the airplane longitudinal response which develops <u>earlier</u> during maneuvering. Many of the objections to both very high and very low short period natural frequencies are due to deficiencies in <u>initial response</u>. Therefore, requirements for short period natural frequency have been expressed as a function of $\frac{n}{\alpha}$; these requirements are designed to maintain $\frac{\left(\omega_{n_{sp}}\right)^2}{n/\alpha}$ essentially constant (see Figures 4.96 a, b, and c). Of course, short period damping is always extremely important and must also be maintained within acceptable limits.

---

[7] Assumptions of constant airspeed and a "high-frequency" control systems are made.

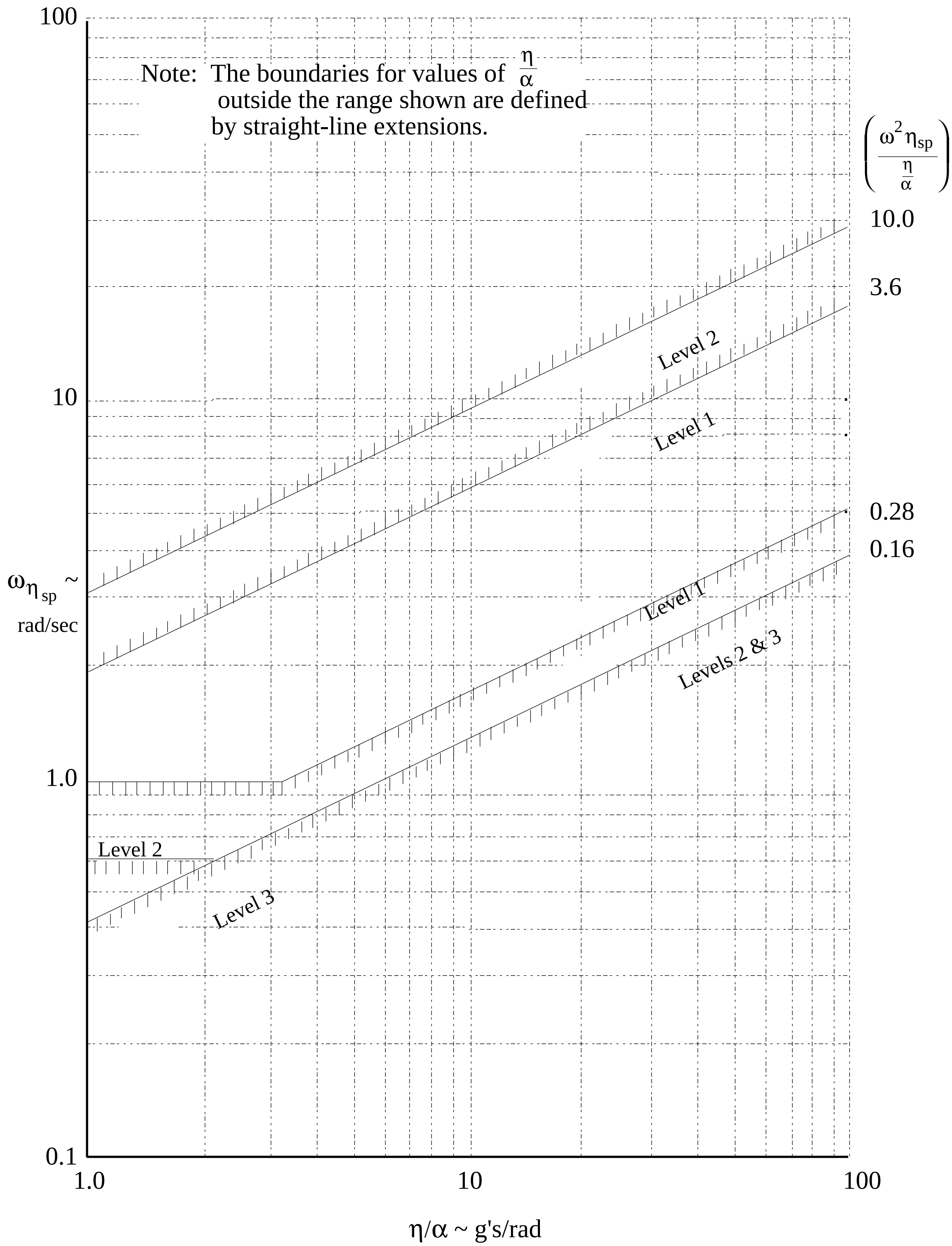


Figure 4.96a
MIL-F-8785B

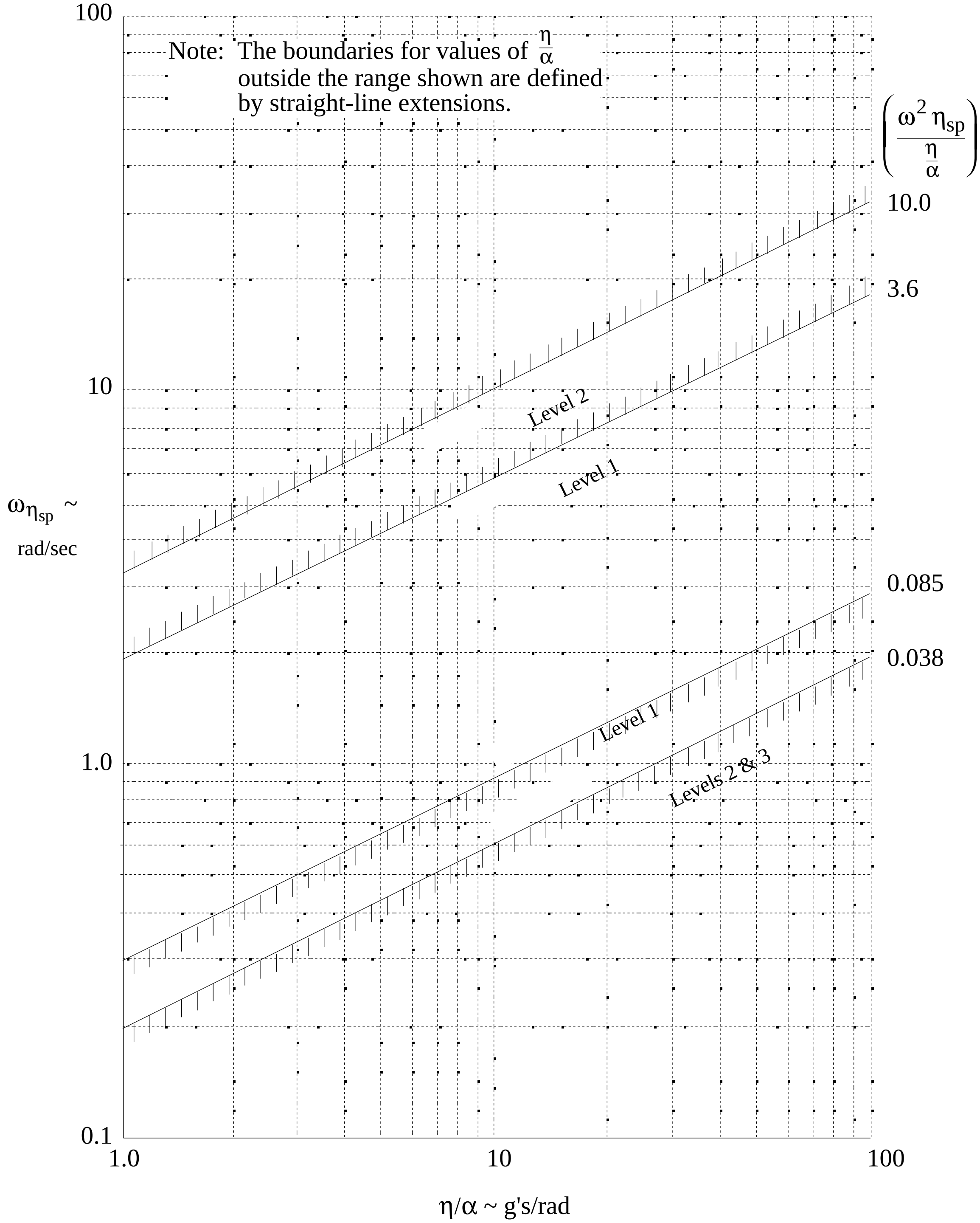


Figure 4.96b
MIL-F-8785B

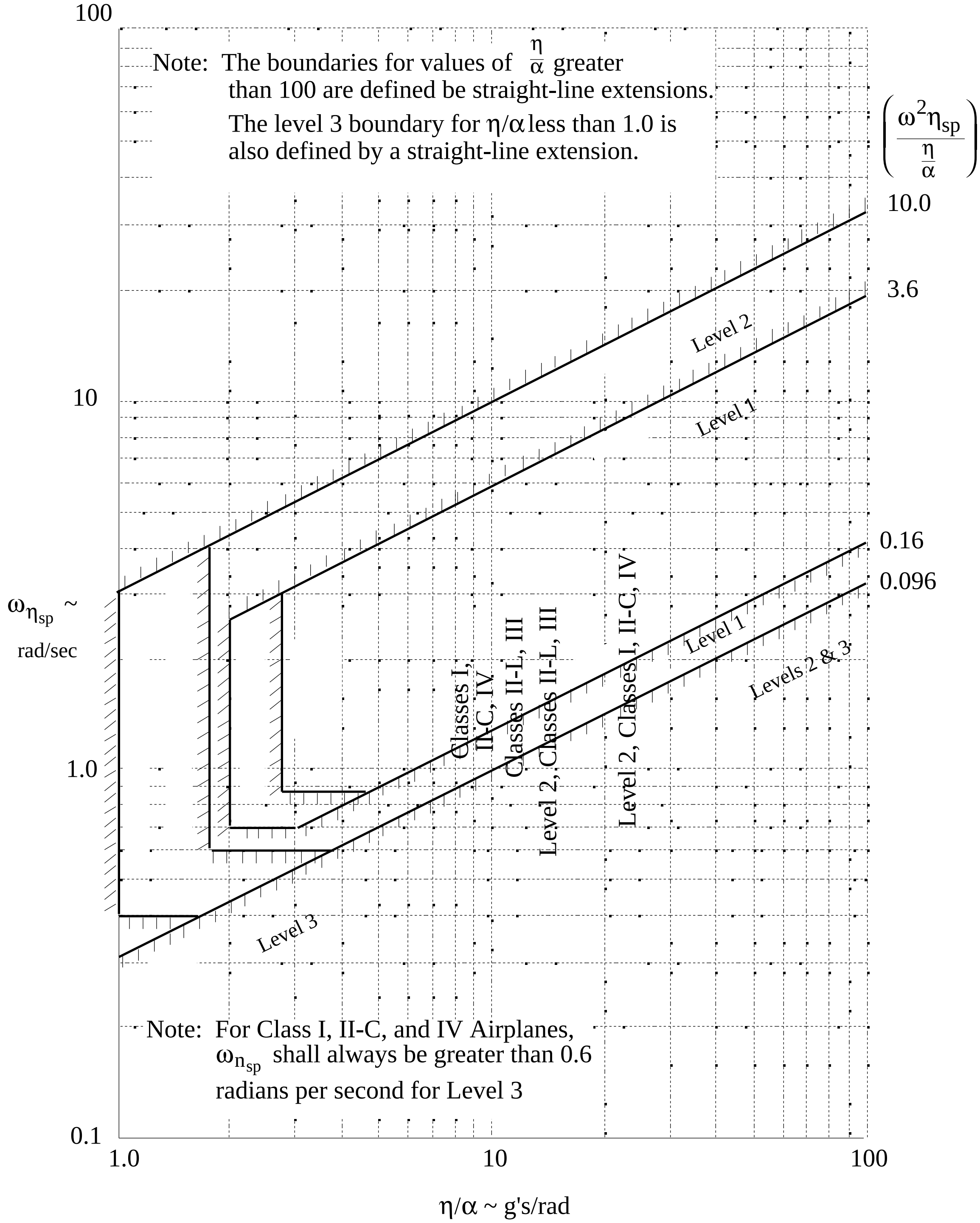


Figure 4.96c
MIL-F-8785B

4.8.2.3.6 The Doublet, Pulse, and 2 g Pull-Up

Three methods will be introduced for obtaining quantitative short period characteristics. The method utilized for a particular flight test will depend on the characteristics of the airplane, the requirements against which tested, and the preference of the individual test pilot.

The "doublet input" excites the short period motion nicely, while suppressing the phugoid. It is generally considered to be the optimum means of exciting the short period motion of any airplane. The doublet input manufactures a deviation in pitch attitude in one direction (nose-down), then cancels it with a deviation in the other direction (nose-up). The total deviation in pitch attitude from trim at the end of a doublet is zero. Thus, the phugoid mode is suppressed. However, the short period motion will be evident since the doublet generates deviations in pitch rate, normal acceleration, and angle of attack at a constant airspeed. Short period characteristics may be determined from the manner in which these parameters return to the original trimmed conditions.

The doublet is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, activate them before initiating any deviation from trim. (The first part of the trace then serves as a "trim shot.")

2. With a smooth, but fairly rapid motion, apply airplane nose-down longitudinal control to decrease pitch attitude a few degrees, then reverse the input to nose-up longitudinal control to bring the pitch attitude back to trim. As pitch attitude reaches trim, return the longitudinal cockpit control to trim and release it (controls-free short period) or restrain it in the trim position (controls-fixed short period). (Both methods should be utilized.) At the end of the doublet input, pitch attitude should be at the trim position (or oscillating about the trim position) and airspeed should be exactly trim airspeed. The doublet input and various other significant parameters are shown in Figure 4.97.

3. Obtaining quantitative information on short period characteristics from cockpit instruments is difficult and will be almost impossible if the motion is heavily damped. However, if a sensitive accelerometer and/or sensitive angle of attack indicator are available in the cockpit, and if the motion is not too heavily damped, the test pilot may be able to see enough of the free oscillation to obtain a half-cycle amplitude ratio. From this parameter, an approximate damping ratio can be quickly obtained[8]. The time required for a half-cycle may be measured with a one - or three - second sweep stopwatch. Doubling this time yields the approximate damped period of the short period motion. From this parameter, approximate values for damped frequency and undamped natural frequency may be computed, if desired[8]. If the pilot cannot see enough of the motion to measure and time a half-cycle amplitude ratio, the short period motion should be qualitatively described as essentially deadbeat.

4. If automatic recording devices (oscillograph or magnetic tape, etc.,) are available, the entire doublet input and short period response may be recorded and analyzed[8] later for accurate quantitative information.

5. The frequency with which the doublet input is applied depends on the frequency and response characteristics of the airplane. The test pilot must adjust the doublet input to the particular airplane. The maximum response amplitude will be generated when the time interval for the complete doublet input is approximately the same as the period of the undamped short period oscillation (see Figure 4.97).

6. The amplitude of the doublet input must be large enough to generate a large enough short period response to analyze. Ease and accuracy of analysis increases with size of the short period response. It is judicious to make small amplitude inputs until familiarity is gained with the response characteristics. This is particularly important for a low altitude, high speed flight condition or any high dynamic pressure flight condition.

---

[8] See "Analysis of Second Order Responses" in the introduction of this manual.

7. The doublet input may be made by first applying aft stick, then reversing to forward stick. However, this results in less than 1g normal acceleration at the completion of the doublet and is more uncomfortable for the pilot.

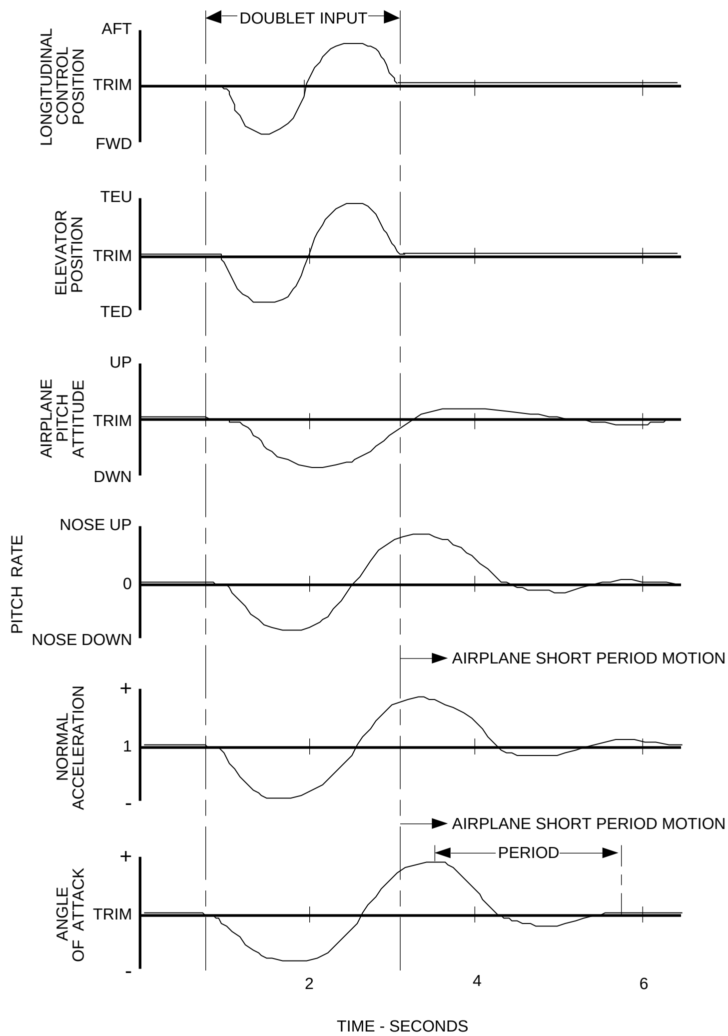


Figure 4.97
Doublet Input and Airplane Short Period Response

The pulse input also excites the short period nicely; however, it also tends to excite the phugoid mode. This confuses data analysis, since the response of the airplane through the phugoid may be taken as a part of the short period response. This is particularly true for low-frequency, slow responding airplanes. Therefore, the pulse can usually only be utilized for high-frequency, quick responding airplanes in which the short period motion subsides before the phugoid response can develop. The pulse can always be used for a quick, qualitative look at the form of the short period motion. It is performed as follows:

1. Stabilize and trim in the desired configuration at the desired flight condition. Actuate the automatic recording devices, if available, before initiating any deviation from trim.

2. With a smooth, but fairly rapid motion, apply airplane nose-up longitudinal control to generate pitch rate, normal acceleration, and angle of attack changes, then return the longitudinal control stick to the trim position. The short period motion may then be observed while restraining the control stick at the trim position (controls-fixed short period) or with the control stick free (controls-free short period).

3. The pulse is actually the last half of a doublet input. The parameters shown in Figure 4.97 at the completion of a doublet input will be the same at the completion of the pulse input <u>except airplane pitch attitude will be different from trim and will return to trim only through the phugoid motion</u>.

4. Pulses may also be performed by first applying airplane nose-down longitudinal control.

The <u>2g pull-up</u> excites the short period motion nicely and suppresses the phugoid if performed correctly. It requires more time and effort than either the doublet or pulse inputs. However, it is useful for investigating short period characteristics in low frequency, slow responding airplanes. It may also be used in any airplane which exhibits heavy short period damping and a large amplitude motion is desired for analysis. The 2g pull-up is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, take a trim shot, then turn the devices off.

2. Decrease airspeed by increasing pitch attitude, then apply airplane nose-down longitudinal control to enter a dive.

3. Trim altitude should be approached in a fairly steep nose-down attitude. As airspeed increases toward the trim airspeed, actuate the automatic recording devices and apply airplane nose-up longitudinal control to establish pitch rate, normal acceleration, and angle of attack changes.

4. As the airplane pitch attitude approaches the initial trim pitch attitude, airspeed should be trim airspeed, and normal acceleration should be approximately 2g. As the pitch attitude reaches trim, smartly return the cockpit control stick to the trim position and restrain it there or release it.

5. Observe appropriate short period characteristics and deactuate the automatic recording devices after the motion has subsided.

4.8.3 Pilot-Induced Oscillations

The pilot-induced oscillation (PIO) can be defined as sustained oscillations or instabilities resulting from the pilot being in the control loop. These oscillations would not occur if the pilot had not closed the loop, since with few exceptions the airplane alone is dynamically stable. It follows that control system dynamics as well as airframe and pilot dynamics enter into this phenomenon. In other words, it is the total system that must be considered when evaluating PIO.

Several open loop type flight tests have been developed to identify characteristics that tend to contribute to PIO. One test is the sinusoidal pumping of the elevator control at frequencies up to the short period. The phase angle between the elevator and stick is used to assure adequate performance of the elevator servo. For this same type of pumping at all input frequencies, the ratio of peak stick forces to peak load factors is used as an indicator of how bobweights, augmentation systems, and basic control systems interact. If this ratio decays significantly below steady state stick force per g, the airplane tends to become either very sensitive in maneuvering or PIO may be encountered. Tests on the artificial feel system including centering, breakout, friction, freeplay, and damping may also yield information about control system characteristics that contribute to the problem.

All of these open loop tests may be used to point out areas of the flight envelope where PIO tendencies exist; however, the PIO is a closed loop phenomenon and tasks involving close pilot control of the airplane must be included. One such test is high-speed formation flying where the pilot attempts to hold a precise wing position. Others involve precise pitch tracking of either another aircraft or small ground and cloud targets. If PIO is encountered, the pilot should get out of the control loop since the natural stability of the airplane will normally damp out the oscillations. This can be done by simply releasing the cockpit controls or by "clamping" the stick in the neutral position with both hands. Obviously, if the PIO is encountered at very low altitude, the best recovery technique is to smoothly but positively apply a pull force and commence a climb before releasing or restraining the stick.

Since the PIO involves a closed loop where the short period mode is driven divergent, it is obvious that a lightly damped free oscillation short period may contribute to PIO. Also, since the dynamics of the pilot are involved, the higher frequency airplanes are usually more prone to PIO. However, it does not necessarily follow that the lightly damped, relatively high frequency airplane will exhibit PIO tendencies. Thorough testing of both open and closed loop response characteristics in all flight phases is necessary to fully define PIO tendencies.

There are no rigorous test techniques with which to investigate PIO tendencies. The straight-forward approach is probably the best - fly the airplane in the flight conditions where PIO tendencies are predicted and see if any are encountered. This approach should, of course, be made in gradual steps, building up to the most critical conditions as experience is gained. The following general guidelines are offered in planning and conducting a test program for investigating pilot-induced-oscillation tendencies:

1. The airplane should be maneuvered as it would be while making precise corrections in pitch attitude and normal acceleration. Close formation flying and precise gunsight tracking tasks are "tight spots" where PIO tendencies may be readily apparent.

2. A properly planned test program using various frequencies of “sinusoidal stick pumping” would practically force the test pilot to experience a PIO if the tendency were present. Sinusoidal stick pumping was previously introduced in the discussion of transient maneuvering control forces.

3. Extreme caution should be exercised in attempting a test program of this nature of the LAHS flight regime. The amplitude of normal acceleration variation during a PIO in this regime could precipitate dramatic and sudden structural failure of the airframe and possibly incapacitate the pilot so that escape would be impossible.

4. If longitudinal stability augmentation is installed, the effect of its failure on PIO tendencies must be investigated with a careful build-up program.

4.8.4 Postflight Procedures

As soon as possible after returning from the flight, the pilot should write a brief, rough qualitative report of the longitudinal flying qualities exhibited during the mission tasks under evaluation. This report should be written while the events of the flight are fresh in the test pilot’s mind. Qualitative pilot opinion, appropriately related to the mission tasks under evaluation, will be the most important part of the final report.

Appropriate data should be selected to substantiate the pilot’s opinion. Several suggested means of presenting data will be introduced. No matter what method is used, it should be clear, concise, and complete.

4.8.4.1 MECHANICAL CHARACTERISTICS OF THE LONGITUDINAL CONTROL SYSTEM

Mechanical characteristics may be presented as shown previously in the discussion of "Test Procedures and Techniques - Nonmaneuvering Tasks."

4.8.4.2 LONGITUDINAL MANEUVERING STABILITY CHARACTERISTICS

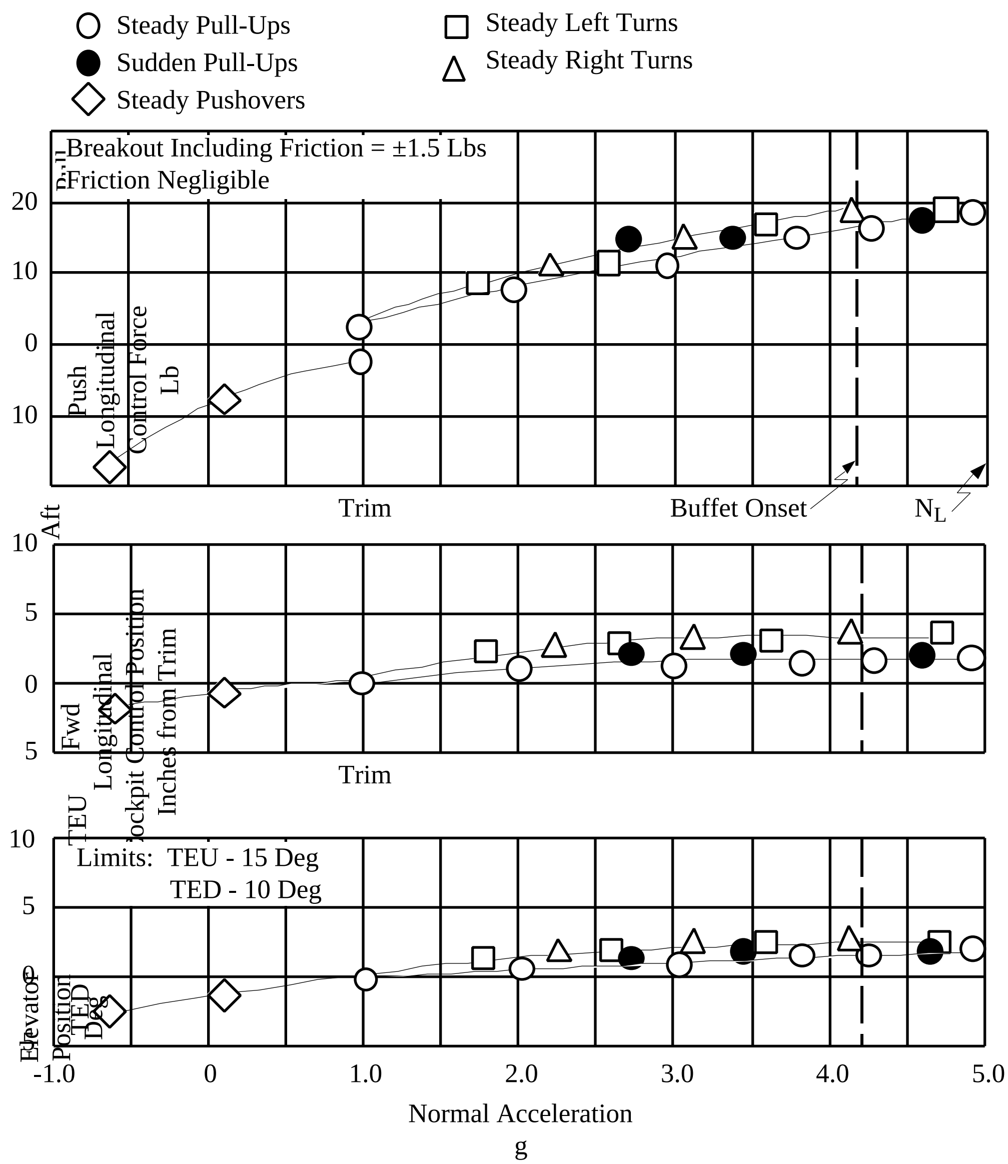
Longitudinal maneuvering stability characteristics may be presented as plots of longitudinal control force, elevator position, and longitudinal cockpit control position versus normal acceleration at a constant trim airspeed. This presentation shows the linearity or any nonlinearities of the relationships. If $\frac{n}{\alpha}$ is linear applicable specification limits for the local $\frac{F_s}{n}$ gradient can be plotted on the longitudinal control force curves as an aid in determining specification compliance (i.e., when local $\frac{F_s}{n}$ gradient is steeper than minimum or shallower than maximum specification requirement). Longitudinal control system breakout forces, including friction should be considered when fairing curves through the data points and when drawing specification limits. The $\frac{F_s}{n}$ curves and the specification limit lines should originate at the longitudinal control force value corresponding to breakout, including friction in either the push or pull direction. Typical plots are shown in Figure 4.98.

(NOTE: For the example shown, no difference in the characteristics and be detected between left and right steady turns. For single engine, single rotation propeller airplanes, the difference may be significant enough to fair curves for both left and right turns.)

If enough data points are obtained during the maneuvering stability tests, and/or if no significant differences can be detected for steady turns and steady pull-ups, it is not necessary to attempt to fair a curve through the data. In this case, the data points themselves define the curves. Typical plots are shown in Figure 4.99. An additional plot helpful in determining and presenting specification compliance is shown in Figure 4.100.

Wind-up turns at constant airspeed yield as many data points as the engineer desires to obtain from the automatic recording traces. If automatic data reduction facilities are available, enough data points can be obtained to "shot gun" the data. A typical example of this method of data presentation is shown in Figure 4.101 for the "stick force per g" characteristics only.

If maneuvering stability tests are performed at various CG positions for the same configuration and essentially the same flight conditions, the data may be presented as shown in Figure 4.102.



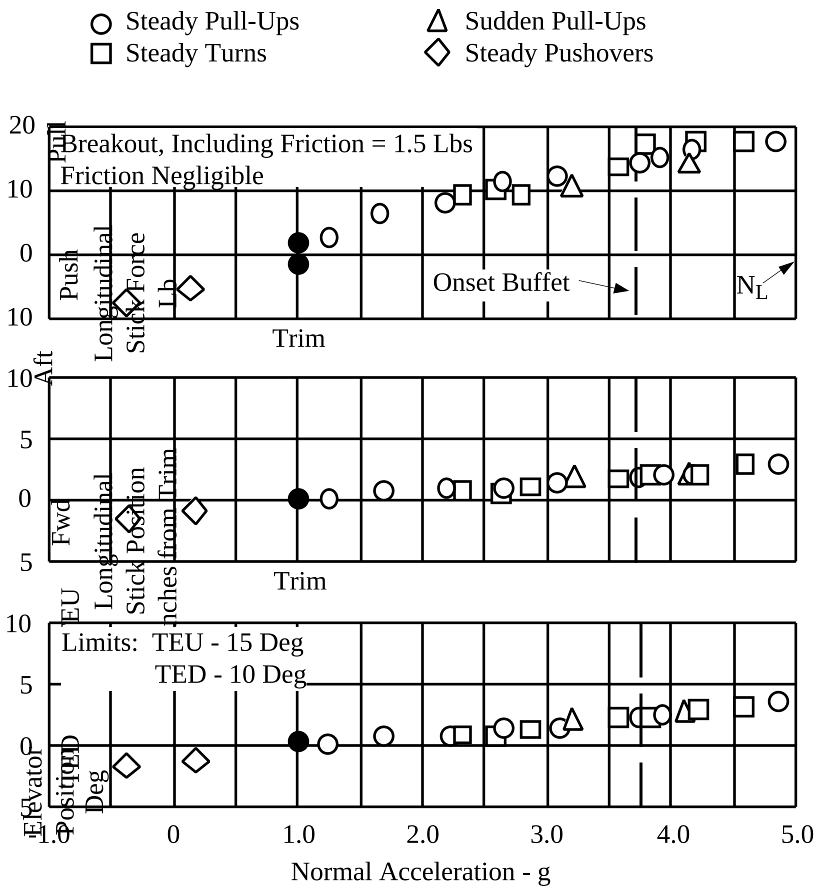
Model_________Airplane

BuNo________

Configuration: Power
Loading: Long Range Attack
CG: 20.7% MAC

Trim: 400 KIAS, 10,000 Ft.
Gross Weight: 18,240 Lb.
Longitudinal Trim: 0.63º and
Stab Aug: On

Figure 4.98
Longitudinal Maneuvering Stability Characteristics



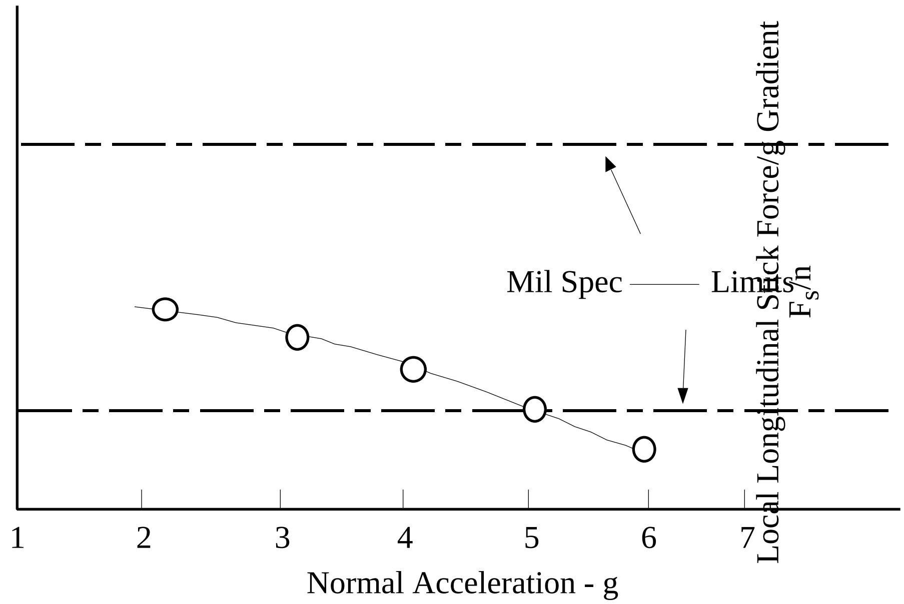
Model_________Airplane

BuNo________

Configuration: Cruise
Loading: C
CG: 21.6% MAC

Trim: 350 KIAS, 5,000 Ft.
Gross Weight: 17,130 Lb.
Longitudinal Trim: 0.55º and
Stab Aug: Off

Figure 4.99
Longitudinal Maneuvering Stability Characteristics



Model__________Airplane

BuNo_________

Configuration: Power
Loading: C
CG: 21.6% MAC

Trim: 450 KIAS, 5,000 Ft.
Gross Weight: 18,340 Lb.
Stab Aug: On

Figure 4.100
Longitudinal Maneuvering Stability Characteristics

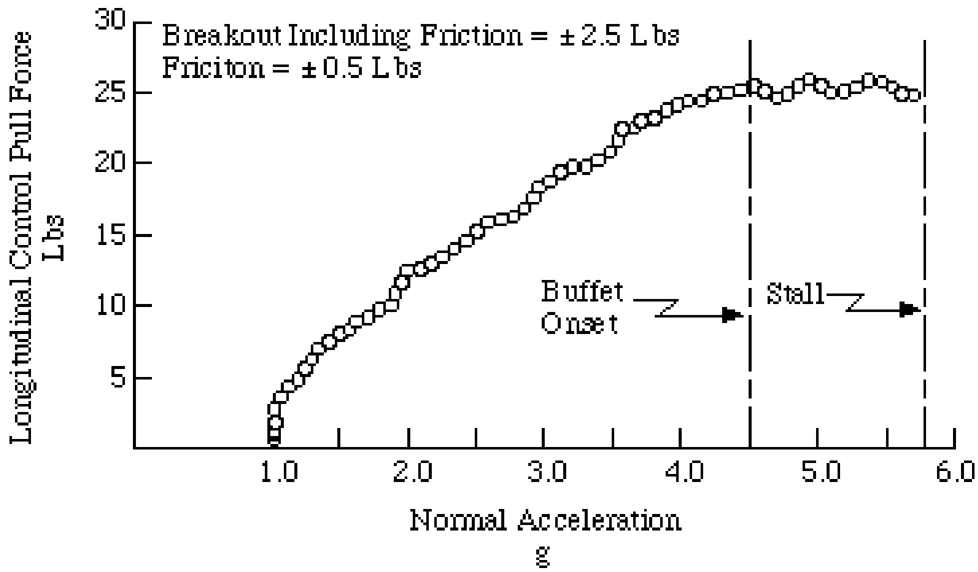


Figure 4.101
Possible Means of Presenting "Stick Force per g"
Data from a Wind-Up Turn

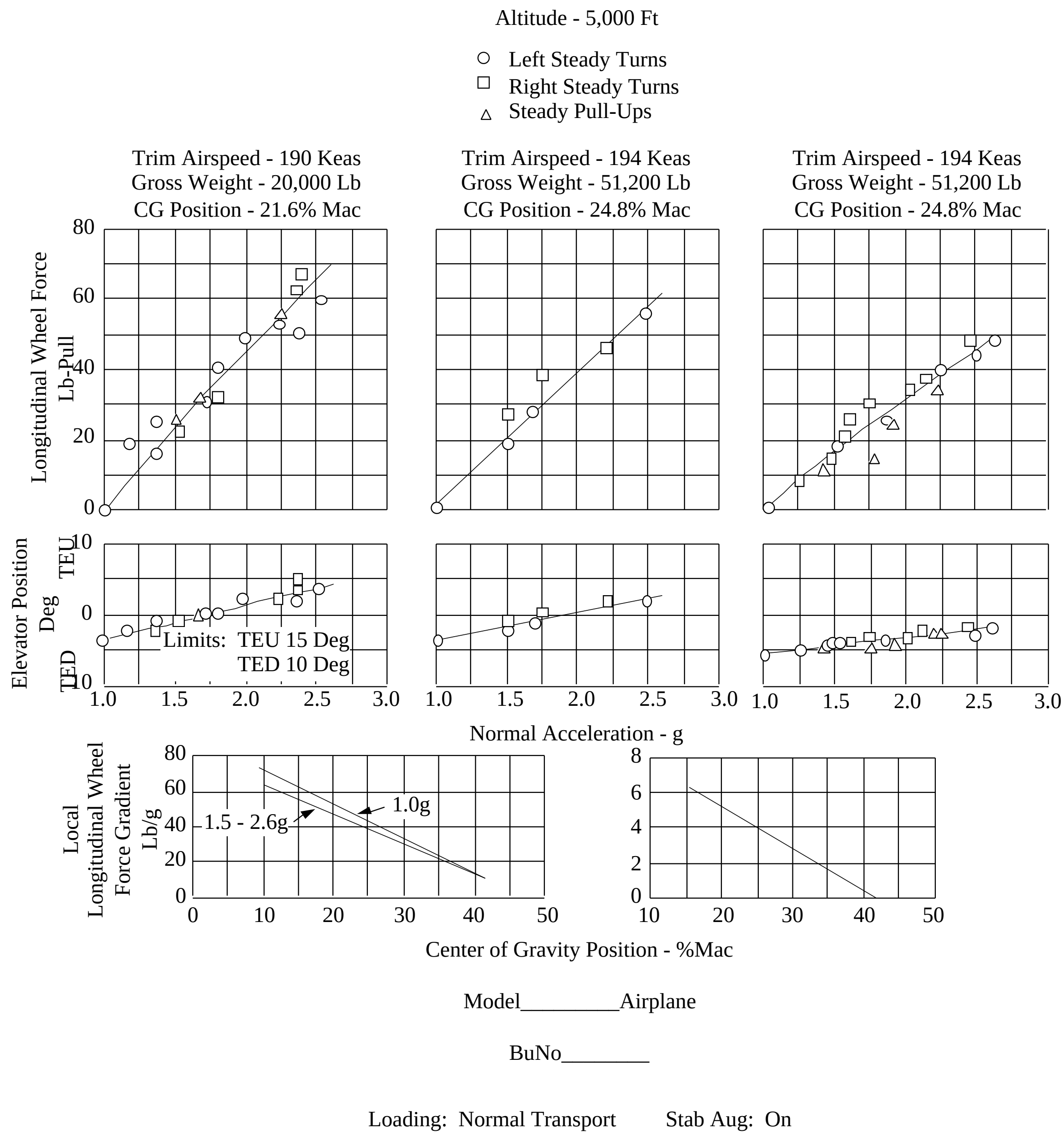


Figure 4.102
Longitudinal Maneuvering Stability Characteristics in Configuration Cruise

Longitudinal maneuvering control force data is sometime presented in tabular form when many loadings, configurations, altitudes, trim airspeeds, and CG positions have been utilized. An example is presented in Figure 4.103.

		CG			Trim	Control Force

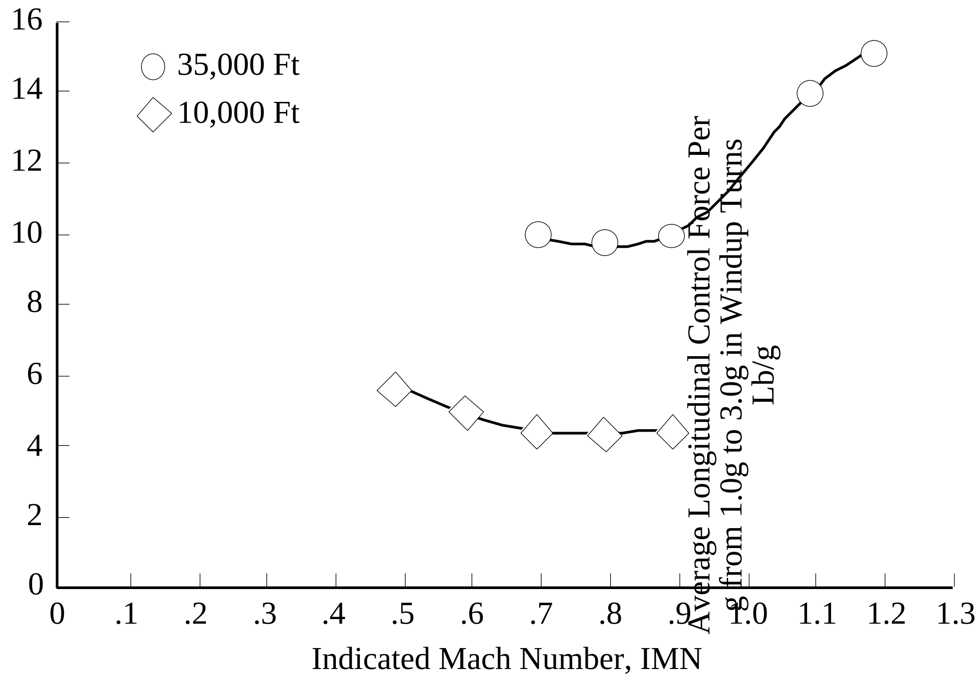
Loading	Configuration	Position (%MAC)	Gross Wt. (lb)	Altitude (Ft)	Airspeed (KCAS/M)	Gradient (1) (Lb/g)
A	CR	17.1	14,310	40,000	181/.62	23.6
A	CR	21.1	14,155	40,000	221/.77	11.1
A	P	23.2	15,240	40,000	274/.90	9.2
C	CR	20.6	16,875	30,000	235/.64	9.2
C	P	25.1 (2)	18,380	30,000	324/.81	6.9
A	P	18.3	14,850	10,000	521/.92	6.8
A	P	21.3	14,500	10,000	515/.915	5.0
A	CR	24.2	15,830	10,000	433/.775	5.2
A	CR	24.1	15,775	10,000	289/.52	5.6
C	CR	19.4 (2)	16,565	10,000	346/.63	11.0
C	P	24.9 (2)	18,240	10,000	445/.79	4.9
A	PA	25.0	15,690	10,000	129	16.6
C	PA	22.9	17,500	10,000	150	19.0

(1) Average maneuvering control force gradient measured from one g to maximum g attained in a wind-up turn.

(2) See CG- Cross Weight relationship shown in Appendix IV, Figure 1.

Figure 4.103
Longitudinal Maneuvering Stability Table

The flight test team may desire to presented the variation on the "stick force per g" gradient with altitude and airspeed (or Mach number.) If so, average gradients of stick force per g can be determined for different trim airspeeds and altitudes and presented on the same plot (Figure 4.104). In general, the average gradient should be computed for the same normal acceleration increment at each trim point; i.e., "average gradient between 1.0 and 3.0g" or "average gradient between 1.0 g and buffet onset." At any rate, the normal acceleration increment used to compute the average gradient should be presented on the plots.



Model_________Airplane

BuNo________

Configuration Cruise - Power
Loading: Normal Fighter

Gross Weight: 16,000 - 20,000 Lbs
CG: 20.0% - 25% MAC

Figure 4.104
Variation of Stick Forces per g with Altitude and Mach Number

4.8.4.3 AIRPLANE SHORT PERIOD CHARACTERISTICS

The presentation of airplane short period characteristics will be dictated by the amount of data available. If the scope of the evaluation is limited, short period characteristics are effectively presented in tabular form. An example is shown in Figure 105. If the short period motion was recorded on oscillograph, magnetic tape, or telemetry, the actual trace, approximately annotated, may be presented in the report. Angle of attack is the most desirable parameter to use in analyzing the test results, since it exhibits the pure short period response better than any other parameter, particularly at low speeds. However, normal acceleration or pitch rate may also be utilized in obtaining approximate quantitive short period characteristics.

Configuration	Altitude (Ft)	Trim Airspeed (KIAS, IMN)	CG (%MAC)	Stab Aug	$\omega_{n_{sp}}$ (Rad/ Sec)	ζ_{sp}	$\frac{C_1}{10}$
Power (P)	40,000	1.2	35.0	On	3.9	0.3	1.0
Power (P)	5,000	550, 0.9	35.5	On	5.0	0.6	0.5
Power Approach	5,000	130	26.0	Off	Motion Essentially Dead Beat		

Figure 4.1705
Airplane Short Period Characteristics

When the flight test team desires to shown the variation of airplane short period characteristics with airspeed or Mach number, a plot similar to that shown in Figure 4.107 may be utilized.

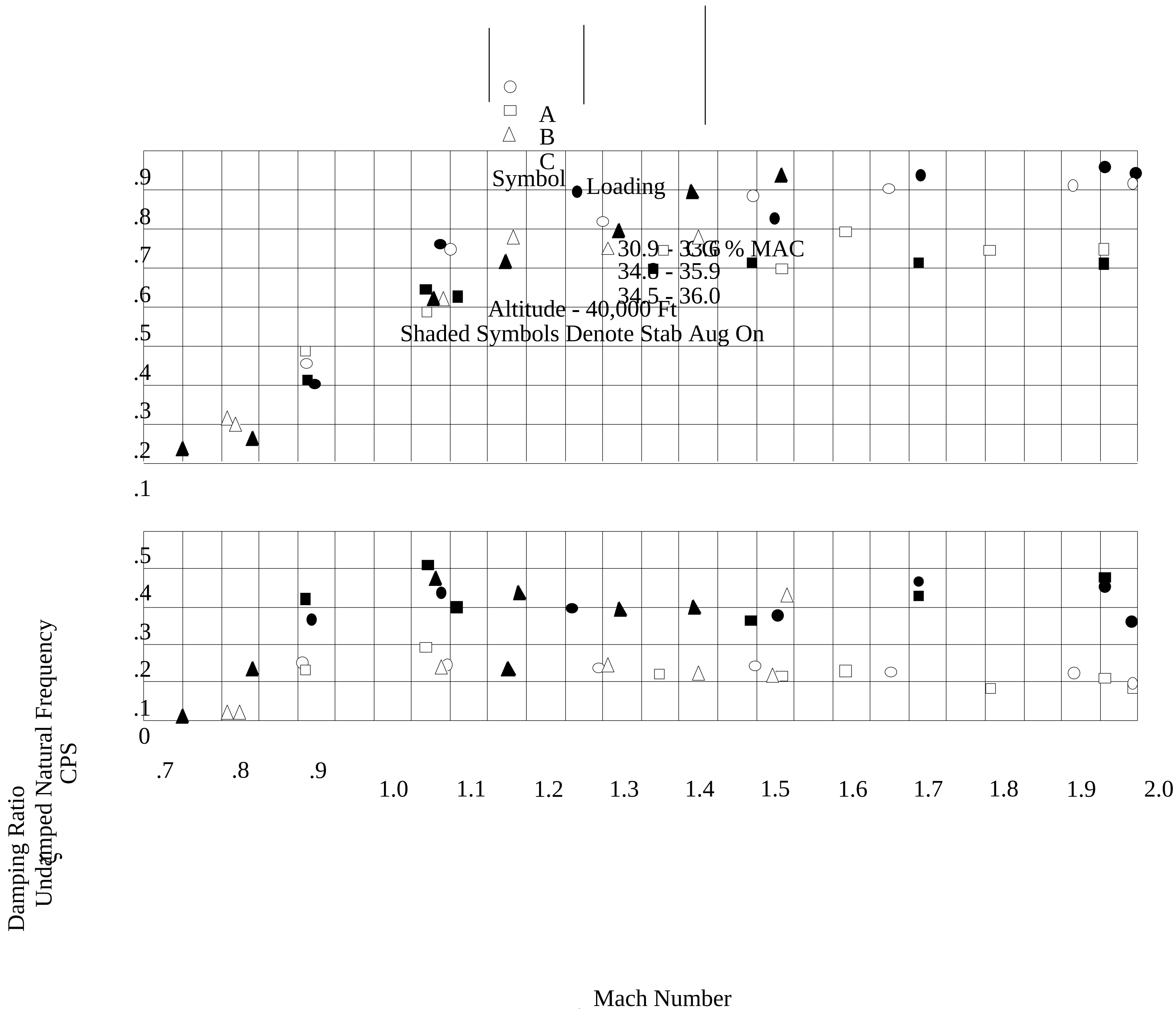


Figure 4.106
Variation of Airplane Short Period Characteristics with Mach Number

4.8.4.4 PILOT-INDUCED OSCILLATIONS

Any tendency of the pilot-airplane combination toward PIO during any of the maneuvering tasks must, of course, be thoroughly discussed in the technical report. This discussion may be a separate section within the report or integrated into other sections of the report, such as mechanical characteristics, maneuvering stability, or short period motion discussions.

4.9 SPECIFICATION REQUIREMENTS

Requirements for static and dynamic longitudinal flying qualities during maneuvering tasks are contained in the following applicable paragraphs of Military Specification MIL-F-8785B(ASG), of 7 August 1969, hereafter referred to as the Specification.

3.2.2 Longitudinal maneuvering characteristics.

3.2.3.2 Longitudinal control in maneuvering flight.

3.5.2 Mechanical characteristics.

3.5.3 Dynamic characteristics.

3.5.4 Augmentation systems.

The requirements of the Specification may be modified by the applicable airplane Detail Specification. Comments concerning only those portions of the Specification which require some interpretation are presented below.

3.2.2.2.1 Control Forces in Accelerated Flight

The terms "local gradient" and "average gradient" are not defined in the Specification. The definitions presented earlier in this section should be utilized in the analysis of test results. Note that limit local factor, n_L, is defined as the symmetrical flight limit load factor for a given Airplane Normal State, based on structural considerations. If this information is not defined by the contractor, utilize the maximum allowable load factor for the particular configuration, store loading, etc., in computation of Specification limits for maximum and minimum local force gradients. Interpretation of the $\frac{F_s}{n}$ limits of table V. Because the limits on $\frac{F_s}{n}$ are a function of both n_L and $\frac{n}{\alpha}$, table V is rather complex. To illustrate its use, the limits are presented on Figure 4.107 for an airplane having a center-stick controller and n_L + 7.0.

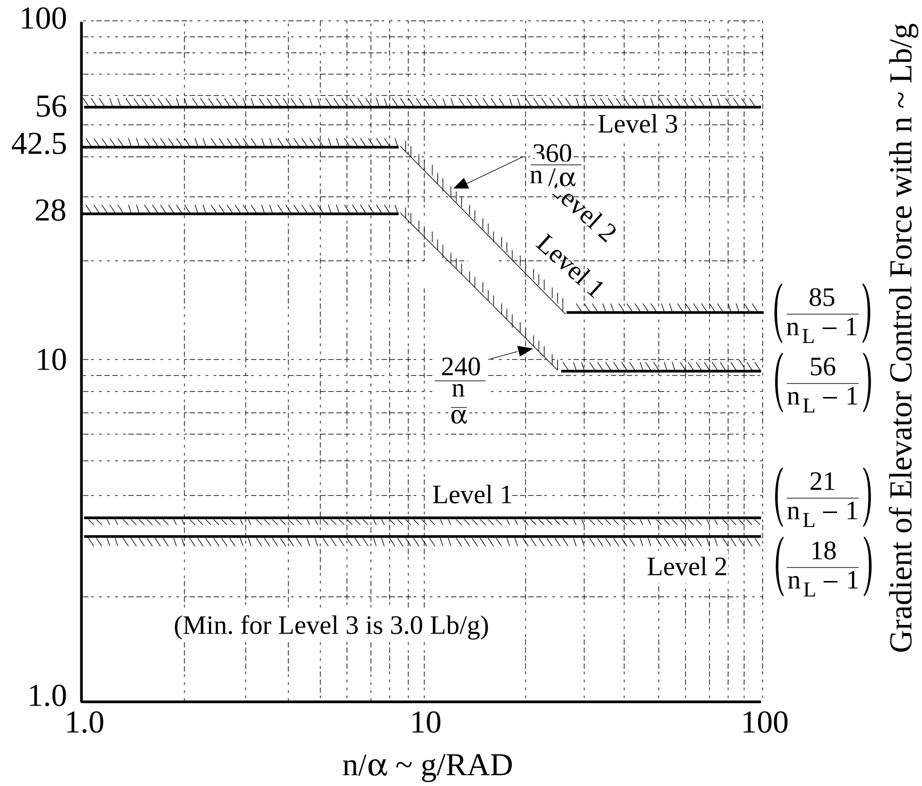


Figure 4.107
MIL-F-8785B

4.10 MANEUVERING TASKS - GLOSSARY

Pitch Rate Damping	Pitching moment created because of the angular rotation of the airplane in pitch during curvilinear flight. Sometime called "damping in pitch" or "viscous damping in pitch."
Stick-Fixed Maneuvering Neutral Point	The location of the center of gravity of an aircraft for which the gradient of elevator position versus normal acceleration at constant airspeed would be zero. Sometimes called the "elevator position maneuvering neutral point."
Stick-Free Maneuvering Neutral Point	The location of the center of gravity of an aircraft for which the gradient of longitudinal; control force versus normal acceleration at a constant airspeed would be zero. Sometime called the "longitudinal control force maneuvering neutral point."
Maneuvering Tasks	Those tasks which result in accelerated flight conditions; during these tasks, transitions for one equilibrium flight condition to another are made quickly, and possibly, somewhat roughly.
Local Stick Force per g Gradient	Slope of the tangent to the curve of longitudinal control force versus normal acceleration at any point.
Average Stick Force per g Gradient	Slope of a line drawn from the 1g point where breakout, including friction is overcome to the point under consideration on the curve of longitudinal control force versus normal acceleration.
Pilot-Induced-Oscillation (PIO)	A divergent oscillation of the pilot - airplane combination where the airplane alone exhibits at least some degree of dynamic stability.

4.11 MANEUVERING TASKS - REFERENCES

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control*, by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Analysis of Frequency and Damping Characteristics of Airplane Longitudinal Short Period Oscillations*, by William B. Rhodes, Flight Test Division, Naval Air Test Center, Patuxent River, Maryland, FT-TM-1-64, 28 December 1964.

4. *Calculation of the Response of a Fighter Aircraft to Turbulent Air and a Comparison with Flight Measurements*, by C.G.B. Mitchell, Royal Aircraft Establishment, Ministry of Aviation, Farnborough, Hants, RAE Technical Report 66275, August 1966.

5. *Computer Programmers to Calculate the Response of Flexible Aircraft to Gusts and Control Movements*, by C.G.B. Mitchell, Ministry of Technology, Aeronautical Research Council, London, Current Paper No. 957, October 1966.

6. *Considerations in the Determination of Stability and Control Derivatives and Dynamic Characteristics from Flight Data*, by Chester H. Wolowicz, NASA Flight Research Center, Edwards Air Force Base, California, AGARD Report 549 - Part 1, 1966.

7. *Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training*, by F. D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F-1, September, 1962.

8. *Dynamics of Flight, Stability and Control*, by Bernard Etkins, Professor of

Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

9. *Experience With Shinbrot's Method of Transient Response*, by B.R.A. Burns, British Aircraft Corporation, Preston Division, Warton Aerodrome, Preston, Lancashire, AGARD Report 549 - Part II, 1966.

10. *Flight Investigation of Longitudinal Short Period Frequency Requirements and PIO Tendencies*, by Dante A. Difranco, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-66-163, June 1967.

11. *Flight Test Evaluations of Airplane Dynamic Longitudinal Stability in the Low Altitude High Speed Region*, Mr. Thomas M. McPherson and Captain Kenneth W. Weir, USMC, Flight Test Division, Naval Air Test Center, Patuxent River, Maryland.

12. *Flight Testing of Aircraft*, by M. G. Kotik, et al., NASA Technical Translation F-442, April, 1967.

13. *Frequency Response Method of Determining Aircraft Longitudinal Short-Period Stability and Control System Characteristics in Flight*, by Henry A. Klung, Jr., Captain, USAF, Aerospace Engineer, AFFTC-TR-66-24, August 1966.

14. *Handling and Performance Exercises Test Techniques*, Empire Test Pilots' School, Flying Department Notes.

15. *Investigation of Low Level, High Speed Flight Problems*, by Richard C. Adams, Commander, USN, Flying Qualities and Performance Branch, Flight Test Division, Naval Air Test Center, Patuxent River, Maryland, Society of Experimental Test Pilots 8th Annual Symposium, 25-26 September 1964.

16. *Low Altitude, High Speed Handling and Riding Qualities*, by Ralph C. A'Harrah, AGARD Flight Mechanics Meeting, Turin, Italy, April 1963.

17. *Maneuverability and Gust Response Problems Associated with Low Altitude, High Speed Flight*, By R.C. A'Harrah, AGARD Consultant, AGARD Report 556, October 1967.

18. *Military Specifications, Flying Qualities of Piloted Airplanes,* MIL-F-8785C, of 5 November 1980.

19. *Minimum Flyable Longitudinal Handling Qualities of Airplanes*, by Gifford Bull, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Report No. TB-13130-F-1, December, 1959.

20. *Modern Flight Dynamics*, by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

21. *National Aeronautics and Space Administration Aeronautical Dictionary by Frank D. Adams*, United States Government Printing Office, Washington, 1959.

22. *Stability and Control*, Empire Test Pilots' School Ground School Notes, Book "C".

23. *Stability and Control*, USAF Aerospace Research Pilot School, Edwards Air Force Base, California, FTC-TIJ-64-2004, revised November, 1964.

24. *Stability and Control of Airplanes and Helicopters*, by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

25. *Study of Some Hand-Computing Techniques to Determine the Approximate Short Period Mode From Airplane Response*, by Benjamin H. Dolbin, Jr., Cornell Aeronautical Library, Inc., Buffalo, New York, CA FSDM-371, March 1966.

26. *Substantiation Information for Selected Longitudinal Requirements*, Cornell Aeronautical Laboratory Report No. BM-2238-F-1, May 1967.

4.12 MISCELLANEOUS LONGITUDINAL TESTS

4.12.1 Introduction

There are numerous miscellaneous longitudinal tests which have not been discussed previously and which are conveniently presented as a group. These tests are the subject of this section.

4.12.2 Longitudinal Trim Changes

Since changing power, lowering the landing gear, extending the flaps, opening the speed brakes, and movement of other external variable position devices cause pitching moments, the magnitude of the forces involved in opposing these moments must be determined. The specification contains a table of the most commonly encountered "configuration changes" along with the trim speeds to be used and the initial configurations to be set up prior to varying the configuration. Each of the configurations represents conditions of flight under which the configuration change would logically be made. Consider the following example:

Flight Phase	Initial Trim Condition					Configuration Change	Parameter to be Held Constant
	Altitude	Speed	Landing Gear	High-Lift Devices & Wing Flaps	Thrust		
Approach	$h_{0_{min}}$	Normal Pattern Entry Speed	Up	Up	TLF	Gear Down	Altitude and Airspeed*

*Throttle setting may be changed during the maneuver.

In this case, the airplane is being prepared for an approach. The trim speed (zero stick force) is the normal gear extension speed. (Reference should be made to table 1 of the specification which defines $h_{o_{min}}$.) Power applied is sufficient to maintain level flight at the trim speed. The configuration change is lowering the landing gear. The peak forces involved in holding altitude and airspeed constant for a 5-second period after initiation of the configuration change are the data desired to determine specification compliance. A greater period of time may be of interest for mission suitability considerations.

The specification allows considerable latitude in this particular test. The table set forth in the specification is only a recommended scope of investigation. It states specifically that the changes should be made under conditions of flight representative of operational procedures. A majority of the objections to the table result from the speed brake section which is not too realistic in that the effect of extension and retraction of brakes in dives is not included. Accordingly, a recommended addition to the scope of the tests for Class IV airplanes is an investigation of the effect of extending speed brakes in a dive with the parameter to be held constant being attitude (or the aiming point of the dive prior to extension). The trim changes resulting from speed brake extension and retraction at different airspeeds as well as during jet instrument penetrations, GCA approaches, and landing approaches should also be assessed.

Another item to be considered when conducting this test is the airplane's response to the changes in configuration. With modern control systems, the forces are usually low, even when the pitching moments are considerable. Therefore, the response of the airplane is quite important in determining suitability from a pilot's point of view. In particular, when putting wheels and flaps down under instrument flight conditions, a rapid pitch-up or down is highly undesirable even though the forces required to maintain altitude constant are of a low order. Another condition which arises frequently is a roll or yawing moment which results from asymmetric extension or retraction of flaps or wheels. This is annoying as pilot attention must be directed to directional and/or lateral trim changes in addition to the longitudinal change normally expected.

For each trim change condition, an attempt should be made to trim out the final forces resulting from the change in configuration. In some cases, when holding altitude constant is the requirement, the additional drag caused by extension of wheels or flaps may cause the speed to reduce to stall. In such a case, there is no final value, only a peak force. Usually upon completion of a card, examination of the data will reveal that several of the configuration changes which could logically be changed simultaneously will require application of longitudinal force in the same direction. For example, if two configuration changes require a push force individually, most likely their combined effects would be additive. The test pilot should determine which combination of configuration changes would be encountered in the airplane's mission and investigate them thoroughly.

4.12.3 Longitudinal Trimming Device Irreversibility

It is highly desirable that the longitudinal trimming device maintain a given setting indefinitely, unless changed intentionally; or to put it differently, it should be irreversible. In order to test a longitudinal trimming device for irreversibility, it is necessary that the tab or variable stabilizer be subjected to a high angle of attack at as great a "q" (dynamic pressure) as is feasible. This is down by carefully trimming the airplane at an airspeed, entering a dive, and increasing the airspeed to close to the maximum permissible airspeed of the airplane without retrimming. A high g, symmetrical pull-out is executed at a lower altitude (at least 10,000 feet) and without retrimming, a check is made at the original trim airspeed, altitude, and power setting. If the longitudinal trimming device is irreversible, the force required to maintain the trim speed will be zero. If, however, the tab is reversible, a push or pull force will be required to maintain the trim speed.

The tab indicator itself is not reliable inn determining movement of the tab. If control system friction is high, a change in tab setting may not be apparent through the airplane's stick-free stability. In such cases, it will be necessary to instrument for tab position if the reversibility is actually annoying. Normally, if there is not apparent change in trim (as reflected by zero F_S) at the original trim speed, the fact that the tab has moved is not too objectionable, unless the tab movement under some other condition of flight is bothersome.

4.12.4 Ground Effect Hold-Off

For airplane configurations with the horizontal tail behind the wing, "ground effect" reduces the angle of downwash at the tail. Since the downwash behind a wing not in ground effect reduces the tail angle-of-attack (α_t),the change caused by ground effect results in an increase in the tail angle-of-attack and a corresponding increase in tail lift. This effect is illustrated in Figure 4.108.

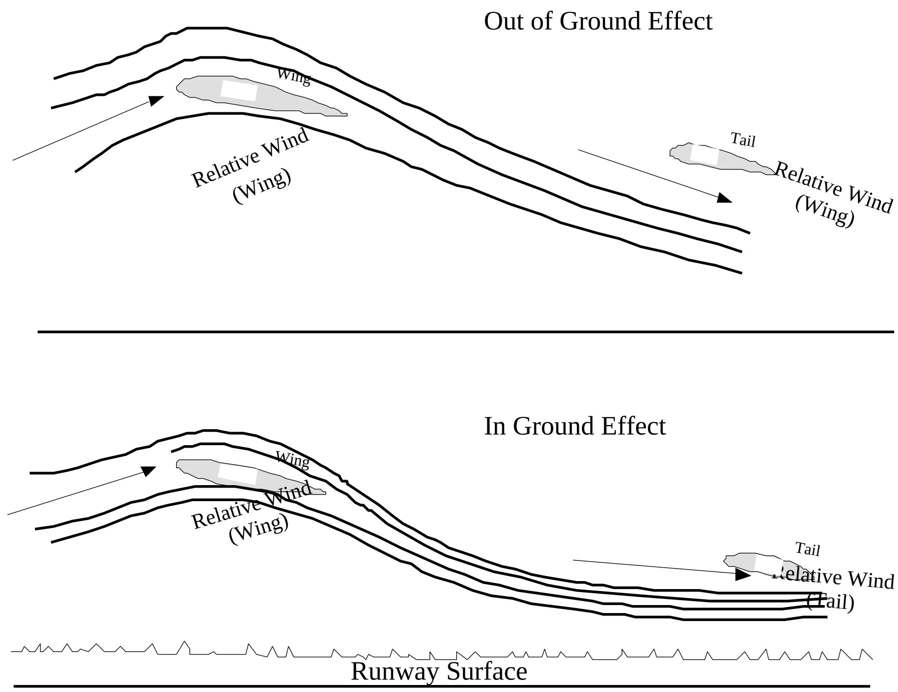


Figure 4.108
Ground Effect

Since tail lift is increased by ground effect, more up elevator is required to maintain a given lift coefficient (C_L) in ground effect than out; if follows that because the elevator deflection is physically limited, its capacity to produce nose-up pitching moments is lessened by ground effect. In some airplanes, we find that the elevator effectiveness is insufficient to obtain an aerodynamic stall. Now, if the airplane is flown in ground effect, the effectiveness of the elevator is further reduced, and very likely it is not possible to obtain the minimum guaranteed landing speed or even the minimum speed that was attainable at altitude.

Approach and landing speeds are becoming more and more critical and should not be limited by longitudinal control effectiveness. Because of this, we test for what is called the "ground effect hold-off speed;" i.e., the minimum speed attainable in ground effect in configuration L and the stick force required to maintain the required elevator deflection at this speed. It is important to understand that this test is conducted to evaluate longitudinal control effectiveness in close proximity to the ground and is not intended to be a test of landing characteristics with idle power/thrust.

4.12.4.1 METHOD OF TEST

With a forward critical CG loading, the airplane is trimmed in configuration PA at normal approach speed and at low altitude (1000 to 2000 feet). It is then flown to the field traffic pattern and, without changing trim, a fairly long, flat straight-in approach is made at an airspeed 20-30 knots above normal approach speed. As the end of the runway is crossed at roughly 20 feet altitude, power (thrust) is reduced to "idle." The airplane is leveled off 2-3 feet above the runway and, as the speed decreases, up elevator deflection is increased so as the maintain a constant height above the runway. Care should be exercised to avoid changes in height during this phase; if a rate-of-descent is set up, more up elevator is required to stop the rate-of-descent and hold a given speed than would have been required to hold the same speed at a constant height. If an aerodynamic stall does not precede application of full-up elevator, the airplane will settle to the runway soon after the elevator has reached full-up deflection. The speed at which the airplane touches down and the elevator force required to maintain a stick position just short of the "stop" is noted. If the airplane is not instrumented for automatic recording of longitudinal stick force, it is normally necessary to make two approaches, recording the minimum hold-off speed during one approach and the longitudinal stick force with a hand-held force indicator during the next.

Caution should be exercised during the later phase of the hold-off. There is a definite tendency to neglect some of the more important aspects of landing an airplane when concentrating on the test being conducted. For one thing, the test should <u>not</u> be conducted in a strong cross-wind for obvious reasons. Further, a touch-and-go type landing will be planned from the start of the approach, unless runway length is extensive, as a large portion of the runaway is usually used during the hold-off.

4.12.4.2 ALTERNATE METHOD OF TEST (GEOMETRY LIMITED)

The geometry of certain airplanes precludes attainment of a stall attitude during a hold-off landing. For example, the tail of the F-8 will strike the runway before either a stall or full aft stick is reached. In such cases, the specification requires that longitudinal control effectiveness in ground effect be sufficient to maintain the geometry-limited touchdown attitude in level flight.

The test is performed in the same manner as for the nongeometry limited airplane except that the maneuver is terminated when the predetermined geometry-limited attitude (or angle of attack) is reached in level flight. It is especially important that the hold-off be performed in close proximity to the runway (2-3 feet) for acceptable data and to prevent hard landings. After the test pilot has reached the predetermined attitude and noted the appropriate data, he should concentrate on making a normal touch-and-go landing or a wave off. If automatic recording devices are installed the "event marker" should be actuated to denote the predetermined attitude. Longitudinal trim must remain at the initial setting (normal approach speed). After the final landing elevator position and force (if the system is irreversible) required for level flight in ground effect are recorded.

4.12.5 Nosewheel Touchdowns

The nosewheel touchdown speed is defined as the speed at which the nosewheel touches the runway following a landing in which the elevators are held in the full up position during the rollout. There is no specification requirement for this characteristic at the present time; however, it is considered useful information and worthy of investigation. It is common practice to slow airplanes equipped with tricycle landing gears by holding the nose-up as long as possible during the landing rollout. This creates more drag than would be present with the nosewheel on the runway because of the higher angle of attack. By using such a technique, it is possible to minimize the use of brakes in slowing the airplane to a safe speed before turning off the runway.

4.12.5.1 METHOD OF TEST

Using the trim settings normally used during the approach, a smooth landing is made on the two main wheels. The nosewheel is held off the runway by application of up elevator. As the full elevator deflection is attained, the airplane will nose over and the nosewheel will contact the runway. The speed at which this occurs is the nosewheel touchdown speed.

4.12.6 Nosewheel Lift-Off

As explained earlier in the "Ground Effect Hold-Off" section, the effectiveness of the elevator is reduced when in close proximity to the runway. In extreme cases, a lack of elevator effectiveness can compromise take-off performance. For example, a given airplane can produce enough lift at 95 knots to take off, provided sufficient angle of attack can be attained. The elevator, however, may not be effective enough to rotate the airplane to this angle of attack. Therefore, the airplane lifts off at some speed greater then 95 knots. This penalizes the airplane in that a longer take-off run is required to attain the speed necessary to lift clear of the runway. This test is of particular importance for carrier airplanes in that insufficient longitudinal control effectiveness will probably result in serious pitch control problems during the normal rotation required during catapult launches or bolters.

Since the speed at which the nosewheel can be lifted from the runway is essentially the speed at which take-off attitude can be obtained, we determine nosewheel lift-off speed and use it as an indication of elevator effectiveness in take-off. If the nosewheel lift-off speed is determined to be equal to or less than .9 V_{min} (as defined in paragraph 3.1.8.2 of the specification), then elevator effectiveness does not compromise take-off performance. The loading tested should be the CG position that produces the greatest load on the nosewheel. For carrier airplanes, elevator control effectiveness must be sufficient to prevent an undesirable nose-up or nose-down pitch between the minimum catapult end-speed (as published in the Launch Bulletins) and 30 knots above this speed.

If the airplane is equipped with a trimmable stabilizer, or a unit horizontal tail (stabilator) with deflection limits which vary with trim setting, the take-off trim used can have a profound effect on the minimum nosewheel lift-off speed obtained. Increased nose-up trim will produce increased nose-up pitching moments attainable with full aft longitudinal control deflection. The trim setting used when testing airplanes equipped with tab trimmed elevators affects only the forces present during the run. Caution should be exercised when checking the effect of applying additional nose-up trim, because some airplanes "break ground" rapidly once the nosewheel begins to lift. For this reason, a "build-up" should be employed, making a series of runs at increasing nose-up stabilizer settings. The amount of nose-up stabilizer that can be tolerated will depend upon its effect on the forces encountered during the take-off and transition to the "clean" condition. For both land and carrier-based operations, satisfactory take-offs should not be dependent on the use of the trimmer during take-off or on complicated control manipulation by the pilot. All forces encountered during take-off and the ensuing acceleration phase should be low enough to be handled easily with one hand. If raising the landing gear and flaps causes a nose-up pitching moment, the force required to overcome this pitching moment should be considered when deciding upon the trim setting to be used.

The landing gear will usually be retracted immediately after becoming airborne. Consequently, if a push force is encountered during the initial acceleration after take-off and a push force results from retraction of the gear, the combined forces will be additive. While the longitudinal trim setting used is optional, it must be held constant throughout the take-off run and subsequent acceleration to $V_{max_{TO}}$ (as defined in paragraph 3.1.8.1 of the specification). Normally it will be a setting which will give zero stick forces just after becoming airborne and will result initially in a pull force (if the airplane has positive stick force stability), which will lessen throughout the take-off run.

This test should be conducted only on a smooth runway as nose-up pitching moments applied through the nosewheel strut can cause the nosewheel to lift off prior to attaining a speed at which the elevator effectiveness is sufficient to rotate the airplane. Rapid application of nose-up elevator control should also be avoided. A careful check of inflation of the nosewheel strut should be made to insure that the prescribed pressure is used. Usually this test is conducted as a matter of course during each take-off; i.e., usually flights are not exclusively devoted to determining the nosewheel lift-off speed. Although the forces encountered during the take-off are of interest, unless the airplane is provided with a means of automatic recording of longitudinal control forces and speed, the most that can be hoped for are the forces at nosewheel lift-off, take-off, and $V_{max_{TO}}$ plus, of course, qualitative comments as to the acceptability of the forces.

4.12.6.1 METHOD OF TEST

The airplane, loaded at the forward critical CG loading, is aligned with the runway heading and take-off power (or thrust) is applied. With the stick held in the full aft position, the brakes are released. The speed at which the nosewheel leaves the runway is noted and the run is aborted or a take-off is accomplished. If it is decided to continue the take-off, a normal take-off attitude should be established after the nosewheel lift-off speed is noted. Recent experience indicates that it is possible with full-up elevator to hold certain airplanes in nose-up attitudes which preclude effecting a take-off due to drag effects. Caution must be exercised in some airplanes to prevent damage to the aft fuselage or tail section by overrotation.

4.12.7 Thrustline Level

This test is the nosewheel lift-off test's opposite number for determining elevator effectiveness during the take-off run of airplanes equipped with tail wheel type landing gear. Because there is less drag present with the thrustline level than in a "three-point" attitude, acceleration will be greater during take-off if the thrust line is level. Further, the distance required to take off is directly related to the acceleration. This leads to the conclusions that the initial phase of a minimum distance take-off should be made with the thrustline level. And, in a majority of cases, this is true. The normal technique used is to raise the tail wheel as soon as the elevator effectiveness permits and then accelerate with the thrustline level until the take-off speed is approached. At this point, the stick is eased to the rear in such a manner as to reach a "three-point" attitude simultaneously with attaining take-off speed. Therefore, the ability of the elevator to raise the tail to a thrustline level attitude and the speed at which this is possible is of interest since it affects take-off performance.

In the nosewheel lift-off test, ground effect lessened elevator effectiveness. The converse is true in this test where the increasing α_t due to "ground effect" aids the elevators in lifting the tail. As would be expected, the critical loading is the normal service loading that provides the greatest tail heavy moment. It should be pointed out that CG position alone does not define this condition. Loadings are possible wherein the effect of increasing weight predominates over CG position. For example, a lightweight, aft CG position loading will probably not cause as great a weight on the tail wheel as a much heavier loading will, at a CG position slightly forward of the light loading CG position. The longitudinal trim setting is optional, but should be a setting which does not result in excessive forces during the take-off run or immediately after becoming airborne. As in the nosewheel lift-off test, trimmable stabilizers can be adjusted so as to provide a nose down pitching moment so long as the elevator forces encountered during take-off are not excessive.

4.12.7.1 METHOD OF TEST

The pilot aligns the airplane with the runway heading and locks the brakes. The flaps are extended to the optimum take-off position and the power is increased until the tail wheel is just lifting from and returning to the runway surface. As much power should be applied as possible, without the risk of nosing over. (If the airplane begins such a maneuver as the power is increased, release the brakes immediately. As you will find out from running this thrustline level test, it is impossible to nose an airplane over, even with full forward stick and full power applied unless the brakes are used.) Next, the brakes are released, take-off power is applied, and the stick is eased to the full forward position. Remember the order in this sequence of events; first, release the brakes; second, apply full power; third, ease the stick full forward. Full forward stick position is maintained until the thrustline is level. The indicated airspeed is noted and the force required is determined. The test should be repeated until reliable data is obtained. The major problem is that of determining when the thrustline is level. It is recommended that a given nose attitude be arbitrarily chosen as the equivalent of thrustline level. This is the only way that repeatable data will be obtained unless an outside observer transmitting a "mark" is utilized.

4.12.8 Speed Brake Effectiveness

Investigation of the effectiveness of the speed brakes is a qualitative task. Speed brakes can be used for airplane deceleration, dive speed limitation, allowing higher engine speed during penetrations and approaches, etc. The particular functions desired from the speed brakes will depend on the airplane's mission. A thorough test plan encompassing all possible uses of the speed brakes should be formulated. In general, for the particular airplane mission, the speed brakes should be sufficient to provide adequate control of airspeed, flight path, etc. at any flight conditions within the Operational Flight Envelope.

4.12.9 Longitudinal Control Forces in Dives

The purpose of this test is to determine the magnitude and rate of change of longitudinal control forces in dives to maximum airspeeds and the ease with which these forces can be maintained near zero by retrimming.

4.12.9.1 METHOD OF TEST

The airplane is trimmed for V_{MRT} (level flight) at high altitude. Without changing power, a dive is entered so as to reach maximum operational airspeed at a low altitude. The maximum longitudinal control force required at maximum operational airspeed is noted. During a similar dive, the ability of the pilot to keep longitudinal control forces near zero by retrimming is determined.

4.12.10 Longitudinal Trim System Effectiveness and Failures

The purpose of this test is to determine the capability of the longitudinal trim system to reduce control forces to zero at all operational airspeeds. Longitudinal trim system failures should also be investigated.

4.12.10.1 METHOD OF TEST

The airplane is flown from minimum airspeeds to maximum operational airspeeds in all configurations, and the ability to trim the longitudinal control forces to zero is evaluated.

Trim system failures are simulated at representative operational airspeeds by running the longitudinal trim in the nose-up and nose-down direction until full control deflection is reached, excessive control forces are required, or the maximum trim travel limits are reached. The controllability of the airplane is then evaluated.

CHAPTER FIVE

LATERAL - DIRECTIONAL FLYING QUALITIES

CHAPTER FIVE

FIGURES

CHAPTER FIVE

EQUATIONS

		PAGE
$Y_{\beta_v} = -a_v \left(\beta - \sigma\right) q_v S_v$	*eq 5.1*	5.8
$C_{y_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w}$	*eq 5.2*	5.9
$Y_{\delta_r} = a_v \tau_v \delta_r q_v S_v$	*eq 5.3*	5.11
$C_{y_{\delta_r}} = a_v \tau_v \eta_v \frac{S_v}{S_w}$	*eq 5.4*	5.12
$C_{\ell_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} \frac{Z_v}{b}$	*eq 5.5*	5.17
$L_{\delta_r} = Y_{\delta_r} Z_v$	*eq 5.6*	5.19
$C_{\ell_{\delta_r}} = C_{y_{\delta_r}} \frac{Z_v}{b}$	*eq 5.7*	5.19
$L_{\delta_a} = 2a_w \tau_a q_a \int_{y_1}^{y_2} c\, y\, dy$	*eq 5.8*	5.20
$C_{\ell_{\delta_a}} = \frac{-2a_w \tau_a \eta_a}{S_w b} \int_{y_1}^{y_2} c\, y\, dy$	*eq 5.9*	5.21
$N_{\beta_v} = -Y_{\beta_v} \ell_v = a_v \left(1 - \frac{d\sigma}{d\beta}\right) q_v S_v \ell_v$	*eq 5.10*	5.24

$$C_{n_{\beta_V}} = -C_{y_{\beta_v}} \frac{\ell_v}{b} = a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} \frac{\ell_v}{b}$$ *eq 5.11* 5.24

$$C_{n_\beta} = C_{n_{\beta_{w,\,F,\,N}}} + C_{n_{\beta_v}}$$ *eq 5.12* 5.25

$$N_{r_v} = -a_v S_v q_v \frac{{\ell_v}^2}{V}$$ *eq 5.13* 5.26

$$C_{n_{r_v}} = \frac{\partial C_n}{\partial\left(\frac{rb}{2V}\right)} = -2a_v \frac{S_V}{S_w} \eta_v \frac{{\ell_v}^2}{b^2}$$ *eq 5.14* 5.26

$$N_{\delta_r} = -a_v \tau_v q_v S_v \ell_v$$ *eq 5.15* 5.27

$$C_{n_{\delta_r}} = -a_v \tau_v n_v \frac{S_v}{S_w} \frac{\ell_v}{b}$$ *eq 5.16* 5.27

<u>SIDEFORCE</u> $$C_{y_0} + C_{y_\beta} \beta + C_{y_{\delta_r}} \delta_r + C_L \phi = 0$$

eq 5.17 5.34

<u>YAWING MOMENT</u> $$C_{n_0} + C_{n_\beta} \beta + C_{n_{\delta_r}} \delta_r + C_{n_{\delta_a}} \delta_a = 0$$

eq 5.18 5.34

<u>ROLLING MOMENT</u> $$C_{\ell_0} + C_{\ell_\beta} \beta + C_{\ell_{\delta_r}} \delta_r + C_{\ell_{\delta_a}} \delta_a = 0$$

eq 5.19 5.34

<u>SIDEFORCE</u> $$C_{y_\beta} + C_{y_{\delta_r}} \frac{d\delta_r}{d\beta} + C_L \frac{d\phi}{d\beta} = 0$$

eq 5.20 5.34

<u>YAWING MOMENT</u> $$C_{n_\beta} + C_{n_{\delta_r}} \frac{d\delta_r}{d\beta} + C_{n_{\delta_a}} \frac{d\delta_a}{d\beta} = 0$$

eq 5.21 5.34

ROLLING MOMENT $$C_{\ell_\beta} + C_{\ell_{\delta_r}} \frac{d\delta_r}{d\beta} + C_{\ell_{\delta_a}} \frac{d\delta_a}{d\beta} = 0$$

eq 5.22 5.34

$$\frac{d\delta_r}{d\beta} = \frac{-\dfrac{C_{n_\beta}}{C_{n_{\delta_r}}}\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \dfrac{C_{\ell_\beta}}{C_{n_\beta}}\right\}}{\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}}$$

eq 5.23 5.35

$$\frac{d\delta_a}{d\beta} = \frac{-\dfrac{C_{\ell_\beta}}{C_{\ell_{\delta_r}}}\left\{1 - \dfrac{C_{\ell_{\delta_a}}}{C_{n_{\delta_a}}} \dfrac{C_{n_\beta}}{C_{\ell_\beta}}\right\}}{\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}}$$

eq 5.24 5.35

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} + C_{y_{\delta_r}} \frac{d\delta_r}{d\beta}\right\}$$

eq 5.25 5.35

$$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}}$$

eq 5.26 5.35

$$\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}$$

eq 5.27 5.36

$$1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \frac{C_{\ell_\beta}}{C_{n_\beta}}$$

eq 5.28 5.36

$F_r = K_1 \Delta\delta_r$ (linear feel spring system)

eq 5.29 5.37

$F_r = K_2 q \Delta\delta_r$ ("q – feel" system)

eq 5.30 5.37

$$\delta_{r_{Float}} = -\frac{C_{h_{\beta_v}}}{C_{h_{\delta_r}}} \beta_v$$

eq 5.31 5.38

$$F_r = -K\, C_{h_{\delta_r}}\, q_v S_r \bar{c}_r \left\{\delta_{r_{Equilibrium}} - \delta_{r_{Float}}\right\}$$

eq 5.32 5.39

$\frac{dF_r}{d\beta} = -K\ C_{h_{\delta_r}}\ q_v S_r \bar{c}_r \left\{\frac{d\delta_r}{d\beta} - \frac{d\delta_{r_{Float}}}{d\beta}\right\}$	*eq 5.33*	5.39
$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}}$ (if the assumption $C_{n_{\delta_a}} = 0$ is valid)	*eq 5.34*	5.39
$\frac{d\delta_{r_{Float}}}{d\beta} = -\frac{C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\left(1 - \frac{d\sigma}{d\beta}\right)$	*eq 5.35*	5.39
$\frac{dF_r}{d\beta} = -\ K\frac{C_{h_{\delta_r}}}{C_{n_{\delta_r}}}\ q_v S_r \bar{c}_r \left\{-C_{n_\beta} + \frac{C_{n_{\delta_r}}\ C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\left(1 - \frac{d\sigma}{d\beta}\right)\right\}$	*eq 5.36*	5.40
$\frac{C_{n_{\delta_r}}\ C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\left(1 - \frac{d\sigma}{d\beta}\right)$	*eq 5.37*	5.40
$\frac{d\delta_a}{d\beta} = \frac{-\frac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}}\left\{1 - \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\ \frac{C_{n_\beta}}{C_{\ell_\beta}}\right\}}{\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_r}}}\ \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}}$	*eq 5.38*	5.43
$\frac{d\delta_a}{d\beta} = -\frac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}}$	*eq 5.39*	5.44
$\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\ \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}$	*eq 5.40*	5.44
$\left\{1 - \frac{C_{\ell_{\delta_a}}}{C_{n_{\delta_a}}}\ \frac{C_{n_\beta}}{C_{\ell_\beta}}\right\}$	*eq 5.41*	5.44
$\frac{d\phi}{d\beta} = -\ \frac{1}{C_L}\left\{C_{y_\beta} + C_{y_{\delta_r}}\ \frac{d\delta_r}{d\beta}\right\}$	*eq 5.42*	5.47
$\frac{d\delta_r}{d\beta} = -\ \frac{C_{n_\beta}}{C_{n_{\delta_r}}}$	*eq 5.43*	5.47
$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\ C_{n_\beta}\right\}$	*eq 5.44*	5.47

$$C_{y_\beta} = C_{y_{\beta_{\text{Wing, Fuselage, Nacelles}}}} + C_{y_{\beta_{\text{Vertical Tail}}}}$$ *eq 5.45* 5.47

$$C_{n_\beta} = C_{n_{\beta_{\text{Wing, Fuselage, Nacelles}}}} + C_{n_{\beta_{\text{Vertical Tail}}}}$$

$$C_{n_\beta} = C_{n_{\beta_{W,F,N}}} - C_{y_{\beta_v}} \frac{\ell_v}{b}$$ *eq 5.46* 5.47

$$\frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} = \frac{C_{y_{\delta_r}}}{-C_{y_{\delta_r}} \frac{\ell_v}{b}} = -\frac{b}{\ell_v}$$ *eq 5.47* 5.47

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_{\beta_{W,F,N}}} + C_{y_{\beta_v}} + \frac{b}{\ell_v}\left(C_{n_{\beta_{W,F,N}}} - C_{y_{\beta_v}} \frac{\ell_v}{b}\right)\right\}$$

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_{\beta_{W,F,N}}} + \frac{b}{\ell_v} C_{n_{\beta_{W,F,N}}}\right\}$$

eq 5.48 5.48

$$L_\beta = 0 \quad L_r = 0$$ *eq 5.49* 5.51

$$S\left(S - L_p\right)\begin{vmatrix} S - Y_\beta & 1 \\ -N_\beta & S - N_r \end{vmatrix} = 0$$ *eq 5.50* 5.53

$$S\left(S - L_p\right)\left\{S^2 + \left(-Y_\beta - N_r\right) S + \left(N_\beta + Y_\beta N_r\right)\right\} = 0$$ *eq 5.51* 5.53

$$\frac{g}{u_0}\left\{L_\beta N_r - N_\beta L_r\right\}$$ *eq 5.52* 5.54

$$p_{SS} = -\frac{L_{\delta_a}}{L_p}\delta_a = -\frac{C_{\ell_{\delta_a}}}{C_{\ell_p}} \frac{2V}{b} \delta_a$$ *eq 5.53* 5.55

$$L_{\delta_a} = \frac{\partial L / \partial \delta_a}{I_{XX}} = C_{\ell_{\delta_a}} \frac{qSb}{I_{XX}}$$ = rolling moment due to lateral control deflection (lateral control power) term.

eq 5.54 5.55

$$\tau_R = -\frac{1}{L_p}$$ *eq 5.55* 5.56

$$\left\{S^2 + \left(-Y_\beta - N_r\right) S + \left(N_\beta + Y_\beta N_r\right)\right\} = 0$$ *eq 5.56* 5.57

$$\omega_{n_{DR}} \doteq M\sqrt{C_{n_\beta} \frac{\gamma P_a Sb}{2 I_{ZZ}}}$$ *eq 5.57* 5.57

$$\zeta_{DR} = C_{n_r}\sqrt{\frac{\rho Sb^3}{8 C_{n_\beta} I_{ZZ}}}$$ *eq 5.58* 5.59

$$\frac{\phi}{\delta_a}(S) = \frac{L_{\delta_a}\left[S^2 + 2\zeta_\phi \omega_{n_\phi} S + \omega_{n_\phi}^2\right]}{\left(S + \frac{1}{\tau_s}\right)\left(S + \frac{1}{\tau_R}\right)\left[S^2 + 2\zeta_d \omega_{n_d} S + \omega_{n_d}^2\right]}$$ *eq 5.59* 5.67

$$g \sin\phi = Vr$$ *eq 5.60* 5.75

SIDEFORCE $$C_{y_\beta}\beta + C_{y_r}\left(\frac{rb}{2V}\right) + \frac{W}{qS}\sin\phi = \frac{W}{g}\frac{Vr}{qS}$$

eq 5.61 5.76

YAWING MOMENT $$C_{n_\beta}\beta + C_{n_r}\left(\frac{rb}{2V}\right) = 0$$ *eq 5.62* 5.76

ROLLING MOMENT $$C_{\ell_\beta}\beta + C_{\ell_{\delta_a}}\delta_a + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0$$

eq 5.63 5.76

$$\delta_{a_{\text{Equilibrium}}} = -\frac{1}{C_{\ell_{\delta_a}}}\left\{C_{\ell_r}\left(\frac{rb}{2V}\right) + C_{\ell_\beta}\beta\right\}$$ *eq 5.64* 5.76

$$\beta = -\frac{C_{n_r}}{C_{n_\beta}}\left(\frac{rb}{2V}\right)$$ *eq 5.65* 5.76

$$\frac{d\delta_{a_{\text{Equilibrium}}}}{d\left(\frac{rb}{2V}\right)} = -\frac{1}{C_{\ell_{\delta_a}} C_{n_\beta}}\left\{C_{\ell_r} C_{n_\beta} - C_{\ell_\beta} C_{n_r}\right\}$$ *eq 5.66* 5.77

SIDEFORCE $$C_{y_\beta}\beta + C_{y_{\delta_r}}\delta_r + C_{y_r}\frac{rb}{2V} + \frac{W}{qs}\sin\phi = \frac{W}{g}Vr$$

eq 5.67 5.81

YAWING MOMENT $C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_r}\ \frac{rb}{2V} = 0$ *eq 5 .68* 5.81

ROLLING MOMENT $C_{\ell_\beta}\ \beta + C_{\ell_r}\ \frac{rb}{2V} = 0$ *eq 5.69* 5.81

$$\delta_{r_{\text{Equilibrium}}} = -\frac{1}{C_{n_{\delta_r}}}\left\{C_{n_r}\left(\frac{rb}{2V}\right) + C_{n_\beta}\ \beta\right\}$$ *eq 5.70* 5.81

$$\beta = -\frac{C_{\ell_r}}{C_{\ell_\beta}}\ \frac{rb}{2V}$$ *eq 5.71* 5.81

$$\frac{d\delta_{r_{\text{Equilibrium}}}}{d\left(\frac{rb}{2V}\right)} = \frac{1}{C_{n_{\delta_r}}\ C_{\ell_\beta}}\left\{C_{\ell_r}\ C_{n_\beta} - C_{\ell_\beta}\ C_{n_r}\right\}$$ *eq 5.72* 5.81

SIDEFORCE $C_{y_\beta}\ \beta + C_{y_{\delta_r}}\ \delta_r + C_{y_r}\left(\frac{rb}{2V}\right) = 0$

$\left(\text{Since } \frac{W}{qS}\ \sin\ \phi = \frac{W}{g}\ \frac{Vr}{qS}\right)$ *eq 5.73* 5.82

YAWING MOMENT $C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_{\delta_a}}\ \delta_a + C_{n_r}\left(\frac{rb}{2V}\right) = 0$

eq 5.74 5.82

ROLLING MOMENT $C_{\ell_\beta}\ \beta + C_{\ell_{\delta_r}}\ \delta_r + C_{\ell_{\delta_a}}\ \delta_a + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0$

eq 5.75 5.82

$$\delta_{r_{\text{Equilibrium}}} = \frac{\begin{vmatrix} C_{y_\beta} & -C_{y_r} \\ C_{n_\beta} & -C_{n_r} \end{vmatrix}}{\begin{vmatrix} C_{y_\beta} & C_{y_{\delta_r}} \\ C_{n_\beta} & C_{n_{\delta_r}} \end{vmatrix}}\left(\frac{rb}{2V}\right)$$ *eq 5.76* 5.82

$$\frac{d\delta_{r_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = -\frac{C_{n_r}}{C_{n_{\delta_r}}}\left\{\frac{C_{y_\beta} - \frac{C_{y_r}}{C_{n_r}} C_{n_\beta}}{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} C_{n_\beta}}\right\}$$ *eq 5.77* 5.82

$$\delta_{r_{Equilibrium}} = \frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right)$$

$$C_{n_{\delta_a}} = 0$$ *eq 5.78* 5.83

$$C_{n_\beta}\,\beta + C_{n_{\delta_r}}\left\{-\frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right)\right\} + C_{n_r}\left(\frac{rb}{2V}\right) = 0$$

$$\beta = \frac{\left(C_{n_r} - C_{n_r}\right)\left(\frac{rb}{2V}\right)}{C_{n_\beta}}$$

$$\beta = 0$$ *eq 5.79* 5.83

$$\delta_{r_{Equilibrium}} = -\frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right) \qquad \beta_{Equilibrium} = 0$$ *eq 5.80* 5.83

$$C_{\ell_{\delta_r}}\left\{-\frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right)\right\} + C_{\ell_{\delta_a}}\,\delta_{a_{Equilibrium}} + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0$$

$$\frac{d\delta_{a_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = -\frac{C_{\ell_r}}{C_{\ell_{\delta_a}}}\left\{1 - \frac{C_{\ell_{\delta_r}} C_{n_r}}{C_{\ell_r} C_{n_{\delta_r}}}\right\}$$ *eq 5.81* 5.84

Level 1 $-\Delta\zeta_d\ \omega_{n_d} = .014\left(\omega_{nd}^2\ |\phi/\beta|_d - 20\right)$

Level 2 $-\Delta\zeta_d\ \omega_{n_d} = .009\left(\omega_{nd}^2\ |\phi/\beta|_d - 20\right)$

Level 3 $-\Delta\zeta_d\ \omega_{n_d} = .005\left(\omega_{nd}^2\ |\phi/\beta|_d - 20\right)$

with ω_{n_d} in rad/sec. *eq 5.82* 5.109

$$\psi_\beta = \frac{-360}{T_d}\,t_{n_\beta} + (n-1)\,360 = \frac{-360}{3.5}\,(2.95) = -303^\circ$$ *eq 5.83* 5.116

SIDEFORCE $$C_{y_\beta}\,\beta + C_{y_{\delta_r}}\,\delta_r + C_L \sin\phi = \frac{mu_0\dot{\beta} + mu_0\, r}{qS}$$

eq 5.84 5.131

YAWING MOMENT	$C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_{\delta_a}}\ \delta_a + C_{n_r}\ \frac{rb}{2V} + C_{n_p}\ \frac{pb}{2V} = \frac{1}{qSb}\ I_{yy}\dot{r}$	*eq 5.85*	5.131
ROLLING MOMENT	$C_{\ell_\beta}\ \beta + C_{\ell_{\delta_r}}\ \delta_r + C_{\ell_{\delta_a}}\ \delta_a + C_{\ell_r}\ \frac{rb}{2V} + C_{\ell_p}\ \frac{pb}{2V} = \frac{1}{qSb}\ I_{yy}\dot{p}$	*eq 5.86*	5.131

$C_{\ell_\beta}\ \beta\ ;\ C_{\ell_{\delta_r}}\ \delta_r\ ;\ C_{\ell_r}\ \frac{rb}{2V}$	*eq 5.87*	5.131
$\dot{p}\ \frac{I_{XX}}{qSb} - C_{\ell_p}\ \frac{pb}{2V_T} - C_{\ell_{\delta_a}}\ \delta_a = 0$	*eq 5.88*	5.132
$\dot{p} - \frac{C_{\ell_p}\,qSb}{I_{XX}}\ \frac{b}{2V_T}\ p - \frac{C_{\ell_{\delta_a}}\,qSb}{I_{XX}}\ \delta_a = 0$	*eq 5.89*	5.132
$\dot{p} - L_p\ p - L_{\delta_a}\ \delta_a = 0$	*eq 5.90*	5.132
$p\,(t) = \frac{L_{\delta_a}\delta_a}{L_p}\left\{e^{L_p t} - 1\right\}$	*eq 5.91*	5.132
$p\,(t) = p_{ss}\left\{1 - e^{-t/\tau_R}\right\}$	*eq 5.92*	5.133
$\dot{p} - L_p\ p - L_{\delta_a}\ \delta_a = 0$	*eq 5.93*	5.133
$\dot{p}_t = 0 = L_{\delta_a}\ \delta_a$	*eq 5.94*	5.133

$$\dot{p} - L_p\ p - L_{\delta_a}\ \delta_a = 0$$

however, when $\left|L_p\ p\right| = \left|L_{\delta_a}\ \delta_a\right|$, $\dot{p} = 0$, and $p = p_{ss}$, thus:

$$p_{ss} = -\frac{L_{\delta_a}\ \delta_a}{L_p}$$ *eq 5.95* 5.135

$$\left(\frac{pb}{2V_T}\right)_{Max} = -\frac{C_{\ell_{\delta_a}}}{C_{\ell_p}}\ \delta_{a_{Max}}$$ *eq 5.96* 5.136

$$\tau_R = -\frac{1}{L_p}$$ *eq 5.97* 5.137

$$p_{ss} = -\frac{L_{\delta_a}}{L_p}\ \delta_a$$ *eq 5.98* 5.138

$$\tau_R = -\frac{1}{L_p} \text{ or } \tau_R = \frac{4\ I_{xx}}{C_{\ell_p}\rho\ V_T S b^2} \text{ or } \tau_R = \frac{4\ I_{xx}}{C_{\ell_p}\sqrt{\sigma\rho_{ssl}}V_e S b^2}$$ *eq 5.99* 5.138

$$P_{CRIT_1} \sim \sqrt{\frac{\frac{C_{m_\alpha}\ qS\bar{c}}{I_{yy}}}{\frac{I_{xx} - I_{zz}}{I_{yy}}}}$$ *eq 5.100* 5.148

$$P_{CRIT_2} \sim \sqrt{\frac{\frac{C_{n_\beta}\ qSb}{I_{zz}}}{\frac{I_{yy} - I_{xx}}{I_{zz}}}}$$ *eq 5.101* 5.148

$$F_a = K_1\ \Delta\delta_a \text{ (linear feel spring system)}$$ *eq 5.102* 5.156

$$F_a = K_2\ q\Delta\delta_a \text{ ("q – feel" system)}$$ *eq 5.103* 5.156

$$C_{h_\alpha} = C_{h_{\delta_\alpha}}\ \delta_\alpha + C_{h_\alpha}\ \Delta\alpha_{Ave}$$ *eq 5.104* 5.157

$$\Delta\alpha_{Ave} = \frac{py'}{V}$$ *eq 5.105* 5.157

$$\delta_{a_{Float}} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}}\ \frac{py'}{V} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}}\left(\frac{2y'}{b}\right)\left(\frac{pb}{2V}\right)$$ *eq 5.106* 5.157

$F_a = -K\, C_{h_{\delta_a}}\, q\, S_a\, \bar{c}_a \left\{\delta_{a_{Equilibrium}} - \delta_{a_{Float}}\right\}$	*eq 5.107*	5.158
$\delta_{aEquilibrium} = -\frac{C_{\ell_p}}{C_{\ell_{\delta_a}}}\frac{pb}{2V}$ (steady state roll)	*eq 5.108*	5.159
$\delta_{a_{Float}} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}}\left(\frac{2y'}{b}\right)\left(\frac{pb}{2V}\right)$	*eq 5.109*	5.159
$F_a = V_p \left\{-\frac{K}{4}\, C_{h_{\delta_a}}\, \rho\, S_a \bar{c}_a\, b\right\}\left\{-\frac{C_{\ell_p}}{C_{\ell_{\delta_a}}} + \frac{C_{h_\alpha}}{C_{h_{\delta_a}}}\,\frac{2y}{b}\right\}$	*eq 5.110*	5.159
$F_a = K_1\, V_p$	*eq 5.111*	5.159
$pV = K_2$	*eq 5.112*	5.159
$p\,(t) = p_{ss}\left\{1 - e^{-t/\tau_R}\right\}$	*eq 5.113*	5.190
$p\,(t) = p_{ss} - p_{ss}\, e^{-t/\tau_R}$	*eq 5.114*	5.190
$X\,(t) = p_{ss} e^{-t/\tau_R}$	*eq 5.115*	5.191
$\ln X(t) = \ln p_{ss} - \frac{t}{\tau_R}$	*eq 5.116*	5.192
$\ln X(t) = K_1 - \frac{t}{K_2}$	*eq 5.117*	5.192
$X(t) = p_{ss}\, e^{-t/\tau_R}$	*eq 5.118*	5.192
$\tau_R = t_2 - t_1$	*eq 5.119*	5.194

CHAPTER FIVE

LATERAL-DIRECTIONAL FLYING QUALITIES

5.1 INTRODUCTION

The investigation of lateral-directional stability and control involves the study of characteristics exhibited in the airplane's planes of asymmetry. These planes of asymmetry divide the airplane into unsymmetrical parts of contain components of motion only along the Y axis and about the X and Z axes (see Figure 5.1).

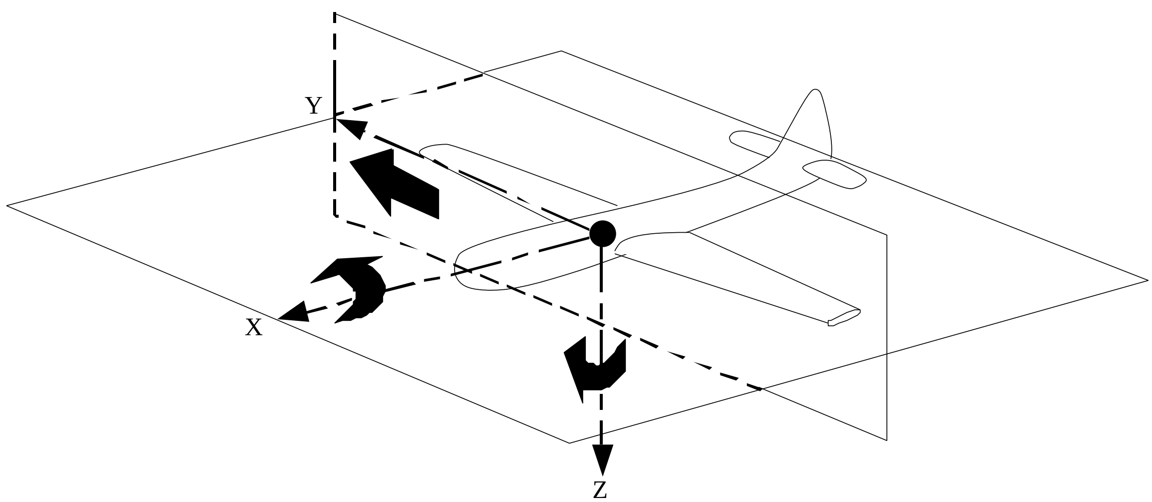


Figure 5.1
Airplane Axis System and Planes of Asymmetry

Airplane motion in the planes of asymmetry or lateral-directional motion generally results in some motion in the plane of symmetry or longitudinal motion. For part of the study of lateral-directional flying qualities, this longitudinal motion will be considered fairly insignificant. However, in some flight conditions, lateral-directional motion can generate significant longitudinal motion and vice versa. These conditions will be discussed thoroughly in a subsequent section.

Lateral-directional flying qualities must be investigated from equilibrium and nonequilibrium flight conditions. From equilibrium conditions, the static lateral-directional characteristics may be determined. These characteristics are:

1. Variation of directional control forces and rudder positions with sideslip angle in steady heading flight at a constant trim airspeed (static directional stability characteristics).

2. Variation of lateral control forces and aileron positions with sideslip angle in steady heading flight at a constant trim airspeed (static lateral stability characteristics or dihedral characteristics).

3. Variation of bank angle with sideslip angle in steady heading flight at a constant trim airspeed (sideforce characteristics).

Dynamic lateral-directional flying qualities are investigated from nonequilibrium flight conditions. This requires study of the characteristics of the three lateral-directional modes of motion - the Dutch roll mode, the spiral mode, and the roll mode - which are suppressed in equilibrium flight. Two of the lateral-directional modes differ from the longitudinal modes in that the pilot does not usually deliberately excite the Dutch roll or spiral modes. Excitation of these modes is not required to maneuver the airplane under normal flight conditions. However, the Dutch roll and spiral modes are continually inadvertently excited by the pilot or by external perturbations. Therefore, the characteristics of these modes greatly affect the pilot's opinion of the airplane during all phases of mission accomplishment. During certain special flight conditions, such as flight with asymmetric power or landing with a crosswind; the pilot may deliberately utilize the Dutch roll mode to generate sideslip changes in order to maintain steady heading flight. Since the Dutch roll mode is a second order response generally involving both lateral and directional motion, the characteristics of this mode to be investigated are:

1. Frequency or period of the motion.

2. Damping of the motion or lack of it.

3. The relative magnitude of the lateral part of the motion to the directional part of the motion, or simply, the "roll to yaw ratio".

4. The degree of excitation of the Dutch roll mode during uncoordinated, aileron only turns.

Since the spiral mode is a first order motion which may be convergent, divergent, or neutral, the characteristics of this motion to be investigated are:

1. The nature of the motion; i.e., whether it is divergent, neutral, or convergent.

2. The time required for the amplitude of the first order motion to double or half.

Obviously, the pilot will deliberately excite the roll mode in order to make bank angle changes required in all phases of mission accomplishment. Characteristics of the roll mode have a significant influence on the pilot's opinion of the airplane. The roll mode is an essentially first order response and is usually heavily damped. Therefore, the characteristics of the roll mode to be investigated are:

1. The roll mode time constant.

2. Steady state roll rates obtainable with various lateral control inputs.

3. The nature and amount of yawing motion generated during rolling maneuvers.

The pilot's opinion of lateral-directional flying qualities depends on <u>all</u> the static and dynamic characteristics mentioned above <u>plus the characteristics of the lateral-directional control system</u>. In addition, other parameters influence lateral-directional flying qualities because of inseparable interaction phenomenon, such as roll response to a directional control input, and yaw response to a lateral control input and roll rate. Therefore, it is not possible to state which of the aforementioned characteristics are dominant in a particular flight condition. They are all important to varying degrees and must all be investigated to determine their influence on every piloting task. In the interest of simplicity, lateral-directional flying qualities will be presented in four sections:

1. "Normal" lateral-directional flying qualities.

2. Rolling performance and roll coupling.

3. Asymmetric power.

4. Miscellaneous lateral-directional tests.

Lateral-directional stability and control characteristics profoundly affect the pilot's ability to perform both maneuvering and nonmaneuvering tasks required in mission fulfillment. Satisfactory lateral-directional characteristics allow the pilot to trim the airplane easily and simply, maneuver the airplane safely and precisely without excessive effort, and maintain adequate control of the airplane under all possible operational flight conditions. Thus, mission tasks can be performed safely, simply, and precisely, and overall mission effectiveness is correspondingly enhanced.

5.2 THEORY - NORMAL LATERAL-DIRECTIONAL STABILITY AND CONTROL

5.2.1 Static Lateral-Directional Stability and Control

It is now necessary to study the stability characteristics of the airplane when its flight path deviates from the plane of symmetry. This means that the relative wind will be making some angle to the airplane's plane of symmetry; this angle is referred to as the sideslip angle, β (see Figure 5.2). The angle of sideslip differs from the angle of attack in that it lies in a different plane and its action is quite different. Angle of attack determines the airplane's lift coefficient and, therefore, its airspeed (for unaccelerated flight conditions) or normal acceleration (at a constant airspeed). However, sideslip is generally quite useless to the pilot. It can be used to increase drag and increase rate of descent during landing, or it can be used to make airplane heading the same as runway heading during crosswind landings, or it can be used to ease the pilot's workload during flight with asymmetric power, etc. Nevertheless, it can be stated that, in general, it is advantageous to maintain zero sideslip in almost all flight conditions. Therefore, the problem of directional stability and control is to insure that the airplane tends to maintain zero sideslip and to insure that the pilot is provided a suitable means of controlling sideslip during maneuvers that tend to produce sideslip, or during maneuvers in which he wishes to deliberately induce sideslip.

Obviously, the pilot must also be provided with satisfactory control of the airplane's angle of bank, ϕ; this control is necessary to provide a force to accelerate the flight path in the horizontal plane; i.e., to turn the airplane. A subsequent section on Rolling Performance will discuss thoroughly the subject of lateral control. However, since sideslip generally induces a rolling moment, it is necessary to introduce here the concept of dihedral effect or lateral stability. This is really not a static stability in the true sense of the term since the rolling moments created are not a result of bank angle but are a result of sideslip. However, dihedral effect has a significant influence on the pilot's opinion of the airplane's lateral-directional flying qualities.

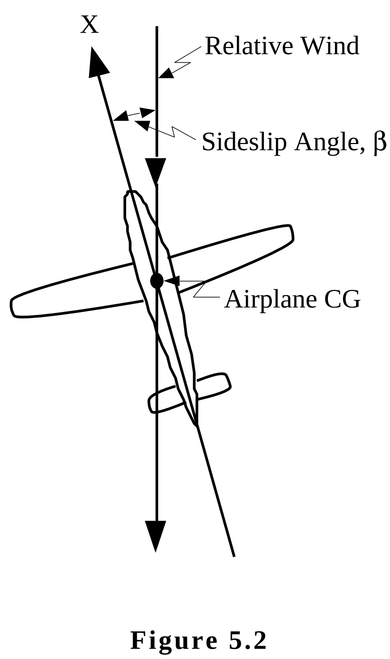


Figure 5.2
The Sideslip Angle

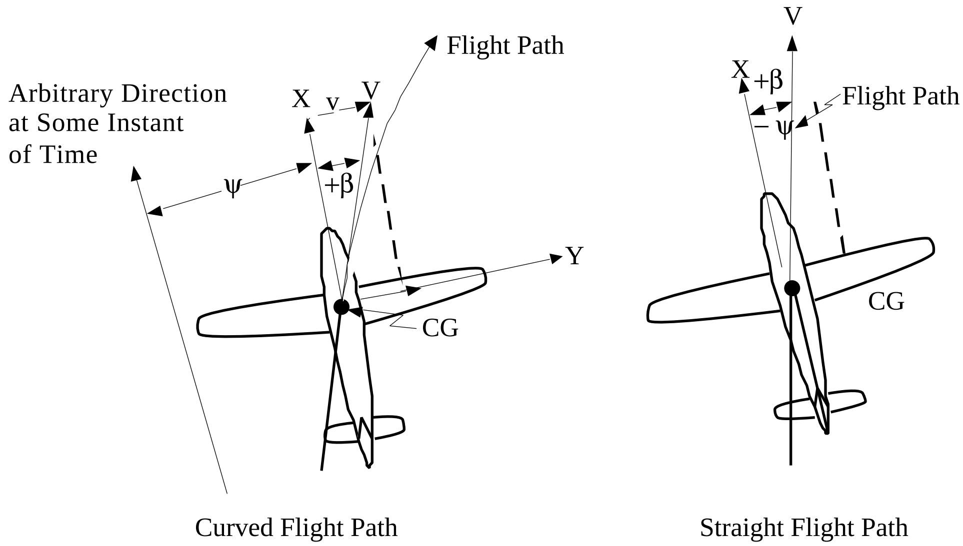
In order to discuss the theory of later-directional flying qualities, it is necessary first to develop some terminology which will permit a complete description of the forces and moments acting on the airplane during any maneuver that involves sideslip.

5.2.1.1 LATERAL-DIRECTIONAL TERMINOLOGY

The angle of sideslip, β, is equal of the arcsin $\left(\frac{v}{V}\right)$ or for small angles encountered in normal flight, $\beta = \frac{v}{V}$ (Figure 5.3). The sideslip angle is arbitrarily given a positive sign when the relative wind is to the right of the geometric longitudinal axis of the airplane. The angle of yaw, ψ, is defined as the angular displacement of the airplane's geometric longitudinal axis in the horizontal plane from some arbitrary direction taken as zero at some instant of time. Note that for a curved flight path, yaw angle does not equal sideslip angle. In a 360 degree turn, the airplane yaws through 360 degrees but may develop no sideslip during the maneuver if the turn is perfectly coordinated. If the airplane is sideslipped to maintain a straight path, the angle of yaw is equal in magnitude but opposite in sign to the angle of sideslip.

Considerable confusion can arise if the terms sideslip and yaw are misunderstood. In this manual, sideslip will always be used to describe the angle generated by the relative wind not being perfectly aligned with the airplane's geometric longitudinal axis in the XY plane.[1] However, the term yaw will also be used in describing <u>rates</u> and <u>moments</u> in many cases. This is necessary since, for instance, the airplane can exhibit a <u>yaw rate</u> with <u>zero sideslip</u> or can have <u>yawing moments</u> generated at <u>zero sideslip</u>. With the meanings of sideslip and yaw in mind, other lateral-directional stability derivatives may now be developed.

[1] Sideslip is invariably used to describe this angle in flight test work. However, yaw is almost invariably used in wind tunnel work and in many theoretical works to describe the same angle.



V = Velocity of the Airplane Tangential to the Flight Path at any Time

v = Component of V along the Y Axis of the Airplane

Figure 5.3
Sideslip Angle and Angle of Yaw

5.2.1.2 LATERAL-DIRECTIONAL STABILITY DERIVITIVES-SIDEFORCES AND SIDEFORCE DERIVATIVES

Whenever a sideslip angle is imposed on the airplane, sideforces are developed by the fuselage, wings, and the vertical tail. The main contributions come from the fuselage and the vertical tail. Wing interference with the flow of air over the wing-fuselage combination makes the estimation of the sideforce due to the fuselage alone very difficult. The portion of the sideforce that is due to the vertical tail is more predictable (Figure 5.4).

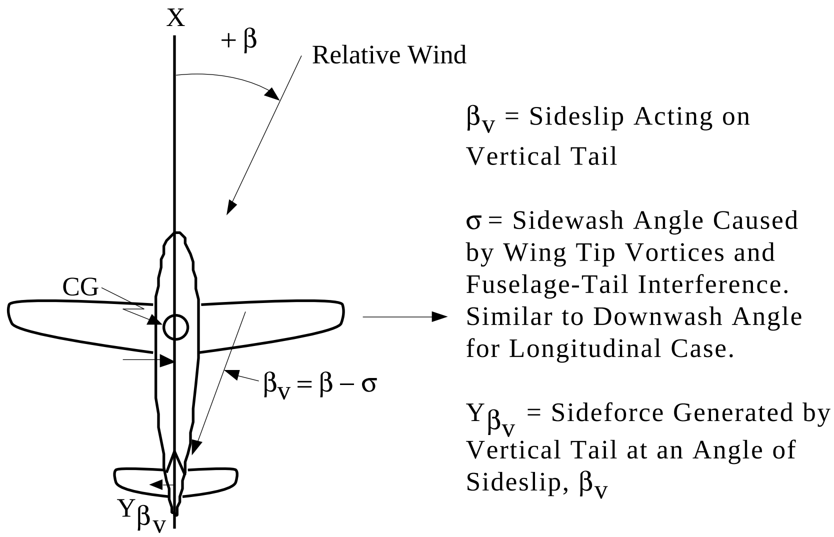


Figure 5.4
Generation of Sideforce Due to Sideslip by the Vertical Tail

5.2.1.2.1 Vertical Tail Contribution, Y_{β_v} or $C_{y_{\beta_v}}$

When the airplane is subjected to a sideslip angle, β, the vertical tail is subjected to a sideslip angle, β_V, which is generally less than β because of the sidewise inflow of air which occurs prior to the airflow impinging on the vertical tail. Thus, the vertical tail develops a force which is directed sideways with respect to the airplane. The magnitude of this sideforce may be expressed as:

$$Y_{\beta_v} = -a_v \left(\beta - \sigma\right) q_v \; S_v \qquad \textit{eq 5.1}$$

Where:

a_V = lift curve slope of the vertical tail, a vertical tail design parameter.

β = airplane sideslip angle.

σ = sidewash angle, a measure of the change in direction of the relative wind in the XY plane due to interference from airplane components, best determined from wind tunnel studies.

q_v = dynamic pressure at vertical tail in pounds per square foot.

S_v = area of the vertical tail in square feet.

The sign of the sideforce created by the vertical tail is <u>negative</u> since the sign convention utilized in this manual is that forces acting out the right wing of the airplane are positive. Thus, since positive sideslip generates vertical tail sideforces acting to the left, Y_{β_v} carries a negative sign. Similarly, the nondimensional derivative form, $C_{y_{\beta_v}}$, also carries a negative sign and may be expressed as follows:

$$C_{y_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} \qquad \textit{eq 5.2}$$

Where:

η_v = vertical tail efficiency factor, $\frac{q_v}{q}$, where q is free stream dynamic pressure.

S_w = area of the wing in square feet.

5.2.1.2.2 Sideforce Due to Roll Rate, Y_p *or* C_{y_p}

A sideforce is developed at the vertical tail whenever the airplane is rolling. As the airplane rolls, every point on the vertical tail that is not on the rolling axis is subjected to a side velocity. This side velocity creates a sideslip at the vertical tail, even though the airplane may have zero sideslip (Figure 5.5). This contribution, called sideforce due to roll rate, Y_p, is generally small unless the airplane has a very high vertical tail. It carries a negative sign, since a positive (right) roll rate, p, creates sideforce acting to the left.[1] Similarly, the nondimensional derivative form, C_{y_p}, also carries a negative sign.

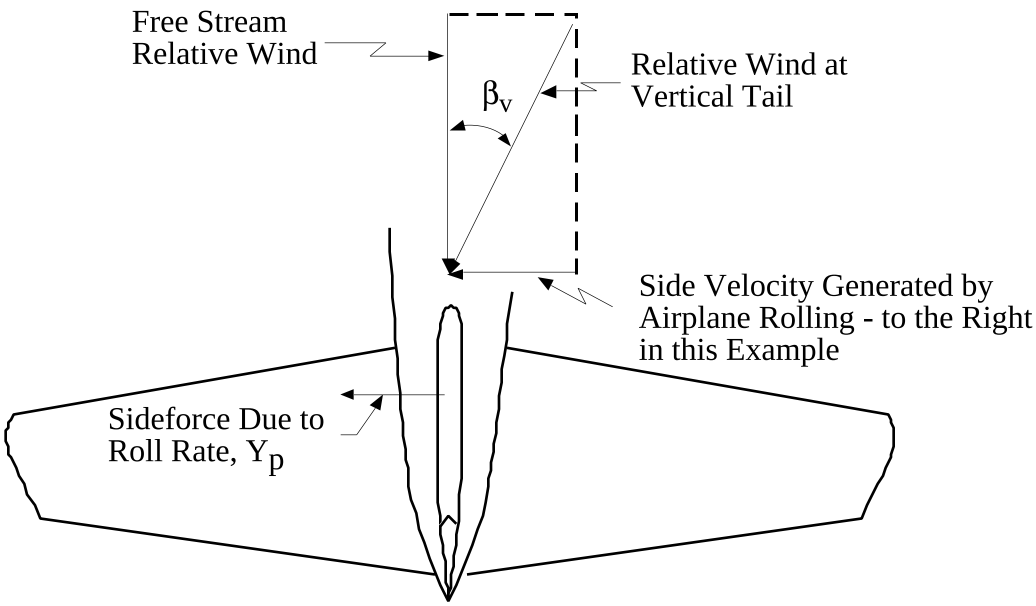


Figure 5.5
Generation of Sideforce by the Vertical Tail Due to Roll Rate

5.2.1.2.3 Sideforce Due to Yaw Rate, Y_r *or* C_{y_r}

Whenever the airplane exhibits a yaw rate, r, another sideforce is developed. This sideforce, Y_r, is developed as a result of side velocity due to the yaw rate (Figure 5.6). The sideforce due to yaw rate is generally fairly small. The vertical tail and other surfaces aft of the CG develop a positive Y_r or C_{y_r}; areas forward of the CG develop a negative Y_r or C_{y_r}. The vertical tail contribution is generally dominant, therefore, Y_r and C_{y_r} usually carry a positive sign.

---

[1] This is the "right hand rule" convention. If the right hand is placed along the airplane axis under consideration such that the thumb points in the positive direction, the fingers of the hand will curl in the positive rotational direction. Positive axis directions are defined as (from the airplane CG) toward the right wing, forward, and down.

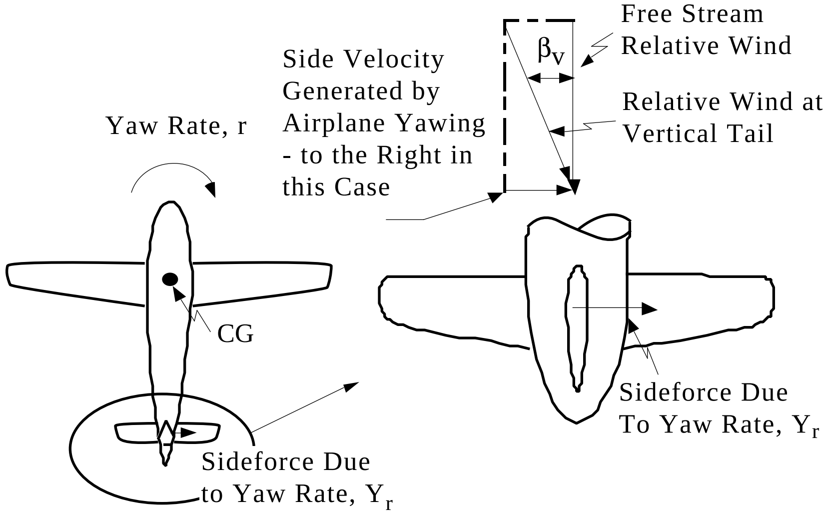


Figure 5.6
Generation of Sideforce Due to Yaw Rate

5.2.1.2.4 Sideforce Due to Rudder Deflection,

Y_{δ_r} *or* $C_{y_{\delta_r}}$

A deflection of the rudder control surface makes the vertical tail-rudder combination a cambered airfoil. This generates a sideforce from the vertical tail (Figure 5.7). The magnitude of this sideforce, Y_{δ_r}, may be expressed as follows:

$$Y_{\delta_r} = a_v \ \tau_v \ \delta_r \ q_v \ S_v \qquad \text{eq 5.3}$$

Where:

τ_v = rate of change of effective tail sideslip angle with rudder deflection, sometimes written $\frac{d\beta_v}{d\delta_r}$. It is a function of the ratio of the area of the rudder to the area of the entire vertical tail; for the all-moveable vertical tail, $\tau_v = 1.0$.

δ_r = rudder surface deflection.

The sideforce due to rudder deflection carries a positive sign, since trailing edge left rudder deflection, considered positive rudder deflection, generates a sideforce acting toward the right. Similarly, the nondimensional derivative form, $C_{y_{\delta_r}}$, also carries a positive sign and may be expressed as follows:

$$C_{y_{\delta_r}} = a_v \, \tau_v \, \eta_v \frac{S_v}{S_w} \qquad \text{eq 5.4}$$

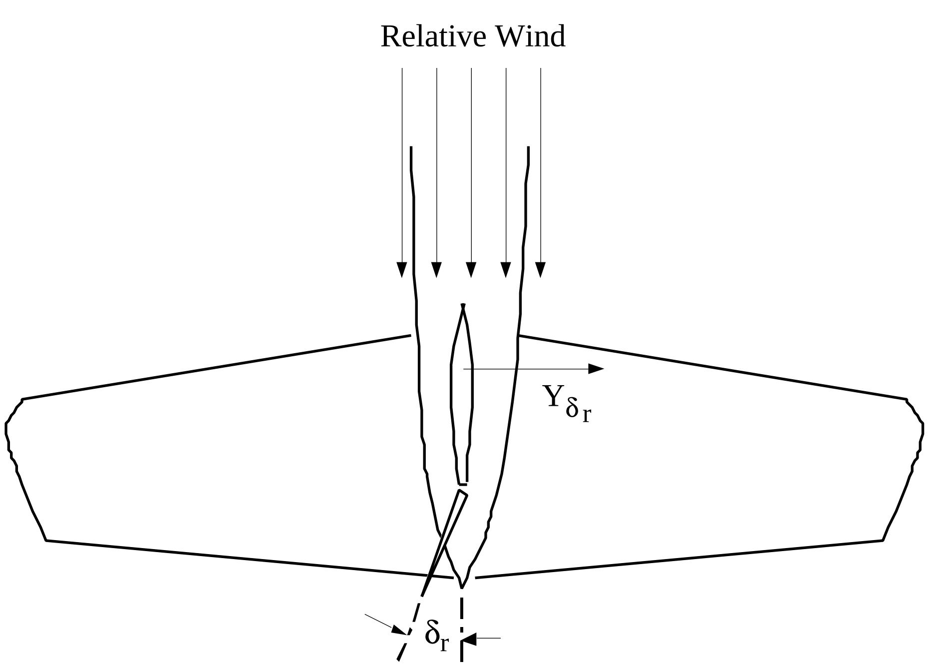


Figure 5.7
Generation of Sideforce Due to Rudder Deflection

5.2.1.2.5 Sideforce Due to Gravity, W sin ϕ *or* C_{L_ϕ}

Whenever the airplane is banked, there is a gravitational component of sideforce. The magnitude of this sideforce is dependent on the magnitudes of the bank angle and the weight vector of the airplane. It is the projection of the weight vector on the Y axis of the airplane (Figure 5.8) and may be expressed as W sin ϕ or mg sin ϕ .[1] In nondimensional form for "small" bank angles, it may also be expressed as C_{L_ϕ}. Since the gravitational component of sideforce acts through the airplane CG, it generates no rolling or yawing moments.

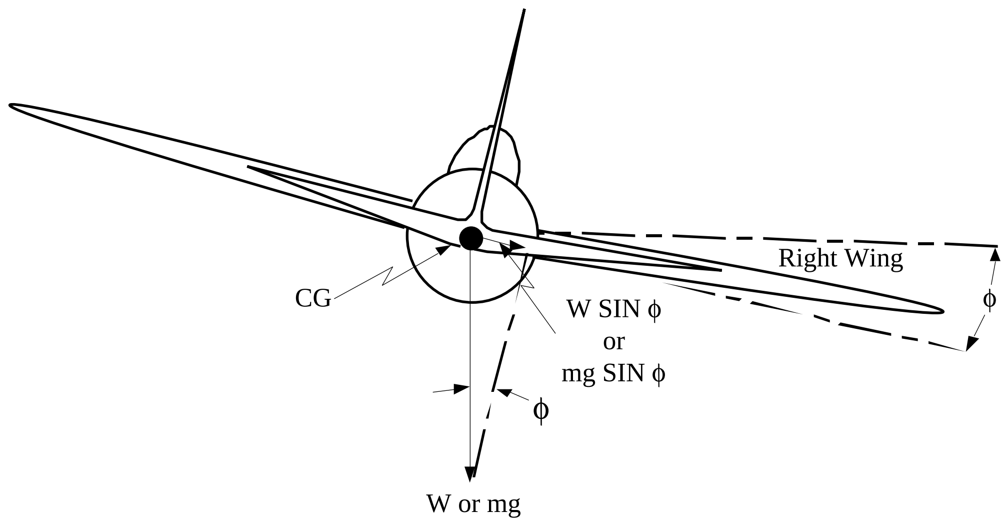


Figure 5.8
Gravitational Component of Sideforce Due to Bank Angle

5.2.1.2.6 Sideforce Due to Lateral Control Deflection, Y_{δ_a} *or* $C_{y_{\delta_a}}$

If the lateral control devices are located close inboard on the wings, their deflection may disturb the airflow around the fuselage enough to generate a sideforce. In supersonic flight, shock wave formation due to lateral control deflection may cause a similar effect. This sideforce is generally very small and, therefore, is usually neglected.

[1] Where W is the weight of the airplane, m is the mass, and ϕ is the bank angle.

5.2.1.3 LATERAL-DIRECTIONAL STABILITY DERIVATIVES - ROLLING MOMENTS AND LATERAL STABILITY DERIVATIVES

5.2.1.3.1 Rolling Moment Due to Sideslip, Dihedral Effect L_β *or* C_{ℓ_β}

Whenever a sideslip angle is imposed on the airplane, rolling moments are generally developed as a direct result of the sideslip. This rolling moment due to sideslip is called dihedral effect. Dihedral effect is generally referred to as being positive if the airplane tends to roll in a direction opposite to the imposed sideslip. However, the sign of the moment, L_β or nondimensional derivative, C_{ℓ_β}, is negative for "positive" dihedral effect since a positive (right) sideslip angle would result in a negative (left) rolling moment. Dihedral effect is mainly influenced by geometric wing dihedral, wing sweep, wing placement on the fuselage, and vertical tail height, as well as power for propeller-driven airplanes, and flap deflection if the flap hinge line is swept. If the airplane exhibits a positive wing dihedral angle, Γ (tip chord above the root chord), the airplane tends to roll away from any sideslip which is developed because the forward wing is subjected to a higher effective angle of attack than the trailing wing (Figure 5.9). Thus, positive geometric wing dihedral contributes to positive dihedral effect.

Wing sweepback tends to contribute to positive dihedral effect since the forward wing's effective sweep angle is reduced and the trailing wing's effective sweep angle is increased (Figure 5.10). Neglecting compressibility effects and assuming that the airplane is not operating close to its stall angle of attack, the lift coefficient on the forward wing is thus increased and the lift coefficient on the trailing wing is decreased.[1] Therefore, wing sweepback contributes to positive dihedral effect. Additionally, dihedral effect of the airplane with swept wings is directly related to lift coefficient (Figure 5.11) Airplanes with sweepback will exhibit increasing positive dihedral effect with increasing lift coefficient; these airplanes may tend to have excessive positive dihedral at low airspeeds.

---

[1] See Figure 2.3, Page 2.6 of the "STALLS" section.

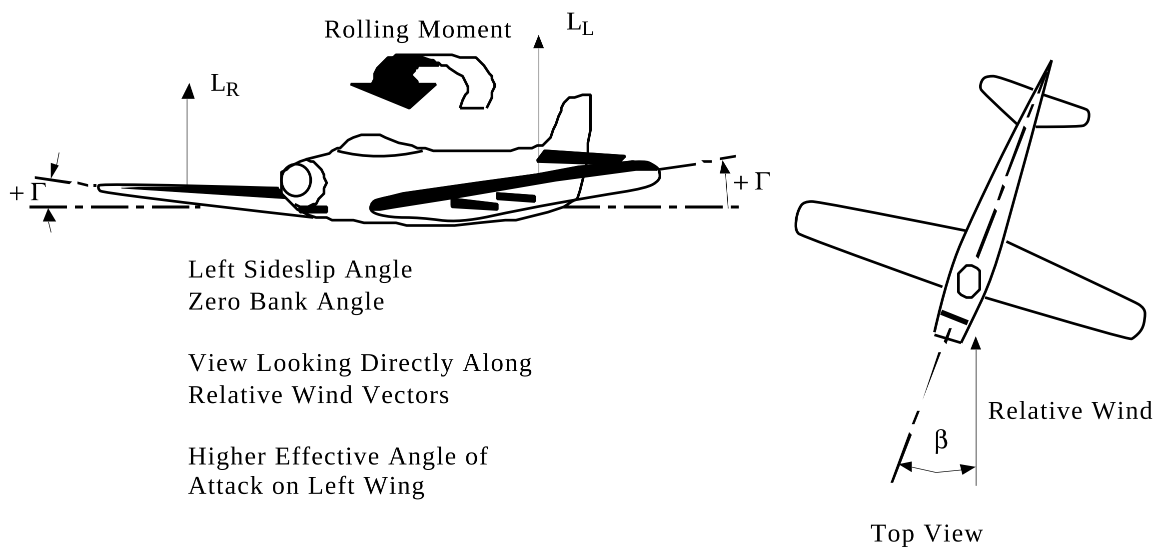


Figure 5.9
Rolling Moment Generated by Sideslip and Geometric Dihedral

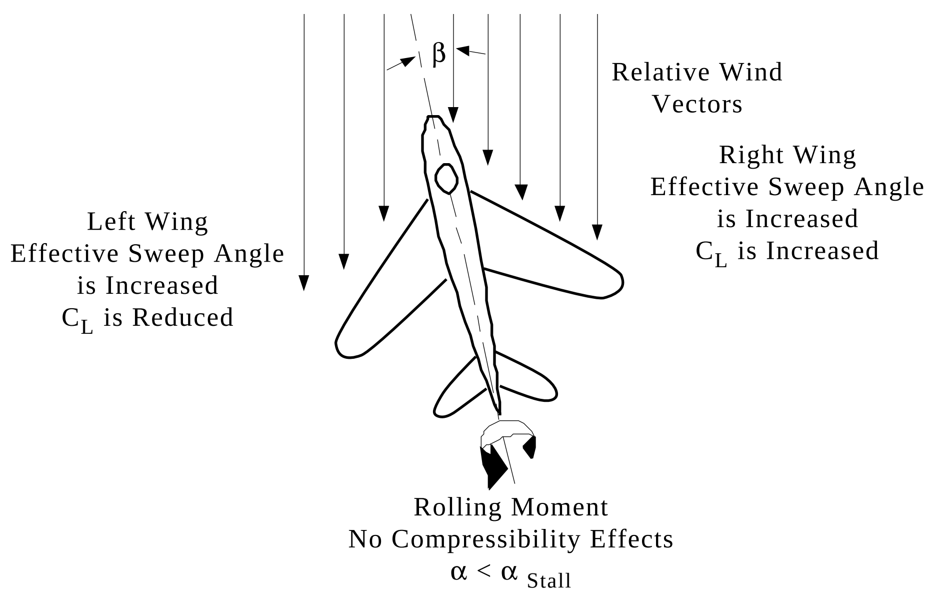


Figure 5.10
Rolling Moment Generated by Sideslip and Wing Sweep

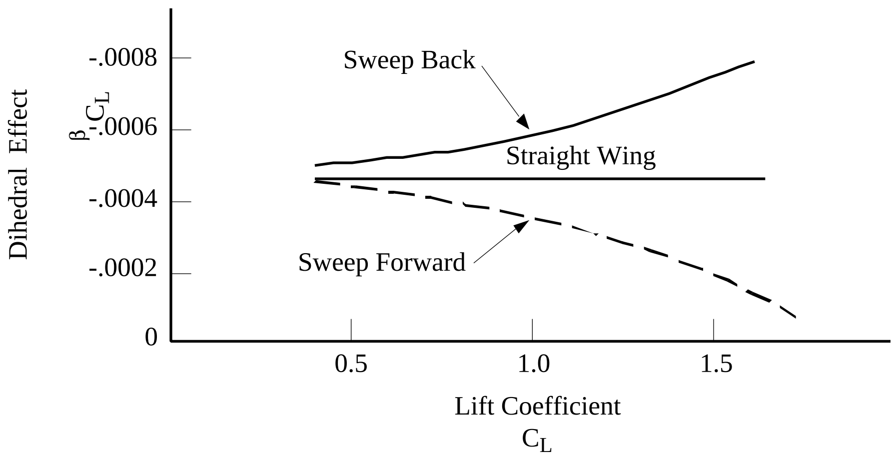


Figure 5.11
Typical Variation of Dihedral Effect with Lift Coefficient

Interference effects between the wing, fuselage, and vertical tail influence the airplane dihedral effect; however, the influence is bothersome and difficult to analyze. In general, a high wing placement on the fuselage tends to increase dihedral effect, a mid wing design has negligible influence, and a low wing design tends to decrease dihedral effect (Figure 5.12).

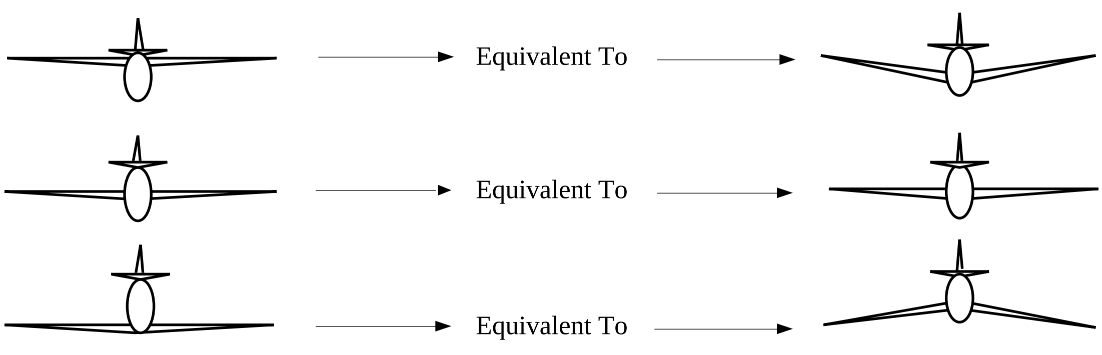


Figure 5.12
General Relationships Between Wing Placement and Effective Dihedral

The vertical tail can contribute to the dihedral effect if it is located above the airplane's center of gravity. This contribution is quite easily computed from the sideforce created by the vertical tail when sideslip is imposed on the airplane (Figure 5.13). Since positive sideslip generates a rolling moment from the vertical tail which tends to roll the airplane in the direction opposite to the sideslip, the vertical tail contributes to positive dihedral effect. The magnitude of this rolling moment contribution may be presented as follows in nondimensional derivative form:

$$C_{\ell_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} \frac{Z_v}{b} \qquad \textit{eq 5.5}$$

Where:

Z_v = Vertical distance between center of pressure of the vertical tail and airplane center of gravity in feet.

b = airplane wing span in feet. (b appears in <u>moment</u> relationships in order to nondimensionalize the moment derivatives).

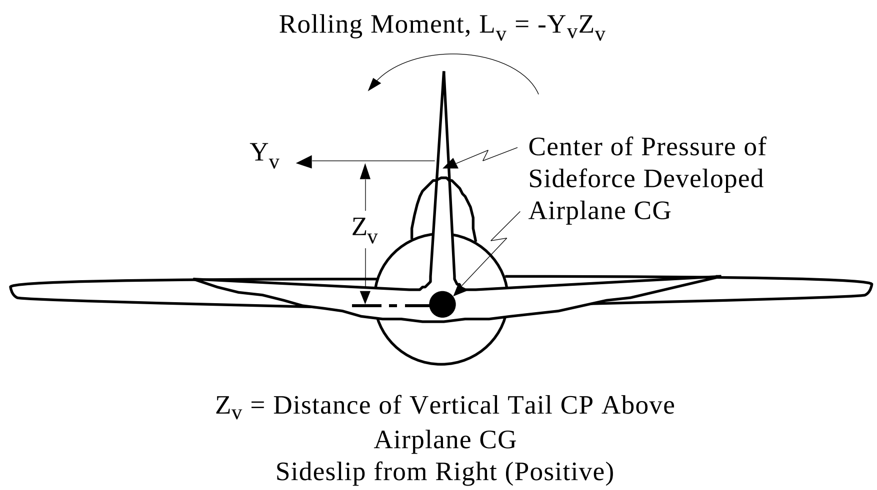


Figure 5.13
Influence of Vertical Tail on Dihedral Effect

The influence of power on dihedral effect is apparent only if the airplane is equipped with one or more reciprocating or turboprop engines. This influence arises because of the displacement of the propeller slipstream in a sideslip, resulting in one wing being immersed in the slipstream to a greater extent than the other (Figure 5.14). This tends to cause the airplane to roll in a direction toward the sideslip; thus power <u>tends</u> to cause negative dihedral effect in propeller driven airplanes. This power influence is largest in full-power, low-airspeed flight conditions where the ratio of slipstream velocity to free steam velocity is the greatest. The "destabilizing" influence of power on dihedral effect may be extremely pronounced with inboard flaps extended, since the flap on one wing will be immersed in the slipstream more than the flap on the other wing. If the flap hinge line is swept forward, the contribution toward negative dihedral effect is increased (Figure 5.14).

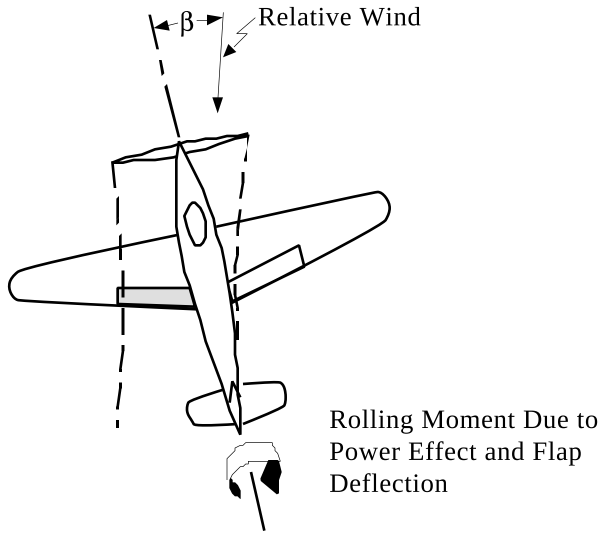


Figure 5.14
Possible Influence of Power and Flap Deflection on Dihedral Effect

5.2.1.3.2 Rolling Moment Due to Yaw Rate, L_r *or* C_{ℓ_r}

A rolling moment is generated if the airplane is yawing; this is the "roll due to yaw rate," L_r, or in nondimensional derivative form, C_{ℓ_r}. This rolling moment is composed of contributions from the vertical tail and the wing. When the airplane is yawing, a sideforce is developed by the vertical tail (see Figure 5.6). A positive (right) yaw rate, r, generates a positive (right) rolling moment contribution from the vertical tail. The left wing of an airplane yawing to the right advances into the air slightly more rapidly than the right wing. As a result, the left wing develops slightly more lift, which generates a rolling moment to the right. This rolling moment is proportional to lift coefficient, C_L, and the magnitude of this contribution increases with decreasing speed. A positive yaw rate generates a positive rolling moment contribution from the wing. Thus, both the vertical tail and wing contribute positively to the rolling moment due to yaw rate. Therefore, L_r and C_{ℓ_r} carry positive signs.

5.2.1.3.3 Rolling Moment Due to Rudder Deflection, L_{δ_r} *or* $C_{\ell_{\delta_r}}$

A deflection of the rudder produces a sideforce as shown earlier in Figure 5.7. This sideforce generates a rolling moment which may be expressed as follows:

$$L_{\delta_r} = Y_{\delta_r} Z_v \qquad \textit{eq 5.6}$$

or in nondimensional derivative form:

$$C_{\ell_{\delta_r}} = C_{y_{\delta_r}} \frac{Z_v}{b} \qquad \textit{eq 5.7}$$

As shown in Figure 5.15, a positive rudder deflection generates a positive rolling moment, therefore, L_{δ_r} and $C_{\ell_{\delta_r}}$ carry positive signs. If the airplane has a very high vertical tail, this derivative can be substantial.

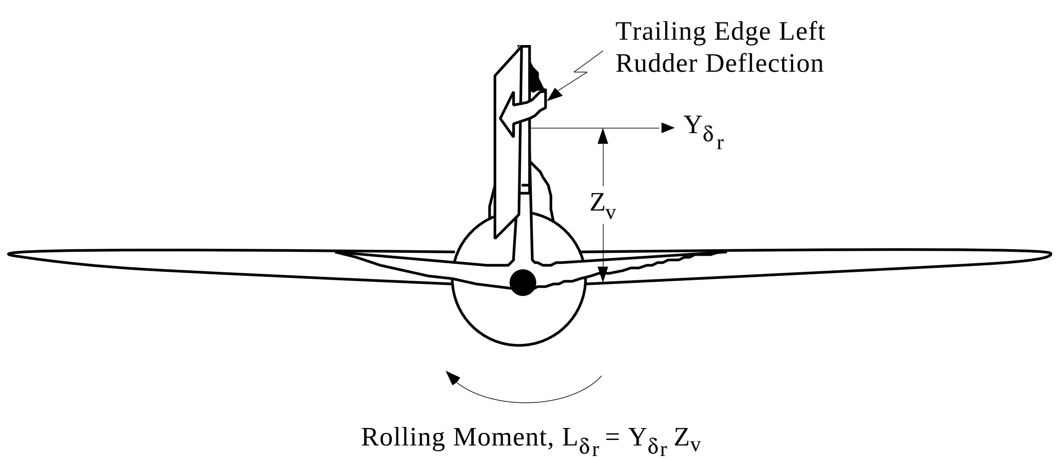


Figure 5.15
Rolling Moment Generated by Rudder Deflection

5.2.1.3.4 Rolling Moment Due to Lateral Control Deflection, L_{δ_a} *or* $C_{\ell_{\delta_a}}$

From the previous discussion, it can be argued that the pilot can maintain control over bank angle by use of simple rudder control. However, this method is not practical because of the lack of precision of control and, in many instances, lack of any bank angle control at all by use of the rudder. Therefore, the major means of controlling rolling moments and bank angle is through the use of lateral control devices - ailerons or spoilers or a combination of the two. The ailerons on each wing deflect asymmetrically, thereby so altering the spanwise load distribution that a rolling moment is created (Figure 5.16). Wing spoilers act in a similar manner.

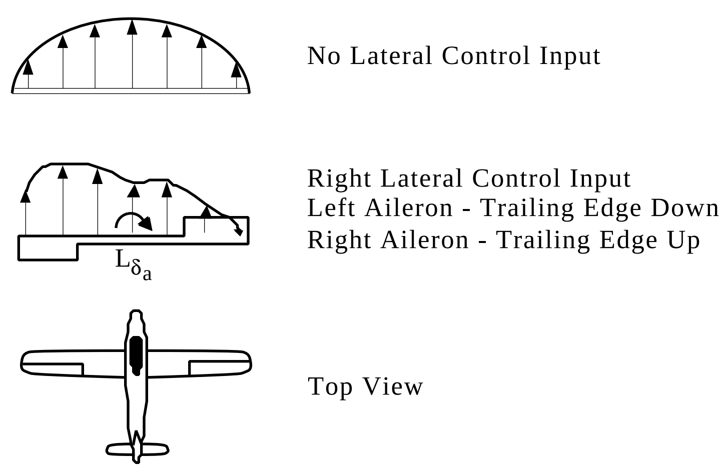


Figure 5.16
Generation of Rolling Moment from Aileron Deflection

The magnitude of the rolling moment due to aileron deflection, L_{δ_a}, or the lateral control power coefficient, $C_{\ell_{\delta_a}}$, may be evaluated by the strip integration method (Figure 5.17) and expressed as follows:

$$L_{\delta_a} = -2a_w \, \tau_a \, q_a \int_{y_1}^{y_2} c \, y \, dy \qquad \textit{eq 5.8}$$

$$C_{\ell_{\delta_a}} = \frac{-\,2a_w\ \tau_a\ \eta_a}{S_w\ b} \int_{y_1}^{y_2} c\,y\,dy \qquad \textit{eq 5.9}$$

Where:

τ_a = rate of change of effective wing angle of attack with aileron deflection.

δ_a = aileron deflection, right wing trailing edge down considered positive.

c = wing local chord.

η_a = aileron efficiency factor, $\frac{q_a}{q}$ where q is freestream dynamic pressure.

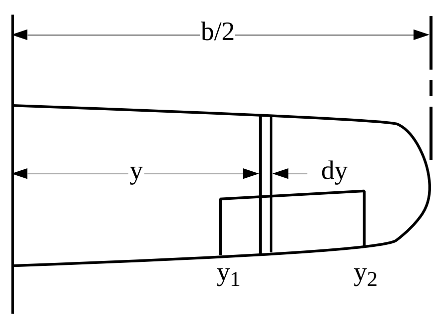


Figure 5.17
Strip Integration for L_{δ_α} and $C_{L_{\delta_\alpha}}$

The sign convention used here is such that right wing trailing edge down aileron is considered positive aileron deflection, and this aileron deflection generates a rolling moment in the negative (left) direction, L_{δ_a} and $C_{\ell_{\delta_a}}$ normally carry negative signs.

5.2.1.3.5 Rolling Moment Due to Roll Rate, Roll Damping, L_p *or* C_{ℓ_p}

As the airplane rolls, a modified wing lift distribution will be created that is a function of the rolling velocity, p. On the downgoing wing, the effective angle of attack is increased, on the upgoing wing, the effective angle of attack is decreased. The resultant changes in lift distribution generate a rolling moment, L_p, which opposes the rolling moment due to aileron deflection, L_{δ_a} (Figure 5.18). In rolling maneuvers, it is this opposing rolling moment, L_p, which determines the final steady state roll rate for a given aileron deflection. Therefore, L_p and the nondimensional derivative form C_{ℓ_p} are commonly referred to as "roll damping" or "damping in roll". It can readily be seen that roll damping depends almost exclusively on wing design parameters such as lift curve slope and wing span, however, a high vertical tail can have some contribution. Since roll damping usually acts to oppose the rolling moment generated by lateral control deflection, L_p and C_{ℓ_p} generally carry negative signs. If the airplane is flying at low airspeeds near stall, an increased angle of attack on the downgoing wing may stall that wing. This causes C_{ℓ_p} to become positive and the wing will "autorotate"[1] unless corrective action is taken by the pilot.

---

[1] Uncontrolled rolling motion, as in a spin.

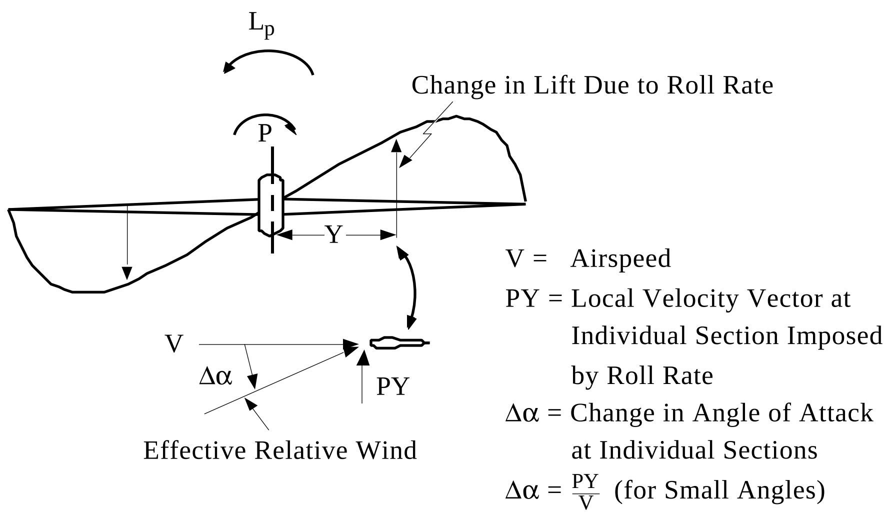


Figure 5.18
Generation of Roll Damping

5.2.1.4 LATERAL-DIRECTIONAL STABILITY DERIVATIVES - YAWING MOMENTS AND DIRECTIONAL STABILITY DERIVATIVES.

5.2.1.4.1 Yawing Moment Due to Sideslip, Directional Stability, N_β *or* C_{n_β}

Whenever a sideslip angle is imposed on the airplane, yawing moments are generally developed as a result of the sideslip. If the yawing moments tend to reduce the imposed sideslip angle, the airplane exhibits directional stability or "weathercock stability". The strength of this directional stability is one of the most important characteristics of the airplane. The main contributions to this characteristic come from the fuselage, nacelles, and the vertical tail. The wing contribution is usually negligible since a sideslip angle creates only very small sideforces on the wing. However, the contributions of the fuselage and nacelles to the yawing moment due to sideslip are usually significant and unstable. The vertical tail must be designed to offset the destabilizing fuselage-nacelle contribution and provide some level of directional stability for the airplane. The contribution of the vertical tail to the yawing moment due to sideslip is usually very powerful and is generated as a result of the sideforce created by the vertical tail (Figure 5.19). This contribution N_{β_v} may be expressed as:

$$N_{\beta_v} = -Y_{\beta_v}\ \ell_v = a_v\left(1 - \frac{d\sigma}{d\beta}\right) q_v S_v \ell_v \qquad \textit{eq 5.10}$$

or in nondimensional derivative form:

$$C_{n_{\beta_V}} = -C_{y_{\beta_v}}\ \frac{\ell_v}{b} = a_v\left(1 - \frac{d\sigma}{d\beta}\right) \eta_v\ \frac{S_v}{S_w}\ \frac{\ell_v}{b} \qquad \textit{eq 5.11}$$

Where:

ℓ_v = horizontal distance between center of pressure of the vertical tail and the airplane center of gravity in feet.

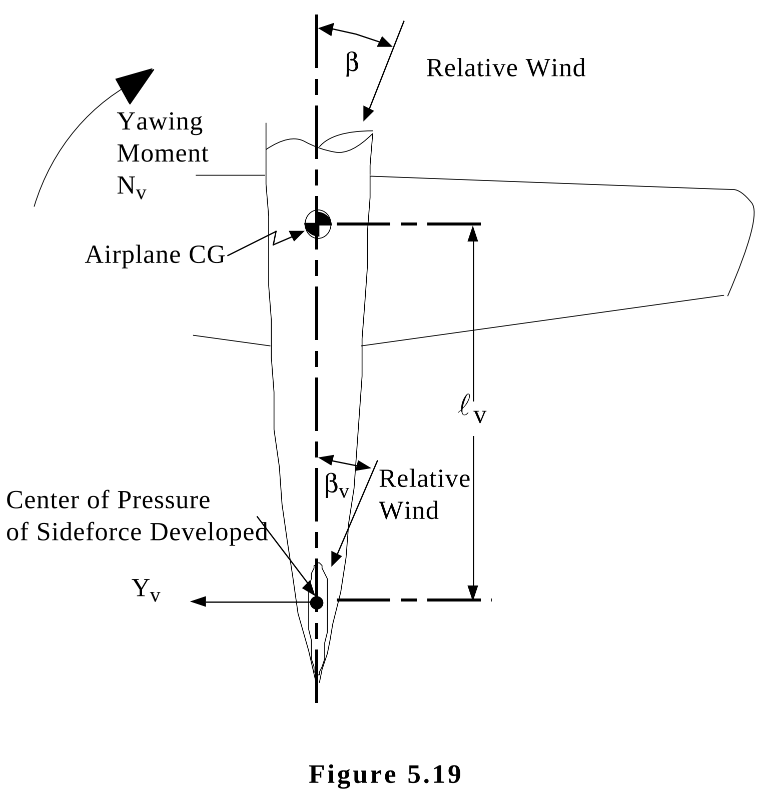


Figure 5.19
Contribution of the Vertical Tail to Directional Stability

The airplane's total yawing moment due to sideslip, or directional stability, is the sum of the contributions from the wing, fuselage, nacelles, and vertical tail:

$$C_{n_\beta} = C_{n_{\beta_{W,\,F,\,N}}} + C_{n_{\beta_V}} \qquad \textit{eq 5.12}$$

If C_{n_β} is positive, i.e., positive sideslip generates a positive yawing moment, the airplane is directionally stable. Obviously, N_β and C_{n_β} normally carry positive signs.

The running propeller can have large effects on the airplane directional stability. A sideforce is generated as a result of the airflow passing through the propeller disc at a sideslip angle. This effect is sometimes called the "propeller fin effect." From a study of Figure 5.20, it may be rationalized that the effect of propeller operation on directional stability depends on the location of the propeller with respect to the airplane CG. If the propeller is positioned ahead of the CG, propeller effects are destabilizing. The influence of jet engine operation on directional stability is the same.

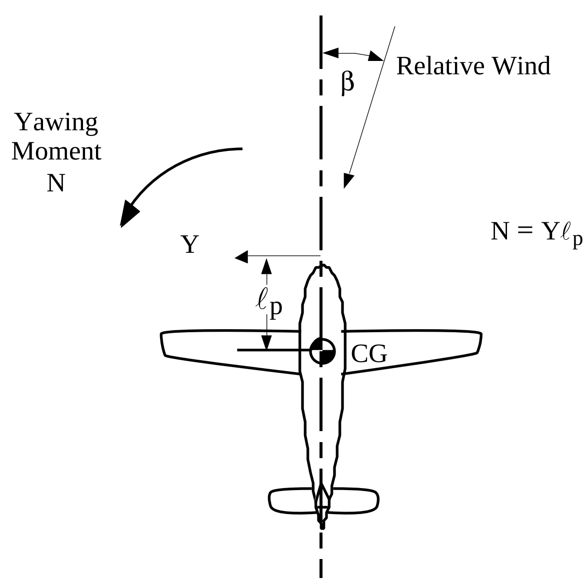


Figure 5.20
Propeller Power Influence on Directional Stability

5.2.1.4.2 Yawing Moment Due to Yaw Rate Damping, N_r *or* C_{n_r}

The sideforces created when the airplane is yawing about its CG with some yaw rate, r, generate yawing moments which generally tend to oppose the motion. This "yaw rate damping" or "damping in yaw" derives most of its magnitude from the vertical tail. The vertical tail is subjected to a sideslip angle, β_V, due to yaw rate even if the airplane sideslip angle is zero[1] (see Figure 5.6). The resultant sideforce generates a yawing moment, N, which opposes the yawing motion (Figure 5.21). The magnitude of the yaw rate damping contribution from the vertical tail may be expressed as follows:

$$N_{r_V} = -a_V \; S_V \; q_V \; \frac{\ell_V^2}{V} \qquad \textit{eq 5.13}$$

or in nondimensional derivative form:

$$C_{n_{r_V}} = \frac{\partial C_n}{\partial\left(\frac{rb}{2V}\right)} = -2a_V \; \frac{S_V}{S_w} \; \eta_V \; \frac{\ell_V^2}{b^2} \qquad \textit{eq 5.14}$$

Where:

$\frac{rb}{2V}$ is called the "nondimensional yaw rate"

V = true airspeed

(Note that vertical tail "arm length," ℓ_V, has a powerful influence on the magnitude of the yaw rate damping.)

Yaw rate damping is increased through a wing contribution particularly at high lift coefficients. This contribution arises because the "outer" wing is traveling forward at a slightly higher airspeed than the wing to the inside of the turn. The outer wing therefore develops more profile and induced drag, which generates an additional yawing moment opposing the yaw rate.

---

[1] This situation is approached in a perfectly coordinated turn.

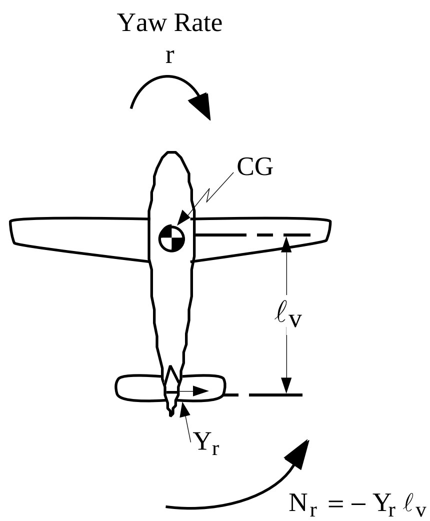


Figure 5.21
Generation of Yaw Rate Damping from the Vertical Tail

Yaw rate damping, N_r, or C_{n_r}, normally are negative quantities since their action is in opposition to the established yaw rate.

5.2.1.4.3 Yawing Moment Due to Rudder Deflection, N_{δ_r} or $C_{n_{\delta_r}}$

The sideforce generated at the vertical tail due to rudder deflection produces a yawing moment about the airplane CG (Figure 5.22). The magnitude of this yawing moment is a measure of the rudder effectiveness or "rudder control power". The yawing moment due to rudder deflection may be stated as:

$$N_{\delta_r} = -a_v \ \tau_v \ q_v \ S_v \ \ell_v \qquad \textit{eq 5.15}$$

or in nondimensional coefficient form:

$$C_{n_{\delta_r}} = -a_v \ \tau_v \ n_v \ \frac{S_v}{S_w} \ \frac{\ell_v}{b} \qquad \textit{eq 5.16}$$

N_{δ_r} and $C_{n_{\delta_r}}$ normally are negative quantities since a positive (trailing edge left) rudder deflection generates a negative yawing moment.

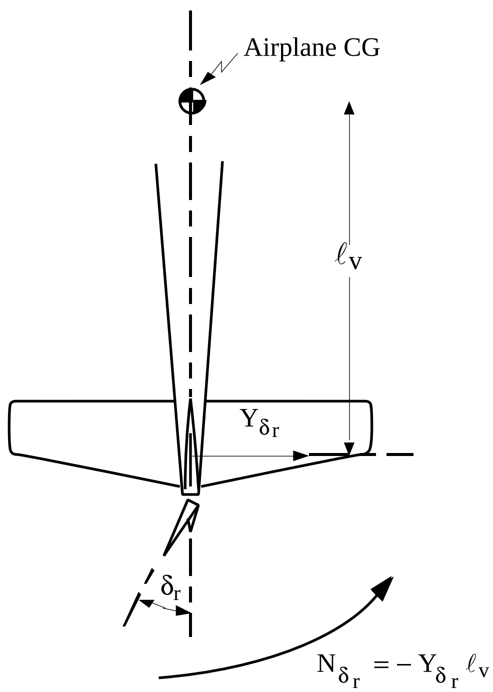


Figure 5.22
Yawing Moment Due to Rudder Deflection-Rudder Power

5.2.1.4.4 Yawing Moment Due to Lateral Control Deflection, N_{δ_a} *or* $C_{n_{\delta_a}}$

The lateral control surfaces - ailerons and/or spoilers - usually generate yawing moments when deflected. If the yawing moments created act so as to rotate the nose of the airplane opposite to the direction of roll, the yawing moment is termed "adverse." If the yawing moments act so as to rotate the nose in the same direction as the roll, the yawing moment is termed "proverse." (These terms do not, in themselves, devote unfavorable or favorable flying qualities. In some cases, adverse yaw contributes more to good flying qualities than proverse yaw!) For airplanes equipped with ailerons, the yawing moments created with aileron deflection are usually <u>adverse</u>. This is due to an increase in drag due to the increase in lift on the wing with the aileron trailing edge down and vice versa (Figure 5.23). The yawing moment due to aileron deflection, N_{δ_a} or, in nondimensional derivative form, $C_{n_{\delta_a}}$, will be positive if the yawing moment is adverse. (Right aileron trailing edge down is positive; left roll is generated; if yawing moment is adverse, a right (positive yawing moment results.)

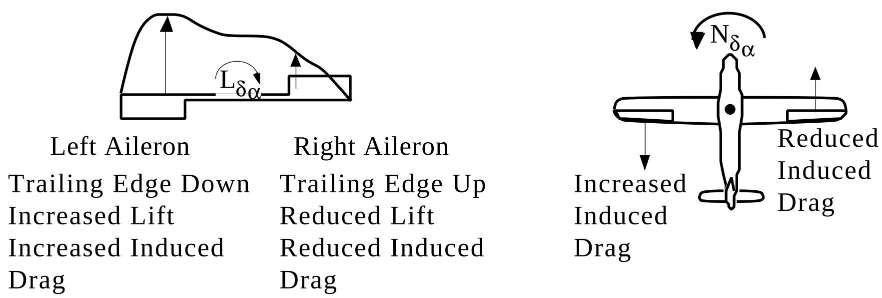


Figure 5.23
Generation of Adverse Aileron Yawing Moment

For airplanes equipped with spoiler type lateral control devices, the yawing moment due to lateral control deflections may be proverse or adverse. This is because spoilers generate changes in profile drag as well as induced drag. If the changes due to profile drag are predominant, the yawing moments due to spoiler deflection are proverse; if changes due to induced drag predominate, the yawing moments are adverse (Figure 5.24). Therefore, at high airspeeds, spoilers tend to generate proverse (negative) yawing moments, while at low airspeeds, spoilers tend to generate adverse (positive) yawing moments.

For airplanes equipped with differential horizontal stabilizers, the yawing movements due to lateral control deflections are usually proverse. This is because the intensified pressure field generated above the stabilizer on the downgoing side is also felt by the vertical tail.

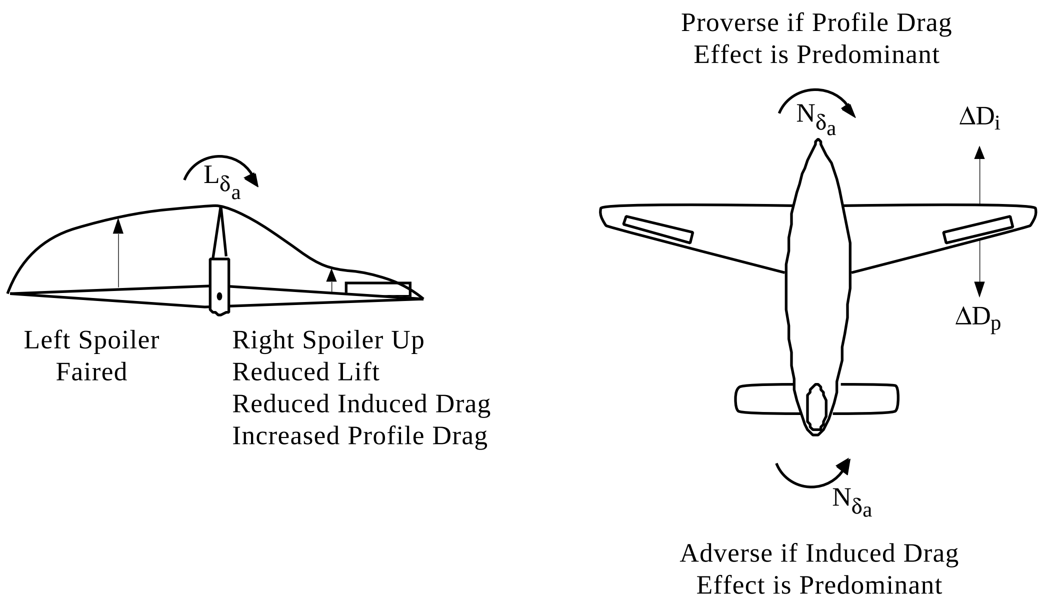


Figure 5.24
Yawing Moments Generated by Spoiler Deflection

5.2.1.4.5 Yawing Moment Due to Rolling Velocity, N_p *or* C_{n_p}

Yawing moments are also generated by rolling velocity. The main contributing factor to this yawing moment, N_p, or in nondimensional derivative form, C_{n_p} is the wing, although large span horizontal and vertical tails can have appreciable effects. The wing contribution is due to the changes in effective angles of attack on the downgoing and upgoing wing during rolling (see Figure 5.18). These changes result in tilting of the lift and drag vectors on the respective wings (along with changes in the magnitudes as well). As shown in Figure 5.25, the modifications to the lift vectors generally are predominant, resulting in a yawing moment which tends to yaw the airplane opposite to the direction of roll. The influence of the horizontal tail would be the same. The sideforces generated at the vertical tail due to roll rate (see Figure 5.5) would result in a yawing moment which tends to yaw the airplane in the direction of roll, which is exactly opposite to the wing effect. The total yawing moment due to roll rate is the sum of the contributions of the wing, horizontal tail, and vertical tail. Since the wing contribution is normally predominant, N_p and C_{n_p} are normally negative quantities, since a right (positive) roll rate would result in a left (negative) yawing moment due to the roll rate.

The influence of the yawing moment due to roll rate is frequently incorrectly included in the effects of aileron adverse or proverse yaw, $C_{n_{\delta_a}}$. This is due to the fact that, in a rolling maneuver, the yawing moments generated are a result of both $C_{n_{\delta_a}}$ and C_{n_p}. It is never possible to completely separate the two during flight test investigations. The test pilot and engineer should be careful to use correct technical terminology when describing the phenomenon observed during flight tests.

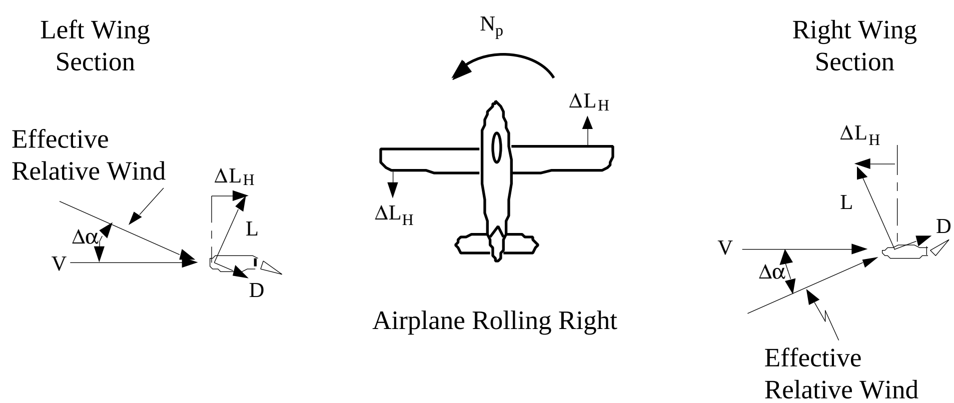


Figure 5.25
Wing Contribution to Yawing Moment Due to Roll Rate

A tabular presentation of the lateral-directional stability derivatives is presented in Figure 5.26.

Force or Moment	Symbol	Non-Dimensional Derivative Form	Normal Signs	Prime Contributing Factors	Remarks
Sideforce Due to Sideslip	Y_β	C_{y_β}	-	Vertical Tail, Fuselage	Contributes to Positive Dihedral Effect, Positive Directional Stability
Sideforce Due to Roll Rate	Y_p	C_{y_p}	-	Vertical Tail	Contributes to Roll Damping, Yawing Moment Due to Roll Rate
Sideforce Due to Yaw Rate	Y_r	C_{y_r}	+	Vertical Tail	Contributes to Roll Due to Yaw Rate, Yaw Rate Damping
Sideforce Due to Rudder Deflection	Y_{δ_r}	$C_{y_{\delta_r}}$	+	Vertical Tail-Rudder Combination	Contributes to Roll Due to Rudder, Rudder Control Power
Sideforce Due to Gravity	$W \sin \phi$	$C_L \phi$	±	Whole Airplane	Generates No Rolling or Yawing Moments
Sideforce Due to Lateral Control Deflection	Y_{δ_a}	$C_{y_{\delta_a}}$	None	Aileron, Spoiler Location	Usually Neglected
Rolling Moment Due to Sideslip-Dihedral Effect	L_β	C_{ℓ_β}	-	Wing Design, Although a High Vertical Tail and Power (Recip) Can Influence	"Positive" Dihedral Effect Corresponds to a Negative Stability Derivative
Rolling Moment Due to Yaw Rate	L_r	C_{ℓ_r}	+	Wing, Vertical Tail	Magnitude of Wing Contribution Increases with Decreasing Airspeed
Rolling Moment Due to Rudder Deflection	L_{δ_r}	$C_{\ell_{\delta_r}}$	+	Vertical Tail-Rudder Combination	Of Significant Magnitude if Airplane has a High Vertical Tail
Rolling Moment Due to Lateral Control Deflection, Lateral Control Power	L_{δ_a}	$C_{\ell_{\delta_a}}$	-	Aileron, Spoiler	
Rolling Moment Due to Roll Rate-Roll Damping	L_p	C_{ℓ_p}	-	Wing, Although a High Vertical Tail Can Influence	L_{δ_a} and L_p Determine the Steady State Roll Rate for a Given Lateral Control Deflection

Figure 5.26
Lateral-Directional Stability Derivatives Summary

Yawing Moment Due Sideslip-Directional Stability	N_{β}	$C_{n_{\beta}}$	+	Vertical Tail, Fuselage, Nacelles	Vertical Tail Must be Designed to Offset Destabilizing Influence of Fuselage-Nacelle Combination
Yawing Moment Due to Yaw Rate-Yaw Rate Damping	N_r	C_{n_r}	-	Vertical Tail, Although the Wing Has Some Influence	Wing Contribution is Largest at High Angles of Attack
Yawing Moment Due to Rudder Deflection-Rudder Control Power	N_{δ_r}	$C_{n_{\delta_r}}$	-	Vertical Tail-Rudder Combination	
Yawing Moment Due to Lateral Control Deflection-Adverse or Proverse Yaw	N_{δ_a}	$C_{n_{\delta_a}}$	±	Lateral Control Device Design	Ailerons Generally Produce Adverse Yaw. Spoilers May Generate Adverse or Proverse Yaw.
Yawing Moment Due to Rate of Roll	N_p	C_{n_p}	-	Wing, Although a High Vertical Tail Can Have Some Influence	Sometimes Incorrectly Included in Aileron Adverse or Proverse Yaw.

Figure 5.26
Lateral-Directional Stability Derivatives Summary (Cont'd)

5.2.1.5 STEADY HEADING SIDESLIPS

The steady heading sideslip is a common maneuver to all pilots. It is sometimes utilized to maintain airplane heading equal to runway heading during crosswind landings, or to increase drag and steepen the glide path in light airplanes. However, the primary reason for studying steady heading sideslips is not to determine the feasibility of utilizing the maneuver in normal operations, although this determination is obviously a by-product of the study. The steady heading slideslip requires the pilot to balance the forces and moments generated on the airplane by the sideslip with appropriate lateral and directional control inputs and bank angle. Since these control forces and positions and bank angles are at least indicative of the sign (if not the magnitude) of the generated forces and moments (and therefore of the associated stability derivative), the steady heading sideslip is a convenient flight test technique.

The equilibrium equation for the steady heading sideslip will not be presented in terms of the stability derivatives. Since the steady heading sideslip results in zero yaw rate and zero roll rate, all force and moment coefficients due to yaw rate or roll rate are eliminated from the equations. Also, the sideforce due to aileron deflection, $C_{y_{\delta_a}}$, will be neglected, since it is usually very small. Thus, the steady heading sideslip equations for sideforce, yawing moment, and rolling moment may be written as follows:

<u>SIDEFORCE</u> $$C_{y_0} + C_{y_\beta}\,\beta + C_{y_{\delta_r}}\,\delta_r + C_L\,\phi = 0 \qquad \textit{eq 5.17}$$

<u>YAWING MOMENT</u> $$C_{n_0} + C_{n_\beta}\,\beta + C_{n_{\delta_r}}\,\delta_r + C_{n_{\delta_a}}\,\delta_a = 0 \qquad \textit{eq 5.18}$$

<u>ROLLING MOMENT</u> $$C_{\ell_0} + C_{\ell_\beta}\,\beta + C_{\ell_{\delta_r}}\,\delta_r + C_{\ell_{\delta_a}}\,\delta_a = 0 \qquad \textit{eq 5.19}$$

Where:

C_{y_0}, C_{n_0}, and C_{ℓ_0} are sideforce yawing moment, and rolling moment derivatives when rudder deflection, aileron deflection, bank angle, and sideslip angle are zero and are <u>constants</u>. These constants could result from asymmetric configurations, loadings, etc.

The variables in the steady sideslip equations can only be bank angle, rudder position, aileron position, and sideslip angle, By differentiating the last equation with respect to sideslip angle, the constants can be eliminated and useful relationships for the variation of rudder position, aileron position, and bank angle with sideslip angle in steady heading flight can be developed. Differentiation yields:

<u>SIDEFORCE</u> $$C_{y_\beta} + C_{y_{\delta_r}}\frac{d\delta_r}{d\beta} + C_L\frac{d\phi}{d\beta} = 0 \qquad \textit{eq 5.20}$$

<u>YAWING MOMENT</u> $$C_{n_\beta} + C_{n_{\delta_r}}\frac{d\delta_r}{d\beta} + C_{n_{\delta_a}}\frac{d\delta_a}{d\beta} = 0 \qquad \textit{eq 5.21}$$

<u>ROLLING MOMENT</u> $$C_{\ell_\beta} + C_{\ell_{\delta_r}}\frac{d\delta_r}{d\beta} + C_{\ell_{\delta_a}}\frac{d\delta_a}{d\beta} = 0 \qquad \textit{eq 5.22}$$

Solving the last three equations for the variables $\frac{d\delta_r}{d\beta}$, $\frac{d\delta_a}{d\beta}$, and $\frac{d\phi}{d\beta}$ yields the following important classical relationships† which will be analyzed in turn:

$$\frac{d\delta_r}{d\beta} = \frac{-\frac{C_{n_\beta}}{C_{n_{\delta_r}}}\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\frac{C_{\ell_\beta}}{C_{n_\beta}}\right\}}{\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}} \qquad \textit{eq 5.23}$$

$$\frac{d\delta_a}{d\beta} = \frac{-\frac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}}\left\{1 - \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\frac{C_{n_\beta}}{C_{\ell_\beta}}\right\}}{\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}} \qquad \textit{eq 5.24}$$

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} + C_{y_{\delta_r}}\frac{d\delta_r}{d\beta}\right\} \qquad \textit{eq 5.25}$$

5.2.1.5.1 Rudder Position Required in Sideslip

The variation of rudder position with sideslip angle reduces to the following simple relationship if $C_{n_{\delta_a}}$ is zero:

$$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}} \qquad \textit{eq 5.26}$$

Thus, a positive variation of rudder position with sideslip angle indicates <u>positive directional stability</u> since $C_{n_{\delta_r}}$ is invariably a negative quantity. Obviously, $\frac{d\delta_r}{d\beta}$ yields absolutely no information as to the magnitude of the directional stability C_{n_β}, unless the magnitude of the derivative, $C_{n_{\delta_r}}$, is known.

† Several mathematical manipulations have been omitted.

Even if $C_{n_{\delta_a}}$ is not zero, the gradient $\frac{d\delta_r}{d\beta}$ still has a major influence on the pilot's opinion of lateral-directional flying qualities. The denominator:

$$\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\} \qquad \textit{eq 5.27}$$

can usually be considered to be one and can be neglected in analysis.

However, the numerator:

$$1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \frac{C_{\ell_\beta}}{C_{n_\beta}} \qquad \textit{eq 5.28}$$

can have a significant influence on the rudder positions required in sideslips. For example, a swept wing airplane in approach configuration can have a ratio of dihedral effect to directional stability $\left(\frac{C_{\ell_\beta}}{C_{n_\beta}}\right)$ as high as three. If these airplanes exhibit significant aileron adverse yaw, $C_{n_{\delta_a}}$, in this flight condition, the gradient $\frac{d\delta_r}{d\beta}$ may be reduced drastically even though directional stability, C_{n_β}, may still be strong. Thus, adverse aileron yaw and high dihedral effect can combine to reduce significantly the pilot's opinion of the directional "stiffness" of the airplane.

Obviously, some level of positive rudder position variation with sideslip angle (Figure 5.27) is required in any airplane. This variation is sometimes referred to as <u>rudder position directional stability</u> or <u>directional stability, rudder-fixed</u>.

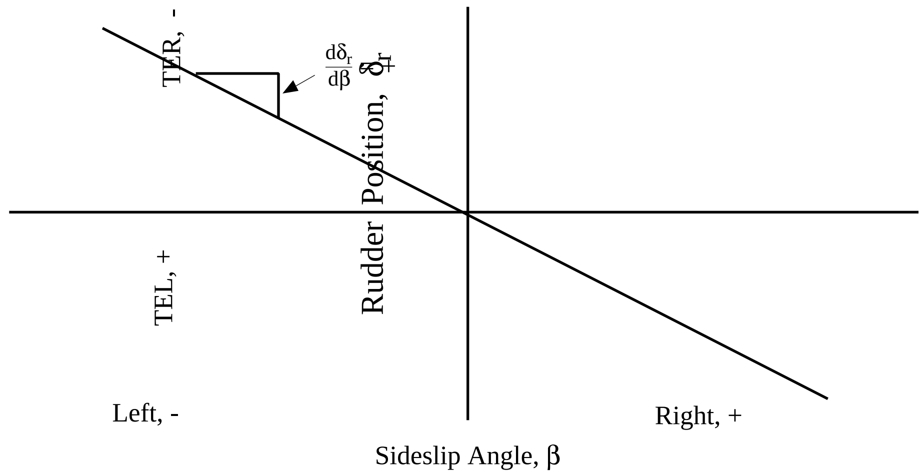


Figure 5.27
Rudder Position Required in Steady Heading Sideslips

5.2.1.5.2 Rudder Forces Required in Sideslips

Of great importance to the pilot is the variation of rudder pedal forces required in steady heading sideslips. Rudder forces are generated by the requirement for the pilot to move the rudder to the position for equilibrium. If the directional control system is irreversible, rudder forces are merely a function of rudder position, i.e.:

$$F_r = K_1 \Delta\delta_r \text{ (linear feel spring system)} \qquad \textit{eq 5.29}$$

or:

$$F_r = K_2 q \Delta\delta_r \text{ ("q – feel" system)} \qquad \textit{eq 5.30}$$

Where:

K_1 and K_2 are constants describing the characteristics of the system, such as strength of the feel spring, gearing ratio, etc.

However, if the directional control system is reversible, the rudder is free to float in response to hinge moments developed. This floating tendency can have a large influence on the directional stability of the airplane. If the rudder floats so as to align itself with the relative wind at the vertical tail, the restoring moment generated at the vertical tail, $C_{n_{\beta_v}}$

(and therefore C_{n_β}), will be decreased (Figure 5.28). Conversely, if the rudder floats opposite to the relative wind, C_{n_β} will be increased. The floating angle of the rudder can be expressed analytically as follows:

$$\delta_{r_{Float}} = -\frac{C_{h_{\beta_V}}}{C_{h_{\delta_r}}}\beta_V \qquad \textit{eq 5.31}$$

Where:

$C_{h_{\beta_V}}$ = rudder hinge moment coefficient variation with sideslip angle at zero deflection, normally carries a positive sign.

$C_{h_{\delta_r}}$ = rudder hinge moment coefficient variation with rudder deflection at zero sideslip angle, and carries a negative sign.

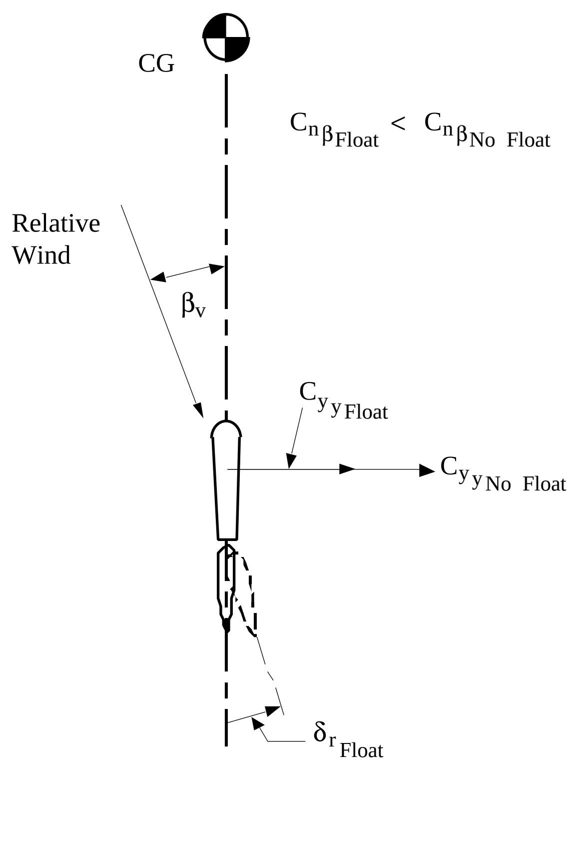


Figure 5.28
The Vertical Tail Contribution to C_{n_β} May be Reduced by Rudder Float

Since the pilot must apply rudder forces to move the rudder from its float position to the position for equilibrium, the analytical expression for rudder forces in the reversible system is as follows:

$$F_r = -K\, C_{h_{\delta_r}}\, q_v S_r \bar{c}_r \left\{ \delta_{r_{\text{Equilibrium}}} - \delta_{r_{\text{Float}}} \right\} \qquad \textit{eq 5.32}$$

Where:

K = a constant describing the characteristics of the system, radians per foot.

q_v = dynamic pressure at vertical tail, pounds per square foot.

S_r = area of the rudder, square feet.

$\bar{c}_r$ = average rudder chord, feet.

The variation of rudder forces with sideslip angle is obtained by differentiating the last equation with respect to sideslip:

$$\frac{dF_r}{d\beta} = -K\, C_{h_{\delta_r}}\, q_v S_r \bar{c}_r \left\{ \frac{d\delta_r}{d\beta} - \frac{d\delta_{r_{\text{Float}}}}{d\beta} \right\} \qquad \textit{eq 5.33}$$

Now since:

$$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}} \text{ (if the assumption } C_{n_{\delta_a}} = 0 \text{ is valid)} \qquad \textit{eq 5.34}$$

and

$$\frac{d\delta_{r_{\text{Float}}}}{d\beta} = -\frac{C_{h_{\beta_v}}}{C_{h_{\delta_r}}} \left(1 - \frac{d\sigma}{d\beta}\right) \qquad \textit{eq 5.35}$$

the gradient of rudder forces with sideslip angle for the reversible system may be expressed as[†] :

$$\frac{dF_r}{d\beta} = -K \frac{C_{h_{\delta_r}}}{C_{n_{\delta_r}}} q_v S_r \bar{c}_r \left\{ -C_{n_\beta} + \frac{C_{n_{\delta_r}} C_{h_{\beta_v}}}{C_{h_{\delta_r}}} \left(1 - \frac{d\sigma}{d\beta}\right) \right\} \qquad \textit{eq 5.36}$$

The expression enclosed in braces in the above equation is sometimes referred to as "directional stability, rudder-free." The destabilizing or stabilizing (depending on the sign of $C_{h_{\beta_v}}$) influence of rudder float is represented by the expression:

$$\frac{C_{n_{\delta_r}} C_{h_{\beta_v}}}{C_{h_{\delta_r}}} \left(1 - \frac{d\sigma}{d\beta}\right) \qquad \textit{eq 5.37}$$

Obviously, the pilot always desires some positive gradient[7] of rudder pedal force with sideslip angle (Figure 5.29). If the gradient is shallow or zero through zero sideslip, the pilot experiences difficulty in maintaining zero sideslip during maneuvering.   If the gradient is high, the airplane feels "stiff" directionally to the pilot and he is generally quite satisfied from this point of view.

[†] Several mathematical manipulations have been omitted.

[7] Left rudder pedal force is considered positive. For a positive rudder force gradient, left rudder pedal forces are required with right sideslip angles, and vice versa.

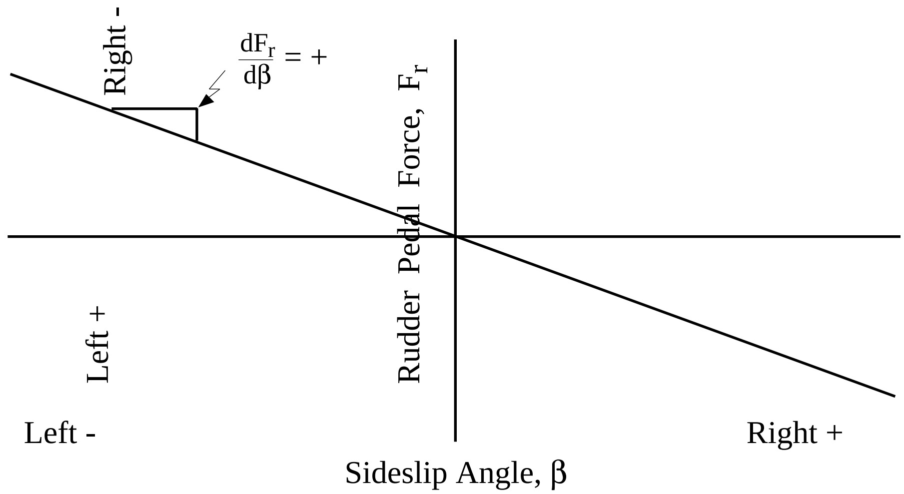


Figure 5.29
Rudder Forces Required in Steady Heading Sideslips

5.2.1.5.3 Rudder Lock

If the directional control system is reversible, the pilot may be confronted with nonlinearities in rudder pedal forces at high sideslip angles even though the rudder position variation is linear. This situation, depicted in Figure 5.30, is due to an increase in rudder float angle at large sideslip angles because of changes in rudder hinge moment characteristics. If the float angle equals the rudder angle required for equilibrium, the rudder forces required will be zero. Any increase in sideslip beyond this point results in the rudder floating all the way to the stops unless the pilot applies opposite rudder force. This situation, called "rudder lock," is obviously unsatisfactory if it is encountered in a flight condition representative of an operational situation. Rudder lock may also be generated by nonlinearities in the rudder position gradient with sideslip angle, i.e., a directional stability problem. This situation, depicted in Figure 5.31, results in the same rudder pedal force variation with sideslip angle but can be differentiated from the "float-induced rudder lock" because the rudder position gradient is nonlinear.

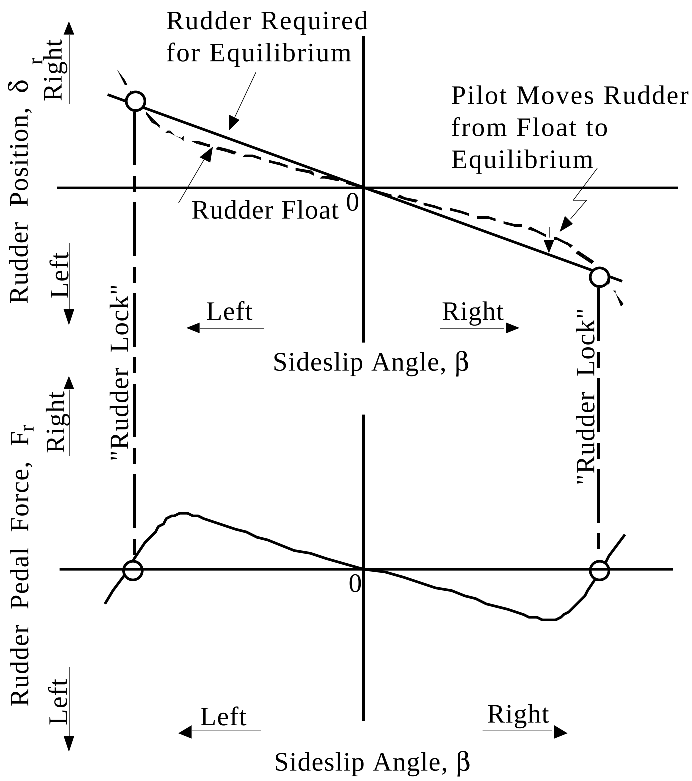


Figure 5.30
"Rudder Lock" Generated by Float Characteristics

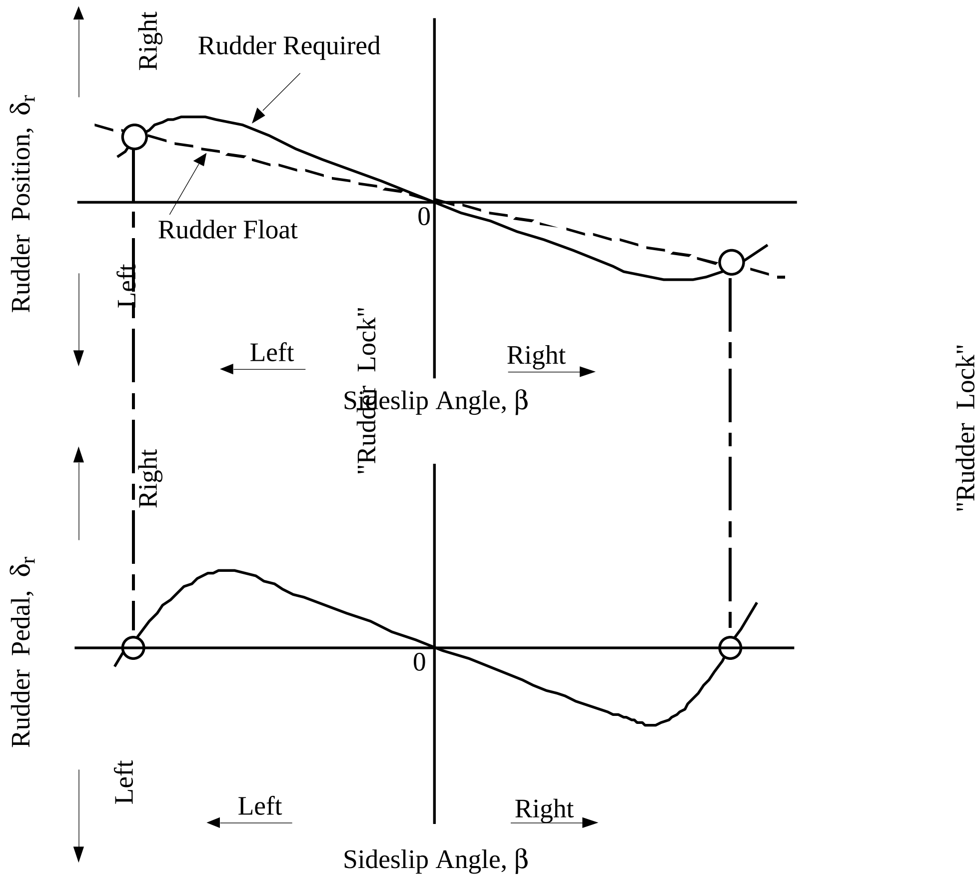


Figure 5.31
"Rudder Lock" Generated by Directional Stability Problem

5.2.1.5.4 Aileron Position Required for Sideslips

The relationship for aileron position for equilibrium in steady heading sideslips is rewritten for convenience:

$$\frac{d\delta_a}{d\beta} = \frac{-\frac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}}\left\{1-\frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\frac{C_{n_\beta}}{C_{\ell_\beta}}\right\}}{\left\{1-\frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}} \qquad \textit{eq 5.38}$$

If roll due to rudder deflection, $C_{\ell_{\delta_r}}$, is zero, the above relationship reduces to the following simple form:

$$\frac{d\delta_a}{d\beta} = -\frac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}} \qquad \textit{eq 5.39}$$

Thus, a variation of aileron position with sideslip angle such that $\frac{d\delta_a}{d\beta}$ is negative indicates positive dihedral effect since $C_{\ell_{\delta_a}}$ is invariably a negative quantity. Obviously, $\frac{d\delta_a}{d\beta}$ yields absolutely no information as to the magnitude of the dihedral effect, C_{ℓ_β}, unless the magnitude of the aileron control power, $C_{\ell_{\delta_a}}$, is known.

Even if $C_{\ell_{\delta_r}}$ is not zero, the gradient $\frac{d\delta_a}{d\beta}$ still has a major influence on the pilot's opinion of the lateral-directional flying qualities. Again, the denominator:

$$\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}} \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\} \qquad \textit{eq 5.40}$$

can usually be considered to be one and can be neglected in analysis. However, the numerator:

$$\left\{1 - \frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}} \frac{C_{n_\beta}}{C_{\ell_\beta}}\right\} \qquad \textit{eq 5.41}$$

can have a significant influence on the aileron positions required in sideslips. For example, a swept wing aircraft with a high vertical tail in supersonic flight can have a ratio of directional stability to dihedral effect $\frac{C_{n_\beta}}{C_{\ell_\beta}}$ as high as two. The ratio of roll due to rudder to rudder control power $\frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}$ is equal to the ratio of vertical tail height (Z_V) to vertical tail length (ℓ_V), which may be about one-half for a high vertical tail. Therefore, in high speed flight, the rudder can provide a significant amount of the rolling moment required for equilibrium in steady heading sideslips.

The steady heading sideslip test conducted under these conditions would yield little useful information as to the ratio of dihedral effect to aileron control power. However, the results of the test would still be very important from a flying qualities standpoint, particularly in the analysis of lateral-directional trimmability and the ease of controlling bank angle with rudder.

The desired magnitude of aileron position variation with sideslip angle has never been clearly determined. Analysis of many flying qualities investigations reveals that, in general, pilots prefer <u>some</u> positive gradient such that right lateral control position is required in right sideslips[8] , and vice versa (Figure 5.32). However, if the gradient is <u>too</u> steep, lateral-directional flying qualities may be degraded also. This variation is sometimes called <u>control-fixed dihedral effect</u> or <u>dihedral effect (aileron position)</u>.

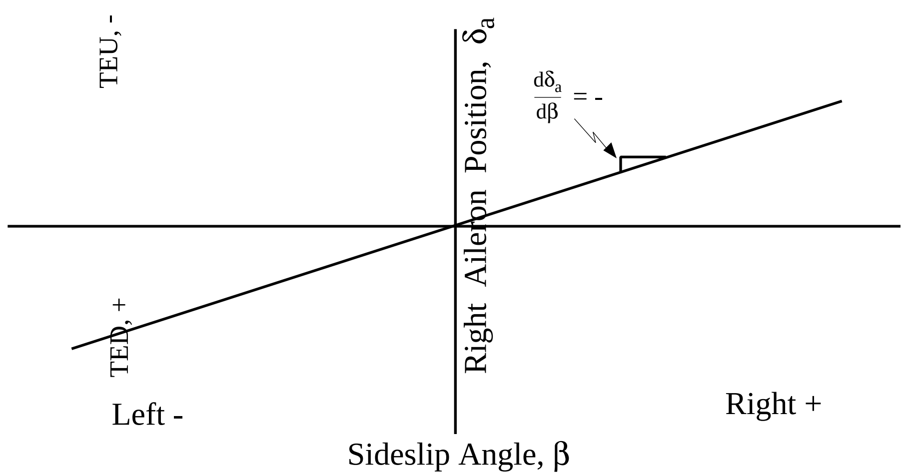


Figure 5.32
Aileron Position Required in Sideslips

---

[8] This results in $\frac{d\delta_a}{d\beta}$ being a negative quantity since right (positive) sideslip would require right aileron trailing edge up (negative) deflection.

5.2.1.5.5 Aileron Forces Required in Sideslips

Aileron forces in the steady heading sideslip are generated by the requirement for the pilot to move the aileron to the position required for equilibrium. If the lateral control system is irreversible, aileron control forces are merely a function of aileron position. If the lateral control system is reversible, the aileron is free to float in response to hinge moments developed. However, aileron float is generally very small during steady heading sideslips. In some cases, aileron float has been observed in airplanes with high geometric dihedral[9] such that a positive gradient of aileron position is exhibited with a flat, or zero, gradient of aileron control force with sideslip. Some positive gradient of aileron control force with sideslip angle is desired such that right lateral control forces are required in right sideslips, and vice versa. This results in $\frac{dF_a}{d\beta}$ being a negative quantity (Figure 5.33). The variation of lateral control forces with sideslip angle is sometimes referred to as control-free dihedral effect or dihedral effect (aileron force). Lateral control forces will be discussed more thoroughly in the section on Rolling Performance.

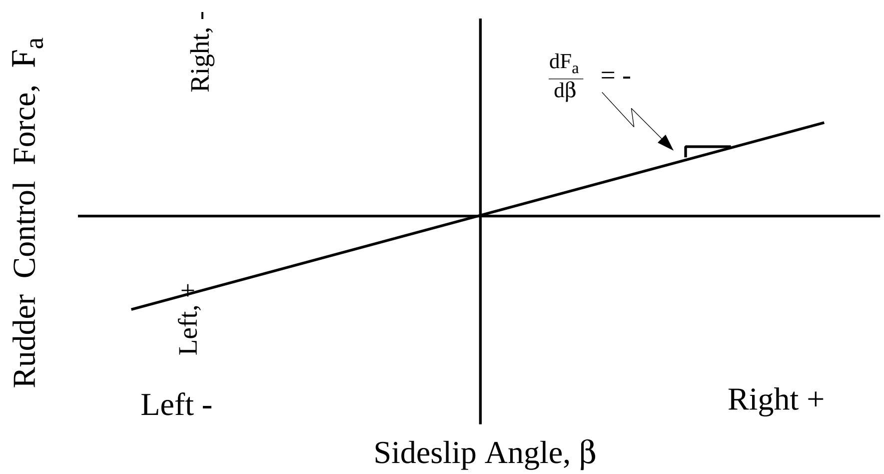


Figure 5.33
Lateral Control Forces Required in Steady Heading Sideslips

[9]Aileron float in steady sideslips appears to increase with geometric dihedral and sweepback, although reversible lateral control systems are seldom utilized on airplanes with high sweepback.

5.2.1.5.6 Bank Angle Required in Sideslips

The classical relationship for bank angle variation with sideslip angle, developed earlier, is as follows:

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} + C_{y_{\delta_r}}\frac{d\delta_r}{d\beta}\right\} \qquad \textit{eq 5.42}$$

If yawing moments due to aileron deflection, $C_{n_{\delta_a}}$, can be assumed negligible, the rudder requirement with sideslip can be written:

$$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}} \qquad \textit{eq 5.43}$$

Therefore, making the above substitution:

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} C_{n_\beta}\right\} \qquad \textit{eq 5.44}$$

However, C_{y_β} can be expressed as follows:

$$C_{y_\beta} = C_{y_{\beta_{\text{Wing, Fuselage, Nacelles}}}} + C_{y_{\beta_{\text{Vertical Tail}}}} \qquad \textit{eq 5.45}$$

and C_{n_β} can be written:

$$C_{n_\beta} = C_{n_{\beta_{\text{Wing, Fuselage, Nacelles}}}} + C_{n_{\beta_{\text{Vertical Tail}}}}$$

$$C_{n_\beta} = C_{n_{\beta_{\text{W, F, N}}}} - C_{y_{\beta_v}}\frac{\ell_v}{b} \qquad \textit{eq 5.46}$$

(the b is required to nondimensionalize the vertical tail length.) and $\frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}$ can be written:

$$\frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} = \frac{C_{y_{\delta_r}}}{-C_{y_{\delta_r}}\frac{\ell_v}{b}} = -\frac{b}{\ell_v} \qquad \textit{5.47}$$

Therefore, making the above substitutions:

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_{\beta_{W,F,N}}} + C_{y_{\beta_v}} + \frac{b}{\ell_v}\left(C_{n_{\beta_{W,F,N}}} - C_{y_{\beta_v}}\frac{\ell_v}{b}\right)\right\}$$

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_{\beta_{W,F,N}}} + \frac{b}{\ell_v}C_{n_{\beta_{W,F,N}}}\right\} \qquad \textit{eq 5.48}$$

The point of the last tedious derivation is to emphasize the fact that if $C_{n_{\delta_a}}$ is zero, the bank angle requirement in steady heading sideslips is dependent strictly on "tail-off" parameters. Also, if $C_{n_{\delta_a}}$ is zero, $\frac{d\phi}{d\beta}$ is not included by <u>dihedral effect</u>. Even if $C_{n_{\delta_a}}$ is not zero, the bank angle requirement in steady heading sideslips is primarily dependent on the sideforces and yawing moments created by wing, fuselage, and nacelles. The gradient, $\frac{d\phi}{d\beta}$, is most often referred to as the <u>sideforce characteristic</u>. It is a rather important flying quality, since it provides a cue to the pilot if sideslip is developed. If $\frac{d\phi}{d\beta}$ is zero, the pilot will probably find that the airplane is easy to trim up in a <u>wings level sideslip</u>. Some positive gradient of $\frac{d\phi}{d\beta}$ - right bank angle required in right sideslips and vice versa - is desirable for satisfactory lateral-directional flying qualities (Figure 5.34).

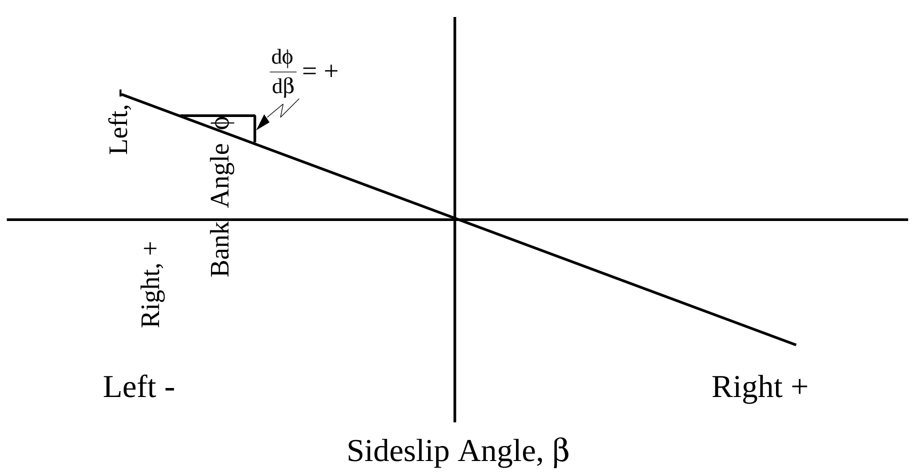


Figure 5.34
Sideforce Characteristics

5.2.1.5.7 Miscellaneous Characteristics Observed in Steady Heading Sideslips

There are several other characteristics of lesser importance which can be determined from the steady heading sideslip test. The first of these is the indicated airspeed error induced by sideslip. Subjecting the production airspeed source to a sideslip angle can result in erroneous airspeed indications in the cockpit. Generally, boom-mounted airspeed sources utilized for sensitive airspeed read outs are not as susceptible to airspeed error with sideslip, particularly if the boom head swivels so as to align itself with the relative wind. Large indicated airspeed errors with sideslip angles normally employed in operational usage is obviously an undesirable, if not dangerous, characteristic. Airspeed errors may vary with the direction of sideslip.

Pitching moments are generated by sideslip angles due to changes in airflow characteristics and effective angles of attack at the wing and horizontal tail. Propeller-driven airplanes commonly exhibit a nose-down pitching moment with sideslip from one side and a nose-up moment with sideslip from the opposite side. This is due to the phenomenon of moving the horizontal tail in and out of the high energy propeller slipstream when sideslips are induced. These pitching moments or longitudinal trim changes are manifested to the pilot through the elevator deflection and longitudinal control force required to maintain airspeed constant in the steady heading sideslips. Excessive longitudinal trim changes with sideslip angels normally utilized in operational flight conditions would place excessive demands on pilot attention and coordination.

Sideslip angle almost invariably result in increases in airplane drag coefficient. Thus, to maintain airspeed constant without changing power setting, the pilot notes an increase in rate of descent during sideslipping flight. This characteristic has been utilized to increase the landing approach flight path angle in light airplanes. However, excessive rate of descent with sideslip angles normally utilized in operational flight conditions would be undesirable.

5.2.1.5.8 Steady Heading Sideslip Review

The forces and moments which must be balanced in the steady heading sideslip are summarized in Figure 5.35.

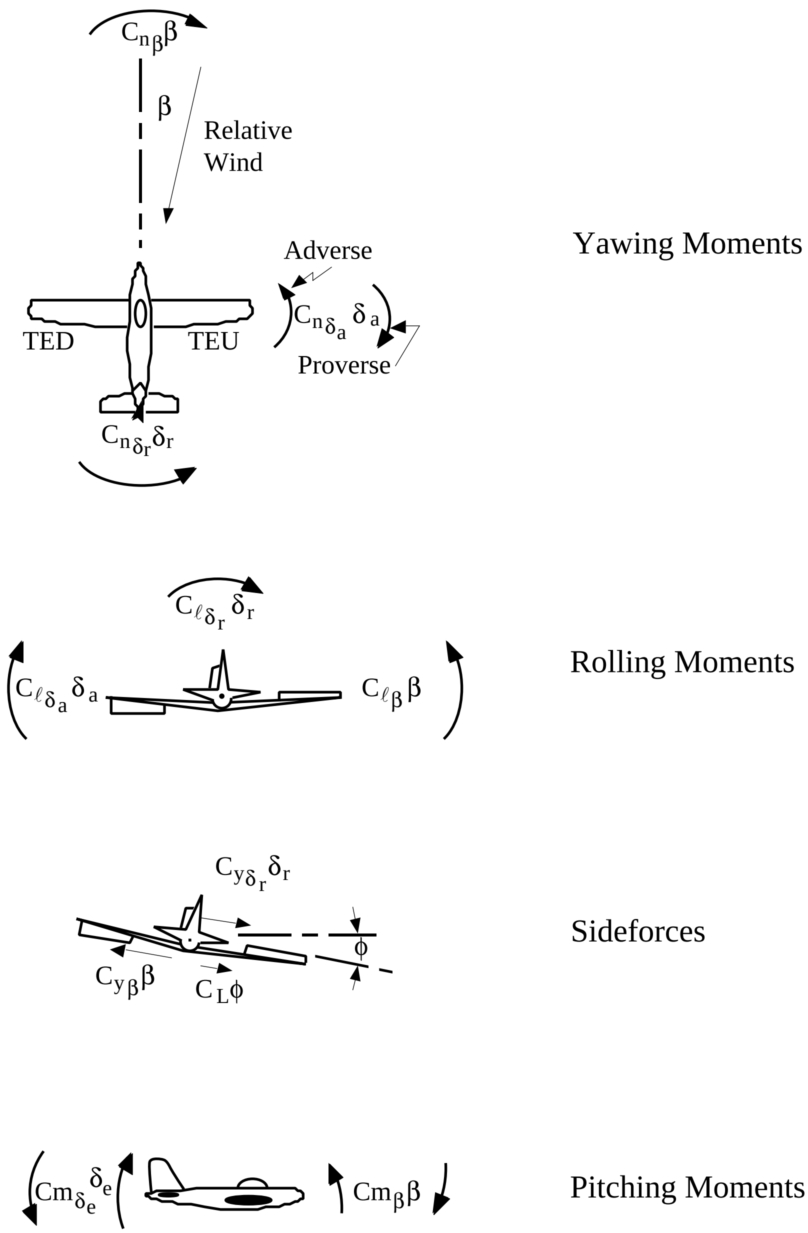


Figure 5.35
Forces and Moments Balanced in Steady Heading Sideslips

5.2.2 Dynamic Lateral-Directional Stability and Control

The previous discussion of lateral-directional stability has been concerned only with equilibrium flight conditions. The discussion will now be exp anded to study the means by which one equilibrium flight condition is changed to another equilibrium flight condition, either by pilot control inputs or by external perturbations. The study of dynamic lateral-directional stability and control characteristics will require the investigation of nonequilibrium flight conditions.

The origin, characteristics, and parameters affecting the lateral-directional modes of motion will now be introduced.

5.2.2.1 ORIGIN OF THE LATERAL-DIRECTIONAL MODES OF MOTION

Without derivation, which can be found in appropriate literature, the determinant of the transformed lateral-directional equation of motion for "small" disturbances may be written as shown in Figure 5.36. (Note: The lateral-directional determinant is presented with the assumption that the effect of lateral-directional inertia cross-coupling is included in each of the terms of the equation. This cross-coupling is a result of the product of inertia in roll and yaw, I_{XZ}. The assumption that I_{XZ} effects are "built into" each of the stability derivatives is deemed valid for qualitative analysis. However, the use of complete equations, which show the influence of the produce-of-inertia on each term, is recommended for quantitative analysis. These equations may be found in appropriate technical literature.)

The solutions of the lateral-directional determinant will provide much useful information about the lateral-directional modes of motion. In order to easily obtain the first solution, the assumption is made that dihedral effect and roll due to yaw rate are zero:

$$L_{\beta} = 0 \quad L_{r} = 0 \qquad \textit{eq 5.49}$$

(The effect of varying these parameters will be introduced later.)

$$\begin{array}{l} \text{Sideforce Characteristics} \rightarrow \\ \text{Rolling Moment Characteristics} \rightarrow \\ \text{Yawing Moment Characteristics} \rightarrow \end{array} \begin{vmatrix} S - Y_{\beta} & -\frac{g}{\mu_0} & 1 \\ -L_{\beta} & S\left(S - L_p\right) & -L_r \\ -N_{\beta} & -N_p S & S - N_r \end{vmatrix} = 0$$

Terms Generated by Changes in Sideslip Angle

Terms Generated by Changes in Bank Angle

Terms Generated by Changes in Yaw Rate

S = Laplace Operator.

g = acceleration due to gravity.

μ = horizontal velocity (u_0 = initial horizontal velocity).

I_{XX} = airplane moment of inertia in roll.

I_{ZZ} = airplane moment of inertia in yaw.

$Y_{\beta} = \frac{\partial Y / \partial \beta}{mu_0} = C_{y_{\beta}} \frac{qS}{mu_0}$ = sideforce due to sideslip term.

$L_{\beta} = \frac{\partial L / \partial \beta}{I_{XX}} = C_{\ell_{\beta}} \frac{qSb}{I_{XX}}$ = rolling moment due to sideslip (dihedral effect) term.

$L_p = \frac{\partial L / \partial p}{I_{XX}} = C_{\ell_p} \frac{qSb^2}{2u_0 I_{XX}}$ = rolling moment due to roll rate (roll damping) term.

$L_r = \frac{\partial L / \partial r}{I_{XX}} = C_{\ell_r} \frac{qSb^2}{2u_0 I_{XX}}$ = rolling moment due to yaw rate term.

$N_{\beta} = \frac{\partial N / \partial \beta}{I_{ZZ}} = C_{n_{\beta}} \frac{qSb}{I_{ZZ}}$ = yawing moment due to sideslip (directional stability) term.

$N_p = \frac{\partial N / \partial p}{I_{ZZ}} = C_{n_p} \frac{qSb^2}{2u_0 I_{ZZ}}$ = yawing moment due to roll rate term.

$N_r = \frac{\partial N / \partial r}{I_{ZZ}} = C_{n_r} \frac{qSb^2}{2u_0 I_{ZZ}}$ = yawing moment due to yaw rate (yaw rate damping) term.

Figure 5.36
The Lateral-Directional Determinant

The lateral-directional determinant thus reduces to:

$$S\left(S - L_p\right)\begin{vmatrix} S - Y_\beta & 1 \\ -N_\beta & S - N_r \end{vmatrix} = 0 \qquad \textit{eq 5.50}$$

Solving the above yields:

$$S\left(S - L_p\right)\left\{S^2 + \left(-Y_\beta - N_r\right)S + \left(N_\beta + Y_\beta N_r\right)\right\} = 0 \qquad \textit{eq 5.51}$$

The last equation describes the three lateral-directional modes of motion if L_β and L_r are both equal to zero. These mode of motion are the spiral mode - indicated by $\left\{S + \left(-Y_\beta - N_r\right)S + \left(N_\beta + Y_\beta N_r\right)\right\}$. These "classic" lateral-directional roots are presented on the convenient root locus plot in Figure 5.37. As previously developed, the spiral root lies at the origin, indicating neither a divergence or convergence when excited. The roll mode is characterized by a large negative real root, corresponding to a heavily damped, nonoscillatory rolling motion. The Dutch roll mode is characterized by a complex pair of roots, indicating an oscillatory second-order type of motion.

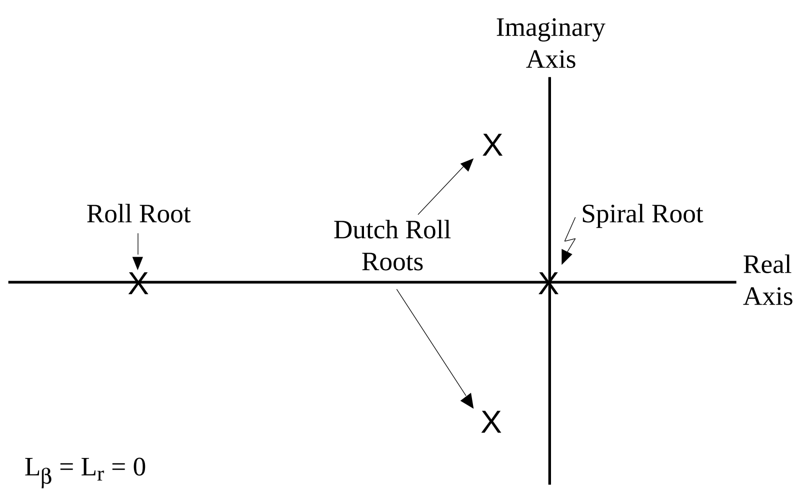


Figure 5.37
Complex Plane Representation of Classic Lateral-Directional Modes of Motion

5.2.2.2 CHARACTERISTICS OF THE LATERAL-DIRECTIONAL MODES OF MOTION

5.2.2.2.1 The Spiral Mode

The spiral mode is a first order nonoscillatory mode of motion. It can be described as a bank angle divergency or convergency after a bank angle disturbance from wings level flight with controls restrained in the position for wings level flight. Actually, the spiral mode can be convergent, divergent, or neutral. The pilot does not normally excite the spiral mode intentionally; however, it is excited any time the bank angle of the airplane is disturbed from a trimmed wings level condition. Therefore, it has some influence on the airplane's lateral-directional flying qualities. The nature of the spiral mode, i.e., whether it is convergent, divergent, or neutral, depends on the sign of the following combination of stability derivatives[10] :

$$\frac{g}{u_0}\left\{L_\beta\ N_r - N_\beta\ L_r\right\} \qquad \textit{eq 5.52}$$

If $L_\beta\ N_r$ is larger than $N_\beta\ L_r$, the spiral mode will be convergent; if $N_\beta\ L_r$ is larger than $L_\beta\ N_r$, the spiral mode will be divergent (Figure 5.38). In general, strong directional stability (large N_β) promotes a divergent spiral mode; while high positive dihedral effect (large negative L_β) promotes a convergent spiral mode. At any rate, the magnitude of the spiral root tends to vary inversely with airspeed (see the last equation). Thus, the magnitude of the spiral, which corresponds to the rate of convergency or divergence of the motion, tends to be large at slow airspeeds and small at high airspeeds.

---

[10] Sometimes referred to as the "E coefficient" of the lateral-directional characteristic equation.

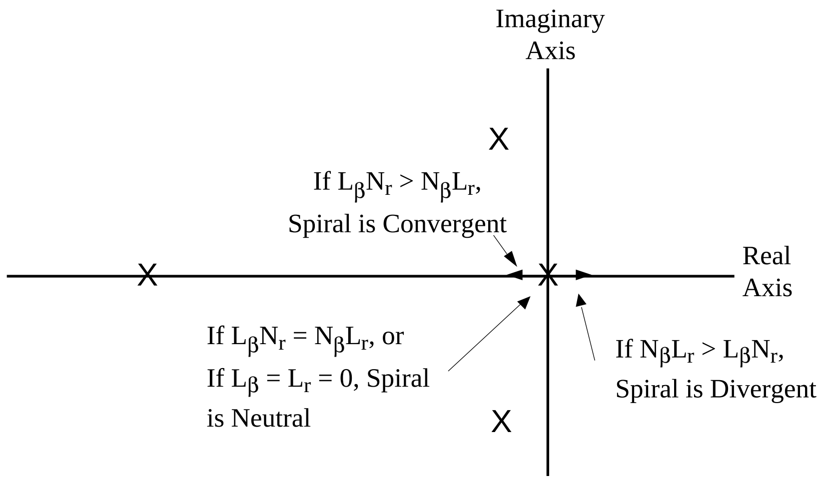


Figure 5.38
Nature of the Spiral Mode

5.2.2.2.2 The Roll Mode

The classic roll mode is a heavily damped, first order, nonoscillatory mode of motion manifested in a build-up of roll rate to a steady state value for a given lateral control input. This rolling motion (depicted in Figure 5.39) is utilized by the pilot to vary and control bank angle. Thus, the characteristics of this rolling motion have a major influence on the pilot's opinion of the maneuvering capabilities of the airplane. For now, the roll response will be considered to be a "single degree of freedom" motion; i.e., the airplane is free only to roll (not yaw or pitch) in response to a lateral control input. For this single degree of freedom roll, the steady state roll rate for a given lateral control deflection can be expressed as:

$$p_{SS} = -\frac{L_{\delta_a}}{L_p}\,\delta_a = -\frac{C_{\ell_{\delta_a}}}{C_{\ell_p}}\,\frac{2V}{b}\,\delta_a \qquad \textit{eq 5.53}$$

Where:

$$L_{\delta_a} = \frac{\partial L/\partial \delta_a}{I_{XX}} = C_{\ell_{\delta_a}}\frac{qSb}{I_{XX}}$$ = rolling moment due to lateral control deflection (lateral control power) term.

eq 5.54

The rate at which the roll rate builds up to steady state, or decelerates to zero, is governed by the "roll mode time constant," τ_R. The roll mode time constant is defined as the time required for the roll rate to reach 63.2 percent of the steady state roll rate following a step input of lateral control (Figure 5.39). The roll mode time constant is inversely proportional to the roll damping parameter:

$$\tau_R = -\frac{1}{L_p} \qquad \textit{eq 5.55}$$

Actual rolling motion is generally contaminated by yawing motion which results in the roll rate being somewhat oscillatory. Actual rolling motion will be discussed more completely in the section on Rolling Performance.

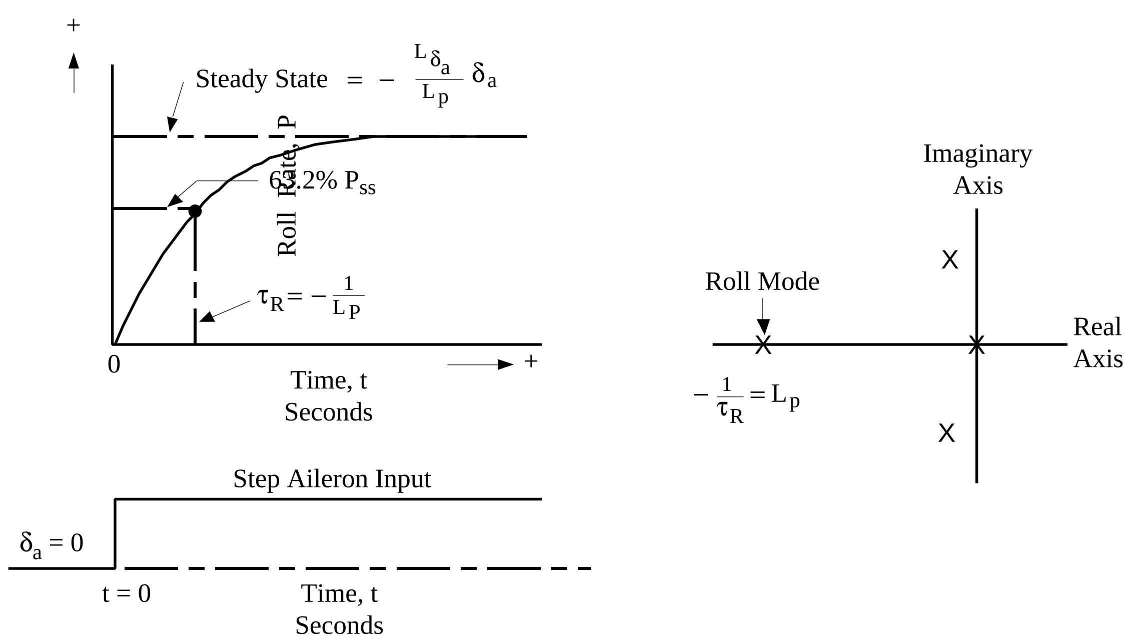


Figure 5.39
Single Degree of Freedom Roll Response

5.2.2.2.3 The Dutch Roll Mode

The remaining two roots of the lateral-directional solution form a complex pair corresponding to a classic second-order, oscillatory mode of motion. This is the "Dutch roll" or "lateral-directional oscillation." The Dutch roll mode is sometimes referred to as a "nuisance" or "annoyance" mode since the motion is not normally deliberately excited in normal flying. However, it is inadvertently excited almost continually by pilot control inputs or by external disturbances. Therefore, the characteristics of this mode of motion greatly influence the pilot's opinion of the airplane during all phases of mission accomplishment. (During certain special flight conditions, such as flight with asymmetric power or landing with a crosswind, the pilot may utilize the Dutch roll mode to generate sideslip changes in order to maintain steady heading flight. In addition, bank angle control by use of the rudders is manifested through the Dutch roll mode of motion.)

There are no simple approximations for the frequency and damping ratio of the Dutch roll motion. However, rough approximations may be made by assuming that L_β and L_r are both zero. This reduces the Dutch roll motion to "two degrees of freedom" - rotation about the Z axis of the airplane and lateral translation (the Dutch roll for these conditions is a pure directional oscillation - a pure "snaking" motion). The portion of the characteristic lateral-directional equation which describes this motion is:

$$\left\{S^2 + \left(-Y_\beta - N_r\right) S + \left(N_\beta + Y_\beta N_r\right)\right\} = 0 \qquad \textit{eq 5.56}$$

From this relationship, a rough approximation for the undamped natural frequency of the Dutch roll oscillation can be derived and written as follows:†

$$\omega_{n_{DR}} \doteq M\sqrt{C_{n_\beta} \frac{\gamma P_a Sb}{2 I_{ZZ}}} \qquad \textit{eq 5.57}$$

Where:

M = Mach number

C_{n_β} = directional stability derivative

γ = a constant, generally taken as 1.4

P_a = absolute pressure, pounds per square foot

† Several mathematical manipulations have been omitted.

S = Wing area, square feet

b = Wing span, feet

I_{ZZ} = Moment of inertia in yaw

Several important relationships can be gathered from a study of this equation for $\omega_{n_{DR}}$:

1. The undamped natural frequency of the Dutch roll motion increases as Mach number increases; thus, the <u>period</u> decreases with increasing Mach number. (The "quickness" of the motion increases).

2. The undamped natural frequency of the Dutch roll motion increases with an increase in directional stability, and decreases as C_{n_β} is decreased. This is analogous to strengthening or weakening the spring in the spring-mass-damper system.

3. The undamped natural frequency of the Dutch roll motion decreases with an increase in pressure altitude at a constant Mach number. (The "quickness" of the motion decreases at high altitude if all other parameters are constant.)

4. The undamped natural frequency of the Dutch roll motion decreases with an increase in moment of inertia in yaw. This is analogous to increasing the mass in the spring-mass-damper system. (Large airplanes with large I_{ZZ} parameters have low frequency Dutch roll motions. They are therefore slow in responding to gust disturbances or rudder inputs.)

In order to obtain an approximation for the damping ratio of the Dutch roll motion, it is assumed that Y_β and N_r are approximately the same value. The damping ratio may then be developed as follows†:

$$\zeta_{DR} = C_{n_r} \sqrt{\frac{\rho \ S b^3}{8 \ C_{n_\beta} \ I_{zz}}} \qquad \textit{eq 5.58}$$

Where:

C_{n_r} = yaw rate damping derivative.

ρ = density, slugs per cubic foot.

Certain important effects are visible from this relationship:

1. Damping of the Dutch roll motion is largely dependent on yaw rate damping. Changing yaw rate damping is analogous to changing the viscosity of the damper in the spring-mass-damper system.

2. Damping of the Dutch roll motion decreases with increasing altitude because of the reduction in density.

3. Increasing directional stability decreases Dutch roll damping.

4. Increasing the yawing moment of inertia decreases Dutch roll damping.

5. Damping of the Dutch roll mode of motion is not a direct function of airspeed or Mach number.

† Several mathematical manipulations have been omitted.

If the airplane exhibits some level of dihedral effect and/or roll due to yaw rate, there will obviously be rolling motion generated by the sideslip and yaw rate excursions during the Dutch roll oscillation. The degree of rolling motion exhibited, expressed in terms of "roll-to-yaw" ratio $\left(\frac{\phi}{\beta}\right)$, has a significant influence on the acceptability of a particular combination of Dutch roll damping and frequency. In general, more damping is required as the roll-to-yaw ratio is increased.

The real Dutch roll oscillation generally consists of yawing, sideslipping, and rolling. The phasing of the rolling and yawing depends on whether positive or negative dihedral effect is present. The Dutch roll oscillation shown in Figure 5.40 is presented assuming the airplane exhibits positive dihedral effect.

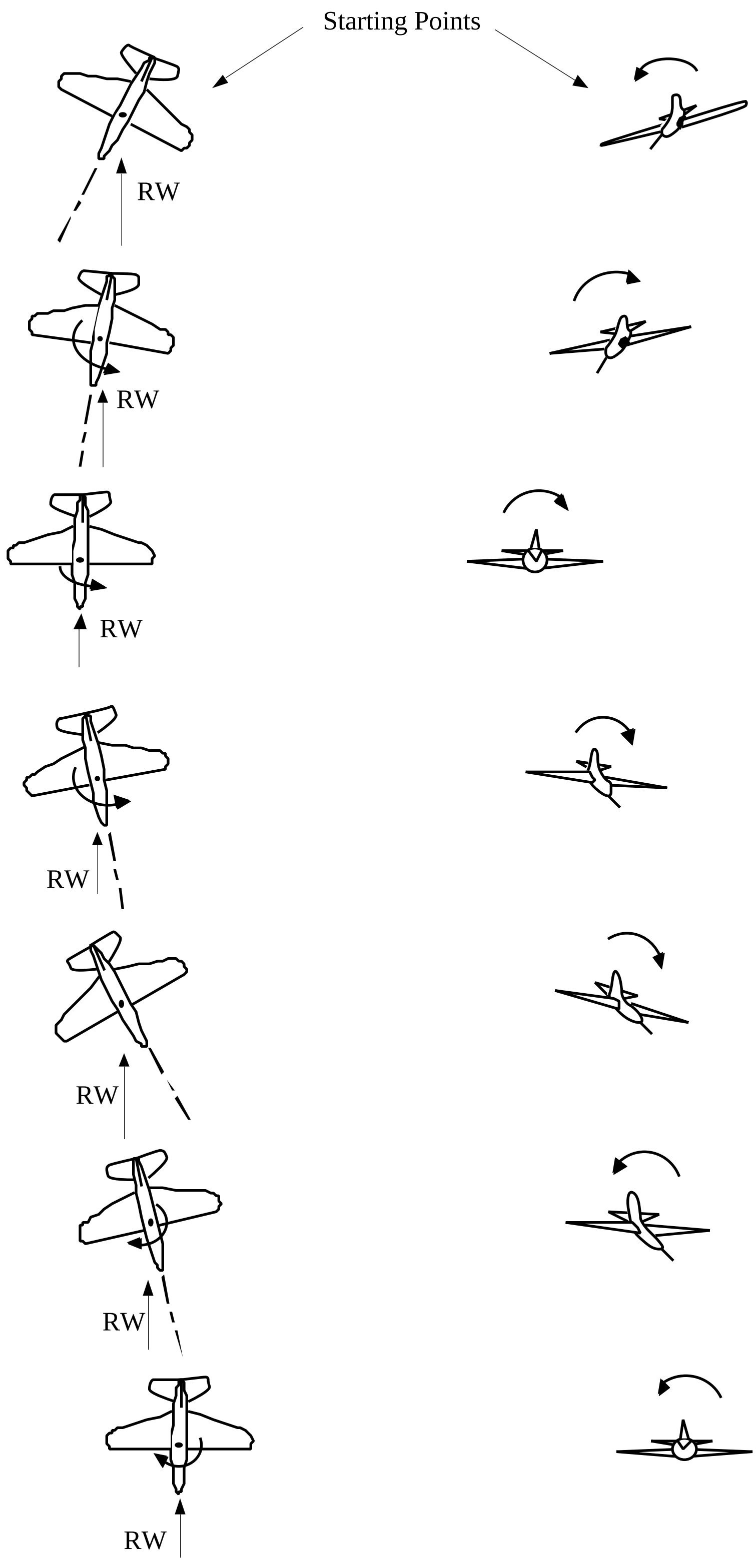


Figure 5.40
Typical Dutch Roll Motion

5.2.2.3 EFFECTS OF VARIOUS PARAMETERS ON THE LATERAL-DIRECTIONAL MODES OF MOTION

The influence of varying several parameters on lateral-directional dynamics will now be presented using the convenient root locus plots.

The characteristic roots for zero dihedral effect and roll due to yaw rate are as shown in Figure 5.41.

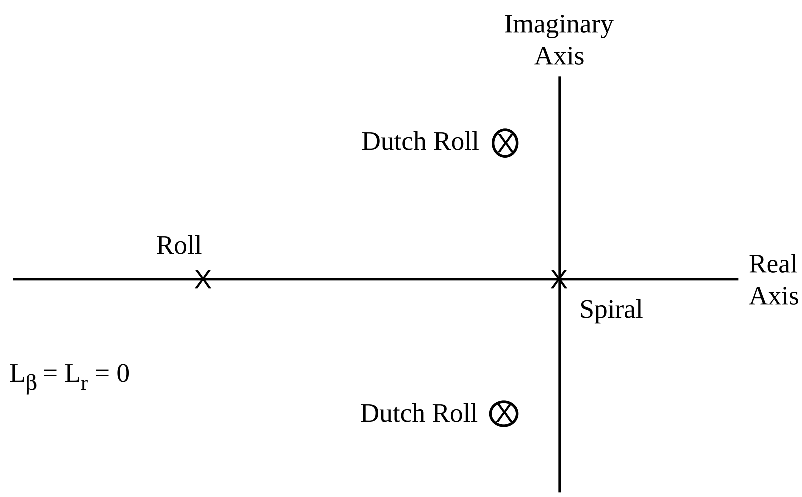


Figure 5.41
Complex Plane Representation of Classic Lateral-Directional Roots

The following discussion and plots are introduced to show the influence of certain derivatives on the characteristic roots.

5.2.2.3.1 Dihedral Effect, L_β

Fixing the values of roll due to yaw, L_r, and yaw due to roll, N_p, at zero, the influence of dihedral effect is as shown in Figure 5.42.

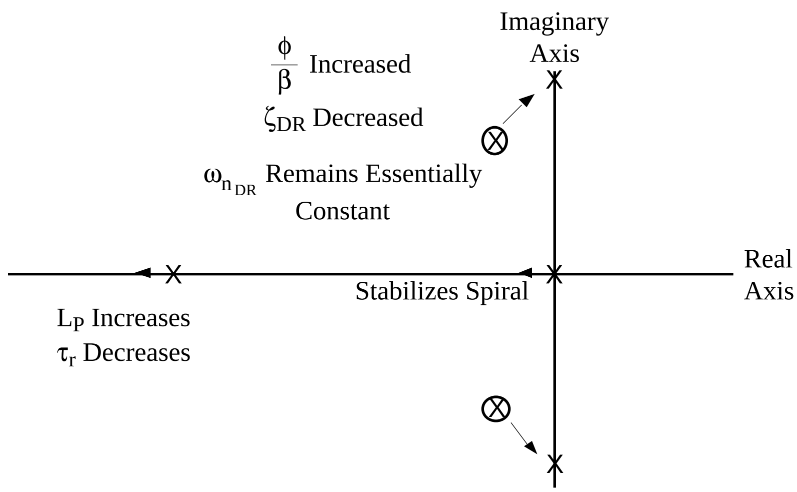


Figure 5.42
Influence of Adding Positive Dihedral Effect

The consequences of introducing dihedral effect are:

1. Total roll damping is increased.

2. Damping of Dutch roll is decreased.

3. Spiral mode tends to become stable.

5.2.2.3.2 Rolling Moment Due to Yaw Rate, L_r

Assuming L_β and N_p are fixed at zero, the influence of L_r is as shown in Figure 5.43.

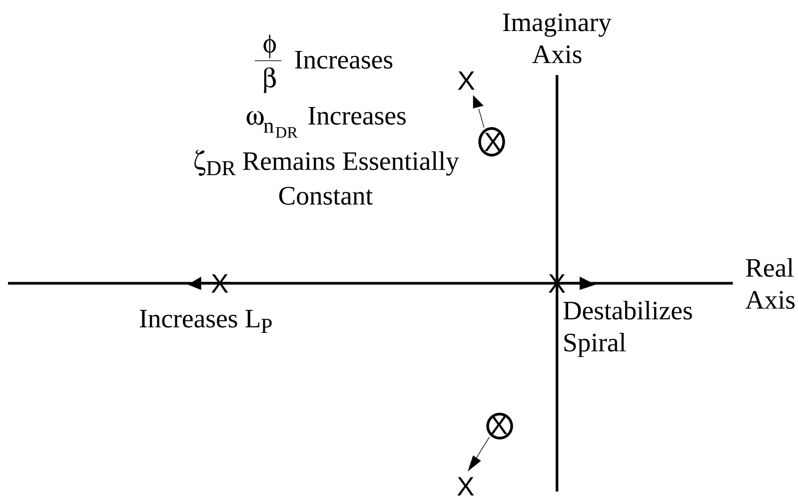


Figure 5.43
Effect of Adding Roll Due to Yaw Rate, L_r

Roll due to yaw rate effects on root locations are:

1. The Dutch roll frequency tends to increase.

2. The spiral mode tends to destabilize.

3. Roll damping increases.

5.2.2.3.3 Yawing Moment Due to Roll Rate, N_p

The general trend of the influence of N_p on a <u>typical</u> airplane is discussed. For nominal negative values of N_p and typical values of L_β and L_r, the Dutch roll damping is reduced and the total roll damping may increase or decrease depending on how much dihedral effect is present. The spiral mode is practically unchanged but tends to move toward the origin. Positive values of N tend to have the reverse effect.

5.2.2.3.4 Directional Stability, N_β

The frequency of the Dutch roll is influenced most strongly by directional stability. There is a minor influence on the spiral mode and roll mode. Figure 5.44 illustrates the effect of increasing directional stability.

The following is a summary of the effects of directional stability:

1. Increasing N_β will increase the Dutch roll frequency, ω_{n_d}, and decrease the damping ratio, ζ_d.

2. Increasing N_β will tend to drive the spiral mode divergent but has limited influence on the magnitudes of the mode.

3. Increasing N_β tends to drive the roll mode toward the single degree of freedom root location, L_p.

4. Decreasing N_β towards zero would finally drive the Dutch roll to a dead beat mode and the lateral-directional characteristic equation would be made up of four real roots.

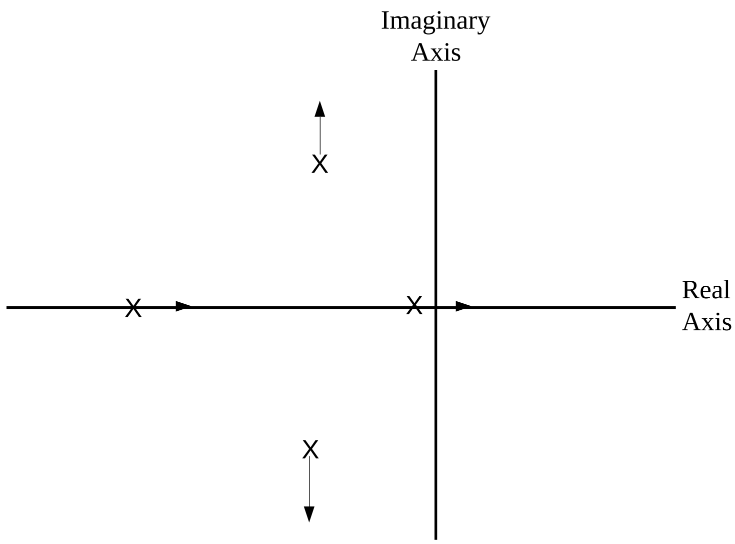


Figure 5.44
Influence of Changing Directional Stability on Two Degree of Freedom Dutch Roll

5.2.2.3.5 Yaw Rate Damping, N_r

The influence of yaw rate damping, N_r, is primarily on the Dutch roll damping. The influence on the spiral mode and roll mode is limited to a very minor effect. Figure 5.45 illustrates the trends.

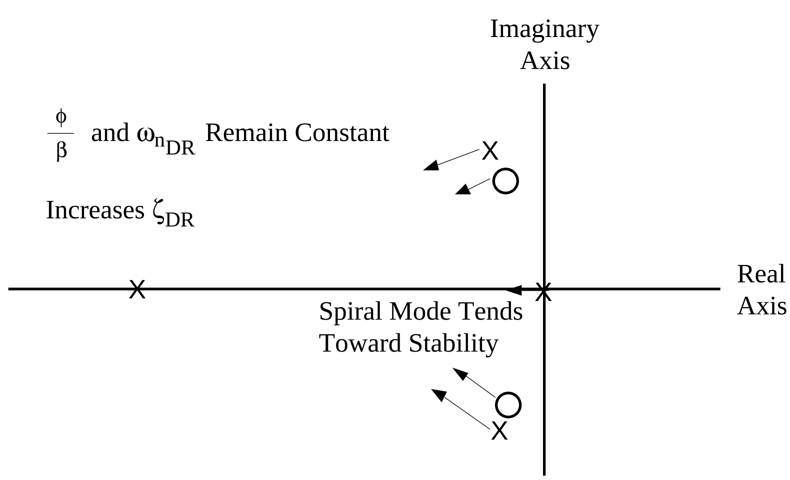


Figure 5.45
Effects of Increasing Yaw Rate Damping, $\mathbf{N_r}$

The following is a summary of the typical influence of a nominal increase in yaw rate damping:

1. The real part of the Dutch roll is increased.

2. The spiral mode tends toward a convergent mode if positive dihedral effect is present.

3. The roll mode root is slightly increased if positive dihedral effect is present.

5.2.2.3.6 Bank Angle Control and Adverse Yaw

In the previous paragraphs, the influence of various stability derivatives on the roots of the characteristic equation was discussed. Yaw due to aileron deflection, N_{δ_a}, was not among them simply because it has <u>no</u> affect on root location. However, it is obvious that it has an effect on <u>handling qualities</u> and the following discussion to show that effect.

The bank angle transfer function can be expressed as:

$$\frac{\phi}{\delta_a}(S) = \frac{L_{\delta_a}\left[S^2 + 2\zeta_\phi\ \omega_{n_\phi}\ S + \omega_{n_\phi}{}^2\right]}{\left(S + \frac{1}{\tau_s}\right)\left(S + \frac{1}{\tau_R}\right)\left[S^2 + 2\ \zeta_d\ \omega_{n_d}\ S + \omega_{n_d}{}^2\right]} \qquad \textit{eq 5.59}$$

where the roots of the denominator are the roots of the characteristic equation that have previously been discussed. Recall that the Dutch roll roots were the two degree of freedom roots as modified by dihedral effect, roll due to yaw, etc. These roots will now be referred to as the three degree of freedom Dutch roll. The numerator roots are also the two degree of freedom Dutch roll roots but as modified by adverse yaw and dihedral effect. Figure 5.46 shows the movement of the three degree of freedom Dutch roll roots as a result of increased dihedral effect and roll due to yaw rate for nominal values of the remaining terms and the movement of the numerator roots as either adverse yaw or dihedral effect is increased. How much these roots are separated is indicative of the amount of Dutch roll response that will be present in the total roll response or how much oscillatory roll is added to the average roll.

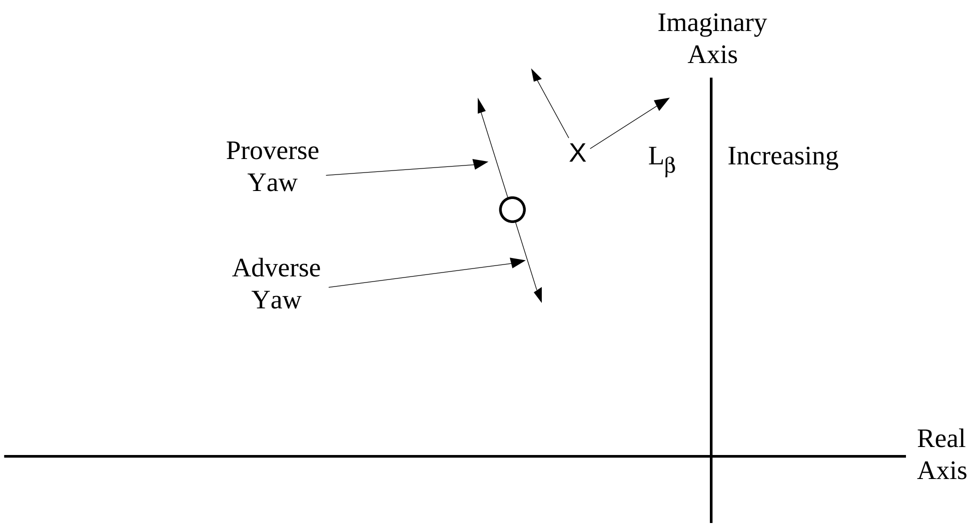


Figure 5.46
Dutch Roll Root and Zero Combinations

The zero locations become very important in the closed loop task of bank angle control. With the pilot in the loop and tasked with precise bank angle control, the roots of the system tend to move as indicated in Figures 5.47 and 5.48.

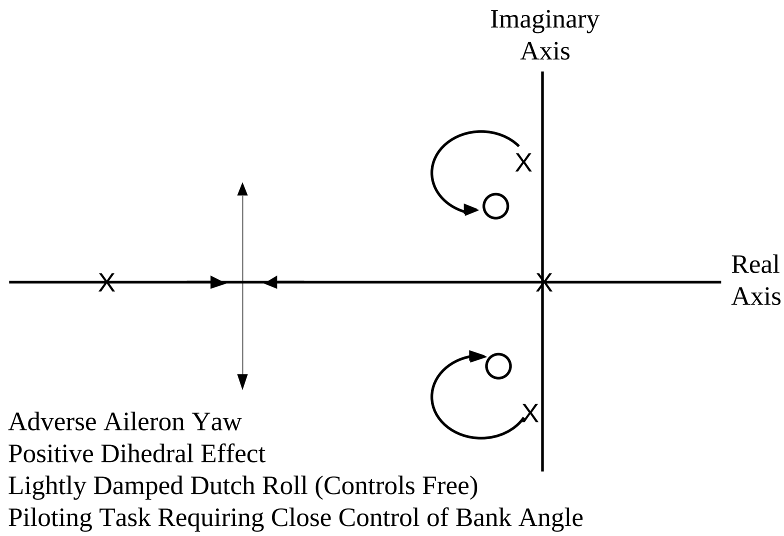


Figure 5.47
Path of the Lateral-Directional Roots with Increase in Tightness of Bank Angle Control

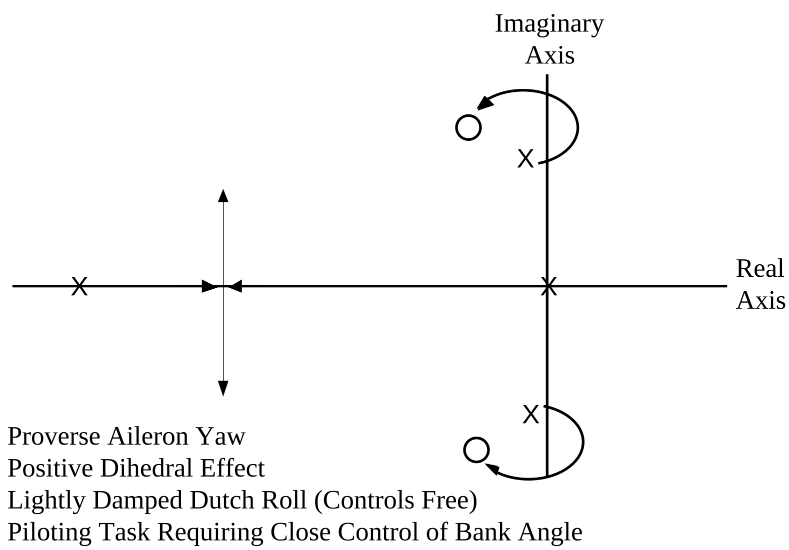


Figure 5.48
Paths of the Lateral-Directional Roots with Increase in Tightness of Bank Angle Control

Figure 5.50 is for a typical case where a nominal value of adverse yaw is present. Note that the roll/spiral mode is stabilized and the closed loop damping of the Dutch roll is increased. Figure 5.49 is for a case of proverse yaw and, for the case shown, the Dutch roll is driven unstable. The pilot will not permit this and will have to change his control technique. The pilot would obviously rate this system as unsatisfactory. It is important to note that the cause was zero location and not the locations of the characteristic roots.

In the roll rate response to a step input of aileron, the phase angle between roll rate and sideslip is indicative of the relative angular positions of the Dutch roll root and the zeros of the roll transfer function.

The expression, $\frac{\omega_\phi}{\omega_{DR}}$, is frequently used as an indication of the roll disturbance at the Dutch roll natural frequency due to aileron inputs. If $\frac{\omega_\phi}{\omega_{DR}} = 1$, the yawing moment due to aileron deflection, N_{δ_a}, is zero and there is little or no Dutch roll motion in the roll response to aileron inputs. If $\frac{\omega_\phi}{\omega_{Dr}} > 1$, the yawing moment due to aileron deflection is proverse $\left(N_{\delta_a} < 0\right)$ and the damping of the Dutch roll motion during precise bank angle tracking tasks may decrease. The Dutch roll motion will be excited during the roll response of the airplane to a lateral control input if $\frac{\omega_\phi}{\omega_{Dr}} > 1$. When $\frac{\omega_\phi}{\omega_{Dr}} < 1$, the yawing moment due to lateral control deflection is adverse $\left(N_{\sigma_a} > 0\right)$ and the damping of the Dutch roll motion during precise bank angle tracking tasks generally is better than the damping exhibited by the airplane with controls free. The Dutch roll motion will be excited to some extent whenever the airplane rolls in response to a lateral control input when $\frac{\omega_\phi}{\omega_{Dr}} < 1$.

There are two parameters used to quantitatively evaluate the Dutch roll influence on roll performance in "tightly controlled" tracking tasks. These parameters are:

1. The ratio of the oscillatory component of roll rate to the average component of roll rate following a rudder-pedals-free step aileron control command, $\frac{P_{osc}}{P_{av}}$ (see Definitions section of MIL-SPEC 8785B for further definition).

2. The phase angle is a cosine representation of the Dutch roll component of sideslip, ψ_β.

5.2.2.4 EFFECT OF ψ_β ON FLYING QUALITIES

Since ψ_β is a rather abstract parameter, it is well to consider its physical implications and significance to the piloting of an airplane. Very simply, ψ_β can be considered as an indication of those closed-loop stability characteristics of an airplane that are related to the lateral-directional coupling derivatives; and of the difficulty a pilot will experience in coordinating a turn entry. Further clarification can be obtained by discussing the variation of the specified values of $\frac{P_{osc}}{P_{av}}$ with ψ_β for positive dihedral.

The parameters $\frac{P_{osc}}{P_{av}}$ and ψ_β have been used to specify criteria as a function of Flight Phase Category and Level as shown in Figure 5.49a (Specification Paragraph 3.3.2.2.1, Figure 5.4).

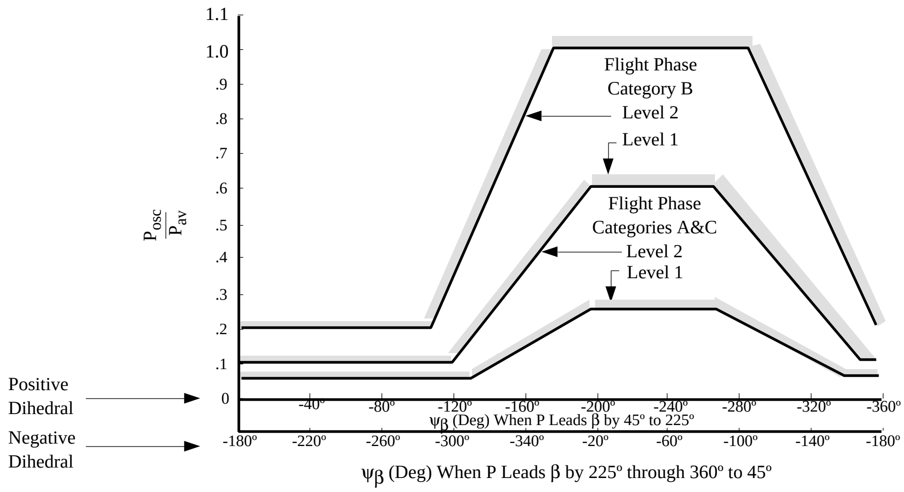


Figure 5.49a
MIL-F-8785B

It should be noted that this figure has two ψ_β scales, one for positive dihedral (p leads β by 45 º to 225 º) and the other for negative dihedral (p leads β by 225º through 360º to 45 º).

From this figure it can be seen that the ratio of roll rate oscillation to steady state roll rate can be much greater for some values of ψ_β than for others. Specifically, the specified values of $\frac{P_{osc}}{P_{av}}$ for $0° \geq \psi_\beta \geq -90°$ are far more stringent than $-180° \geq \psi_\beta \geq -270°$. There are at least three reasons why this is so:

1. Differences in Closed-Loop Stability - From a root locus analysis, it can be shown that when the zero of the $\frac{P}{\delta_{as}}$ transfer function list in the lower left quadrant with respect to the Dutch roll pole, $\left(-180° \geq \psi_\beta \geq -270°\right)$, the closed-loop damping increases when the pilot closes a bank angle error to aileron loop. (See Figure 5.49b.) The reason for this in physical terms is that when the zero lies in the lower left quadrant, aileron inputs proportional to bank angle errors generate yawing accelerations that tend to damp the Dutch roll oscillations. Thus, the Dutch roll damps out more quickly closed-loop than

open-loop, so a pilot will tend to tolerate somewhat more $\frac{P_{osc}}{P_{av}}$. Conversely, it can be shown that when the zero lies in the upper right quadrant with respect to the Dutch roll pole $\left(0^o \geq \psi_\beta \geq -90^o\right)$, the closed-loop damping decreases when the pilot applies aileron inputs proportional to bank angle error. The physical explanation for this is that aileron inputs generate yawing accelerations that tend to excite or sustain the Dutch roll oscillations. Thus, the Dutch roll damps out less quickly closed loop than open loop, and can even go unstable closed loop; that is, pilot-induced oscillations can result. In this case a pilot's tolerance of $\frac{P_{osc}}{P_{av}}$ tends to reduce.

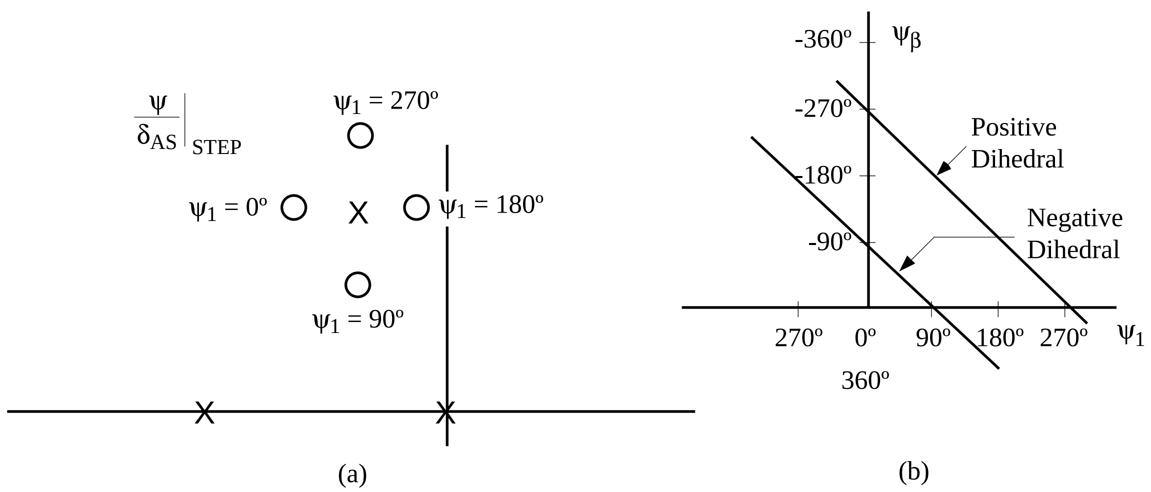


Figure 5.49b
MIL-F-8785B

2. <u>Differences in Difficulty of Rudder Coordination</u> - Significant differences in the $\frac{P_{osc}}{P_{av}}$ requirements also occur because of differences in difficulty of rudder coordination while performing coordinated turn entries or exits. For $-180^o \geq \psi_\beta \geq -270^o$, normal coordination may be effected, that is, right rudder pedal for right rolls. Thus, even if large roll rate oscillations occur in rudder-pedal-free rolls (the conditions under which the $\frac{P_{osc}}{P_{av}}$ tests are conducted), sideslip oscillations can be readily minimized by use of rudder

pedals so that roll rate oscillations do not occur. On the other hand, for $0^o \geq \psi_\beta \geq -90^o$ it is necessary to cross control to effect coordination, that is, left rudder pedal with right aileron. Since pilots do not normally cross control (and if they must, have great difficulty in doing so) for $0^o \geq \psi_\beta \geq -90^o$, oscillations in sideslip, and hence oscillations in roll rate, either go unchecked or are amplified by the pilot's efforts to coordinate.

3. Differences in Average Roll Rate - The third reason why the $\frac{P_{osc}}{P_{av}}$ requirements vary so significantly with ψ_β is that the average roll rate, $\frac{P_{osc}}{P_{av}}$ for a given aileron input, varies significantly with ψ_β. For positive dihedral, adverse yaw-due-to aileron $\left(\psi_\beta \;\; -180^o\right)$ tends to decrease average roll rate whereas proverse yaw-due-to aileron $\left(\psi_\beta \;\; 0^o\right)$ tends to increase average roll rate. As a matter of fact, proverse yaw-due-to aileron is sometimes referred to as "complementary yaw" because of this augmentation of roll effectiveness. Thus, for a given amplitude of P_{osc}, $\frac{P_{osc}}{P_{av}}$ will be greater at $\psi_\beta = -180^o$ than it will be at $\psi_\beta = 0^o$.

In summary, the parameters that have been chosen in the specification to describe and specify the coupling that exists between sideslip and roll for moderate to high $\left|\frac{\phi}{\beta}\right|_d$ response ratios are $\frac{P_{osc}}{P_{av}}$ and ψ_β. These parameters were chosen as being measurable parameters which most simply, directly, and accurately reflect the important flying qualifies considerations. The measurements are taken from the p and β traces which are obtained from sensitive instrumentation.

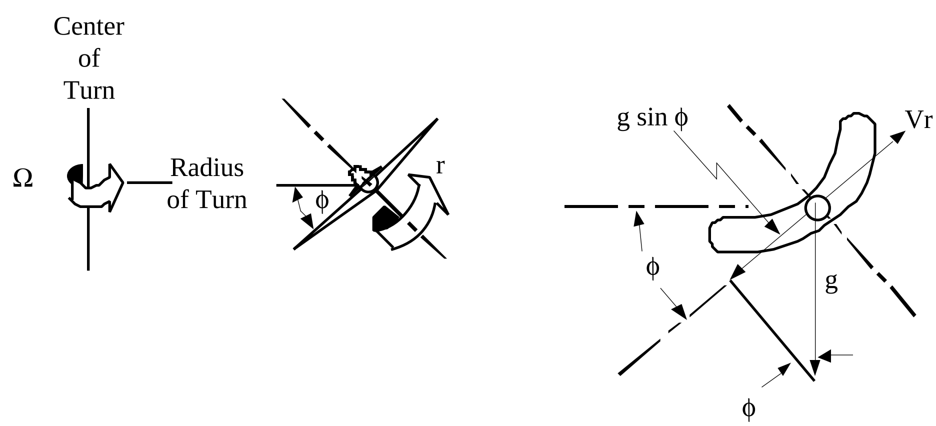
5.2.3 Lateral-Directional Characteristics in Turning Flight

One of the most important considerations in the investigation of lateral-directional flying qualities is the ease with which the pilot can enter and maintain turning flight. The longitudinal flying qualities associated with turning flight were discussed earlier. The lateral-directional characteristic of most concern to the pilot in turning flight is the coordination required of lateral and directional control during turn entries and steady turns. The pilot desires to keep the ball of the needle-ball instrument as close to the center of its race as possible during these evolutions. If the ball remains centered, the pilot, passengers, and objects in the airplane are not subjected to uncomfortable sideward accelerations. As shown in Figure 5.50, a balance of the forces acting on the ball is attained or the airplane is in a coordinated turn, when:

$$g \sin \phi = Vr \qquad \textit{eq 5.60}$$

Where:

- g = acceleration due to gravity, $\mathrm{ft}/\mathrm{sec}^2$.
- ϕ = bank angle, radians.
- V = airplane true airspeed, ft/sec.
- r = airplane yaw rate, $\mathrm{radians}/\mathrm{sec}$.



Coordinated Turn: $Vr = g \sin \phi$

($r = \Omega \cos \phi$, if ϕ is small, $\cos \phi \approx 1.0$; $r \approx \Omega$)

Figure 5.50
The Airplane in a Coordinated Turn

This relationship is appropriate only for fairly small bank angle turns (ϕ =30 degrees or less) since the angular turn rate, Ω, becomes largely pitching motion instead of yawing motion as the bank angle approaches 90º.

With the concept of the coordinated turn in mind, consider now the so-called "two-control turns"; i.e., "aileron only turns" and "rudder only turns."

5.2.3.1 AILERON-ONLY TURNS

In the steady (constant bank angle) aileron-only turn, an expression for the aileron position required for equilibrium can be easily obtained as a function of the nondimensional yaw rate, $\frac{rb}{2V}$. For a symmetrical airplane $\left(C_{n_0} = 0,\ C_{\ell_0} = 0,\ C_{y_0} = 0\right)$, no yawing moments due to lateral control deflection $\left(C_{n_{\delta_a}} = 0\right)$, and no influence of inertia terms on yawing and rolling moments, the equilibrium equations for an aileron-only turn may be written as follows:

SIDEFORCE $$C_{y_\beta}\ \beta + C_{y_r}\left(\frac{rb}{2V}\right) + \frac{W}{qS}\sin\phi = \frac{W}{g}\frac{Vr}{qS} \qquad \textit{eq 5.61}$$

YAWING MOMENT $$C_{n_\beta}\ \beta + C_{n_r}\left(\frac{rb}{2V}\right) = 0 \qquad \textit{eq 5.62}$$

ROLLING MOMENT $$C_{\ell_\beta}\ \beta + C_{\ell_{\delta_a}}\ \delta_a + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0 \qquad \textit{eq 5.63}$$

From the rolling moment equation, the aileron requirement can be derived as follows:

$$\delta_{a_{\text{Equilibrium}}} = -\frac{1}{C_{\ell_{\delta_a}}}\left\{C_{\ell_r}\left(\frac{rb}{2V}\right) + C_{\ell_\beta}\ \beta\right\} \qquad \textit{eq 5.64}$$

In order to eliminate the sideslip variable, an expression for sideslip is obtained from the yawing moment equation:

$$\beta = -\frac{C_{n_r}}{C_{n_\beta}}\left(\frac{rb}{2V}\right) \qquad \textit{eq 5.65}$$

By substituting this expression for sideslip and by appropriate manipulation and differentiation, the aileron required in a steady aileron-only turn may be presented as follows:

$$\frac{d\delta_{a_{\text{Equilibrium}}}}{d\left(\frac{rb}{2V}\right)} = -\frac{1}{C_{\ell_{\delta_a}}\, C_{n_\beta}} \left\{ C_{\ell_r}\, C_{n_\beta} - C_{\ell_\beta}\, C_{n_r} \right\} \qquad \textit{eq 5.66}$$

Notice that the term in braces is the same combination of stability derivatives which dictated whether the spiral mode was convergent, neutral, or divergent. If this term is positive, the spiral mode is divergent, and the pilot will be required to hold "top-aileron" or "out-of-turn-aileron" in an aileron-only turn. If it is negative, the spiral mode is convergent, and the pilot will hold "bottom-aileron" or "in-turn-aileron" in the aileron-only turn. Obviously, if it is zero, the aileron requirement is zero. The balance of forces and moments in the steady aileron-only-turn is presented in Figure 5.51.

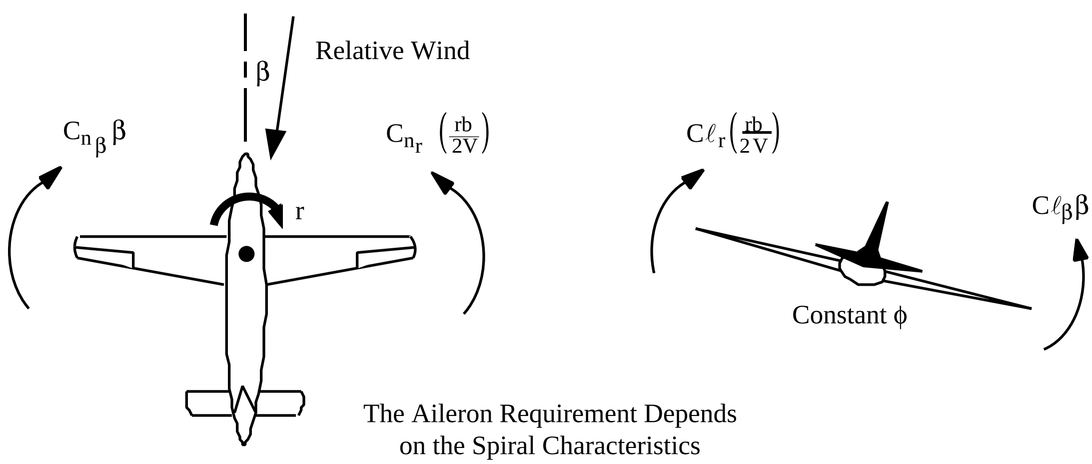


Figure 5.51
The Airplane Established in the Steady Aileron-Only Turn

From this illustration, the following rationalization may be made. If directional stability is strong, the sideslip developed in the aileron-only turn will be fairly small. Therefore, the rolling moment due to sideslip $\left(C_{\ell_\beta}\ \beta\right)$ will be relatively small unless the airplane exhibits a large value of C_{ℓ_β}. Thus, strong directional stability generally results in an unstable, or divergent, spiral mode. Conversely, strong positive dihedral effect (large negative C_{ℓ_β}) generally results in a convergent spiral mode.

During the aileron-only turn, the airplane generally develops sideslip toward the direction of turn (Figure 5.53). If the airplane exhibits normal positive sideforce characteristics[11] , sideforces are generated which "slide" the airplane away from the center of the turn. Thus, the airplane in the aileron-only turn will fly on a larger radius of turn (Figure 5.53). If bank angle and airspeed are maintained constant, the <u>larger radius of turn results in a lower yaw rate</u>. This creates an unbalance in the acceleration components acting on the ball in the needle-ball instrument such that it assumes a position toward the center of the turn ($g \sin \phi > Vr$ in the aileron-only turn). If the sideslip were reduced, the aileron-only turn would approach the coordinated turn in which the ball is centered.

[11] If the airplane exhibits weak sideforce characteristics, such that the derivative C_{y_β} approaches zero, the ball position (and "seat-of-the-pants") loses its significance as an indicator of the perfection of the turn. The pure all-wing tailless airplane exhibits essentially zero sideforce characteristics. In these airplanes, the pilot would have no indication that large sideslip angles had developed during turns unless a sideslip indicator were installed in the cockpit. Thus, the importance of providing good positive sideforce characteristics is again emphasized.

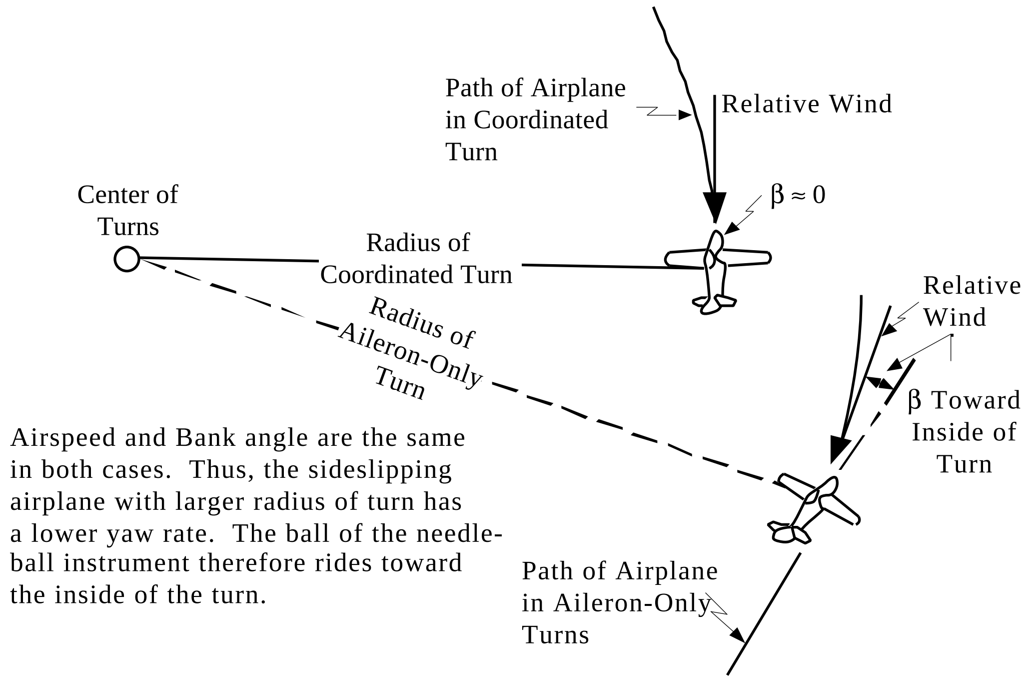


Figure 5.52
Comparison of Typical Coordinated and Aileron-Only Turn

5.2.3.2 RUDDER-ONLY TURNS

Bank angle response to a rudder-only control input depends on the magnitudes and signs of the following stability derivatives:

Dihedral effect, C_{ℓ_β}

Rolling moment due to yaw rate, C_{ℓ_r}

Rolling moment due to rudder deflection, $C_{\ell_{\delta_r}}$

The rolling moment due to rudder deflection is generally so minute that it has no apparent influence on the roll response. However, in airplanes with high vertical tails, the effect of $C_{\ell_{\delta_r}}$ may be detected as an initial hesitation of the roll response or a roll response resulting in a bank angle change opposite to the rudder input (left bank angle change with right rudder, and vice-versa).

Soon after the rudder-only input, the airplane responds through the Dutch roll mode of motion[12] and a yaw rate is developed. Obviously, if the rolling moment due to yaw rate derivative carries its normal sign (positive), the airplane should begin to roll in the same direction as the rudder input. Additionally, sideslip is developed - such that left sideslip is generated by a right rudder input - and the airplane responds in roll due to its dihedral effect. If dihedral effect is positive (C_{ℓ_β} exhibits a negative sign), the rolling moment due to sideslip will cause a bank angle change in the same direction as the rudder input. Thus, the roll response to a rudder-only input is due almost exclusively to the combined effects of the rolling moments generated by yaw rate, C_{ℓ_r}, and sideslip, C_{ℓ_β} (Figure 5.54). Obviously, the influence of these derivatives cannot be separated during the rudder-only turn unless one of the derivative is known to be zero.

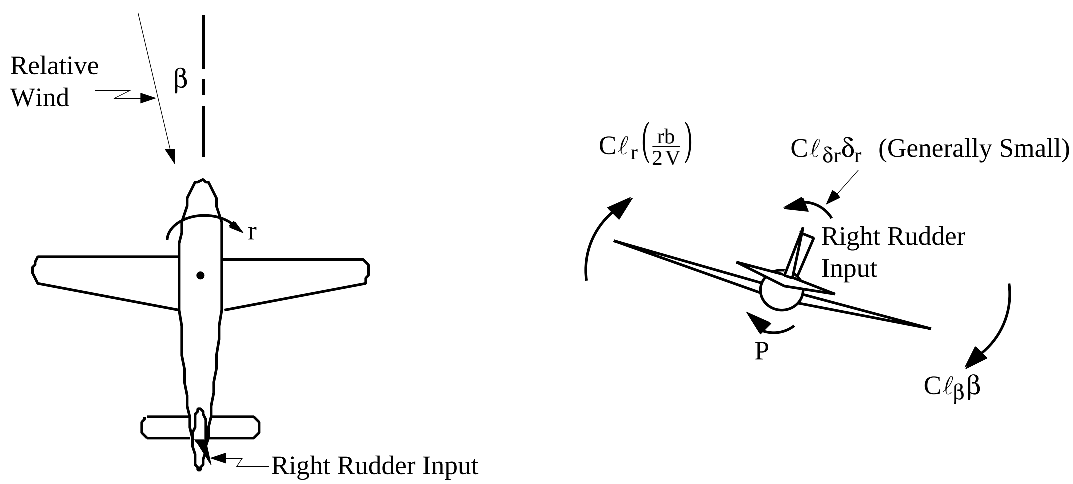


Figure 5.53
Typical Airplane Response to a Rudder-Only Input

In the steady (constant bank angle) rudder-only turn, an expression for the rudder position required for equilibrium may be obtained as a function of the nondimensional yaw rate, $\frac{rb}{2V}$. For a symmetrical airplane $\left(C_{n_0} = 0, C_{\ell_0} = 0, C_{y_0} = 0\right)$, no roll due to

[12] This can be proved mathematically, however, the derivation will not be presented here.

rudder deflection $\left(C_{\ell_{\delta_r}} = 0\right)$, and no influence of inertia terms on the yawing and rolling moments, the equilibrium equations for a rudder-only turn may be written as follows:

SIDEFORCE $\quad C_{y_\beta}\ \beta + C_{y_{\delta_r}}\ \delta_r + C_{y_r}\ \frac{rb}{2V} + \frac{W}{qs}\sin\phi = \frac{W}{g}\ Vr$ *eq 5.67*

YAWING MOMENT $\quad C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_r}\ \frac{rb}{2V} = 0$ *eq 5 .68*

ROLLING MOMENT $\quad C_{\ell_\beta}\ \beta + C_{\ell_r}\ \frac{rb}{2V} = 0$ *eq 5.69*

From the yawing moment equation, the rudder requirement can be derived as follows:

$$\delta_{r_{Equilibrium}} = -\frac{1}{C_{n_{\delta_r}}}\left\{C_{n_r}\left(\frac{rb}{2V}\right) + C_{n_\beta}\ \beta\right\} \qquad \textit{eq 5.70}$$

In order to eliminate the sideslip variable, an expression for sideslip is obtained from the rolling moment equation:

$$\beta = -\frac{C_{\ell_r}}{C_{\ell_\beta}}\ \frac{rb}{2V} \qquad \textit{eq 5.71}$$

By substituting this expression for sideslip, and by appropriate manipulation and differentiation, the rudder required in a steady, rudder-only turn may be expressed as:

$$\frac{d\delta_{rEquilibrium}}{d\left(\frac{rb}{2V}\right)} = \frac{1}{C_{n_{\delta_r}}\ C_{\ell_\beta}}\left\{C_{\ell_r}\ C_{n_\beta} - C_{\ell_\beta}\ C_{n_r}\right\} \qquad \textit{eq 5.72}$$

Notice that the term in braces is again the same combination of stability derivatives which dictated the nature of the spiral mode. If this term is positive, the spiral mode is divergent, and the pilot will be required to hold "top-rudder" or "out-of-turn" rudder-only turn, and vice-versa.

5.2.3.3 COORDINATED TURNS

In the steady (constant bank angle) coordinated turn, expressions for rudder and aileron position and sideslip angle required for equilibrium can be derived as functions of the nondimensional yaw rate, $\frac{rb}{2V}$. Assuming a symmetrical airplane and no influence of inertia terms on the yawing and rolling moments, the equilibrium equations for a coordinated turn may be written as follows:

SIDEFORCE
$$C_{y_\beta}\ \beta + C_{y_{\delta_r}}\ \delta_r + C_{y_r}\left(\frac{rb}{2V}\right) = 0$$
$$\left(\text{Since } \frac{W}{qS}\ \sin\ \phi = \frac{W}{g}\ \frac{Vr}{qS}\right) \qquad \textit{eq 5.73}$$

YAWING MOMENT
$$C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_{\delta_a}}\ \delta_a + C_{n_r}\left(\frac{rb}{2V}\right) = 0 \qquad \textit{eq 5.74}$$

ROLLING MOMENT
$$C_{\ell_\beta}\ \beta + C_{\ell_{\delta_r}}\ \delta_r + C_{\ell_{\delta_a}}\ \delta_a + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0 \qquad \textit{eq 5.75}$$

The rudder requirement in the coordinated turn may be obtained by a determinant solution which reduce to the following if we assume $C_{n_{\delta_a}} = 0$:

$$\delta_{r_{\text{Equilibrium}}} = \frac{\begin{vmatrix} C_{y_\beta} & -C_{y_r} \\ C_{n_\beta} & -C_{n_r} \end{vmatrix}}{\begin{vmatrix} C_{y_\beta} & C_{y_{\delta_r}} \\ C_{n_\beta} & C_{n_{\delta_r}} \end{vmatrix}}\left(\frac{rb}{2V}\right) \qquad \textit{eq 5.76}$$

Solving the determinant, rearranging terms and differentiation yields the following expression:

$$\frac{d\delta_{r_{\text{Equilibrium}}}}{d\left(\frac{rb}{2V}\right)} = -\frac{C_{n_r}}{C_{n_{\delta_r}}}\left\{\frac{C_{y_\beta} - \frac{C_{y_r}}{C_{n_r}}\ C_{n_\beta}}{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\ C_{n_\beta}}\right\} \qquad \textit{eq 5.77}$$

The part of the above expression contained in braces is generally very nearly equal to unity. Therefore, the rudder requirement in a coordinated turn is needed mainly to overcome the yaw rate damping, C_{n_r}, which acts in opposition to the established yaw rate. Obviously, the rudder requirement varies inversely with rudder control power, $C_{n_{\delta_r}}$. For a coordinated turn to the right (positive yaw rate), an analysis of the last expression reveals that the rudder requirement is negative (trailing edge right), which is generated by right rudder pedal deflection. Thus, the pilot applies rudder pedal deflection toward the direction of turn in a coordinated turn; this analytical analysis is substantiated by actual flight experience.

An approximation for the sideslip in the coordinated turn may be obtained from the yawing moment equation by making the following substitution and assumption:

$$\delta_{r_{Equilibrium}} = \frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right)$$

$$C_{n_{\delta_a}} = 0 \qquad \textit{eq 5.78}$$

From the yawing moment equation:

$$C_{n_\beta}\,\beta + C_{n_{\delta_r}}\left\{-\frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right)\right\} + C_{n_r}\left(\frac{rb}{2V}\right) = 0$$

$$\beta = \frac{\left(C_{n_r} - C_{n_r}\right)\left(\frac{rb}{2V}\right)}{C_{n_\beta}}$$

$$\beta = 0 \qquad \textit{eq 5.79}$$

Thus, in the coordinated turn, the sideslip is approximately zero. In reality, the sideslip in the coordinated turn is generally never quite equal to zero, however, it is usually very small.

The aileron requirement in the coordinated turn may be obtained from the rolling moment equation by making the following substitutions:

$$\delta_{r_{Equilibrium}} = -\frac{C_{n_r}}{C_{n_{\delta_r}}}\left(\frac{rb}{2V}\right) \qquad \beta_{Equilibrium} = 0 \qquad \textit{eq 5.80}$$

Thus:

$$C_{\ell_{\delta_r}} \left\{ -\frac{C_{n_r}}{C_{n_{\delta_r}}} \left(\frac{rb}{2V}\right) \right\} + C_{\ell_{\delta_a}} \delta_{a_{Equilibrium}} + C_{\ell_r} \left(\frac{rb}{2V}\right) = 0$$

$$\frac{d\delta_{a_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = -\frac{C_{\ell_r}}{C_{\ell_{\delta_a}}} \left\{ 1 - \frac{C_{\ell_{\delta_r}} C_{n_r}}{C_{\ell_r} C_{n_{\delta_r}}} \right\} \qquad \textit{eq 5.81}$$

The portion of the last expression contained in braces is generally very nearly equal to unity, unless the airplane has a high vertical tail. Therefore, the aileron requirement in the coordinated turn is generated by the necessity to counteract the rolling moment due to yaw rate, C_{ℓ_r}. For a coordinated turn to the right (positive yaw rate), an analysis of the last expression reveals that the aileron requirement is positive (right aileron trailing edge down), which is created by left cockpit control deflection. Thus, the pilot applies lateral cockpit control deflection opposite from the turn direction in the coordinated turn. In actuality, this aileron requirement is generally very small.

5.2.4 Influence of Center of Gravity Movement

Movement of the airplane center of gravity (CG) has rather small effects on lateral-directional characteristics, particularly when compared with the profound effects experienced in the longitudinal case. Of course, yawing moment contributions are slightly modified by CG movements which change the magnitude of the moment arm. However, the parameter measured in flight test work as an indication of the yawing moments generated by the sideslip, $\frac{d\delta_r}{d\beta}$, is not noticeably affected by CG movement. This is due to the fact that a change in CG position alters the rudder control power, $C_{n_{\delta_r}}$, at approximately the same rate that directional stability, C_{n_β} is modified. Therefore, the center of gravity position utilized for flight test investigations of lateral-directional characteristics is not critical; however, if possible, the most aft operational CG positions are generally utilized.

5.3 TEST PROCEDURES AND TECHNIQUES-LATERAL-DIRECTIONAL FLYING QUALITIES

5.3.1 Preflight Procedures

Any rigorous flying qualities investigation must begin with thorough preflight planning. The lateral-directional area of investigation is no exception to this rule. Only by clearly defining the purpose and scope of the investigation can a plan of attack or method of test be formulated.

Preflight planning must begin with research. All available information concerning the airplane's lateral-directional characteristics should be reviewed. The conformation of the airplane should be studied in relation to its influence on lateral-directional flying qualities. Of course, preflight planning must include a thorough study of the lateral-directional control system-encompassing stability and control augmentation if installed. Much useful information may be gained from conferences with pilots and engineers who are familiar with the airplane.

The particular tasks to be investigated must be determined and clearly understood by the flight test team. These tasks, of course, depend on the mission of the airplane. It is particularly important during the investigation of lateral-directional flying qualities to determine if these tasks will be performed in instrument flight (IFR) conditions or merely visual flight (VFR) conditions in operational use. Certain undesirable characteristics can be accepted for VFR missions, but are not acceptable for IFR missions. The availability of an automatic flight control system or autopilot for pilot relief must also be considered. If stability or control augmentation is installed, the consequences of augmentation failures must be given due consideration.

Knowledge of the mission tasks allows determination of appropriate test conditions-configurations, altitudes, centers of gravity, trim airspeeds, and gross weights.

Test conditions must be commensurate with the mission environment of the airplane. Center of gravity position is not particularly critical for lateral-directional tests. Tests at normal operational CG positions for a test loading are generally adequate; however, if feasible, the most aft operational CG positions should be utilized. Lateral-directional characteristics may be altered by various combinations of external stores.

Asymmetric store loadings may seriously degrade lateral-directional flying qualities; these conditions should be investigated on any airplane which may carry asymmetric stores in operational use.

The amount and sophistication of instrumentation will depend on the purpose and scope of the evaluation. A good, meaningful qualitative investigation can be performed with only production cockpit instruments and portable instrumentation-hand held force gauge and stopwatch. Automatic recording devices, such as oscillograph, magnetic tape, and telemetry, are very helpful in rapid data acquisition and may be essential in a long test program of quantitative nature. Special sensitive cockpit instruments are also very useful, not only aiding in rapid data acquisition, but also aiding in stabilization for equilibrium test points. The parameters to be recorded and the ranges and sensitivity of test instrumentation will vary somewhat with each test program.

The final step in preflight planning is the preparation of pilot data cards. An example of a lateral-directional stability and control data card is shown in Figure 5.55. Most test pilots desire to modify data cards to their own requirements or construct data cards for each test. At any rate, the data cards should list all quantitative information desired and should be easy to interpret in flight. Blank cards should be used for appropriate qualitative pilot comments.

LATERAL-DIRECTIONAL STABILITY AND CONTROL RECORD		CARD NUMBER
AIRPLANE TYPE	PILOT	PTR-BIS
BUREAU NUMBER	T. O. GROSS WEIGHT	DATE

T. O. CG GEAR DOWN ____ MAC GEAR UP____ MAC	T. O. TIME_________ LAND TIME ___________
EXTERNAL LOAD NG	CONFIGURATION
TRIM AIRSPEED MACH _______ POWER______	ALT ________ TRIM DIR_______ LAT_______
BREAKOUT FRICTION — CONTROL SYSTEM MECHANICAL CHARACTERISTICS D.R._____ CN_____ LAT_____CN _____	DIR ________________ LAT _______________
CONTROL SYSTEM OSCILLATIONS D.R._____ CN_____ LAT_____CN _____	CENTERING DIR ________________ LAT _______________

RUDDER PEDAL DEP	CN	D_R	STEADY HEADING S DESL PS F_R	O_a	F_a	ϕ	F	F_e	IAS ERROR	R. D. R. C.
TRIM SHOT			0		0	0		0		
zR			R		L	L	L			
zL			L		R	R	R			
			R		L	L	L			
			L		R	R	R			
			R		L	L	L			
			L		R	R	R			
F R			R		L	L	L			
F L			L		R	R	R			

FUEL _____________

TRANSIENT METHOD REMARKS

FUEL ________ CN ________

	SPIRAL STABILITY							
ϕ DEGREES	R							FUEL ____________
TIME SECONDS	O	5	10	15	20	25	30	
ϕ DEGREES	L							

METHOD	DUTCH ROLL CHARACTERISTICS CN	PERIOD	HALF-CYCLE AMPLITUDE RATIO	$\frac{\phi}{\beta}$
TRIM SHOT FUEL _______				

EASE OF TRIM TO ZERO FORCES	TRIMMABILITY TRIM SENSITIVITIES	TRIM RATES
LONG TERM TRIM HOLDING	LOCATION OF TRIM DEVICES	

Figure 5.54
Lateral-Directional Stability and Control Record

5.3.2 Flight Test Techniques

5.3.2.1 THE QUALITATIVE PHASE OF THE EVALUATION

The mere measurement of lateral-directional stability and control characteristics, although important, will have little meaning unless the test pilot can relate the influence of these characteristics on mission accomplishment. Therefore, a portion of the lateral-directional flying qualities evaluation must be devoted to performing or simulating the mission tasks under investigation. While performing these tasks, the test pilot forms the essential qualitative opinion of the lateral-directional flying qualities and should assign a "pilot rating." This opinion will be based on the amount of attention and effort the pilot must devote to "just flying the airplane." Due regard should be given during this phase of the evaluation to the following considerations:

1. Whether the mission task will be performed in VFR and IFR weather or strictly VFR conditions.

2. The amount of time and effort the pilot must devote to duties other than "just flying the airplane" - duties such as setting up a weapons system, coordinatingmultiplane tactics, communicating with other aircraft or a controlling station, etc.

3. The availability of an autopilot or automatic flight control system for pilot relief.

4. If stability or control augmentation are installed, the consequences of their failure.

The test pilot's qualitative opinion of the airplane's lateral-directional flying qualities in relation to selected mission tasks is the most important information to be obtained.

5.3.2.1.1 Lateral-Directional Trimmability

Trimmability is conveniently evaluated during the qualitative phase of the investigation. Lateral-directional trimmability is indicated by the ease with which lateral and directional control forces are reduced to zero in wings-level, steady heading flight and the ability of the airplane to maintain a trimmed condition. It is directly influenced by the major static lateral-directional stability characteristics: directional stability, dihedral effect, and sideforce characteristics. If the airplane is difficult to trim in wings-level, steady-heading flight and does not readily maintain the trimmed condition, the reason could very well be weak or negative static directional stability and/or dihedral effect. Weak sideforce characteristics may result in the pilot inadvertently trimming the airplane into a wings-level, steady-heading sideslip if the ball of the needle-ball instrument is the only cue available for trimming. Of course, lateral-directional trimmability is also influenced by the rate of operation and sensitivity of the lateral and directional trim system as well as the physical location and ease of operation of the trim devices in the cockpit.

Trimmability, and especially long-term trim holding, will be affected by the mechanical characteristics of the airplane's control system. For example, an airplane with poor, nonabsolute lateral control centering will be difficult to trim to wings-level, steady flight, and will not readily return to its trimmed condition following a lateral input.

The trimmability determination is qualitative. The test pilot should attempt to trim the airplane precisely in wings-level, steady-heading flight by using the ball of the needle-ball instrument as a cue for trimming. The inherent sideslip in the trimmed condition should then be noted if a sideslip indicator is available. The controls of the airplane should then be released to evaluate the airplane's ability to maintain the trimmed condition.

After trimming the airplane, the test pilot may utilize two "qualitative test techniques" which should provide further information about the later-directional characteristics. (These techniques may also be utilized for a quick evaluation of a particular configuration if insufficient flight time is available for quantitative measurements.)

5.3.2.1.2 Rudder-Only Turns

The airplane should be trimmed for wings-level, steady-heading flight. Maintaining trim constant, the pilot then enters and maintains turns by use of rudder-only inputs. The bank angles utilized during this evaluation should not be excessive since the directional control system alone cannot be expected to generate and maintain turns at large bank angles where the turn is manifested more by pitching motion than by yawing motion. Turns with bank angles up to 30 degrees are generally considered adequate. Airspeed should be maintained at trim with longitudinal control or trim inputs (if necessary) and allowed to vary for a portion of the evaluation. The lateral control system floating characteristics should be noted, if appropriate. If aileron float is observed, rudder-only turns should be performed with the ailerons restrained in the trim position as well as with controls free.

The information available from the rudder-only turn test is considerable and is presented below. The information generally considered to be most important is indicated by an asterisk.

*1. Suitability of the directional control system as an alternate lateral control system. The pilot may desire or may be required to use rudder-only turns in certain circumstances, such as:

    a. Cruising flight during which the pilot's hands are occupied with other tasks.

    b. Flight conditions during which lateral control inputs generate uncomfortable yawing moments.

    c. Emergency situations involving lateral control system malfunctions.

*2. Strength of the dihedral effect, as indicated by the rolling motion when sideslip is induced by the rudder input. However, the test pilot must keep in mind that the roll response to a rudder input depends not only on dihedral effect, but also on the rolling moments generated by yaw rate $\left(C_{\ell_r}\right)$, and rudder deflection $\left(C_{\ell_{\delta_r}}\right)$. The relative significance of C_{ℓ_r} increases with increasing wing span and decreasing speed. At typical STOL approach speeds, it may be a very important derivative.

3. Qualitative indication of Dutch roll frequency and damping. Since the roll response to a rudder input is manifested through the Dutch roll mode of motion, the pilot must excite the Dutch roll during rudder-only turns.

4. Nature of the spiral mode of motion, as indicated by the rudder position required in the steady (constant bank angle) rudder-only turn.

5.3.2.1.3 Aileron-Only Turns

The airplane should be trimmed for wings-level, steady heading flight. Maintaining trim constant, the pilot then enters and maintains turns with aileron-only inputs. Turns with bank angles up to 45 degrees are generally considered adequate. The rate of control input and amount of control deflection should be varied. Airspeed should be maintained at trim with longitudinal control or trim inputs (if necessary). The directional control system floating characteristics should be observed, if appropriate. If rudder float is observed, aileron-only turns should be performed with rudders fixed and rudders free.

The information available from the aileron-only turn is presented below, with the information generally considered to be most important indicated by an asterisk.

*1. Ease of entering and maintaining coordinated turns. During vigorous maneuvering tasks requiring rapid bank angle changes and turn reversals, the airplane which requires little rudder coordination will, all else being equal, be more acceptable to the pilot than the airplane which requires extensive use of the rudder.

2. Yawing moments generated by lateral control deflection and roll rate, as indicated by the motion of the airplane's nose, the turn needle, the heading indicator, or a sideslip gauge during the entry into the aileron-only turn.

3. Dutch roll excitation during a bank angle control task with aileron-only inputs.

4. Nature of the spiral mode of motion, as indicated by the aileron requirement in the steady (constant bank angle) aileron-only turn.

After performing the qualitative phase of the evaluation, the test pilot should have some ideas as to the particular characteristics which make the airplane easy or difficult to fly. Use of the quantitative techniques described below hopefully allows the test pilot to substantiate his qualitative opinion. The results of all the qualitative and quantitative tests must be <u>correlated</u> in order for the test pilot to accurately analyze the lateral-directional characteristics.

5.3.2.2 MEASUREMENT OF THE MECHANICAL CHARACTERISTICS OF THE LATERAL-DIRECTIONAL CONTROL SYSTEM

Mechanical characteristics of the lateral-directional control system have a major influence on lateral-directional flying qualities. The mechanical characteristics to be evaluated are defined as follows:

1. Breakout, including friction: The lateral or directional cockpit control force from the trim position required to initiate movement of the respective control surface.

2. Freeplay: The lateral or directional cockpit control motion from the trim position required to initiate movement of the respective control surface.

3. Centering: The ability of the lateral or directional control system to return to and maintain the original trimmed position when released from any other position.

4. Control System Oscillations: Oscillations in the lateral-directional control system resulting from external or internal disturbances.

5.3.2.2.1 Breakout Forces, Including Friction

Friction in the lateral and directional control system is unavoidable, however, it should be kept as low as possible. Some amount of breakout force is generally beneficial, but too much results in undesirable characteristics. Breakout forces allow the pilot to rest his hand and feet on the control stick and rudder pedals without introducing inadvertent lateral and directional control inputs - this characteristic is particularly important in turbulent air. However, breakout forces must be suitably matched to the lateral and directional control forces experienced after overcoming the breakout forces.

It should be obvious that breakout force can never be measured alone, unless there is zero friction force. Therefore, breakout forces, including friction, are measured at the trimmed conditions of the test. Directional breakout forces, including friction are measured or estimated in flight by carefully stabilizing at the trim condition, then applying slow and smooth rudder force inputs until movement of the rudder control surface is detected.

Movement of the rudder can be detected by visually observing rudder movement, use of a rudder position indicator, or by observing airplane response. Lateral breakout forces, including friction, are measured in flight with the hand held force gauge by carefully stabilizing at the trim condition, then applying slow and smooth lateral force inputs until movement of the aileron is detected. This movement can be detected by visually observing airplane response. When the airplane response is utilized as a cue for rudder or aileron movement, caution must be exercised because the airplane will require a finite time interval to respond to the control surface movement. If automatic recording devices are utilized, breakout forces, including friction, may be measured from the recording traces.

Breakout, including friction, may be measured on the ground for airplanes equipped with irreversible control systems where control forces are merely functions of control deflection. However, ground measurements should be checked with inflight measurements. It is obvious that inflight measurements at the trim condition are the only means of accurately determining breakout, including friction, for the reversible control system.

5.3.2.2.2 Freeplay

Freeplay is the lateral and directional control systems should be as small as possible. Excessive freeplay may cause difficulty in performing precise maneuvers such as instrument approaches and tracking. Freeplay, expressed in inches or degrees of lateral and directional cockpit control movement, is measured in flight at the trim condition much the same as breakout, including friction, was measured. Ground measurements may also be made for irreversible control systems.

5.3.2.2.3 Centering

The lateral and directional control systems should exhibit positive centering in flight at any stabilized trim condition. Poor centering can result in objectionable tracking characteristics and/or large departures in sideslip or bank angle without constant pilot attention to airplane control. Centering is qualitatively evaluated in flight at the trim condition by smoothly displacing the lateral and directional cockpit controls to various positions and observing their motion upon release. Irreversible control system centering characteristics may be evaluated on the ground.

5.3.2.2.4 Control System Oscillations

Oscillations in the lateral and directional control systems, initiated by either external perturbations or pilot action, should not result in objectionable oscillations in sideslip or roll rate, nor should there be any objectionable structural vibrations created. Damping of the respective control systems is measured in flight by abruptly deflecting and releasing the cockpit controls and observing the resulting motion in the control surface or the cockpit controls. These abrupt inputs may be described as rudder "kicks" and lateral stick "raps." Use of automatic recording devices or a cockpit mounted control position indicator aids in data acquisition. If these are not available, and the pilot is unable to visually observe the aileron or rudder control surfaces, the test pilot must resort to observing the motion of the cockpit control stick or rudder pedals. No objectionable oscillations of either the cockpit controls or the airframe control surfaces should be present during maneuvering flight or in turbulence. It must be remembered that motion of the cockpit controls may or may not be good indications of the motion of the aileron or rudder surfaces. Irreversible control system oscillation characteristics may be checked on the ground; however, these characteristics should be evaluated in flight to insure there is not coupling between airplane motion and control system dynamics.

5.3.2.3 MEASUREMENT OF STATIC-LATERAL-DIRECTIONAL STABILITY CHARACTERISTICS

The steady heading sideslip test technique is conveniently utilized to obtain important relationships which have a major influence on lateral-directional flying qualities. <u>It is not performed primarily to determine the feasibility of the maneuver in operational use</u>, although this determination is obviously a by-product. The information available from the steady heading sideslip test is considerable and can be divided into <u>primary</u> and <u>secondary</u> areas of importance.

The primary parameters to be obtained from the steady heading sideslip test are rudder position, rudder force, aileron position, aileron force, and bank angle. The rudder position and rudder force variations with sideslip angle, sometimes referred to as directional stability, rudder-fixed and directional stability, rudder-free, respectively, have a major influence on the pilot's opinion of the directional "stiffness" of the airplane. These variations provide absolutely no information about the <u>magnitude</u> of the static directional stability derivative, C_{n_β}, unless numerous other parameters, such as directional control power, $C_{n_{\delta_r}}$, are known. However, if the variations are positive, such that right rudder pedal force and trailing edge right rudder deflection are required in left sideslips, and vice versa, the directional stability derivative, C_{n_β}, is known to be at least positive. Positive rudder position and rudder force variations with sideslip angle are a basic airplane design requirement which allows the pilot to perform various mission tasks without entering uncontrollable - and possibly catastrophic - flight conditions. In addition, positive variations contribute to good lateral-directional trimmability and maintenance of the trimmed condition. However, rudder control forces and rudder positions required to induce or control sideslip should not be excessive within the range of sideslip angles required in normal or emergency operational conditions. The pilot is continually confronted with situations during which he must use or control sideslip with rudder control or trim inputs. Some of these situations are: crosswind take-offs and landings, flight with asymmetric external stores or asymmetric power, rudder-only turns, rudder coordination during turns and rolling maneuvers, etc. The particular rudder force and position variations desired in any airplane depend on the mission of the airplane and the multitude of pilot tasks required in mission accomplishment. Plots of rudder force and position versus sideslip angle should be essentially linear and should exhibit stable local gradients within a reasonable sideslip angle variation from trim. For larger sideslip angles, an increase in rudder deflection should always be required for an increase in sideslip angle. Lightening of the rudder forces at high sideslip angles may be acceptable; however, the forces should never reduce to zero or reverse, i.e., no "rudder lock" should be encountered.

Aileron position and aileron force variations with sideslip angle, sometimes referred to as control-fixed dihedral effect and control-free dihedral effect, respectively, have a major influence on lateral-directional trimmability and the ease with which the pilot can control bank angle with rudder inputs. These variations provide absolutely no information about the magnitude of the dihedral effect, C_{ℓ_β}, unless numerous other parameters, such as aileron control power, $C_{\ell_{\delta_a}}$, are known. However, if the variations are negative, such that left lateral control force and left lateral cockpit control position are required in left sideslips, and vice versa, the stability derivative, C_{ℓ_β}, is known to exhibit a negative sign. This results in positive dihedral effect, i.e., the airplane tends to roll opposite to the induced sideslip. Some degree of positive dihedral effect, as indicated by aileron control force and aileron position variation with sideslip angle, is desirable for satisfactory lateral-directional flying qualities. However, positive dihedral effect should not be so strong as to require excessive aileron force or deflection to control bank angle in sideslips. In addition, the variation of aileron control force and position with sideslip angle should be essentially linear.

Bank angle variation with sideslip angle in steady heading sideslips is indicative of the sideforce characteristic. Positive sideforce characteristics, as indicated by left bank angle requirement in left sideslips, and vice versa, are necessary for satisfactory lateral-directional flying qualities. The usefulness of the ball of the needle-ball instrument as a valid reference for trimming the airplane (with little inherent sideslip) in wings-level flight and for performing coordinated turns (with little sideslip) depends on sideforce characteristics. Airplanes exhibiting weak or zero bank angle variation with sideslip angle (in steady heading sideslips) are easily inadvertently trimmed and flown in wings-level, ball-centered steady heading sideslips. The pilot will be perfectly happy with this state of affairs until the increased drag caused by the inherent sideslip begins to upset his fuel-used calculations in a long-range-cruise task, or until difficulty is experienced in aligning the airplane longitudinally with the runway or carrier centerline during night or low-visibility approaches. Additionally, weak or zero sideforce characteristics result in the <u>ball</u> and "seat-of-the-pants" <u>feel</u> losing their significance as indications of <u>turn perfection</u>; as a consequence, large sideslip angles may be developed even though the pilot keeps the ball centered and feels no sidearm accelerations during turns. In general, an increase in right bank angle should accompany an increase in right sideslip and an increase in left bank angle should accompany an increase in left sideslip. It is theoretically possible for a symmetrical airplane to possess negative trimmed sideforce characteristics, but such an airplane is unlikely to be encountered in practice, though airplanes in which the sideforce characteristics are very weak or approach zero do exist. On the other hand, asymmetric thrust or store drag can easily generate a requirement for bank away from the sideslip angle for equilibrium.

Secondary parameters obtained from the steady heading sideslip test are longitudinal control force, rate of descent, and indicated airspeed error. Pitching moments or longitudinal trim changes generated by sideslip are manifested to the pilot through the longitudinal control force (and incremental change in elevator or longitudinal control position) required to maintain airspeed constant in the steady heading sideslip. Excessive longitudinal trim changes with sideslip angles normally utilized in operational usage would place excessive demands on pilot attention and coordination. Large rates of descent and indicated airspeed errors with sideslip angles employed in operational flight procedures would be undesirable, if not dangerous, characteristics.

Two test techniques will now be presented for evaluating static lateral-directional stability characteristics in steady heading sideslips.

5.3.2.3.1 Stabilized Steady Heading Sideslips

The stabilized steady heading sideslip test technique is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, a "trim shot" should be taken Record appropriate trim data such as power setting, trim settings, trim lateral and directional control positions, fuel quantity, rate or climb or descent, and inherent sideslip angle.

2. The first sideslip at each predetermined rudder pedal deflection is performed solely to determine if any indicated airspeed error is induced by sideslip. This is accomplished by establishing the sideslip, stabilizing, then quickly returning the controls to trim and noting any increase or decrease in airspeed as the sideslip angle decreases to trim. Airspeed errors which do not exceed $\pm$ 2% V_{Trim} may be neglected. Indicated airspeed is varied to compensate for significant errors in subsequent sideslips. The proper airspeed to utilize is quite easily recognized during the test.

3. Establish each stabilized steady heading sideslip by smoothly applying the predetermined rudder pedal deflection while simultaneously feeding in lateral and longitudinal control inputs to maintain <u>steady heading</u> and the correct airspeed. If the airplane exhibits positive directional stability, positive dihedral effect, and positive sideforce characteristics, the pilot will observe that, in the stabilized steady heading sideslip, <u>right</u> rudder inputs, <u>left</u> lateral control inputs, and <u>left</u> bank angle are required with <u>left</u> sideslip angles, and <u>opposite</u> parameters are required with <u>right</u> sideslip angles.

4. Maintenance of steady heading in the stabilized sideslip is critical if valid data are to be obtained from this test. There are several cues available to the pilot for maintenance of a steady heading. There are the turn needle, the directional indicator, and visual references. A steady heading sideslip may best and most easily be established by using primarily external references with final cross checking against the directional gyro and the turn needle.

5. If using automatic recording devices, take a picture of the stabilized steady heading sideslip. Appropriate cockpit data should then be recorded. Since the parameters to be recorded are considerable, a suggested order of noting and recording data is as follows:

   a. Rudder pedal force. If estimating rudder forces, mentally note the force as soon as the sideslip is stabilized before the leg becomes fatigued.

   b. Rudder position (either estimated or from control position indicators).

   c. Lateral control force. This parameter may be measured with a hand held force gauge.

   d. Aileron position (if available) or lateral cockpit control position. Cockpit control position may be measured roughly with various portable instrumentation - tape measure, calibrated string, calibrated yoke, etc. - if no cockpit indicator is available.

   e. Bank angle.

   f. Sideslip angle. (A quite accurate method for estimating sideslip if no sideslip indicator is available is presented in paragraph 6.)

   g. Longitudinal control force and vertical speed. Longitudinal control forces may be measured with the hand held force gauge.

6. If a sideslip indicator is not available in the cockpit, the directional indicator may be used to estimate sideslip angels. While stabilized in the steady heading sideslip, note the airplane heading after recording all parameters except sideslip. Then, quickly return the controls to trim and stabilize in the original wings-level condition. Again, note the airplane heading. The difference between the heading noted in the sideslip and the heading noted after releasing the sideslip is approximately the sideslip angle induced in the steady heading sideslip.

7. Steady heading sideslips are normally performed at increments of one-quarter rudder deflection, alternating direction of rudder inputs until full rudder deflection sideslips are performed in both directions. Sideslip angles should be increased until:

    a. Full rudder pedal deflection is reached.

    b. Rudder forces reach 250 pounds.

    c. Maximum aileron control or deflection is reached.

Obviously, if "rudder lock" or other unusual circumstances are encountered, the test may be terminated short of these limits.

8. Altitude should be within $\pm$ 2000 feet of the predetermined test altitude during the measurements. Since increased drag will be generated on the airplane during the sideslips, it may be expeditious to start a series of sideslips above the predetermined test altitude.

5.3.2.3.2 Transient Steady Heading Sideslips

If automatic recording devices are available, the transient technique may be utilized to quickly obtain static lateral-directional characteristics. It is also a good "quick look" qualitative technique even without automatic recording devices. It is performed as follows:

1. Stabilize and trim carefully at the desired flight condition. Record a "trim shot" with the automatic recording devices. Record appropriate cockpit data.

2. Actuate the automatic recording devices and very smoothly and steadily enter the steady heading sideslip. Using constant rate rudder inputs, increase sideslip slowly, simultaneously applying lateral control inputs and banking in order to maintain steady heading. Use longitudinal control inputs, if necessary, to maintain constant trim airspeed.

3. Increase the rudder input to full deflection (or other limiting factors), then decrease rudder input at the same slow rate. After returning the airplane to the original trim condition, continue by applying rudder in the opposite direction to full deflection, then back to trim.

4. Actuate the event marker at various sideslip angles or rudder deflections, if desired, to aid in data reduction. Deactivate the instrumentation after returning to the original trim conditions.

5. For the optimum results from this test, the rate of change of sideslip must be very slow; $\frac{1}{2}$ degree per second or less yields excellent data.

6. Maintenance of steady heading is again critical for this test. The test pilot must continually cross-check all references available in order to keep the heading constant throughout the process.

5.3.2.4 MEASUREMENT OF DYNAMIC LATERAL-DIRECTIONAL STABILITY CHARACTERISTICS - THE SPIRAL MODE

5.3.2.4.1 Spiral Stability

Characteristics of the spiral mode of motion have some influence on overall lateral-directional flying qualities, although the influence is generally small in relation to other contributions. The pilot generally is satisfied if the spiral mode is neutral, convergent, or slightly divergent. However, if the spiral is very divergent, such that bank angle changes resulting from external disturbances or inadvertent control inputs build rapidly, the pilot is required to devote some attention to controlling the spiral motion.[13] Therefore, the time

[13] Obviously, an automatic flight control system or autopilot can relieve the pilot of this duty.

duration which the pilot can devote to other duties are correspondingly shortened. A very divergent spiral mode can make long range cruising flight and instrument approaches or departures extremely frustrating for the pilot, particularly in turbulent air.

A divergent spiral mode normally is accompanied by longitudinal trim changes resulting from airspeed increases. These airspeed changes may cause corresponding changes in some speed-dependent stability derivatives $\left(C_{\ell_\beta}, C_{\ell_r}\right)$ which may alter the spiral characteristics. Different spiral characteristics may be observed if airspeed is allowed to vary instead of restraining it at trim.

Propeller-driven airplanes may exhibit <u>different</u> spiral characteristics in left and right turns. This phenomenon may be attributed to various peculiarities of propeller-power flight, such as the manner in which the slipstream impinges on the components of the airplane and the change in slipstream pattern with sideslip direction.

The characteristics of the spiral mode to be determined are the nature of the motion (convergent, neutral, or divergent) and the rate of convergence or divergence (if applicable). These characteristics may be obtained very simply from a time history of bank angle after the spiral motion is excited. The test is performed as follows:

1. Stabilize and trim very precisely in the desired configuration at the desired flight condition.

2. Restrain the lateral control rigidly in the trim position and establish a small bank angle (at least 5 degrees, but not more than 20 degrees) by one of the following means: a very small rudder input, a slight power reduction on one engine of a multiengined airplane, opening a cowl flap or oil cooler door on one engine of a multiengined airplane. Lateral control inputs are usually not used to establish the bank angle because of the difficulty in returning the lateral control to trim and the subsequent significant rolling moments generated. However, this is not a hard and fast rule as in some airplanes it may be easier to ensure the return of the ailerons to a precise trimmed position than to do the same with the rudder; for example, if the centering is obviously better in the aileron circuit. Clearly the important point is to minimize, or eliminate if possible, any contamination of the spiral mode with residual control deflections. After establishment of the

steady bank angle, return the rudder pedals, power setting, cowl flap, or oil cooler to the original trim position and release the cockpit controls. Commence recording a time history of bank angle, utilizing a 60 second sweep stopwatch and the attitude gyro. When sufficient data is obtained, recover to wings-level trim conditions and repeat the procedure with opposite bank angle.

3. By releasing the cockpit controls completely, the pilot observes the combined effects of the spiral mode, longitudinal trim change, and control system characteristics on the spiral motion. This motion is often seen in the operational environment when the pilot releases the control stick to perform other tasks in the cockpit. If pure spiral mode data is desired, trim airspeed should be maintained with longitudinal control inputs.

4. Precise initial trim, particularly lateral trim and smooth air are necessary if valid quantitative data are to be obtained from this test.

5.3.2.5 MEASUREMENT OF DYNAMIC LATERAL-DIRECTIONAL STABILITY CHARACTERISTICS - THE DUTCH ROLL MODE

Since the airplane responds in yaw through the Dutch roll mode (just as it responds in pitch through the short period oscillation), every time it is disturbed in yaw, either externally or by pilot inputs, the Dutch roll will be excited. The pilot seldom needs to make appreciable rudder inputs except for certain special conditions such as flight with asymmetric power (or stores), crosswind landing, controlling bank angle by means of rudder, etc. Nonetheless turbulence and pilot control inputs (especially lateral inputs if significant aileron yawing moments exist) will continually excite the Dutch roll inadvertently. Therefore, it is frequently referred to as a "nuisance" or "annoyance" mode. When the Dutch roll is excited, the resulting oscillations in sideslip, bank angle, and lateral motion must be suppressed by one or a combination of the following: aerodynamic characteristics of the airplane, stability augmentation, pilot control inputs. Damping, frequency, and roll-to-yaw ratio of the Dutch roll, or lateral-directional oscillation, have profound effects on overall lateral-directional flying qualities. During critical mission tasks requiring precise flight path control, precise tracking, and/or rapid maneuvering, a satisfactory combination of Dutch roll characteristics is mandatory for satisfactory mission accomplishment.

Unfortunately, difficulty is experienced in defining and describing satisfactory combinations of Dutch roll damping, frequency, and roll-to-yaw ratio. The optimum combination for a particular airplane obviously depends on the mission of that airplane and the various mission tasks involved. Additionally, pilot technique has a large influence on the acceptability of particular characteristics since the pilot can use any combination of lateral and directional control inputs. The pilot's technique thus influences the pilot's opinion. The generalizations which follow concerning the influence of various Dutch roll parameters must be accepted as generalizations only, since they may not be appropriate to a particular situation.

5.3.2.5.1 Dutch Roll Damping

The damping of the Dutch roll motion is probably the most important Dutch roll characteristic to be considered. Over a wide range of Dutch roll frequencies and roll-to-yaw ratios, the pilot will probably find the lateral-directional dynamics acceptable if the Dutch roll motion is well damped. The parameter, ζ_d, is the damping ratio of the second-order lateral-directional, or Dutch roll, motion. Its value strongly affects the time or dynamic response of the airplane to a rudder or aileron input or lateral gusts. The following rationalizations may be made concerning the influence of various Dutch roll damping ratios on lateral-directional flying qualities.

1. For very low or low damping ratios - the lateral-directional oscillation may be easily excited by pilot control inputs or external disturbances. Once excited, the yawing, rolling, lateral translational motion tends to persist for a relatively long period of time. This persistent motion can cause extreme discomfort and induce airsickness in passengers and crew of transport airplanes as well as degrade aiming accuracy for high-altitude bombing tasks in heavy or tactical bombers. When the pilot attempts to maneuver vigorously and simultaneously maintain an air-to-air or air-to-ground tracking picture, the lightly damped Dutch roll motion may completely preclude precise weapons delivery, particularly in rough air. During the landing approach in instrument or visual conditions, serious heading control problems can be generated if the damping ratio of the Dutch roll motion is too low. The pilot will probably desire to attempt to suppress the motion with lateral or directional control inputs if the damping ratio approaches zero. The success the pilot realizes in personally damping the Dutch roll will depend

largely on the frequency of the motion and other factors such as the roll-to-yaw ratio and the phase relationship between the roll and yaw components of the motion.

2. For low to moderate Dutch roll damping ratios - the lateral-directional oscillation may be excited but will be very noticeably damped, although still apparent. The pilot should be able to perform maneuvers involving precise heading and bank angle control with little difficulty attributable to Dutch roll damping. During vigorous maneuvering involving large, repeated lateral and directional control inputs, noticeable and possibly objectionable yawing and rolling oscillations may be generated. The pilot will probably feel no need to augment natural Dutch roll damping with control inputs since the motion subsides fairly rapidly if the controls are merely fixed or released.

3. For moderate to heavy Dutch roll damping ratios - the lateral-directional oscillation may not even be apparent to the pilot. The pilot will probably feel very secure in maneuvering the airplane vigorously since no noticeable oscillations in yaw or roll are generated. If the Dutch roll damping is very heavy, the airplane will be slow responding to rudder inputs or external disturbances. The pilot may find this sluggish directional response objectionable during maneuvering. In addition, heavy damping of the Dutch roll and heavy yaw rate damping (large C_{n_r}) are generally analogous, therefore, the pilot may find that the rudder requirement in coordinated turns is excessive if the Dutch roll is too heavily damped.

5.3.2.5.2 Dutch Roll Frequency

The parameter, ω_{d_d}, is the damped frequency of the second-order, lateral-directional mode of motion. If it is a real, positive number, it is directly related to the frequency, or quickness, with which the airplane responds to a rudder input or lateral gust disturbance. Obviously, this damped frequency has a major influence on lateral-directional flying qualities. However, the damped frequency is dependent on damping ratio as well as the undamped natural frequency. Therefore, Dutch roll characteristics are conveniently expressed in terms of the undamped natural frequency, ω_{n_d}, and, of course, the damping ratio, ζ_d. The following generalizations may be made concerning the effect of various Dutch roll natural frequencies on lateral-directional flying qualities.

1. For very low to low ω_{n_d} values - the pilot may experience large sideslip excursions when yawing moments are generated by lateral control inputs, roll rate, rudder inputs, lateral gusts, etc. This is due to the close relationship between Dutch roll natural frequency and directional stability. Since low ω_{n_d} and low directional stiffness are generally analogous, the pilot will probably expend considerable effort in maintaining a desired heading for low frequency Dutch roll situations. Because of the slow initial response associated with the low frequency motion, the pilot may experience difficulty in determining the steady state or final magnitude of the motion. This lack of predictability makes precise and rapid heading corrections virtually impossible. Because of the low directional stiffness, the pilot will use frequent rudder inputs in order to compensate, i.e., the pilot becomes a stability augmenter. Increasing the Dutch roll damping ratio probably will not improve the situation measurably if the frequency is very low; the primary objectionable feature associated with the very low frequency is not the oscillatory motion but the weak directional stiffness. If the frequency is a bit higher, increasing the Dutch roll damping may improve the situation somewhat. For some missions and mission tasks, very low Dutch roll natural frequencies may be tolerable, as long as at least some level of positive directional stability is present, i.e., $\omega_{n_d} > 0$. If the airplane is always maneuvered slowly and smoothly, the pilot will probably not object to the slow initial response. The very large transport, passenger, or heavy bomber airplanes, with very large yawing moments of inertia, may be characterized by very low Dutch roll natural frequencies. Since these airplanes do not require extensive maneuvering for mission accomplishment, low Dutch roll natural frequencies may have no derogating influence.

2. For medium to high ω_{n_d} values - the response of the airplane for turns and heading corrections is generally satisfactory, while the sensitivity to lateral gusts should not be excessive. Because of the corresponding medium to high directional stiffness, the pilot finds that few rudder inputs are required to control

sideslip. Directional trimmability is also enhanced. Every correction made during the trimming tasks takes less time and comes to a completion faster. This gives the pilot the feeling that he knows exactly what trim correction is necessary. In other words, the airplane's directional trim point is well defined and corrections to the trim point are quickly accomplished.

3. The very high Dutch roll natural frequency - may cause the airplane to be oversensitive and responsive directionally to rudder inputs and lateral gusts. During precise maneuvers requiring close control of heading or nose position, the very high frequency Dutch roll can precipitate annoying, uncomfortable, rapid excursions in sideslip, roll, and lateral translation. If the pilot tries to damp a very high frequency, lightly damped Dutch roll motion with control inputs, he is likely to get out of phase with the rapid oscillations, reinforce them, and drive the motion divergent.

5.3.2.5.3 Roll-to-Yaw Ratio

The parameter, $\frac{\phi}{\beta}$, is the ratio of the bank angle envelope to sideslip angle envelope during the Dutch roll motion, or simply the roll-to-yaw ratio. Roll-to-yaw ratio has some influence on pilot technique during bank angle control tasks and rolling maneuvers, and may significantly influence the pilot's opinion of the maneuvering capabilities of the airplane during these tasks. The degree of roll disturbance or the sensitivity of the airplane in roll to rudder inputs and lateral gusts is directly proportional to this parameter. The following generalizations may be made concerning the influence of various magnitudes of roll-to-yaw ratios on overall lateral-directional flying qualities. (Roll-to-yaw ratio of the Dutch roll motion will be further discussed in a subsequent section on rolling performance and roll handling qualities.)

1. If the roll-to-yaw ratio is low - the Dutch roll motion is manifested more in yawing than in rolling. If the ratio is very low, so that the motion approaches pure "snaking," the response of the airplane to lateral gusts will be largely heading changes. The pilot may feel compelled to control this gust response during maneuvers requiring precise heading control, and the rudders will be the

control utilized. With low roll-to-yaw ratios, the rolling moments generated by yaw rate and sideslip angle excursions will be small, therefore, the Dutch roll influence on rolling performance will probably be small.

2. For medium roll-to-yaw ratios - some rolling motion will be generated by yaw rate and sideslip angle excursions. If significant aileron yawing moments or yawing moments due to roll rate exist, the pilot will probably be compelled to coordinate aileron inputs with rudder inputs to keep sideslip excursions small, minimize oscillatory variations in roll rate, and realize maximum rolling performance from the airplane.

3. If the roll-to-yaw ratio is high - considerable rolling moments will be generated by sideslip and yaw rate excursions. Rolling performance and lateral handling qualities may be seriously impaired unless the pilot utilizes rudder coordination effectively during maneuvering. The airplane will be very responsive and sensitive in roll to lateral gusts and rudder inputs; bank angle response to turbulent air may be very objectionable, particularly during maneuvering which requires precise bank angle control. As the roll-to-yaw ratio increases, the pilot will probably demand increased Dutch roll damping. This is due to the pilot usually being more sensitive to roll response then sideslip response.

5.3.2.6 DUTCH ROLL REQUIREMENTS

The results of pilot opinion investigations based on in-flight and ground simulator tests have revealed that there are combinations of Dutch roll frequency, damping, and roll-to-yaw ratios which provide satisfactory lateral-directional flying qualities. These investigations showed that the product of the damping ratio and the undamped natural frequency, $\zeta_d \omega_{n_d}$, should be within the limits listed in Table 5.1.

Table 5.1

Flight Phase					
Level	Category	Class	Minζ_d *	Min$\zeta_d \omega_{n_d}$ * rad/sec	Min ω_{n_d} rad/sec
1	A (CO and GA)	IV	0.4	-	1.0
	A	I, IV	0.19	0.35	1.0
		II, III	0.19	0.35	0.4**
	B	All	0.08	0.15	0.4**
	C	I, II-C, IV	0.08	0.15	1.0
		II-L, III	0.08	0.10	0.4**
2	All	All	0.02	0.05	0.4**
3	All	All	0	-	0.4**

* The governing damping requirement is that yielding the larger value of ζ_d, except that a ζ_d of 0.7 is the maximum required for Class III.

** Class III airplanes may be excepted from the minimum ω_{n_d} requirement, subject to approval by the procuring activity, if the requirements of 3.3.2 through 3.3.2.4.1, 3.3.5, and 3.3.9.4 are met.

Note: Values of ζ_d etc. in MIL-F-8785C are quoted to two decimal places. Such accuracy may be obtainable with special instrumentation; however, that will not usually be the case and, where trace records cannot be read to better than about 5%, results which imply an accuracy of 0.5% are clearly unjustifiable.

The effects of a high $\frac{\phi}{\beta}$ ratio can sometimes have a significant influence on mission tasks. Relatively large bank angle excursions resulting from sideslip can make tracking tasks or level flight in turbulence very difficult especially in airplanes with low Dutch roll frequency and/or damping. In an attempt to improve the analysis of these problems in airplanes with high $\frac{\phi}{\beta}$ ratios, an additional parameter has been analytically determined from the results of pilot opinion investigations. This parameter is the product of the square of the undamped natural frequency of the Dutch roll and the $\frac{\phi}{\beta}$ ratio, $\omega_{n_d}^2 \left(\frac{\phi}{\beta}\right) d$. If the value of $\omega_{n_d}^2 \left(\frac{\phi}{\beta}\right) d$ is greater than 20 (rad/sec)^2, then the minimum values of $\zeta_d \omega_{n_d}$ as shown in Table I should be <u>increased</u> by the following values:

$$\text{Level 1} - \Delta\zeta_d \ \omega_{n_d} = .014\left(\omega_{n_d}^2 \left|\phi/\beta\right|_d - 20\right)$$
$$\text{Level 2} - \Delta\zeta_d \ \omega_{n_d} = .009\left(\omega_{nd}^2 \left|\phi/\beta\right|_d - 20\right)$$
$$\text{Level 3} - \Delta\zeta_d \ \omega_{n_d} = .005\left(\omega_{nd}^2 \left|\phi/\beta\right|_d - 20\right)$$

with ω_{n_d} in rad/sec. *eq 5.82*

5.3.2.6.1 Techniques for Exciting Dutch Roll Motion for Quantitative Measurements

Two methods will be introduced for obtaining quantitative Dutch roll characteristics. The method utilized for a particular flight test will depend on the characteristics of the airplane, the requirements against which tested, and the preference of the individual test pilot.

The <u>rudder pulsing technique</u> excites the Dutch roll motion nicely, while suppressing the spiral mode if performed correctly. In addition, this technique can be used to develop a large amplitude oscillation which aids in data gathering and analysis, particularly if the Dutch roll is heavily damped. It is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic devices, take a "trim shot."

2. Smoothly apply alternating left and right rudder inputs in order to excite <u>and reinforce</u> the Dutch roll motion. Restrain the lateral cockpit control at the trim condition or merely release it. Continue the cyclic rudder pulsing until the desired magnitude of oscillatory motion is attained, then smoothly return the

rudder pedals to the trim position and release them or restrain them in the trim position. Simultaneously, activate the automatic recording devices and observe and record appropriate parameters.

3. The frequency with which the cyclic rudder inputs are applied depends on the frequency and response characteristics of the airplane. The test pilot must adjust the frequency of rudder pulsing to the particular airplane. The maximum Dutch roll response will be generated when the rudder pulsing is in phase with the airplane motion, and the frequency of the rudder pulses is approximately the same as the natural (undamped) frequency of the Dutch roll.

4. The test pilot should attempt to terminate the rudder pulsing so that the airplane oscillates about a wings-level condition. This should effectively suppress the spiral motion.

5. Obtaining quantitative information on Dutch roll characteristics from cockpit instruments and visual observations requires patience, particularly if the motion is heavily damped. However, if a sensitive <u>sideslip indicator</u> is available in the cockpit, the test pilot should be able to obtain a <u>half-cycle amplitude ratio</u> from the sideslip excursions. From this parameter, an approximate <u>damping ratio</u> can be easily obtained.[‡] The <u>time required</u> for a half or full cycle can be measured with a one - or three-second sweep stopwatch. From these times, the <u>approximate</u> damped period, damped frequency, and undamped natural frequency can be derived, if desired[‡]. If a sideslip indicator is not available, the <u>turn needle</u> of the needle-ball instrument can be observed to obtain approximate half-cycle amplitude ratio and damped period.
6. The roll-to-yaw ratio of the Dutch roll motion is extremely difficult to obtain accurately without automatic recording devices. The motions are generally phased so that the maximum roll and sideslip excursions <u>do not occur simultaneously</u>. Therefore, merely noting the maximum roll excursion to

[‡] See "Analysis of Second Order Responses" in the Introduction to this manual.

maximum sideslip excursion for a given cycle will probably yield erroneous results‡ . However, the test pilot should be able to obtain a rough idea of the roll-to-yaw ratio by observing the path of the airplane wingtip on the horizon during the Dutch roll oscillations. (A wing-mounted boom or other protrusion, or a dot drawn on the side of the canopy may also be used.) If the Dutch roll is manifested in pure "snaking", the roll-to-yaw ratio is, of course, zero and the wingtip will move only in the horizontal plane. If the roll-to-yaw ratio if 1:1, the wingtip will describe a circle on the horizon, etc. (Figure 5.58). If the Dutch roll motion is heavily damped, the wingtip will describe some form of distorted spiral. Under these conditions, the best possible estimate of roll-to-yaw ratio may be no better than "greater than 1:1," "approximately 1:1" or "less than 1:1."

7. If automatic recording devices are available, the entire Dutch roll motion should be recorded and analyzed later for accurate quantitative information.

‡ [13] See “Analysis of Second Order Responses” in the Introduction to this manual.

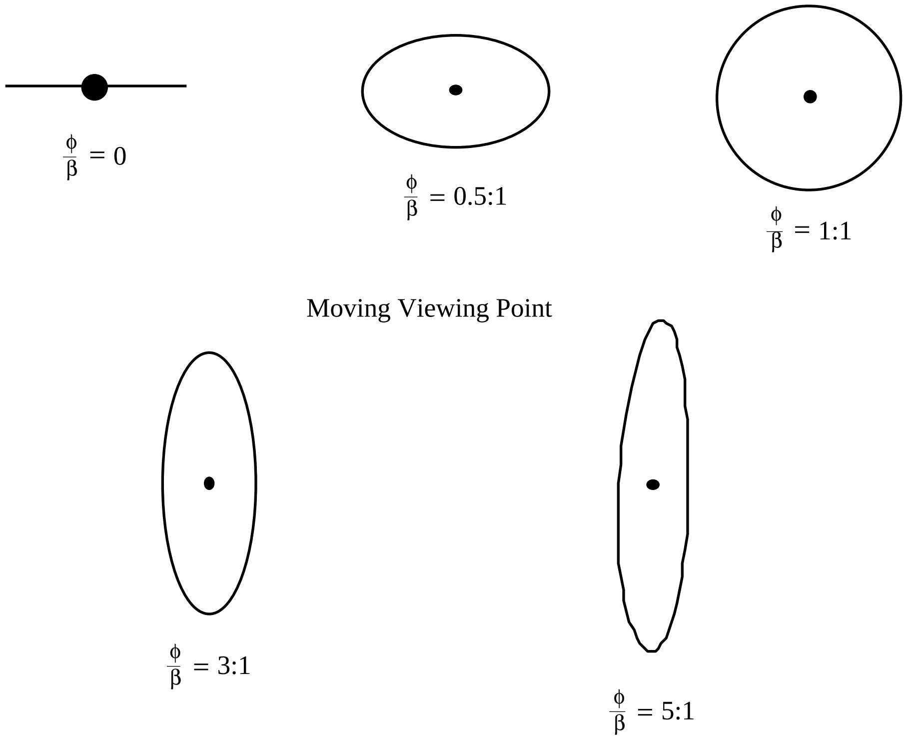


Figure 5.55
Paths of Airplane Wingtip on Horizon During Dutch Roll Motion

The steady sideslip release can also be used to excite the Dutch roll; however, the difficulty in quickly returning the controls to trim and the influence of the spiral mode often precludes the gathering of good quantitative results. The rudder pulsing technique usually produces much better Dutch roll data. The steady sideslip release technique is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, take a "trim shot."

2. Establish a steady heading sideslip of a sufficient magnitude to obtain sufficient Dutch roll motion for analysis. Utilize maximum allowable sideslip, full rudder, or a comfortable rudder force input. Stabilize the sideslip carefully and activate the automatic recording devices. Quickly, but smoothly, return all cockpit controls to trim and release them (controls-free Dutch roll) or restrain them at the trim position (controls-fixed Dutch roll). (Both methods should be utilized.)

5.3.3 Dutch Roll Influence on Roll Performance During Precision Maneuvering

The Dutch roll motion can have a significant influence on roll performance during precision maneuvering. If the Dutch roll influence is severe, the pilot will find it difficult or even impossible to precisely and accurately select a desired bank angle during airborne tracking tasks. The parameter used to quantitatively evaluate this Dutch roll influence is the ratio of the oscillatory component of roll rate to the average component of roll rate, $\frac{P_{osc}}{P_{av}}$ [14]. The phase angle, ψ_β, was discussed previously in the Theory section. The requirements are presented in Figure 5.56a.

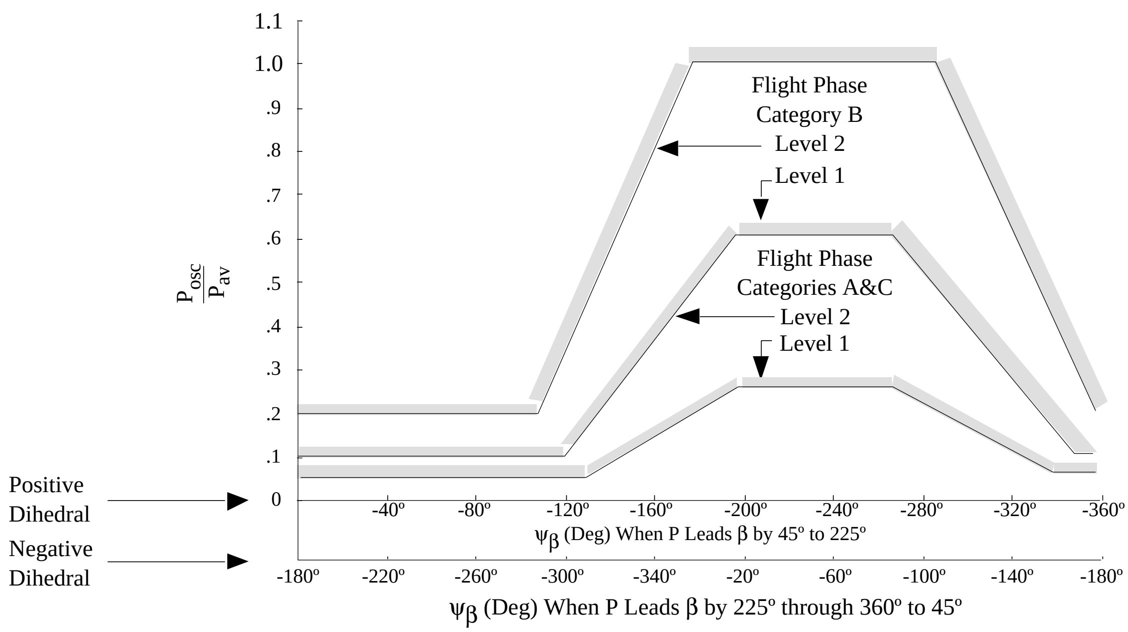


Figure 5.56a
Roll Rate Oscillation Limitations

---

14 $\frac{P_{osc}}{P_{av}}$ - a measure of the ratio of the oscillatory component of roll rate to the average component or roll rate following a rudder-pedals-free step aileron control command:

$$\zeta_d \leq 0.2: \quad \frac{P_{osc}}{P_{Av}} = \frac{P_1 + P_3 - 2P_2}{P_1 + P_3 + 2P_2}$$

$$\zeta_d > 0.2: \quad \frac{P_{osc}}{P_{Av}} = \frac{P_1 - P_2}{P_1 + P_2}$$

where p_1, p_2, and p_3 are roll rates at the first, second, and third peaks, respectively.

Before collecting quantitative data on the influence of the Dutch roll on roll performance during precision maneuvering, preliminary tests must be conducted to accurately determine: (1) the period of the Dutch roll oscillation, t_d, and (2) the amount of aileron deflection required to produce a 60 degree bank angle change in 1.7 t_d seconds. Once these data are known, the test is performed as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition. If using automatic recording devices, take a "trim shot."

2. Activate automatic recording devices and abruptly apply a step aileron command up to the magnitude which causes a 60 degree bank angle change in 1.7 t_d seconds. A chain stop or premeasured block can be used by the pilot in the cockpit to accurately attain the required control stick deflection.

3. The record of the roll should be analyzed after the flight for quantitative data.

The pilot's qualitative evaluation of the Dutch roll influence on roll performance should be performed during typical mission tasks.

5.3.4 Adverse and Proverse Yaw Effects on Precision Maneuvering

Obviously, there must be limits on the amount of proverse or adverse sideslip that is generated during maneuvering flight. Especially during precise tracking tasks, the pilot desires that the airplane be flown without excessive aileron and rudder coordination. The more coordination that is required, the more difficult the pilot's job to stabilize quickly and precisely on a desired flight path.

For small inputs like those used in normal airborne tracking tasks, the parameter used to quantitatively determine the sideslip generated following aileron control commands is the ratio of sideslip, $\Delta\beta$, to a nondimensional constant, k. The denominator k is defined as the ratio of commanded bank angle change in a give time to the required bank angle change, where the time interval and the required bank angle change are obtained from Table IXa, paragraph 3.3.4 of MIL-F-8785C, defining roll performance requirements.

$$k = \frac{[\phi_t] \text{ command}}{[\phi_t] \text{ requirement}}. \quad \text{(see Figure 56b).}$$

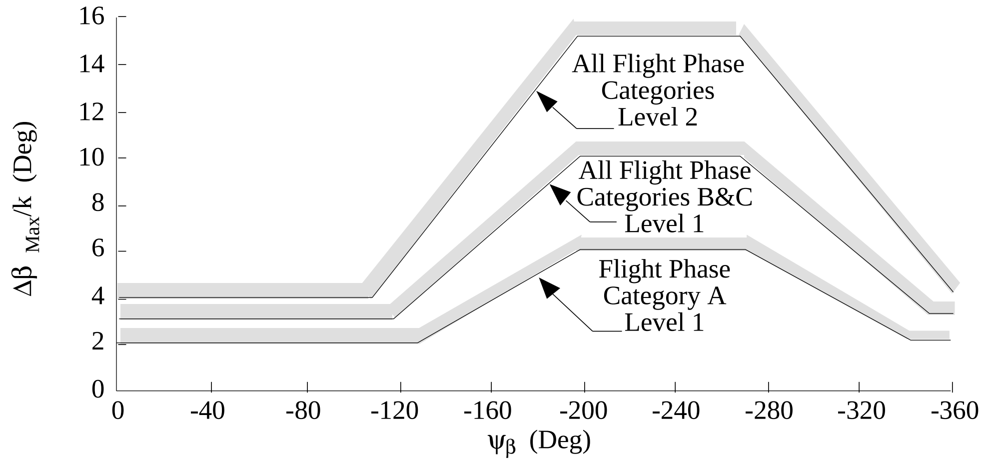


Figure 5.56b
Sideslip Excursion Limitations

A similar technique to that for measuring $\frac{P_{osc}}{P_{av}}$ may be used; however, the specification requirement only holds for small aileron deflections, i.e., up to those required to cause a 60 degree bank angle change in t_d or 2 seconds, whichever is the longer. The constant k is then determined for each maneuver as the ratio of the bank angle actually obtained in time t to the bank angle required by paragraph 3.3.4 of MIL-F-8785C in the same time t. Note that in general this time t will not be equal to t_d or 2 seconds.

The pilot's qualitative evaluation of the adverse or proverse sideslip characteristics should be performed during typical mission tasks.

5.3.5 Determination of Phase Angle from Flight Test Records

An example of how to determine the phase angle, ψ_β, from flight test records is presented in Figure 5.56c.

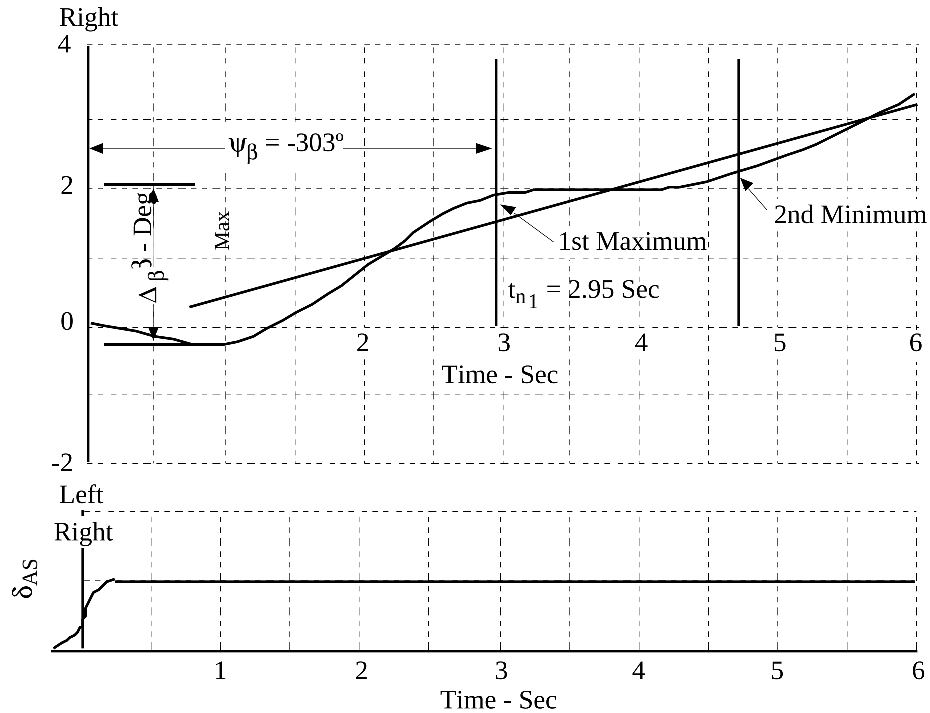


Figure 5.56c
Determination of Phase Angle

Since the first local maximum of the Dutch roll component of the sideslip response occurs at + = 2.95 seconds,

$$\psi_\beta = \frac{-360}{T_d} t_{n_\beta} + (n - 1)\,360 = \frac{-360}{3.5}(2.95) = -303^\circ \qquad \textit{eq 5.83}$$

5.3.6 Postflight Procedures

As soon as possible after returning from the flight, the test pilot should write a brief, rough qualitative report of the lateral-directional flying qualities exhibited during the mission tasks under evaluation. This report should be written while the events of the flight are fresh in the pilot's mind. Qualitative pilot opinion, appropriately related to the mission tasks under evaluation, will be the most important part of the final report.

Appropriate data should be selected to substantiate the pilot's opinion. The data presentation introduced here is only suggested and may be modified as desired by the test activity. No matter what method is used, it should be clear, concise, and complete.

5.3.6.1 MECHANICAL CHARACTERISTICS OF THE LATERAL-DIRECTIONAL CONTROL SYSTEM

Mechanical characteristics may be presented as shown previously in the discussion of "Test Procedures and Techniques - Nonmaneuvering Tasks" in the Longitudinal Flying Qualities Section. Merely modify the presentations to encompass the lateral and directional control systems.

5.3.6.2 STATIC LATERAL-DIRECTIONAL STABILITY CHARACTERISTICS

Static lateral-directional stability characteristics are normally presented as plots of rudder force and position, lateral control force and aileron position, bank angle, and longitudinal control force versus sideslip angle. Rate of climb or descent and airspeed error may be presented, if these parameters are significant. Cockpit control position and/or surface control position may be utilized, depending on the data available. Typical data presentation schemes are shown in Figures 5.57 and 5.58. The data for Figure 5.58 were derived from automatic recording traces of a transient sideslip technique. If automatic data reduction facilities are available, sufficient data points may be extracted to "shot gun" the data. It is apparent that obtaining the same plots by manual data reduction would be extremely laborious.

Discussions of static lateral-directional stability characteristics in the report of the test must be worded with care. The steady sideslip test results indicate the sign, but not the magnitude of directional stability, C_{n_β}, and dihedral effect, C_{ℓ_β}. (Actually, if very unusual stability augmentation or control system gadgetry is utilized, the tests may not even indicate the sign of the stability derivatives.) However, the test results are still extremely important from a flying qualities standpoint. The language of the report should reflect clearly the parameters which were used as indications of the static lateral-directional characteristics. For example, the following introductory sentences might be used in the report: "Directional stability, as indicated by the variations of rudder force and position with sideslip angle in steady heading sideslips, was positive in all configuration tested;" "The variation of lateral control force and aileron position with sideslip angle in steady heading sideslip indicated positive (or negative) dihedral effect," etc.

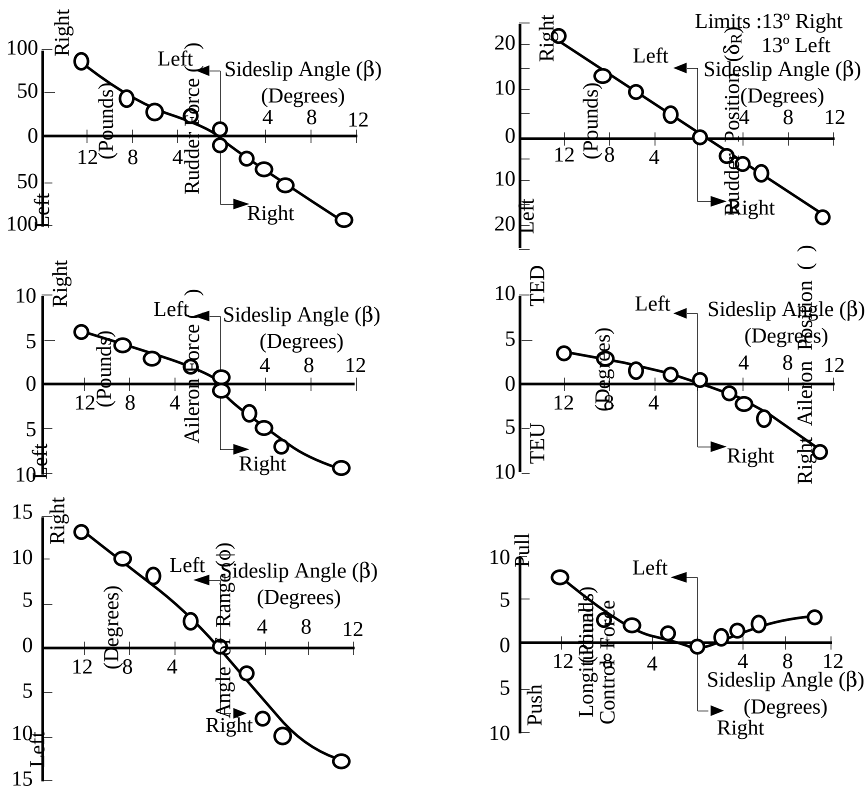


Figure 5.57
Static Lateral-Directional Stability Characteristics

Model ________ Airplane

BuNo ___________

Configuration Power Approach	Gross Weight: 15,450 Lbs
Loading: Long Range Attack	CG: 24.2% MAC
Trim: 125 KIAS, 5,000 Ft	Stab Aug: On

MODEL __________ AIRPLANE

BuNo __________

PILOT: _______________	TRIM: .71M, 30,000FT
CONFIGURATION: CRUISE	GROSS WEIGHT: 16,440 LBS
LOADING: C	CG: 18.2% MAC
DATE: 20 OCTOBER 1968	STAB AUG: OFF

Figure 5.58
Static Lateral-Directional Stability Characteristics

5.3.6.3 DYNAMIC LATERAL-DIRECTIONAL STABILITY CHARACTERISTICS - SPIRAL STABILITY

Spiral stability data are effectively presented as a plot of bank angle versus time. Appropriate specification limits may be shown on the plot. A typical time history is shown in Figure 5.59.

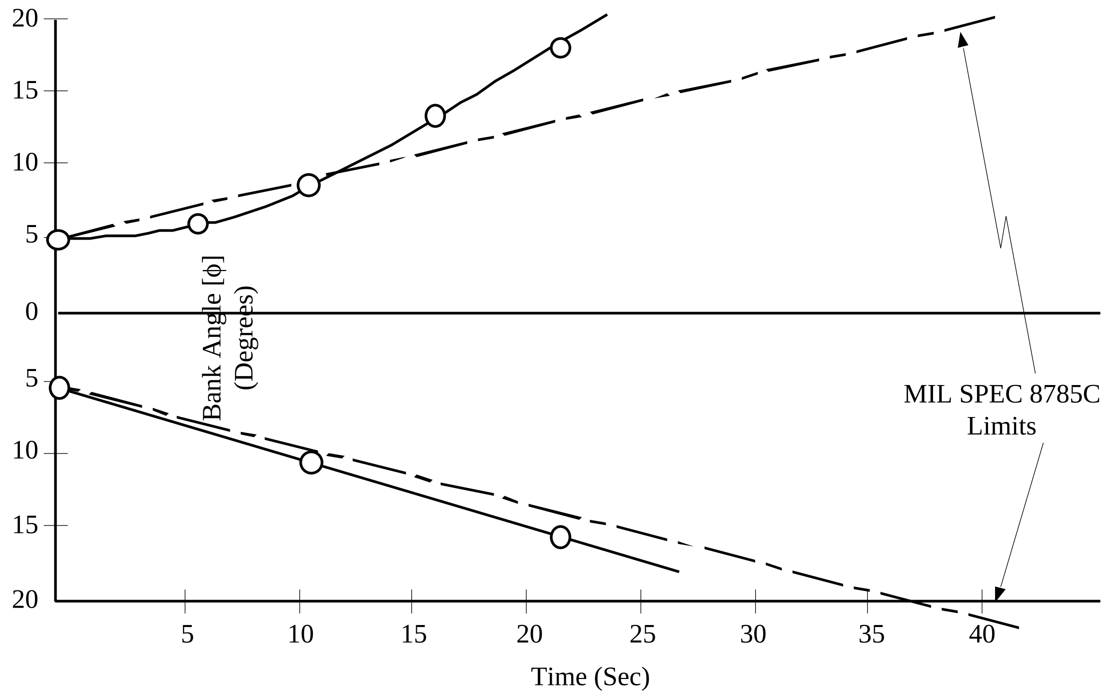


Figure 5.59
Spiral Stability

Model _______Airplane
BuNo ___________

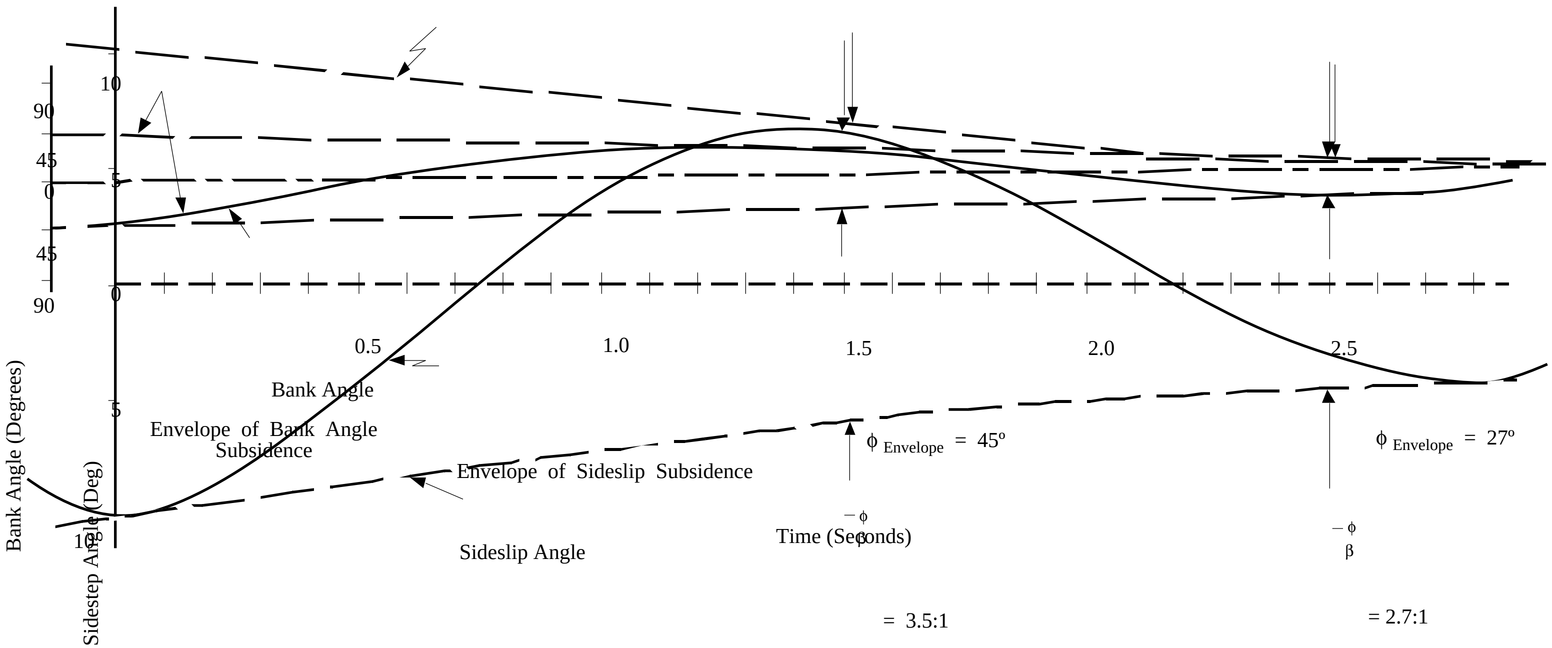
Configuration: Power Approach	Gross Weight: 16,700 Lbs
Loading: Alfa	CG: 27.2% Mac
Trim: 135 KIAS, 7500 Ft	Stab Aug: On

5.3.6.4 DYNAMIC LATERAL-DIRECTIONAL STABILITY CHARACTERISTICS - DUTCH ROLL CHARACTERISTICS

The presentation of Dutch roll characteristics will be dictated by the amount of data available. If the scope of the evaluation is limited, Dutch roll characteristics are effectively presented in tabular form. An example is show in Figure 5.60. If the Dutch roll motion was recorded on oscillograph, magnetic tape, or telemetry, the actual trace, appropriately annotated, may be presented in the report. Sideslip angle is the most desirable parameter to use in determining Dutch roll frequency and damping, since it exhibits the pure Dutch roll response better than any other parameter. However, yaw rate may be utilized if the sideslip trace is not useable. Sideslip angle and bank angle traces should be utilized to determine roll-to-yaw ratio; however, yaw rate and roll rate may be utilized as back-up parameters. In order to clearly illustrate the procedure for determining roll-to-yaw ratio, the bank angle and sideslip angle traces from typical oscillograph traces have been reproduced in Figure 5.60.

Configuration	Altitude (ft)	Trim Airspeed (KIAS, IMN)	CG (%MAC)	Gross Weight (lbs)	Stab Aug	ω_{n_d} (rad/sec)	ζ_d	$\frac{\phi}{\beta}$	$\omega_{n_d}^2 \left(\phi/\beta\right)$ (rad/sec)^2	$\zeta_d\ \omega_{n_d}$ (rad/sec)
Cruise (CR)	30,000	.65	22.8	19000	ON	1.4	0.2	2.5:1	4.8	0.25
Power (P)	10,000	450/.80	23.2	21700	ON	4.3	0.1	1.2:1	21.4	0.60
Power Approach (PA)	10,000	145	19.2	15500	OFF	2.2	0.3	3:1	14.5	0.62

Figure 5.60
Airplane Dutch Roll Characteristics



1. Draw the subsidence envelopes of sideslip and bank angle.

2. Using the envelopes, derive the roll angle and sideslip angle at the same time along the horizontal axis.

3. Compute the roll-to-yaw ratio. (Note that the roll-to-yaw ration changes in this example. This is a common phenomenon which may be influenced by difficulty in drawing precise envelopes.)

4. Note that the roll and sideslip excursions reach maximum values at different times, they are not in phase. Thus, merely noting peak roll to sideslip excursion for a given cycle will not yield the correct roll-to-yaw ratio since the roll-to-yaw ratio is defined as the ratio of the e of roll angle and sideslip angle. (In this example, roll and sideslip are out of phase by only approximately 0.3 second.)

Figure 5.61
Determination of Roll-to-Yaw Ratio of Dutch Roll Motion

If the flight test team desires to show the variation of Dutch roll characteristics with airspeed or Mach number, a plot similar to that shown in Figure 5.62 may be utilized. When discussing Dutch roll characteristics in the body of the report, or when presenting appropriate data, the condition of the stability augmentation, if installed, must be explicitly stated.

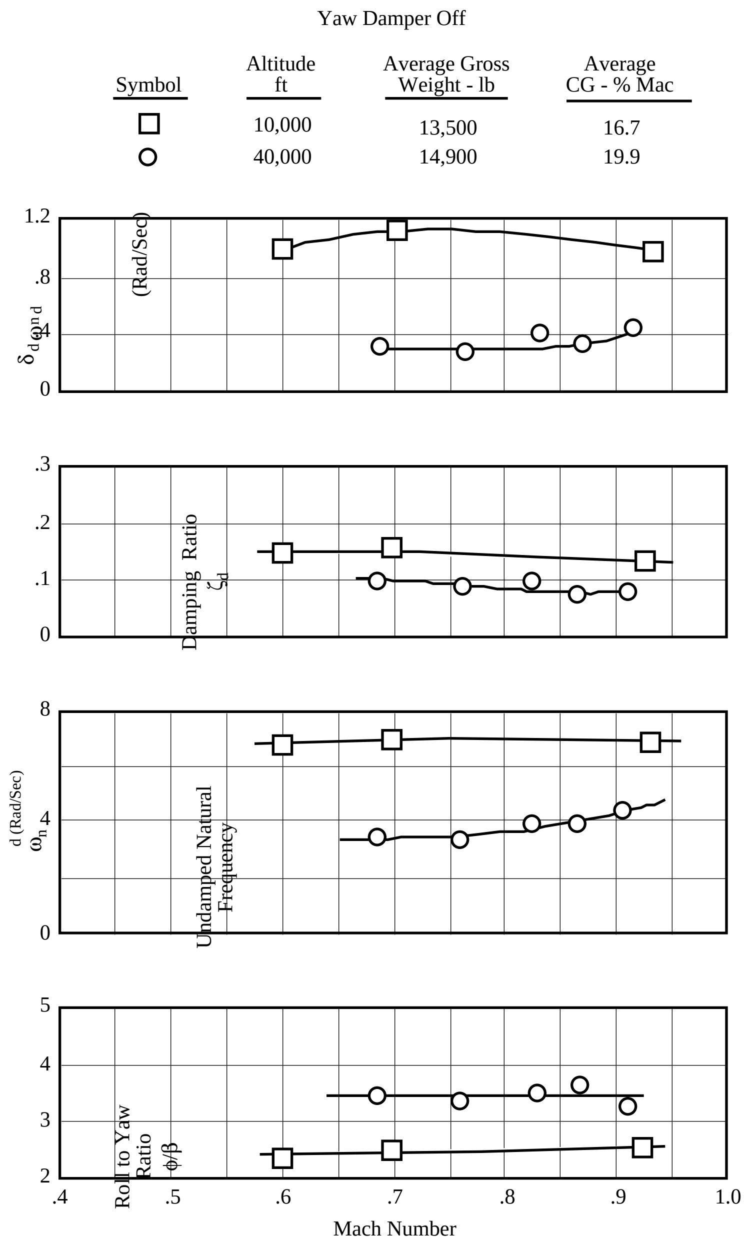


Figure 5.62
Dutch Roll Characteristics

05.3.6.5 LATERAL-DIRECTIONAL TRIMMABILITY, RUDDER-ONLY TURNS, AND AILERON-ONLY TURNS

The determination of trimmability as presented herein is based on the test pilot's qualitative opinion. Therefore, a qualitative discussion of trimmability in the technical report is appropriate. The results of other tests may be used to substantiate the qualitative opinion of lateral-directional trimmability.

The results of rudder-only turn and aileron-only turn tests are probably best blended into discussions of static or dynamic lateral-directional characteristics. The writer should make every effort to correlate the qualitative and quantitative results of all the tests conducted in order to present a meaningful picture of the airplane's lateral-directional flying qualities.

5.4 SPECIFICATION REQUIREMENTS

Requirements for static and dynamic lateral-directional flying qualities are contained in the following applicable paragraphs of Military Specification, MIL-F-8785C, of 5 November 1980, hereafter referred to as the Specification.

3.2.3.7 Longitudinal control in sideslips

3.3.1 Lateral-directional mode characteristics (except 3.3.1.2)

3.3.2 Lateral-directional dynamic response characteristic (subparagraphs 3.3.2.1, 3.3.2.2.1, 3.3.2.4.1, 3.3.2.6)

3.3.3 Pilot-induced oscillations

3.3.4.5 Rudder-pedal-indicted rolls

3.3.5 Directional control characteristics (subparagraphs 3.3.5.1, 3.3.5.2)

3.3.6 Lateral-directional characteristics in steady sideslips

3.3.7 Lateral-directional control in crosswinds

3.3.8 Lateral-directional control in dives

3.5.2 Mechanical characteristics (control system)

The requirements of the Specification may be modified by the applicable airplane Detail Specification. Comments concerning only those portions of the Specification which require some interpretation are presented below.

3.3.2.2.1 The intent of this paragraph is to insure that there will be no objectionable roll oscillations while executing small, precise lateral inputs such as would be used in air-to-air or air-to-ground tracking tasks. The damped period (t_d) of the Dutch roll must be known before quantitative data can be obtained to determine compliance with this paragraph.

3.3.2.4.1 The intent of this paragraph is to limit the amount of adverse or proverse sideslip following small lateral control inputs such as would be used in tracking tasks. Again, the damped period of the Dutch roll must be known to determine Specification compliance.

3.3.7.2.1 This paragraph specifies that satisfactory directional control using either rudder, aileron or a combination of both shall be maintained at 50 knots or above during takeoff and landing rollout.

5.5 LATERAL-DIRECTIONAL FLYING QUALITIES - GLOSSARY

Roll-To-Yaw Ratio — Ratio of bank angle envelope to sideslip angle envelope during Dutch roll oscillation.

Adverse Yaw — Yawing moments created act so as to rotate the nose of the airplane opposite to the direction of roll. The term "adverse" does not, in itself, denote unfavorable flying qualities.

Proverse Yaw — Yawing moments generated act so as to rotate the nose of the airplane toward the direction of roll. The term "proverse" does not necessarily indicate favorable flying qualities.

Roll Mode Time Constant — Time required for the roll rate to reach 63.2 percent of the steady state roll rate following a step input of lateral control.

Coordinated Turn — A turn in which a balance of sideward accelerations acting on objects in the airplane is attained; a "ball-centered" turn.

5.6 LATERAL-DIRECTIONAL FLYING QUALITIES - REFERENCES

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control,* by Courland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control,* by Courland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Analysis of Several Handling Quality Topic Pertinent to Advanced Manned Aircraft,* Systems Technology, Inc., AFFDL-TR-67-2, June, 1967.

4. *Calculation of the Response of a Fighter Aircraft to Turbulent Air and a Comparison with Flight Measurements,* by C. G. B. Mitchell, Royal Aircraft Establishment, Ministry of Aviation, Farnborough, Hants, RAE Technical Report 66275, August, 1966.

5. *Computer Programmes to Calculate the Response of Flexible Aircraft to Gusts and Control Movements,* by C. G. B. Mitchell, Ministry of Technology, Aeronautical Research Council, London, Current Paper No. 957, October, 1966.

6. *Considerations in the Determination of Stability and Control Derivatives and Dynamic Characteristics From Flight Data,* by Chester H. Wolowicz, NASA Flight Research Center, Edwards Air Force Base, California, AGARD Report 549 - Part 1, 1966.

7. *Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training,* by F. D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F-1, September, 1962.

8. *Development of Satisfactory Lateral-Directional Handling Qualities in the Landing Approach,* by Robert L. Stapleford, Donald E. Johnston, Gary L. Teper, and David H. Weir, Systems Technology, Inc., Hawthorne, California, NASA CR-239, July 1965.

9. *Dynamics of Flight, Stability, and Control,* by Bernard Etkin, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

10. *Evaluation of Lateral-Directional Handling Qualities of Piloted Reentry Vehicles Utilizing Fixed-Base and Inflight Evaluations,* by J.I. Meeker, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, NASA CR-778, May, 1967.

11. *Experience with Shinbrot's Method of Transient Response,* by B. R. A. Burns, British Aircraft Corporation, Preston Division, Warton Aerodrome, Preston, Lacashire, AGARD Report 549 - Part II, 1966.

12. *Features of Large Transport Aircraft Affecting Control During Approach and Landing,* by W. J. G. Pinsker, Royal Aircraft Establishment, Bedford, England.

13. *Flight Testing of Aircraft,* by M. G. Kotik, et al., NASA Technical Translation F-442, April, 1967.

14. *Flying Qualities Requirements for United States Navy and Air Force Aircraft,* by William Koven and Richard Wasicko, AGARD Specialist Meeting on Aircraft Stability and Control, Brussels, Belguim, April, 1961.

15. *Handling and Performance Exercises Test Techniques,* Empire Test Pilots' School, Flying Department Notes.

16. *Inflight Evaluation of Lateral-Directional Handling Qualities for the Fighter Mission,* by James I. Meeker and G. Warren Hall, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-98, October 1967.

17. *Inflight Simulation and Pilot Evaluation of Selected Landing Approach Handling Qualities of a Large Logistics Transport Airplane,* by Donald W. Phoads, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-51, July, 1967.

18. *Lateral-Directional Flying Qualities for Power Approach: Influence of Dutch Roll Frequency,* by Edward Seckel, James A. Franklin, and George E. Miller, Princeton University, Princeton, New Jersey, Princeton University Report No. 797, September, 1967.

19. *Lecture Notes on Lateral-Directional Stability*, by F. D. Newell and Michael Parrog, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. TB-1961-F-1, July 1965.

20. *Military Specification, Flying Qualities of Pilot Airplanes MIL-F-8785C,* of 5 November 1980.

21. *Modern Flight Dynamics,* by W. Richard Kokl, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

22. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

23. *Proposal for a Revised Military Specification, "Flying Qualities of Piloted Airplanes" (MIL-F-8785 ASG), With Substantiating Text*, by C. J. Mazza, William Becker, Marshall Cohen, and Alvin Spector, U. S. Naval Air Development Center, Johnsville, Pennsylvania, NADC-ED-6282, January, 1963.

24. *A Review of Recent Research on Handling Qualities, and its Application to the Handling Problems of Large Aircraft*, by P. L. Bisgood, Ministry of Aviation, Aeronautical Research Council, London, R & M. No. 3458, June 1964.

25. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

26. *Stability and Control, Empire Test Pilots' School Ground School Notes, Book "C".*

27. *Stability and Control,* USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

28. *Substantiation Information for Selected Lateral-Directional Requirements*, Cornell Aeronautical Laboratory, Report No. BM-2238-F-1, May, 1967.

5.7 ROLLING PERFORMANCE THEORY

5.7.1 General

The rolling performance and associated roll handling qualities of the airplane directly influence the ease and quickness with which the pilot can make direction changes and wing position corrections. Both these evolutions obviously involve the ability to change and control bank angle. The pilot must be able to change flight direction at will and the most expeditious means of doing so is to roll, then pull back on the stick or yoke. Therefore, the maneuverability of the airplane is directly related to rolling performance as well as longitudinal maneuvering characteristics discussed earlier.

The ability to make wing position changes and corrections is particularly critical in close proximity to the ground. During takeoffs and landings in turbulent and/or crosswind flight conditions, the pilot must be provided with adequate control of wing position. A natural and stringent design requirement on rolling performance is thus in the low airspeed, low altitude, takeoff and landing flight condition.

The pilot's opinion of the rolling performance and associated handling qualities depends on several characteristics; the most important of which are the initial response of the airplane to a lateral control input and the subsequent rolling velocity or roll rate attained. The rolling motion generated by a lateral control input is generally contaminated by yawing and pitching motion which may degrade handling qualities significantly. Obviously, the lateral control forces required in rolling the airplane and mechanical characteristics of the lateral control system also influence pilot opinion.

Much of the language used in this discussion of rolling performance has been previously introduced in the Lateral-Directional Theory section earlier in the manual. The reader should refer to that section for derivations and explanations of stability derivatives and other terms and expressions, if necessary. The classic, single degree of freedom roll mode of motion has also been previously introduced. This discussion of roll response will begin by expanding that discussion of the single degree of freedom roll.

5.7.2 Single Degree of Freedom Roll Response

An expression for the single degree of freedom roll response or the pure roll response to a lateral control input can be derived by considering the simple lateral-directional equations for sideforce, yawing moment, and rolling moment. Since the roll maneuver is a nonequilibrium or nonsteady-state motion during the initial acceleration to steady state roll rate, inertia terms must be included in the expressions:

SIDEFORCE

$$C_{y_\beta}\,\beta + C_{y_{\delta_r}}\,\delta_r + C_L \sin\phi = \frac{mu_0\dot{\beta} + mu_0\, r}{qS} \quad \textit{eq 5.84}$$

YAWING MOMENT

$$C_{n_\beta}\,\beta + C_{n_{\delta_r}}\,\delta_r + C_{n_{\delta_a}}\,\delta_a + C_{n_r}\frac{rb}{2V} + C_{n_p}\frac{pb}{2V} = \frac{1}{qSb} I_{yy}\dot{r} \quad \textit{eq 5.85}$$

ROLLING MOMENT

$$C_{\ell_\beta}\,\beta + C_{\ell_{\delta_r}}\,\delta_r + C_{\ell_{\delta_a}}\,\delta_a + C_{\ell_r}\frac{rb}{2V} + C_{\ell_p}\frac{pb}{2V} = \frac{1}{qSb} I_{xx}\dot{p} \quad \textit{eq 5.86}$$

For the pure roll case, the rolling moment equation is the only equation which need be considered. Additionally the following terms are assumed to be small and can be omitted for the single degree of freedom roll:

$$C_{\ell_\beta}\,\beta\ ;\ C_{\ell_{\delta_r}}\,\delta_r\ ;\ C_{\ell_r}\frac{rb}{2V} \quad \textit{eq 5.87}$$

Thus, the single degree of freedom roll may be expressed as follows:

$$\dot{p}\ \frac{I_{XX}}{qSb} - C_{\ell_p}\ \frac{pb}{2V_T} - C_{\ell_{\delta_a}}\ \delta_a = 0 \qquad \textit{eq 5.88}$$

or

$$\dot{p} - \frac{C_{\ell_p} qSb}{I_{XX}}\ \frac{b}{2V_T}\ p - \frac{C_{\ell_{\delta_a}} qSb}{I_{XX}}\ \delta_a = 0 \qquad \textit{eq 5.89}$$

or

$$\dot{p} - L_p\ p - L_{\delta_a}\ \delta_a = 0 \qquad \textit{eq 5.90}$$

Where:

$\dot{p}$ = rolling acceleration; radians per second per second.

p = roll rate; radians per second.

L_p = rolling acceleration per increment of roll rate, or <u>roll damping</u>; rad/sec^2 per rad/sec, or 1/sec.

L_{δ_a} = rolling acceleration per increment of lateral control deflection, or <u>lateral control sensitivity</u>; rad/sec^2 per radian, or 1/sec^2.

δ_a = lateral control deflection; radians.

These differential equations may be solved[15] to yield a time solution in <u>roll rate</u>, which is presented as follows:

$$p\ (t) = \frac{L_{\delta_a}\delta_a}{L_p} \left\{ e^{L_p t} - 1 \right\} \qquad \textit{eq 5.91}$$

---

[15] The solution is quite laborious and may be found in appropriate technical literature.

Where:

$p(t)$ = the roll rate at any time, t.

This expression describes the classic exponential buildup of roll rate from zero to steady state after a step lateral control input. The fact that the roll rate does buildup at an exponential rate will be used later to analyze the results of in-flight tests. The last expression is sometimes presented as follows:

$$p(t) = p_{ss}\left\{1 - e^{-t/\tau_R}\right\} \qquad \textit{eq 5.92}$$

Where:

p_{ss} = steady state roll rate; radians per second

τ_R = roll mode time constant, or the time for the roll rate to reach 63.2 percent of the steady state roll rate after a step lateral control input; seconds.

This analytical development can now be explained in meaningful, practical terms. First of all, it is important to realize that the lateral control system is utilized to generate <u>roll rate</u>, as the last equations imply, and is not used to <u>directly</u> command <u>bank angle</u>. The bank angle attained depends on the <u>length of time</u> the lateral control input is held (Figure 5.63). The pilot finds this type of roll control <u>quite natural</u> in most conventional airplane designs. Consider now the initial response of the airplane to the lateral control input. At the very <u>instant</u> the pilot makes the control input, there is no roll rate, p, so there is no resistance to the roll acceleration from L_p. Therefore, the initial roll acceleration will be the maximum roll acceleration (Figure 5.64). An expression for this initial roll acceleration may be derived as follows:

$$\dot{p} - L_p\, p - L_{\delta_a}\, \delta_a = 0 \qquad \textit{eq 5.93}$$

however, at t = 0, p = 0, therefore:

$$\dot{p}_t = 0 = L_{\delta_a}\, \delta_a \qquad \textit{eq 5.94}$$

Obviously, this initial roll acceleration is quite important to the pilot's opinion of the maneuverability of the airplane. The initial roll acceleration for maximum lateral control deflection, $L_{\delta_a} \; \delta_{a_{MAX}}$, has been proposed as one criterion by which to measure or classify airplane rolling performance.

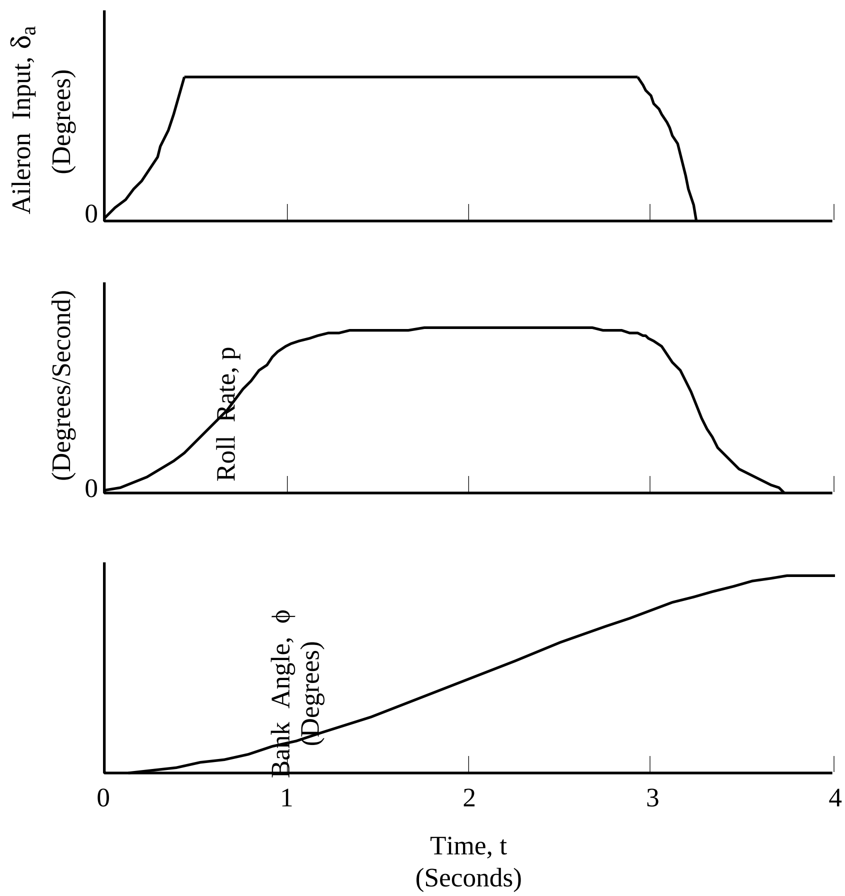


Figure 5.63
Airplane Response to Aileron Input

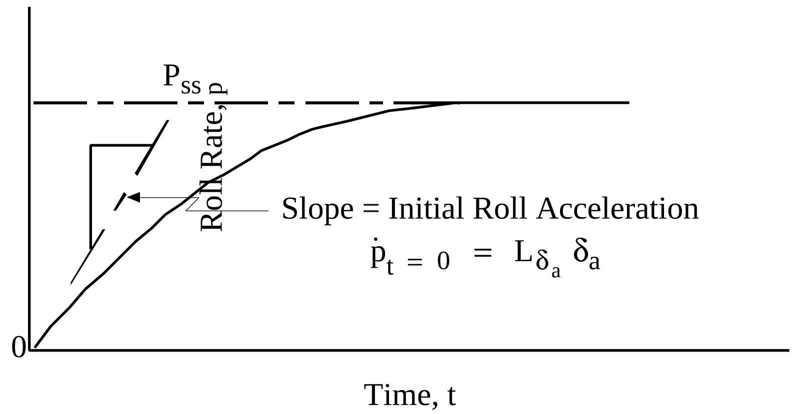


Figure 5.64
Roll Rate Response to Step Lateral Control Input

5.7.2.1 STEADY STATE ROLL RATE

As the roll rate increases after a lateral control input, the resistance to the rolling or roll damping contribution, $L_p p$, increases. When the roll damping contribution equals the lateral control power contribution, $L_{\delta_a}\,\delta_a$, there is no unbalance of accelerating contributions. Therefore, the roll acceleration, $\dot{p}$, is zero, and the steady state roll rate, p_{ss}, is attained (Figure 5.65). An expression for the steady state roll rate is derived as follows:

$$\dot{p} - L_p\,p - L_{\delta_a}\,\delta_a = 0$$

however, when $\left|L_p\,p\right| = \left|L_{\delta_a}\,\delta_a\right|$, $\dot{p} = 0$, and $p = p_{ss}$, thus:

$$p_{ss} = -\frac{L_{\delta_a}\,\delta_a}{L_p} \qquad \textit{eq 5.95}$$

Obviously, the steady state roll rate attained with various magnitudes of lateral control deflection is quite important to the overall rolling performance.

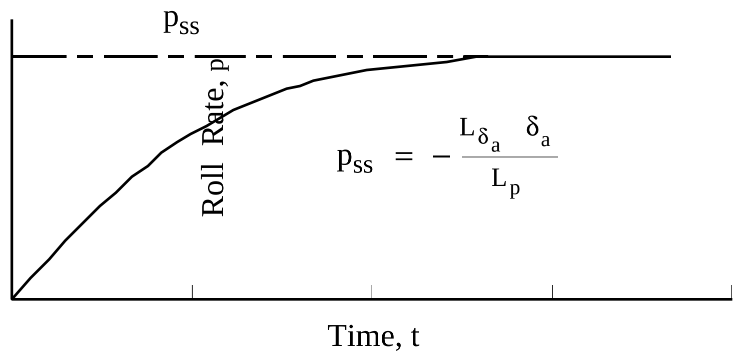


Figure 5.65
Steady State Roll Rate

The <u>maximum</u> steady state roll rate, expressed in <u>nondimensional</u> form[16], was frequently used in the past as a measure of the airplane's rolling performance. It is defined as follows:

$$\left(\frac{pb}{2V_T}\right)_{Max} = -\frac{C_{\ell_{\delta_a}}}{C_{\ell_p}} \delta_{a_{Max}} \qquad \textit{eq 5.96}$$

Where:

p = maximum steady state roll rate attainable with full lateral control deflection; radians per second.

b = wing span; feet

V_T = true airspeed, feet per second

This single criterion for rolling performance is quite poor, at best, since it does not take into consideration rolling acceleration. In addition, it is an unrealistic requirement for airplanes capable of very high true airspeeds (particularly if their wing span is small).

16 $\frac{pb}{2V}$ is actually the helix angle described by the wingtip during a rolling maneuver.

5.7.2.2 ROLL MODE TIME CONSTANT

The roll mode time constant, τ_R, influences the <u>manner</u> in which the roll rate builds up and subsides after lateral control movements are made (Figure 5.66). It is a rather important parameter affecting not only roll acceleration and deceleration but the technique the pilot utilizes in controlling bank angle. Its value is solely dependent on roll damping, L_p:

$$\tau_R = -\frac{1}{L_p} \qquad \textit{eq 5.97}$$

The roll mode time constant is typically 1 second or less. It is a parameter which has been proposed as one criterion for measuring rolling performance. Measurement of the parameter in-flight requires sensitive instrumentation, good piloting technique, and analytical manipulation of the data derived.

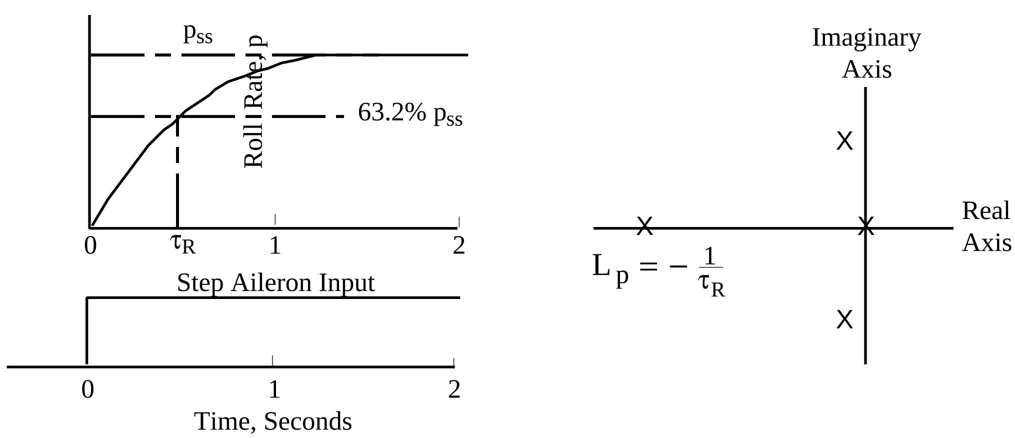


Figure 5.66
Roll Mode Time Constant

5.7.3 Influence of Various Parameters on Roll Characteristics

The variation of p_{ss}, and τ_R, with various parameters will now be presented. It should bee remembered that, for this theoretical discussion, each parameter is varied in turn while holding all other parameters constant. The equations utilized to make the rationalizations which follow are:

$$p_{ss} = -\frac{L_{\delta_a}}{L_p}\,\delta_a \qquad \textit{eq 5.98}$$

$$\tau_R = -\frac{1}{L_p} \text{ or } \tau_R = \frac{4\, I_{xx}}{C_{\ell_p}\rho\, V_T S b^2} \text{ or } \tau_R = \frac{4\, I_{xx}}{C_{\ell_p}\sqrt{\sigma\rho_{ssl}}\, V_e S b^2} \qquad \textit{eq 5.99}$$

5.7.3.1 LATERAL CONTROL DEFLECTION

Steady state roll rate classically varies directly with the magnitude of the lateral control input. Obviously, the roll mode time constant is not influenced by this parameter (Figure 5.67).

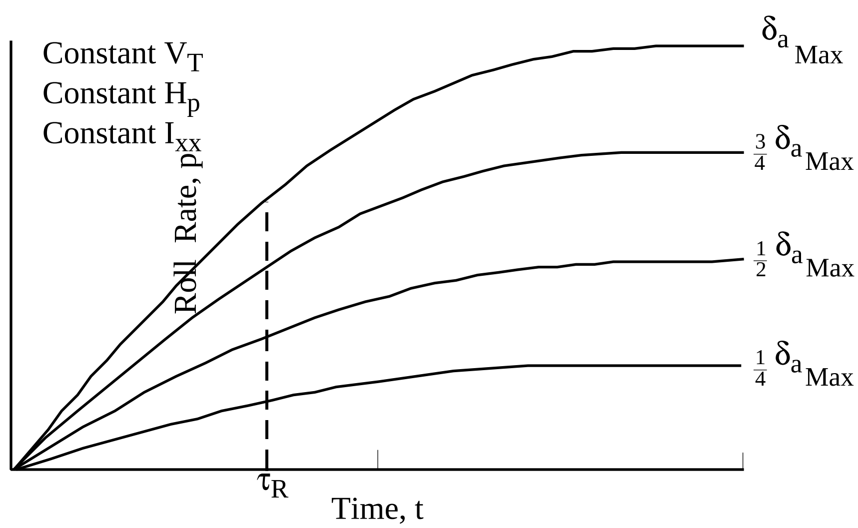


Figure 5.67
Classic Variation of Roll Characteristics with Lateral Control Deflection

5.7.3.2 MOMENT OF INERTIA IN ROLL

It is important to note that variation in rolling moment of inertia have no influence on steady state roll rate. However, the roll mode time constant varies directly with I_{xx} (Figure 5.68). Thus, airplanes with large roll inertias (full tip tanks, heavy wing stores, large wings, etc.) may be capable of significant steady state roll rates yet exhibit poor roll acceleration and deceleration. Steady state roll rate alone is an inadequate indicator of rolling performance.

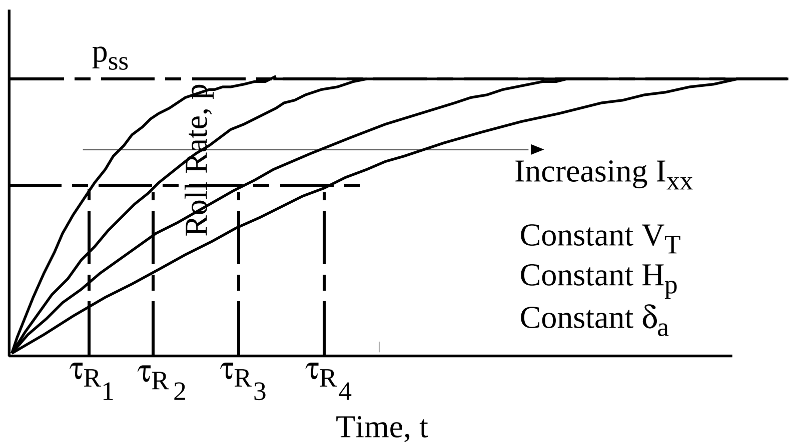


Figure 5.68
Typical Variation of Roll Characteristics with Rolling Moment of Inertia

5.7.3.3 ALTITUDE

If altitude is varied at a constant true airspeed, steady state roll rate remains constant while the roll mode time constant increases with increasing altitude (Figure 5.69).

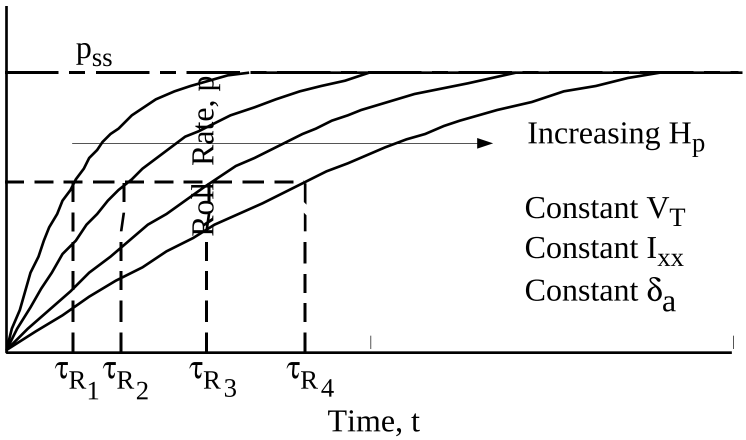


Figure 5.69
Classical Variation of Roll Characteristics with Altitude (Constant True Airspeed)

The influence of varying altitude at a constant equivalent airspeed is quite different as shown in Figure 5.70. Although the roll mode time constant still increases with altitude increase, steady state roll rate also increases because true airspeed increases.

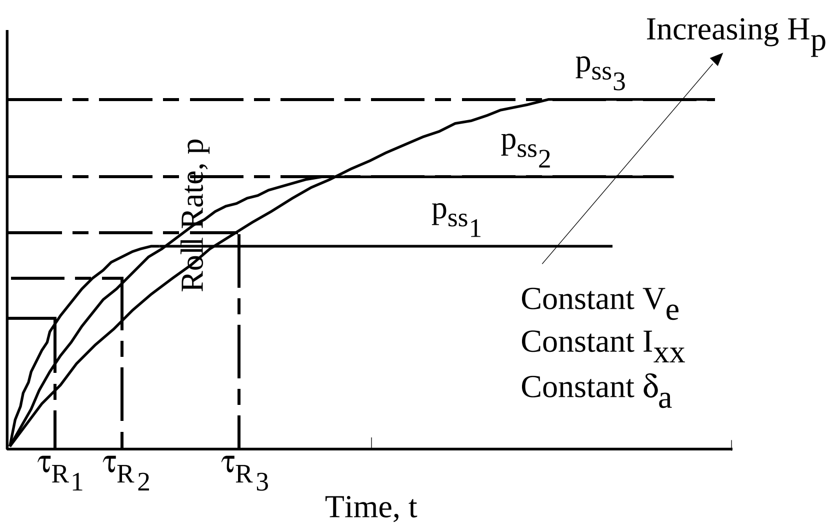


Figure 5.70
Typical Variation of Roll Characteristics with Altitude (Constant V_e)

5.7.3.4 AIRSPEED

Steady state roll rate varies directly with true airspeed and the roll mode time constant varies inversely with true airspeed (Figure 5.71).

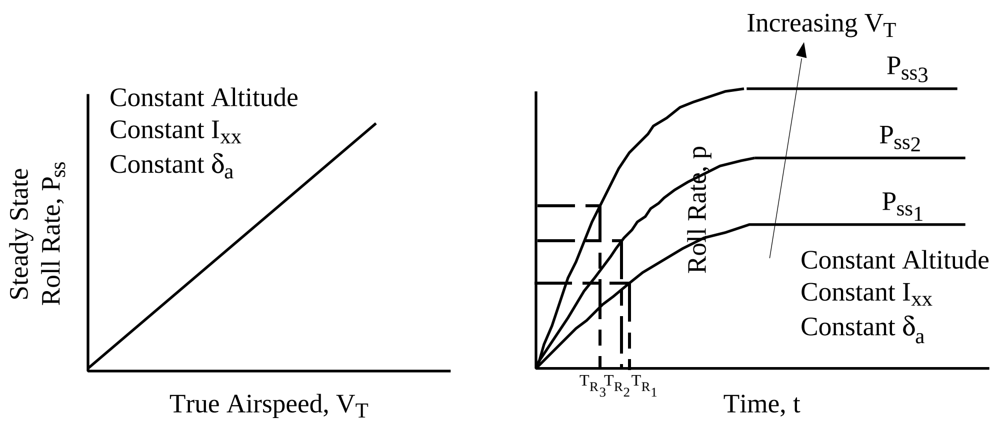


Figure 5.71
Classical Variation of Roll Characteristics with True Airspeed

5.7.4 Real Airplane Response to Lateral Control Inputs

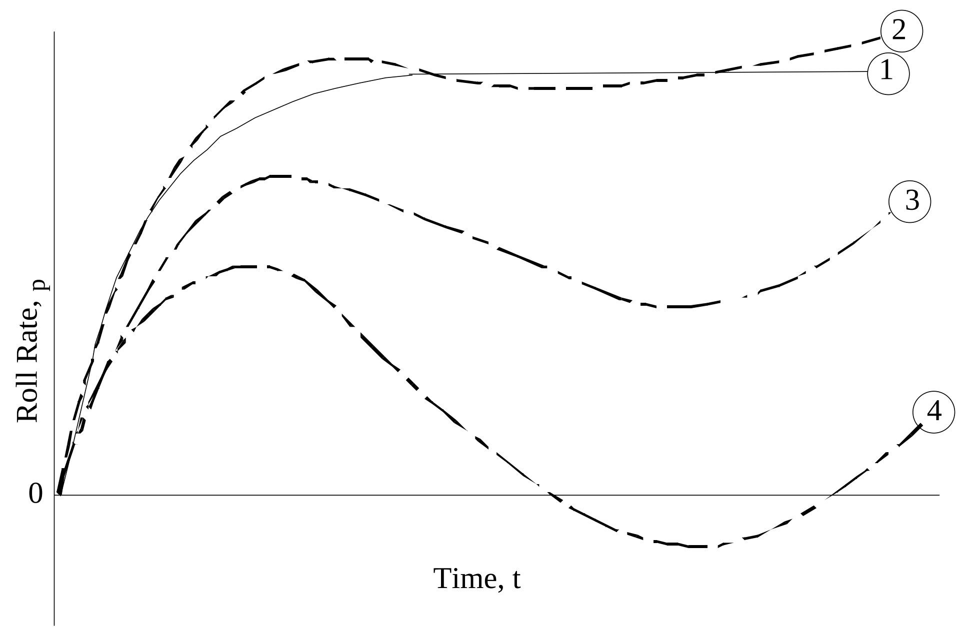
The actual airplane response to a lateral control input is almost invariably somewhat different from that predicted by the classic single degree of freedom analysis. These differences may be attributable to the Dutch roll influence, roll coupling, and aeroelastic effects.

5.7.4.1 DUTCH ROLL EXCITATION DURING ROLLING MANEUVERS

The degree of excitation of the Dutch roll motion and the resultant influence on rolling performance depend on several factors. The most important of these parameters are: directional stability, yawing moments developed with lateral control deflection and roll rate, Dutch roll frequency, damping and roll-to-yaw ratio, and dihedral effect. Obviously, several of these factors are directly related and it is impossible to present discussions of all possible combinations of characteristics. However, several combinations are of particular interest.

5.7.4.1.1 High C_{n_β}*, Low* C_{ℓ_β}*, Low* $C_{n_{\delta_a}}$*, and* C_{n_p}

Initially, consider an airplane with relatively strong directional stability, little adverse or proverse yaw, and weak dihedral effect. The strong directional stability should result in minimum sideslip excursions during rolling maneuvers. The small sideslip angles which are developed have little influence on rolling performance due to weak dihedral effect. (Because of the weak dihedral effect, the roll-to-yaw ratio of the Dutch roll motion will probably be low.) Thus, for this combination of characteristics, Dutch roll influence on rolling motion is insignificant. The airplane may be maneuvered with little rudder coordination, and the pilot will probably find roll response excellent if the steady state roll rate and roll mode time constant are satisfactory. The response curve of this airplane would closely approximate the single degree of freedom case (Figure 5.72).



1 Single degree of freedom roll response.

2 High C_{n_β}, low C_{ℓ_β}, low $C_{n_{\delta_a}}$, low C_{n_p}.

3 Low C_{n_β}, high C_{ℓ_β}, high $C_{n_{\delta_a}}$, high C_{n_p}.

4 Low C_{n_β}, very high C_{ℓ_β}, very high $C_{n_{\delta_a}}$, and C_{n_p}.

Figure 5.72
Typical Influence of Various Characteristics on Roll Response

5.7.4.1.2 Low C_{n_β} *, Low* C_{ℓ_β} *, High* $C_{n_{\delta_a}}$ *, and* C_{n_p}

Next, consider an airplane with relatively low directional stability, high yawing moments generated by lateral control deflection and roll rate, yet relatively low dihedral effect. These conditions would probably result in a low roll-to-yaw ratio. During rolling maneuvers in this airplane, aileron yawing moments, $C_{n_{\delta_a}}$, and yawing moments generated by roll rate, C_{n_p}, would most likely generate significant sideslip excursions. While these excursions would probably have rather small effects on rolling performance per se due to the low dihedral effect, roll handling qualities may be seriously degraded. The pilot may experience such undesirable characteristics as oscillations of the nose on the horizon during turns or by a lag or initial reversal in yaw rate (or "turn rate") during a turn entry. The latter phenomenon would be particularly objectionable in instrument flight conditions.

5.7.4.1.3 Low C_{n_β}*, High* C_{ℓ_β}*, High* $C_{n_{\delta_a}}$*, and* C_{n_p}

The combination of relatively weak directional stability, strong dihedral effect, and significant yawing moments generated by lateral control deflection and roll rate can result in serious roll handling qualities problems. These conditions would probably result in a Dutch roll motion of rather high roll-to-yaw ratio. The Dutch roll motion will be excited during roll maneuvers because of the relatively large adverse or proverse yawing moments. If the yawing moments are adverse, the sideslip generated will have the same sign as the roll rate, i.e., in a roll to the right, the sideslip will be to the right of the airplane's nose, etc. It can readily be seen that adverse yaw combined with large positive dihedral effect (large negative C_{ℓ_β}) can severely degrade rolling performance (Figure 5.72). If the Dutch roll motion is lightly damped, the roll response may be oscillatory (Figure 5.72). In severe cases, very large positive dihedral effect and very large sideslip angles generated by adverse yawing moments, the roll rate may actually be diminished to zero or reverse (Figure 5.72). Acceptable rolling performance may be regained in these cases by appropriate rudder coordination during rolling maneuvers. However, the rudder inputs would have to be large, quick, and precisely timed to alleviate the worst situation shown in Figure 5.72. Under stress of actual operational conditions, the pilot probably would not be able to devote sufficient attention to rudder coordination in this case.

5.7.4.2 ROLL COUPLING

Roll coupling may be defined as pitching and yawing motions which are induced by inertial and kinematic effects during high rate rolls. These motions may be of a sufficient magnitude to overcome the stability of the airplane; spectacular departures into uncontrollable flight conditions are a possible consequence. Under conditions of high dynamic pressure, these excursions may generate forces and accelerations which exceed the design strength of the airframe.

There are actually three factors which tend to destablize the airplane during rolling maneuvers. These factors could never occur singly in actual flight conditions. Their influence may be additive and aggravate the resultant motion or act in opposition and alleviate divergent tendencies. These three factors are: inertia coupling[17], kinematic coupling, and the I_{xz} effect.

The mathematical relationships required to describe such grossly nonlinear motions as roll coupling are extremely complex and are omitted in this presentation. The primary purpose of this discussion is to provide a practical understanding of the factors contributing to roll coupling.

5.7.4.2.1 Inertia Coupling

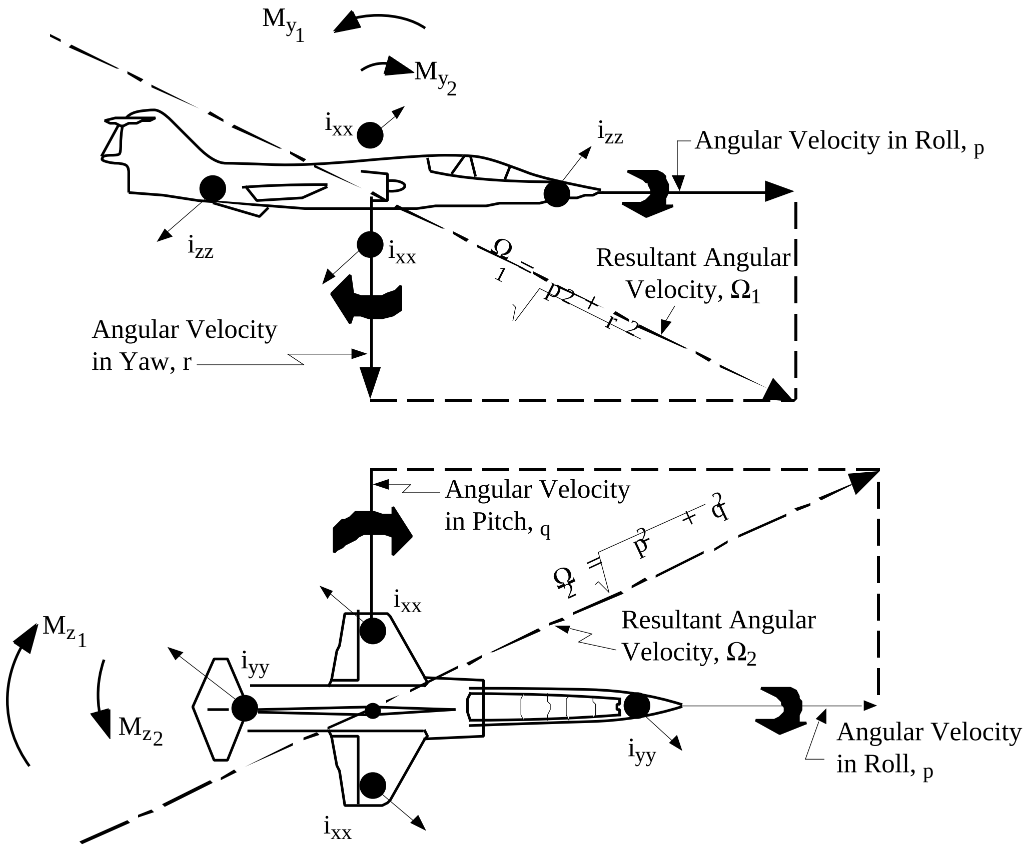
The inertia distribution of an airplane may be represented by pairs of concentrated masses, as shown in Figure 5.73. For clarity in the top pictorial presentation, the rolling inertia pairs (represented by i_{xx}) are shown in the plane of symmetry of the airplane vice in the wing plane. However, this has no effect on the resultant motion. Note that in the top picture of Figure 5.73, if roll rate, p, and yaw rate, r, are present simultaneously, a resultant angular velocity Ω, is created. The airplane then rotates (rolls and yaws) about an axis which is represented by the resultant angular velocity vector, Ω_1. The yawing inertia pairs, represented by i_{zz} are consequently subjected to centrifugal forces which generate a nose up pitching moment, M_{y_1}. The rolling inertia pairs, represented by i_{xx}, are also

[17] Roll coupling has frequently been incorrectly referred to as "inertia coupling," since inertia effects have a predominant influence on the motion. However, the phenomenon is best described simply as "roll coupling."

subjected to centrifugal forces. However, their contribution is a nose down pitching moment, M_{y_2}. The nose up pitching moment, M_{y_1}, is generally predominant since the moment of inertia in yaw, I_{zz}, is usually larger than the moment of inertia in roll, I_{xx}. (If the airplane is rolling left and yawing left, the predominant pitching moment will still be nose up. If the airplane is rolling left and yawing right, or vice versa, the resultant pitching moment will be nose down.)

In the bottom picture of Figure 5.73, the influence of simultaneous roll rate, p, and pitch rate, q, is shown. The resultant angular velocity, Ω_2, is created and the airplane then rotates (pitches and rolls) about the resultant angular velocity vector. The pitching inertia pairs, I_{yy}, are consequently subjected to centrifugal forces which generate a <u>yawing moment to the right</u>, M_{Z_1}. The rolling inertia pairs, represented by I_{xx}, are also subjected to centrifugal forces. However, their resultant yawing moment, M_{Z_2}, is in opposition of M_{Z_1}. The yawing moment, M_{Z_1}, is generally predominant since the moment of inertia in pitch, I_{yy}, is usually larger than the moment of inertia in roll, I_{xx}. (If the airplane is rolling left and pitching nose up, the resultant yawing motion will still be nose right. If the airplane is rolling right and pitching nose up or vice versa, the resultant yawing motion will be nose left.)

It can readily be rationalized that the moments generated by inertia effects could be reduced to zero (see Figure 5.73) if the rolling moment of inertia, I_{xx}, was equal to the yawing and pitching moments of inertia, I_{zz} and I_{yy}, respectively. Thus, inertia coupling depends on <u>mass distribution</u>. The larger the yawing and pitching moments of inertia become in comparison to the rolling moment of inertia, the more fertile become the circumstances in which inertia coupling can develop.



Symbols i_{yy} represent pitching moment of inertia, I_{yy}

Symbols i_{xx} represent rolling moment of inertia, I_{xx}

Symbols i_{zz} represent yawing moment of inertia, I_{zz}

Top airplane is rolling right and yawing right.

Bottom airplane is rolling right and pitching down.

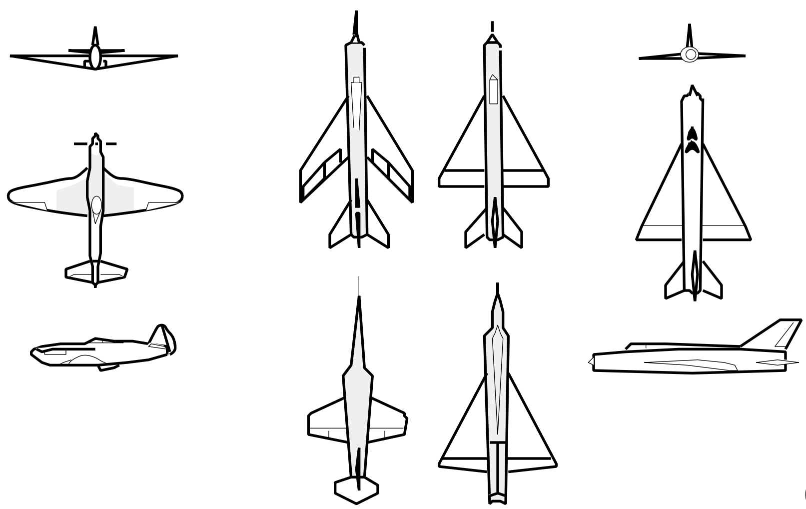
Figure 5.73
Generation of Inertia Coupling Moments

Consider now the evolution of the modern high performance airplane shape and mass distribution (Figure 5.74). In order to accommodate the increasing loads and equipment of the high performance airplane behind a minimum of frontal area, fuselages have become more and more elongated and densely loaded all along their axis. The low aspect ratio, thin wing planforms have been utilized to decrease transonic and supersonic drag. As a result, moments of inertia in pitch and yaw, I_{yy} and I_{zz}, have been increasing steadily with corresponding decreases in moment of inertia in roll, I_{xx}. A comparison of inertia characteristics of World War II vintage airplanes with present day high performance airplanes reveals the following typical ratios:

1939 - 45 Airplanes	Modern High Performance Airplanes
$I_{yy}: I_{xx} = 4:3$	$I_{yy}: I_{xx} = 10:1$
$I_{zz}: I_{xx} = 4:3$	$I_{zz}: I_{xx} = 10:1$

Airplane of
1939 - 45 Period

Modern High Performance
Airplanes



0.

Hatching denotes regions of concentration of primary masses within the airplanes

Figure 5.74
Comparison of Airplane Mass Distribution

Thus, it is easy to realize that inertia coupling has become increasingly more pronounced on modern, high performance airplane designs.

The tendency of the airplane to diverge in yaw or pitch due to inertia coupling is resisted by the directional stability, C_{n_β}, and angle of attack stability, C_{m_α}. Based on inertia coupling considerations only, expressions for critical roll rate, P_{CRIT}, defined as the roll rate at which directional or longitudinal instability will be encountered, may be written as follows:

$$P_{CRIT_1} \sim \sqrt{\frac{\frac{C_{m_\alpha} \quad qS\bar{c}}{I_{yy}}}{\frac{I_{xx} - I_{zz}}{I_{yy}}}} \qquad \textit{eq 5.100}$$

$$P_{CRIT_2} \sim \sqrt{\frac{\frac{C_{n_\beta} \quad qSb}{I_{zz}}}{\frac{I_{yy} - I_{xx}}{I_{zz}}}} \qquad \textit{eq.101}$$

Note that critical roll rate varies directly with angle of attack stability and directional stability. As moments of inertia in pitch and yaw increase proportionately to the moment of inertia in roll, critical roll rate is decreased (the denominator in both expressions approaches one).

5.7.4.2.2 Kinematic Coupling

Kinematic[18] coupling may be considered as an actual interchange of angle of attack and sideslip during a rolling maneuver. This coupling results entirely from geometric considerations.

The interchange of angle of attack and sideslip is illustrated in Figure 5.75. In the top picture of Figure 5.75, the airplane is rolled from an initial positive angle of attack which results in a sideslip of equal magnitude after 90 degrees of roll. The bottom picture of Figure 5.75 shows the result of initiating a roll with inherent sideslip. After 90 degrees of roll, the sideslip is transformed to angle of attack. It is obvious that, in actuality, the aerodynamic stabilities of the airplane in pitch and yaw, C_{m_α} and C_{n_β}, will oppose the introduction of any angle of attack and sideslip excursions which differ from a trimmed, equilibrium condition. However, the airplane's capacity to prevent these excursions

[18] Kinematics is a branch of mechanics which deals with motion in the abstract without reference to the force or mass.

depends on the natural frequencies[19] of the airplane in pitch and yaw as well as the imposed roll rate. The roll rate dictates the rate at which the angle of attack and sideslip interchanges are being imposed on the airplane dynamic system, i.e., the roll rate determines the "disturbance rate." As long as this disturbance rate is relatively low, the airplane dynamic system is able to cope with the disturbance inputs in angle of attack and sideslip, and the motion is stable. However, if the roll rate is high enough in relation to the airplane natural frequencies, the airplane system may be forced to accept cyclic inputs of angle of attack and sideslip with which it cannot contend and a pure divergence in pitch or yaw results. Thus, the relationships between the magnitudes of C_{m_α}, C_{n_β}, and roll rate are of extreme importance in determining the resultant airplane response. As C_{m_α} and C_{n_β} decrease, with resultant decreases in natural frequencies in pitch and yaw, and as roll rate increases, the possibility of saturating the airplane system with cyclic angle of attack and sideslip interchange increases.

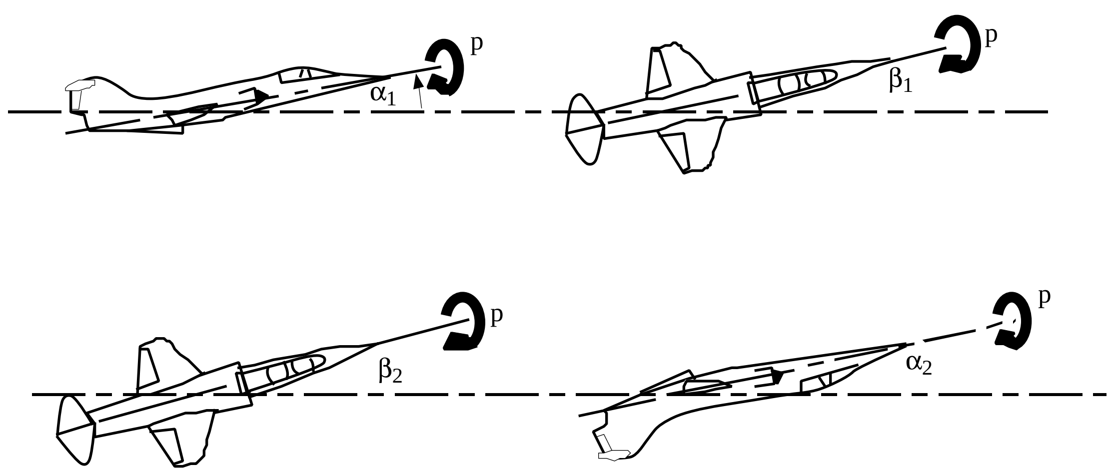
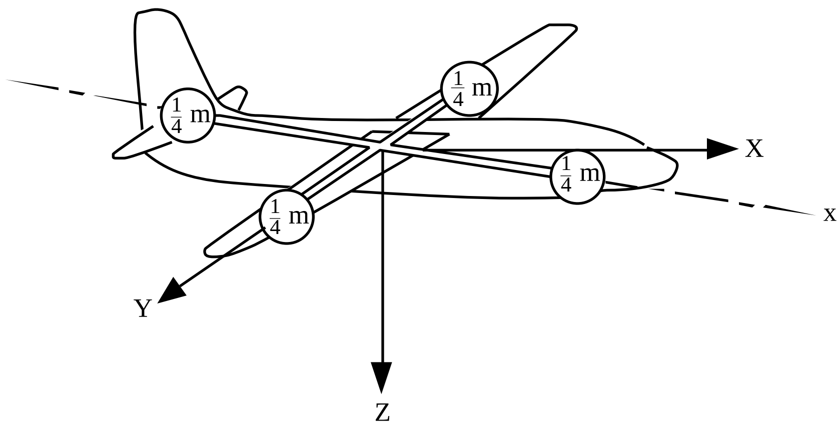


Figure 5.75
Kinematic Coupling

[19] The natural frequencies in pitch and yaw are directly related to angle of attack stability, C_{m_α}, and directional stability, C_{n_β}, respectively.

5.7.4.2.3 The I_{xz} Parameter

Three products of inertia appear in the equations of motion for a rigid airplane. They are I_{xy}, I_{yz}, and I_{xz}. By virtue of typical airplane conformation, the products of inertia, I_{xz} is almost invariably not equal to zero. If I_{xz} is not equal to zero, it means that the X axis of the airplane is not aligned with the principle inertial axis (see Figure 5.76).



The mass distribution of the airplane can be represented by two crossed dumbbells, each bell being a quarter of the total mass. The dumbbells are crossed exactly in their centers at the CG of the airplane. The products of inertia I_{xy}, I_{yz} and I_{xz} can be thought of as measures of the uniformity of the mass distribution about the Y axis, Z axis, and X axis respectively. By virtue of airplane symmetry about the Y and Z axes:

$$I_{xy} = 0$$
$$I_{yz} = 0$$

However, note that the mass distribution about the X axis is not symmetrical, therefore

$$I_{xz} \neq 0$$

There exists an axis, denoted by x, about which the product of inertia, I_{xz}, would be equal to zero. This axis is called the principle inertial axis.

Figure 5.76
Product of Inertia and Principle Inertial Axis

When an airplane is rolled about an axis which does not coincide with its principle inertial axis, centrifugal forces are generated which tend to cause the airplane to diverge (Figure 5.77). The magnitudes of the forces and moments created are functions of the rate of roll, p, and the numerical value of I_{xz} relative to the actual axis of roll. This phenomenon is easily visualized by considering a symmetrical rod to represent the distributed mass along the length of the fuselage. If this rod is subjected to rotation about an axis which is inclined to the rod at some small angle, the centrifugal forces and moments created tend to cause the rod to increase its displacement from the axis of rotation (i.e., become a flyweight).

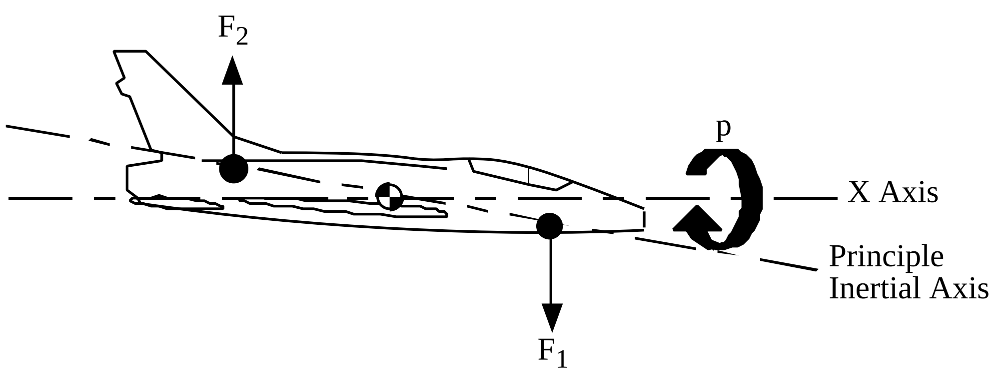


Figure 5.77
I_{xz} Effects on Rolling Motion

Again, the airplane's ability to counteract the influence of mass distribution on roll handling qualities depends on the aerodynamic stabilities of the airplane in pitch and yaw.

5.7.4.2.4 The Pilot's Contribution to Roll Coupling

The pilot is capable of inducing roll coupling difficulties by improper piloting technique or by a lack of appreciation for the flight restrictions imposed on the airplane. Theoretically, the pilot could override the influence of roll coupling during the rolling maneuver by applying suitable coordinated rudder and elevator inputs. A typical computer study program reveals that the pilot would be required to make control inputs as shown if Figure 5.78. Obviously, it would be exceedingly difficult to achieve the required control movements in the short time interval since the control inputs are not related to readily perceptible flight sensations. More than likely, the pilot attempts at coordination would be ill timed and reinforce the departures in angle of attack and sideslip.

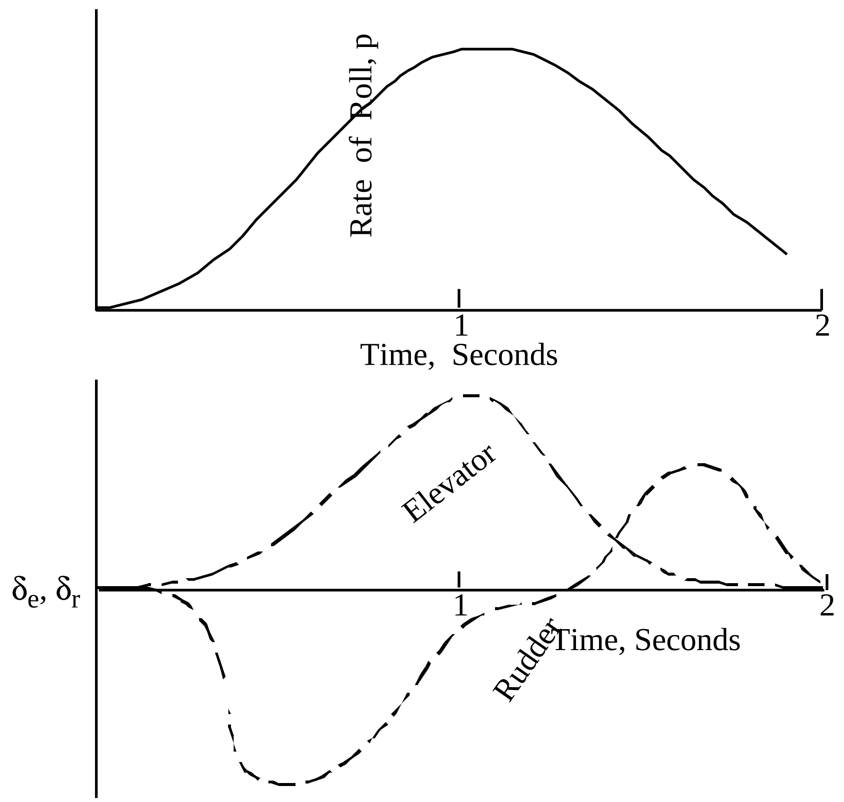


Figure 5.78
Typical Pilot Control Inputs Required to Overcome Roll Coupling Effects

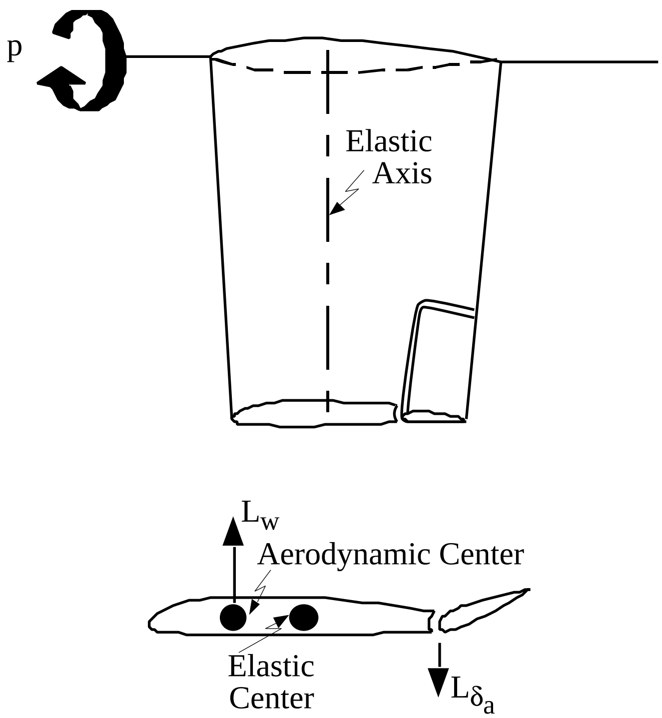
5.7.4.2.4 Conclusions Concerning Roll Coupling

Combinations of inertia coupling, kinematic coupling, and product of inertia effects may result in unstable motions during rolling maneuvers. The problem may be compounded by adverse or proverse yawing moments generated during the rolling maneuver or by improper piloting technique. The pilot cannot reasonably be expected to cope with the sudden divergences in angle of attack and sideslip. Therefore, the only solution to the roll coupling problem is to attempt to prevent the possibilities of encountering it. The airplane designer may provide increased aerodynamic stability in yaw or pitch. This may be done by increasing the size of the vertical tail or by providing suitable artificial stability augmentation. In addition, appropriate airplane operational limitations are published: roll rate limits, bank angle change limits, lateral control deflection limits, rolling pull-out and rolling pushover limits, etc. The operational pilot must be suitably educated in the problems associated with roll coupling; no pilot would intentionally break rolling limitations imposed on his airplane if he understood the possible consequences.

5.7.4.2.5 Aeroelastic Effects

The actual variation of rate of roll with airplane true airspeed may fall short of that predicted by rigid wing theory, especially at high airspeeds. This is due to the effects of wing twist and wing bending. When the trailing edge aileron is deflected, the wing tends to twist so as to unload itself, i.e., attempt to decrease to forces and moments being applied to the wing structure (Figure 5.79). Obviously, wing twist would reduce the rate of roll attainable with a given lateral control deflection.

If the airplane has swept wings, wing bending may be a factor in rolling performance (Figure 5.80). The chord line of the wing with sweepback makes an angle with the airplane centerline, yet the wing bends perpendicular to the airplane centerline. Swept wing bending, in response to moments created by lateral control deflection, thus changes the effective wing angle of attack. These changes tend to reduce the rate of roll attainable with a given lateral control deflection.



L_{δ_a} = force generated by deflecting aileron trailing edge up.

L_w = wing lift vector excluding L_{δ_a}.

Due to the aileron deflection, a moment is created which tends to twist the wing structure (leading edge up in this case). The wing will tend to twist about its <u>elastic center</u>, a point in the section about which torsional deflection occurs. Obviously, if the wing twists in this manner, the rolling moment is somewhat <u>decreased</u>, therefore rolling performance is decreased.

Figure 5.79
Wing Twist During Rolling Motion

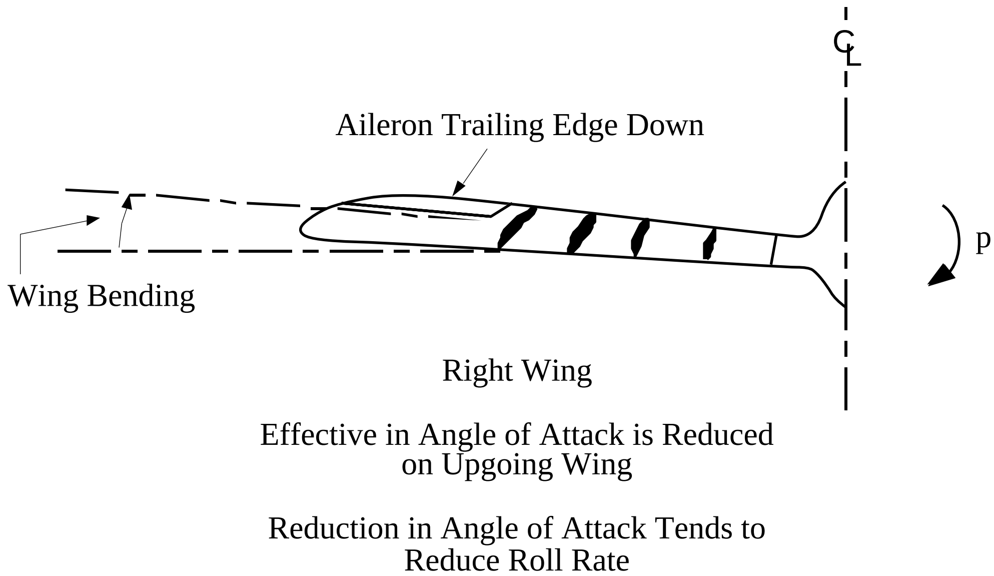


Figure 5.80
Influence of Wing Bending on Rolling Performance of a Swept-Wing Airplane

The typical influence of wing twist and wing bending is shown in Figure 5.81. If the airspeed is high enough, a point may be reached where the combined effects of wing twist and wing bending will counter the rolling moment generated by aileron deflection. This airspeed, at which lateral control is completely negated by aeroelastic effects, is called the "aileron reversal speed", V_r. It is extremely important, obviously, for the airplane designer to insure that the wings are sufficiently rigid to cause V_r to be greater than any airspeed at which the airplane will be operated.

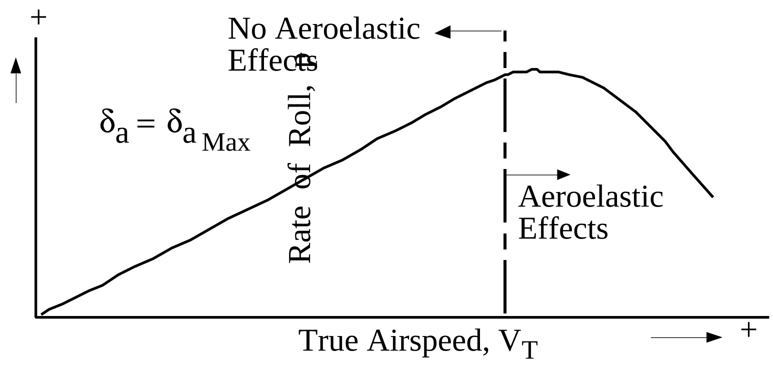


Figure 5.81
Typical Influence of Aeroelastic Effects on Rolling Performance

5.7.5 Lateral Control Forces

Lateral control forces are generated by the requirement for the pilot to move the lateral control system to the position for equilibrium in the rolling maneuver. If the lateral control system is irreversible, lateral control forces are merely a function of lateral control position, i.e.:

$$F_a = K_1 \; \Delta\delta_a \text{ (linear feel spring system)} \qquad \textit{eq 5.102}$$

or

$$F_a = K_2 \; q\Delta\delta_a \text{ ("q – feel" system)} \qquad \textit{eq 5.103}$$

where K_1 and K_2 are constants describing the characteristics of the system, such as strength of the feel spring, gearing ratio, etc.

However, if the lateral control system is reversible, the pilot control force requirements will be a function of hinge moments developed. These hinge moments depend on lateral control system design, dynamic pressure, and the imposed rate of roll. The aileron float angles developed during roll maneuvers may be substantial for the reversible control system. An expression for the floating angle of the aileron may be developed as follows. If the hinge moment coefficients are considered linear functions of angle of attack and aileron deflection, the total hinge moment coefficient may be expressed:

$$C_{h_a} = C_{h_{\delta_a}} \delta_a + C_{h_a} \Delta\alpha_{Ave} \qquad \textit{eq 5.104}$$

Where:

$C_{h_{\delta_a}}$ = aileron hinge moment coefficient variation with aileron deflectionat zero angle of attack

C_{h_α} = aileron hinge moment coefficient variation with wing angle of attack at zero aileron deflection

$\Delta\alpha_{Ave}$ = average increment of angle of attack due to the rolling velocity (Figure 5.83)

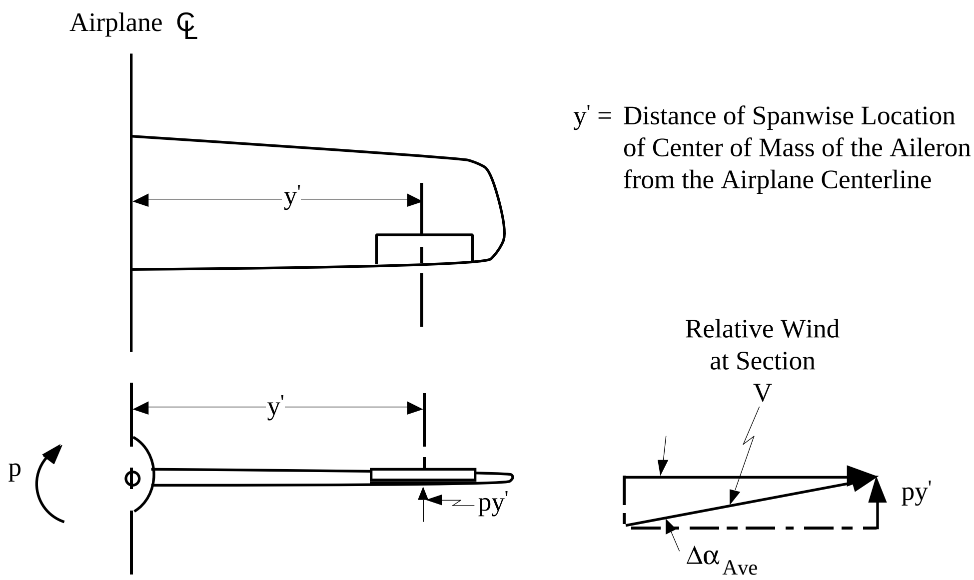
From Figure 5.83:

$$\Delta\alpha_{Ave} = \frac{py'}{V} \qquad \textit{eq 5.105}$$

If C_{h_a} is zero, δ_a is the aileron float angle, $\delta_{a_{Float}}$ which may the be expressed as:

$$\delta_{a_{Float}} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}} \frac{py'}{V} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}} \left(\frac{2y'}{b}\right)\left(\frac{pb}{2V}\right) \qquad \textit{eq 5.106}$$

Thus, float angle in the steady state roll is proportional to roll rate, p.



$$\sin \Delta\alpha_{Ave} = \frac{py'}{V_{Relative}}$$

For Small Angles; $\Delta\alpha_{Ave} = \frac{py'}{V}$

Figure 5.82
Generation of Angle of Attack Changes During Rolling Maneuvers

Since the pilot must apply lateral control forces to move the aileron from its float position to the equilibrium position, the analytical expression for lateral control forces in the reversible system is as follows:

$$F_a = -K\, C_{h_{\delta_a}}\, q\, S_a\, \bar{c}_a \left\{ \delta_{a_{Equilibrium}} - \delta_{a_{Float}} \right\} \qquad \textit{eq 5.107}$$

Where:

K = a constant describing the characteristics of the system, radian per foot.

S_a = area of the aileron, square feet.

$\bar{c}_a$ = average aileron chord, feet.

Now since:

$$\delta_{a_{\text{Equilibrium}}} = -\frac{C_{\ell_p}}{C_{\ell_{\delta_a}}}\ \frac{pb}{2V}\ \text{(steady state roll)} \qquad \textit{eq 5.108}$$

and:

$$\delta_{a_{\text{Float}}} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}}\left(\frac{2y'}{b}\right)\left(\frac{pb}{2V}\right) \qquad \textit{eq 5.109}$$

lateral control forces in the reversible control system may be expressed† as:

$$F_a = V_p\left\{-\frac{K}{4}\ C_{h_{\delta_a}}\ \rho\ S_a\bar{c}_a\ b\right\}\left\{-\frac{C_{\ell_p}}{C_{\ell_{\delta_a}}} + \frac{C_{h_\alpha}}{C_{h_{\delta_a}}}\ \frac{2y'}{b}\right\} \qquad \textit{eq 5.110}$$

or, for a constant altitude, merely:

$$F_a = K_1\ V_p\ (H_p = \text{constant}) \qquad \textit{eq 5.111}$$

Since roll rate varies directly with true velocity, lateral control forces vary essentially as the true velocity squared. Therefore, for the reversible control system, there may very well be some true airspeed beyond which the lateral force gradient exceeds acceptable limits. The airplane designer must be very careful to provide aerodynamic balance which insures reasonable lateral control forces in the usable airspeed range of the airplane. This is a most difficult design problem, thus boosted lateral control systems or completely irreversible lateral control systems are common on airplanes with significant airspeed capabilities.

The last equation can be utilized to show the degree of degradation of rate of roll at high airspeeds where the pilot is unable to apply full lateral control because of excessive forces. For a constant altitude situation, assume the pilot can apply a maximum lateral control force of 30 pounds. For this case:

$$pV = K_2 \qquad \textit{eq 5.112}$$

---

† Several mathematical manipulations have been omitted.

Therefore, the rate of roll will decrease hyperbolically with airspeed after the pilot is denied maximum lateral control deflection by the force restraint (Figure 5.83).

If the response of airplanes with reversible and irreversible lateral control systems is compared, the airplane with the irreversible control system may seem to be more responsive.

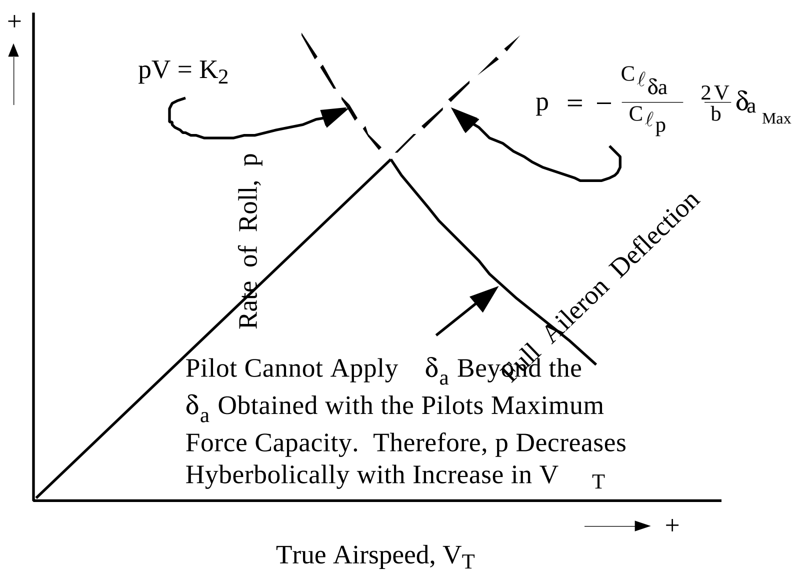


Figure 5.83
Rolling Performance May Be Limited by Force Requirements for the Reversible Control System

Considering only the aerodynamic hinge moments, this is because, for the same lateral force input, the aileron deflection in the reversible system will not be as large because the float angle has not had a chance to develop. The float angle develops as a function of roll rate (Figure 5.84a). In order to generate the same initial response in roll rate, the reversible control system will require more force initially (Figure 5.84b). In any case, the airplane equipped with a reversible lateral control system which exhibits some measure of aileron float may seem less sensitive and responsive to lateral force inputs.

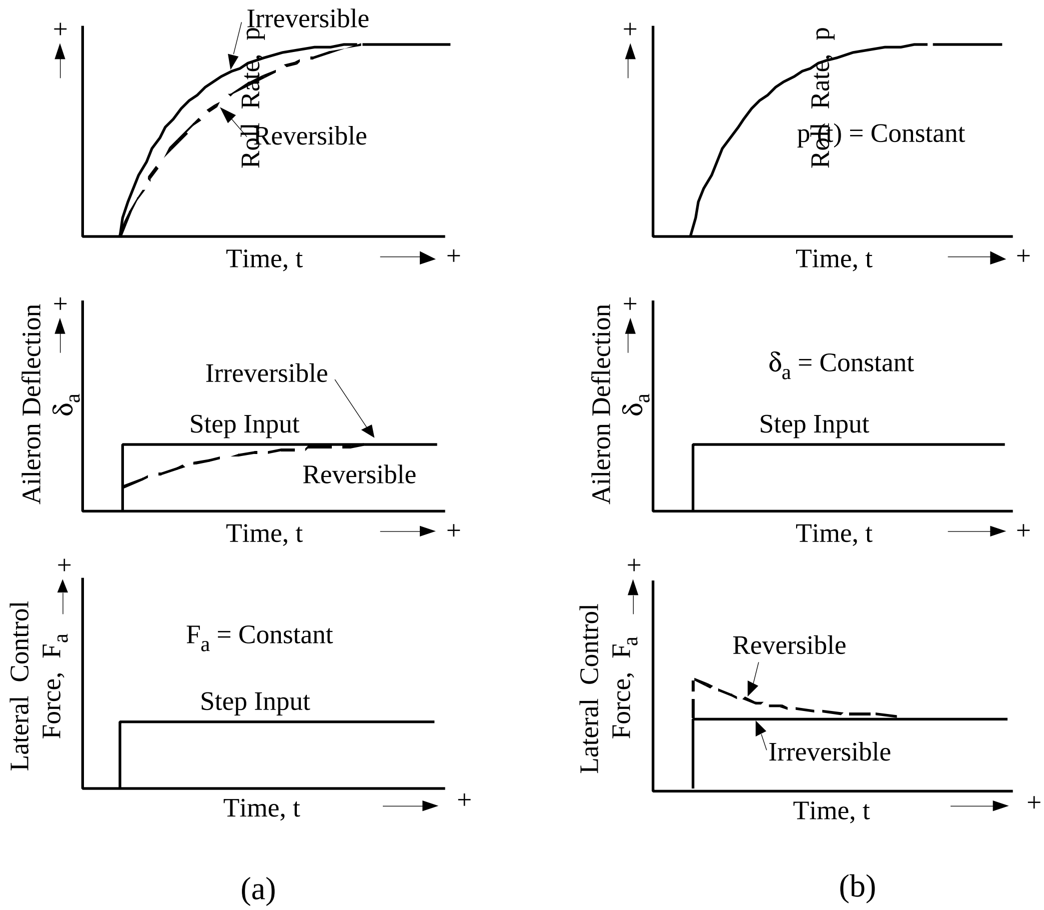


Figure 5.84
Possible Response Characteristics with Reversible and Irreversible Control Systems

5.8 TEST PROCEDURES AND TECHNIQUES ROLLING PERFORMANCE

5.8.1 Preflight Procedures

A thorough investigation of rolling performance and associated roll handling qualities must begin with careful preflight planning. The purpose and scope of the investigation must be clearly defined, then a plan of attack or method of test can be formulated.

Preflight planning must begin with research. This includes a study of the airplane and a thorough study of the lateral control system-including stability and control augmentation if installed. All available information on roll handling qualities should be reviewed. Much useful information may be obtained from pilots and engineers familiar with the characteristics of the airplane.

The flight test team should give due consideration to roll restrictions imposed on the test vehicle. During rolling performance tests, the airplane may be pushed near its boundaries of controllability. Flight testing in suspected regions of roll coupling warrants a cautious, methodical approach and must be accompanied by thorough computer studies that stay current with flight data.

The particular mission tasks to be investigated must be determined and clearly understood by the flight test team. Knowledge of the mission and the associated tasks allows determination of appropriate test conditions - configurations, altitudes, centers of gravity, trim airspeeds, and gross weights. Test conditions must be commensurate with the mission environment of the airplane. Center of gravity position is not particularly critical for rolling performance tests. Tests at normal operational CG positions for a test loading are generally adequate; however, if feasible, the most aft operational CG positions should be utilized. Rolling performance may be altered markedly by various combinations of external stores and/or rolling moments of inertia. The external or internal stores loading which results in the maximum rolling moment of inertia, I_{xx}, should be carefully investigated. Asymmetric store loadings may also seriously impair rolling performance; these conditions should be investigated on any airplane which may carry asymmetric stores in operational use.

The amount and sophistication of instrumentation will depend on the purpose and scope of the evaluation. A good, meaningful qualitative investigation can be performed with only production cockpit instruments and portable instrumentation-hand held force gauge and stopwatch. Automatic recording devices, such as oscillograph, magnetic tape, and telemetry, are very helpful in rapid data acquisition and may be essential in a long test program of quantitative nature. Special sensitive cockpit instruments are also very useful. The parameters to be recorded ant the ranges and sensitivities of test instrumentation will vary somewhat with each test program.

The final step in preflight planning is the preparation of pilot data cards. An example of a rolling performance data card is shown in Figure 5.85. Since the pilot may be recording different parameters for each test condition, the use of different data cards for each test condition may be mandatory. The data cards should list all quantitative information desired and should be easy to interpret in flight. Blank cards should be used for appropriate qualitative pilot comments.

ROLLING PERFORMANCE RECORD		CARD NUMBER
AIRPLANE TYPE	PILOT	PTR-BIS
BUREAU NUMBER	T.O. GROSS WEIGHT	DATE

T.O. CG

GEAR DOWN ____% MAC GEAR UP ____ % MAC | T.O. TIME ________ LAND TIME ________

EXTERNAL LOADING	ALT
CONFIGURATION	POWER

CONTROL SYSTEM MECHANICAL CHARACTERISTICS

BREAKOUT & FRICTION CN __________	FREEPLAY
CONTROL SYSTEM OSCILLATIONS CN __________	CENTERING

ROLLING PERFORMANCE

FUEL START ______________________ END ____________________

V_0 M	CN	DIR OF ROLL	δ_α	Δ_ϕ	P_{SS}* Δt*	F_α	β

OAT___________ OAT___________

v_0__________ v_0__________

*CROSS OUT PARAMETER NOT REQUIRED

Figure 5.85
Rolling Performance Data Card

5.8.2 Flight Test Techniques

5.8.2.1 THE QUALITATIVE PHASE OF THE EVALUATION

Rolling performance and associated roll handling qualities must be evaluated in relation to their influence on various mission tasks. Therefore, the test pilot must devote a portion of the flight test time to performing or simulating the mission tasks which have been selected. The pilot's qualitative opinion of the maneuvering capabilities of the airplane depends, to a substantial extent, on rolling performance. Due consideration should be given during this phase of the test to the following points:

1. Whether the mission tasks will be performed in VFR and IFR weather, or strictly VFR conditions.

2. The availability of an auto-pilot or automatic flight control system for pilot relief.

3. If lateral-directional stability or control augmentation systems are installed, the consequences of their failure.

The test pilot's qualitative opinion of the rolling performance and roll handling qualities in relation to the selected mission task is the most important information to be obtained. Therefore, this phase of the test must not be overlooked. Use of the quantitative test techniques described below hopefully allow the test pilot to substantiate his qualitative opinion.

5.8.2.2 MEASUREMENT OF THE MECHANICAL CHARACTERISTICS OF THE LATERAL CONTROL SYSTEM

Mechanical characteristics of the lateral control system have been previously introduced in the "Test Procedures and Techniques" for lateral-directional flying qualities. Therefore, test techniques for measuring mechanical characteristics of the lateral flight control system will not be restated. This discussion is mainly concerned with the direct influence of mechanical characteristics on roll handling qualities.

5.8.2.2.1 Breakout Forces, Including Friction

Friction in the lateral control system, if substantial, can contribute to poor centering and poor roll sensitivity for small lateral force inputs. Therefore, pure friction should be kept as small as possible. Some lateral control breakout force is generally beneficial. It can contribute to good lateral control centering and it tends to reduce the tendency for the pilot to make inadvertent lateral inputs. Breakout forces tend also to reduce "roll sensitivity" about a trimmed condition, which may eliminate pilot-induced-oscillations in roll for certain flight conditions. However, breakout forces must obviously be maintained at a judicious level.

Breakout forces should be suitably matched to the lateral control force variation with lateral control position. A combination of large lateral breakout force and shallow lateral force gradient results in artificial nonlinearity in lateral control force requirements. This situation generates very poor roll control feel when the pilot attempts to maneuver precisely with small lateral inputs.

5.8.2.2.2 Freeplay

Freeplay in the lateral control system should be as small as possible. Excessive freeplay results in difficulty in performing precise bank angle control tasks with small lateral control inputs.

5.8.2.2.3 Centering

Positive centering of the lateral control system allows the pilot to stop a developed roll rate (in order to establish a desired bank angle) merely by relaxing left or right lateral control force.

5.8.2.2.4 Control System Oscillations

Oscillations of the lateral control surface and lateral control system, initiated by either external perturbations or pilot inputs, should not be noticeable during any bank angle control tasks or rolling maneuvers.

5.8.2.3 FACTORS AND PARAMETERS INDICATING ROLLING PERFORMANCE

The specific technique employed in the flight test program can be easily comprehended if the parameters commonly used as quantitative measures of rolling performance and roll handling qualities are clearly understood. Therefore, these factors and parameters are summarized here.

5.8.2.3.1 Rate of Roll, p

The steady state roll rate p_{SS}, obtainable with various magnitudes of lateral control deflection obviously influences the pilot's opinion of the maneuvering capabilities of the airplane. The time required to make bank angle changes or bank angle corrections is directly related to this characteristic.

A parameter computed using steady state roll rate, $\frac{pb}{2V_T}$, was frequently utilized in the past as a quantitative measure of an airplane's rolling performance. The concept of the roll helix angle, or "non-dimensional roll rate" $\left(\frac{pb}{2V_T}\right)$, as an indicator of rolling performance was justified on the basis that pilots desire an increase in roll rate with faster airspeeds and also desire that small airplanes be capable of higher roll rates than large airplanes. Thus, by specifying a <u>minimum</u> $\left(\frac{pb}{2V_T}\right)$, higher roll rates are required as true airspeed <u>increases</u> and wing span <u>decreases</u>. However, if true airspeed is <u>very high</u> and the airplane's wing span is <u>very small</u>, steady state roll rates which <u>exceed</u> maximum useable roll rates are required to meet the <u>minimum</u> $\frac{pb}{2V_T}$ of some specifications.

Another parameter computed using steady state roll rate, $\frac{pb}{2}$, has also sometimes been utilized as a measure of the rolling performance of fairly large airplanes in the approach and landing phases of mission accomplishment. This parameter is actually the <u>vertical velocity</u> of the wing tip during a rolling maneuver. By stating a <u>minimum</u> $\frac{pb}{2}$, the attempt is made to provide adequate rolling performance to counteract the influence of the <u>maximum</u> vertical gusts experienced in close proximity to the ground.

Steady state roll rate, wing tip helix angle, and wing tip vertical velocity are easily determined or computed from flight tests. However, they are obviously incomplete indicators of rolling performance since the influence of roll acceleration and deceleration is not considered.

5.8.2.3.2 Roll Mode Time Constant, τ_R

The roll mode time constant τ_R, is the time in seconds for the roll rate to build to 63.2 percent of its steady state value following a step input of lateral control deflections. It is obviously a measure of the roll acceleration or deceleration following lateral control position changes, thereby influencing the pilot's opinion of the maneuvering capabilities of the airplane. However, its value can also affect the piloting technique utilized in bank angle control tasks.

For an airplane with a relatively short roll mode time constant, the lateral control system is a "roll rate commanding" system. The pilot applies the lateral control input, the steady state roll rate is quickly attained, and the pilot holds the input until the desired bank angle is approached, then takes it out to stop the roll rate at the desired bank angle. The pilot finds this type of roll control natural and satisfactory from a technique point of view (Figure 5.86).

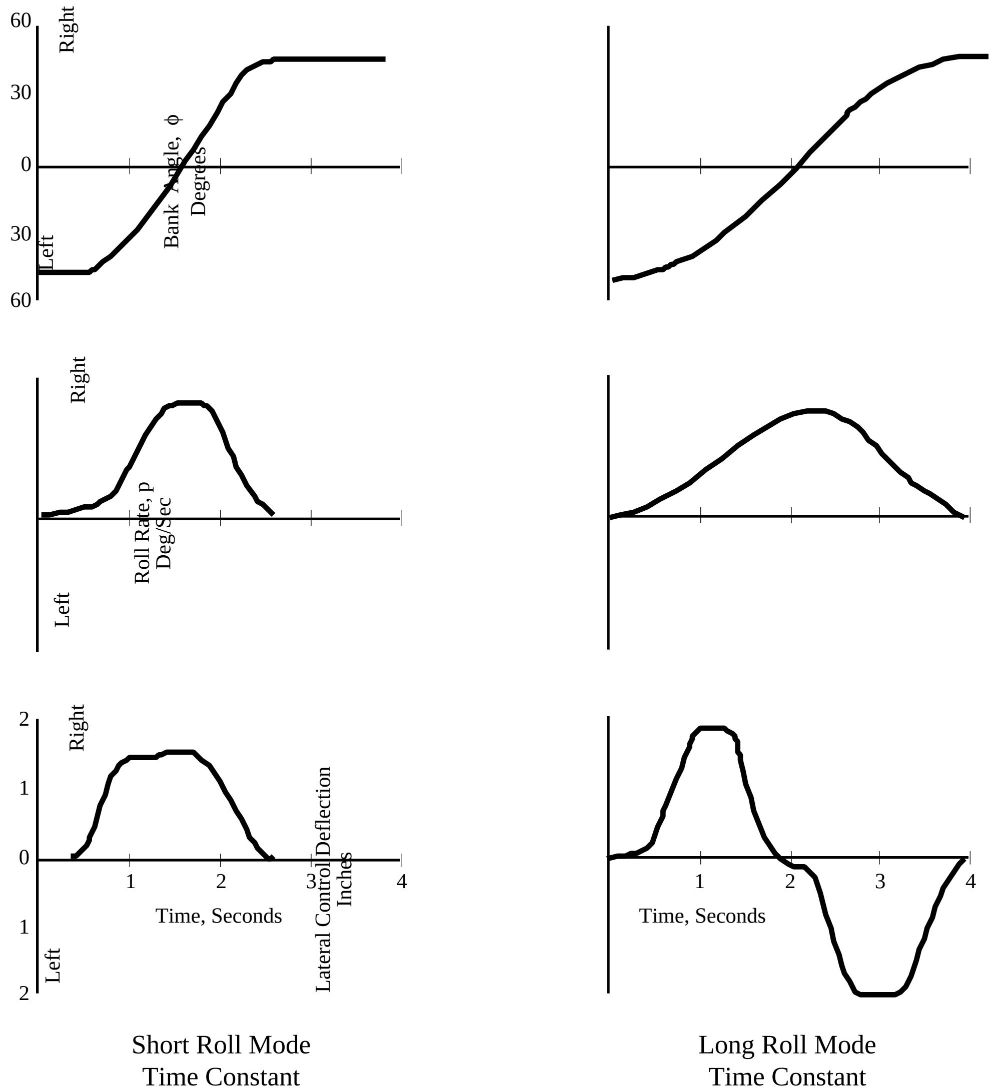


Figure 5.86
Typical Influence of τ_R on Roll Control Techniques

If the roll mode time constant is <u>too long</u>, the initial roll response to a lateral control input may be sluggish, thus impairing quick and precise maneuvering capabilities. The pilot may resort to forcing or driving the <u>initial response</u> he desires by applying large initial inputs of lateral control deflection. However, this technique is generally unacceptable, since the roll rate continues to <u>accelerate</u> or <u>build up</u>, and therefore is difficult to stop. Lateral control inputs essentially command <u>roll acceleration</u> vice <u>roll rate</u> when the time constant is large. During the response time of interest to the pilot - one or two seconds following a lateral control input - the roll rate exhibits a constant change. Therefore, the pilot has an uncomfortable or insecure feeling about both the magnitude of the final roll rate and the magnitude of the bank angle excursion. In flying an airplane with this characteristic, the pilot generally pulses the lateral control system to get the roll started, then leads with a pulse in the opposite direction to stop the roll. This type of roll control requires increased pilot attention and adaptability (Figure 5.86).

The results of flying qualities investigations have revealed that the roll mode time constant should be no greater than one second for high maneuverability airplanes in flight phases or tasks which require precision tracking or precise flight path control. For all airplanes in all phases of mission accomplishment, a roll mode time constant greater than 1.4 seconds generally results in objectionably sluggish roll response and requires a change in piloting technique. Determination of the roll mode time constant from in-flight tests requires special sensitive automatic recording devices (oscillograph, magnetic tape, telemetry, etc.) and a special data analysis procedure to be presented later.

5.8.2.3.3 Bank Angle Change in a Given Time, ϕ_t

The bank angle change in a given time parameter, ϕ_t, is probably the best single indicator of rolling performance. It depends both on steady state roll rate, p_{ss}, and the roll mode time constant, τ_R. In addition, when the bank angle change is timed from the initiation of the pilot's lateral force application, ϕ_t, will reflect any freeplay, lost motion, flexibility, or lag in the lateral control system. By basing the parameter on bank angle changes which are representative of operational maneuvers, rolling performance is expressed in direct and meaningful terms. Determination of the bank angle change in a given parameter can be easily and accurately made from flight tests with automatic recording devices. It can be approximately measured using only cockpit instruments and portable instrumentation.

5.8.2.3.4 Lateral Control Force, F_a

Lateral control forces required to obtain the rolling performance necessary for various mission tasks should be comfortable for the pilot. Forces of too large or too small a magnitude cause objectionable sluggishness or sensitivity in response to small lateral control inputs. The maximum and minimum forces which are acceptable in any airplane depend on the mission of the airplane. In general, lower lateral control forces are desirable for high maneuverability airplanes, and increased lateral control forces are required and desired in low maneuverability airplanes. If the airplane is equipped with a wheel or yoke type cockpit controller, higher lateral control forces may be accepted since the pilot is able to apply both hands, thus larger forces to the control.

The measurement of lateral control forces during rolling maneuvers is difficult unless automatic recording devices are available. If a hand-held force gauge is used to measure lateral control forces, care must be take to ensure that the control does not reach the limit stops during measurement.

5.8.2.3.5 Lateral Cockpit Control Position

The lateral cockpit control movements required to generate rolling performance necessary for various mission tasks should neither be to small or too large. If the pilot is continually striking the lateral control stops in order to obtain acceptable rolling performance, he probably feels insecure and uncertain about the maneuverability of his airplane. However, the pilot should be able to obtain full lateral control deflection if needed without interference from his body or his flight equipment. For airplanes equipped with wheel or yoke type cockpit controls, the lateral cockpit control movement required to generate satisfactory rolling performance should not necessitate inordinate arm motion. More than 60 to 90 degrees of wheel or yoke throw in either direction is generally considered excessive.

5.8.2.3.6 Dutch Roll Influence

The degree of excitation of the Dutch roll motion and the resultant influence on rolling performance and roll handling qualities depends on many factors. For optimum rolling characteristics, the pilot should visually detect little or no Dutch roll motion during rolling maneuvers required in operational mission tasks. If the Dutch roll is excited to a significant degree, it will probably be manifested in one of two ways, depending on the roll-to-yaw ratio of the motion.

For airplanes exhibiting relatively low roll-to-yaw ratios, corresponding to weak dihedral effect, Dutch roll excitation during rolling maneuvers is noted as sideslip excursions. Rolling performance, in terms of steady state roll rate or bank angle change in a given time, is probably not seriously affected by these sideslip excursions. However, the sideslip per se is important since it precipitates oscillations of the nose of the airplane during turns and/or a lag or initial reversal in turn rate during a turn entry. The amount of sideslip acceptable during bank angle control tasks depends on the phase angle of the sideslip, i.e., whether the sideslip is adverse or proverse. Adverse sideslip is actually easier for the pilot to counteract; "natural" rudder coordination - rudder application in the same direction as the lateral control input and turn - can be utilized. However, if the sideslip is proverse, the rudder coordination required to reduce sideslip is difficult and unnatural, since cross controlling is necessary. During rapid maneuvering, most pilots unconsciously apply rudder in the same direction as the lateral control input and turn even though they are aware that proverse yawing moments and proverse sideslip is being generated. This seriously complicates the sideslip excursion problem.

The second way in which the Dutch roll motion is manifested during rolling maneuvers is more direct, and possibly, more detrimental. For airplanes exhibiting relatively moderate to high roll-to-yaw ratios, corresponding to significant dihedral effect, Dutch roll excitation may enhance or impair rolling performance in terms of steady state roll rate or bank angle change in a given time. Whether the rolling performance itself will be enhanced or impaired is dependent on both the sign of the dihedral effect and the direction of the yawing moment and sideslip excursions. Additionally, the excitement of a moderate to high roll-to-yaw ratio , Dutch roll motion during bank angle tracking tasks may result in oscillations in roll rate response (see Figure 5.72.). This oscillatory roll response can severely impair roll handling qualities, generating overshooting of and oscillations about the desired bank angle during rolling maneuvers and generally precluding accurate and precise bank angle control. The degree of Dutch roll excitation which can be accepted again depends on the sign of the effective dihedral and whether the generated yawing moment and sideslip are adverse or proverse. If the airplane exhibits positive dihedral effect, adverse yawing moments and sideslip actually enhance Dutch roll damping during tight, precision bank angle control tasks. Conversely, proverse yawing moments and sideslip tend to cause decreased Dutch roll damping in this situation. Flying qualities investigations have revealed that the degradation in roll handling qualities is proportional to the amount of roll rate oscillation, P_{osc}, about some mean or average value, p_{ave}, and the phase angle, ψ_β, of the Dutch roll component of sideslip, i.e., whether the sideslip is adverse or proverse.

Exact quantitative parameters indicating the Dutch roll influence on rolling performance and roll handling qualities can be measured only with automatic recording devices. However, the test pilot may be able to determine the direction and magnitude of the sideslip excursions during rolling maneuvers from a cockpit mounted sideslip indicator.

5.8.2.4 MEASUREMENT OF ROLLING PERFORMANCE

Although the actual technique utilized in obtaining rolling performance parameters is easily understood, the test pilot will probably realize difficulty in generating accurate, repeatable data without extensive flight test experience and practice. The difficulty associated with obtaining accurate quantitative parameters <u>increases</u> with <u>decrease</u> in available instrumentation. The specific approach to the rolling performance test is determined by several <u>general considerations</u>, which must be discussed prior to proceeding to the actual techniques involved.

1. The <u>time required</u> for the test pilot to establish the lateral control input to initiate the roll must be <u>as small as possible</u>. The input should approximate a <u>step input</u>. If the time required to accomplish the input exceeds <u>one-half second</u>, some of the parameters of interest, such as roll mode time constant and bank angle change in a given time, may be impossible to determine accurately or meaningfully. The test pilot should strive for inputs which are accomplished within a time interval of <u>0.2 second</u> for the most representative test results. The use of both hands on the control stick or yoke may facilitate <u>snapping</u> the lateral control quickly to the desired position.

2. <u>Lateral control deflection</u> utilized for each roll should be increased in incremental steps until full lateral control defection is reached. Obviously, the test should be terminated short of maximum control defection if flight restrictions prohibit use of full control deflection or if unusual airplane responses, such as roll coupling tendencies or excessive sideslip excursions are encountered. The flight test team may desire to utilize some sort of lateral control deflection restrictor if <u>exact partial</u> lateral control deflection points are desired or if severe airplane response is anticipated with large lateral control inputs. In most cases, only full lateral control deflection is required for <u>repeatable</u> cockpit control position and the use of control deflection restrictors is not necessary. Nevertheless, the test pilot should use appropriate partial lateral control deflections as a build up to full deflection rolls at each test condition.

Generally, one-quarter increments are utilized, although one-half of all deflection increments may be used if no unusual airplane response is expected. The test pilot can estimate these incremental lateral control deflections quite accurately, particularly if the partial deflection inputs are practiced on the ground to establish the lateral stick or yoke positions required. Correlation of stick or yoke position with various markings and flight instruments in the cockpit is very helpful.

3. The direction of roll should be alternated so that the influence, if any, of roll direction on rolling performance may be ascertained. This influence is generally more pronounced in propeller-driven airplanes. Therefore, left and right rolls should be performed in these airplanes with the same lateral control input, i.e., left and right rolls with one-half lateral control deflection, then left and right rolls with full lateral control deflection. Since rolling performance in pure jet airplanes is usually not influenced by the direction of roll, left and right rolls may be performed in these airplanes with consistently increasing control deflection.

4. The airplane should be in a trimmed, unaccelerated flight condition prior to the initiation of the roll. The test pilot should make every attempt to keep the longitudinal control position at the trim position throughout the roll. No effort should be made to counteract pitching moments generated during high rate rolls, since the pilot is likely to augment the motion, vice stop it. (Techniques for investigating roll coupling will be presented later.)

5. The use of rudders during the rolling performance tests will vary with the airplane type and mission, as well as the flight conditions under consideration. In general, rudders should remain free for high maneuverability airplanes (Class IV) and for all carrier-based airplanes in Category C flight phases (levels 1 and 2). For low to medium maneuverability airplanes, the rudders may be used to reduce adverse sideslip (sideslip retarding roll rate) providing the rudder inputs are simple, easily coordinated, and consistent with normal piloting technique for the mission and task under evaluation. Rudder inputs should not be employed to produce proverse sideslip which could augment rolling performance.

6. The bank angle change through which the roll is allowed to continue depends on roll restrictions, test conditions, and data desired. In general, rolls should be commenced from wings level, zero roll rate conditions. However, if the bank angle changes is limited by roll restrictions or test conditions (airspeeds near stall, high lift configurations, excessive sideslip excursions), the roll may be initiated from an established bank angle so as to roll through wings level to an opposite bank angle. Thus, the airplane is always upright throughout the maneuver.

7. If the rolling performance tests are to be performed between very low airspeeds near stall to near maximum airspeeds, the test should commence at airspeeds near the center of the spectrum and proceed to the end points. This procedure allows the flight conditions generally considered most critical to be approached with an adequate build-up program.

8. Lateral control forces will be difficult to obtain without automatic recording devices. The hand-held force gauge is usually too cumbersome and annoying to utilize for inflight measurements. Additionally, the hand-held force gauge floating pointer would indicate the transient lateral force applied during the sharp control input vice the steady state control force of interest. For irreversible lateral control systems where lateral force is merely a function of lateral stick position, the lateral control forces may be accurately measured on the ground with the hand-held force gauge. For other types of lateral control systems, the test pilot may need to resort to estimating lateral control force requirements during flight tests. When measuring or estimating forces required for full deflection, the control should be placed just short of the stop. If the control is against the stop, the force measured will be whatever the pilot decides to apply.

9. The test pilot should be able to note the direction of sideslip excursions during rolling maneuvers merely from the response of the airplane. If a cockpit mounted sideslip indicator is available, the magnitude of the adverse or proverse sideslip can be determined. Sensitive automatic recording devices are necessary if parameters such as roll rate oscillations and the phase angle of the Dutch roll component of sideslip are to be determined.

10. Two methods may be employed to obtain the desired trim airspeeds of the test. The primary method is merely to utilize the power setting required for the airspeed and altitude combination. An alternate method may be used in propeller-driven airplanes to eliminate the influence of power variation on roll characteristics. The power used is that required to maintain a level flight trim airspeed midway between the maximum and minimum airspeed of the test spectrum. Other airspeeds are then attained by trimming in climbs and descents with the power setting remaining constant.

11. Altitude variance during rolling performance tests should not exceed ±2000 feet from the selected base altitude for the tests.

The specific method utilized to obtain various rolling performance parameters depends on the parameters themselves as well as the amount of instrumentation available. Specific techniques will now be presented for test conducted with and without extensive instrumentation.

5.8.2.4.1 Automatic Recording Devices

If a complete package of automatic recording instrumentation is available, all rolling performance parameters may be obtained from the recording traces. The test pilot merely performs the rolls as follows:

1. Stabilize and trim carefully in the desired configuration at the desired flight condition.

2. Actuate the automatic recording devices and perform the roll.

3. After completing the roll, recover to erect flight, deactivate the automatic recording devices, and prepare for the next roll.

5.8.2.4.2 Portable Instrumentation and Cockpit Mounted Instruments

Several of the parameters needed to completely describe rolling characteristics cannot be determined with only cockpit instruments and portable instrumentation. However, some approximate, yet meaningful, data can be obtained. The general procedures for the test remain the same, with the following specific procedures recommended for certain situations.

If the parameter required is steady state roll rate, p_{ss}, the following approach may be utilized.

1. When a rate of roll indicator is available in the cockpit, the test pilot merely rolls through a bank angle change large enough for the roll rate to reach steady state. The value of the roll rate is noted from the indicator.

2. If no roll rate indicator is installed, the test pilot must compromise by obtaining an average roll rate computed from a timed bank angle change. The average roll rate will closely approximate the actual steady state roll rate if the bank angle change is as large as possible and if the period during which the roll rate is building up is not timed. For example, a left roll could be initiated from a 45 degree right bank angle. The timing would start at wings level and end after a 360 or 180 degree bank angle change. The attitude indicator and a one- or three-second sweep stopwatch may be employed to obtain $\Delta\phi$ and Δt. The average roll rate can be computed after the flight.

If the parameter required is bank angle change in a given time, ϕ_t, the following approach may be utilized.

1. The parameter, ϕ_t, may be approximately determined with a one- or three-second stopwatch and the attitude gyro. The test pilot may desire to set up for this test so as to roll about a wings level condition. The data required is merely the time necessary to accomplish the bank angle change under consideration. Timing should start at the initiation of the lateral force input. From the $\Delta\phi$ and the Δt data, approximate values of ϕ_t may be extrapolated by assuming a

linear bank angle-time relationship. This is obviously not a correct assumption, since this relationship is nonlinear during the time interval prior to the roll rate reaching steady state. Therefore, the ϕ_t obtained by this method is only approximate.

2. If the purpose of the test is to check against a specification requirement, such as ϕ_t = 90 degrees in 1.3 seconds, it is only necessary to time the roll through the bank angle change under consideration. If the bank angle change is accomplished in a time interval equal to or less than the specified time, the requirement is met.

5.8.2.5 ROLL COUPLING

Violent roll coupling should never be encountered during any maneuver which conceivably could be utilized in a operational mission task. Some pitching and yawing motion, not directly attributable to pilot control inputs or yawing moments generated by lateral control inputs or roll rate, may be expected during rolling maneuvers. However, these motions should not generate sideslip and angle of attack excursions large enough to exceed structural limits or result in uncontrollable flight conditions, such as roll auto-rotation. The yawing and pitching experienced during rolling maneuvers utilized in typical mission tasks should not be severe enough to impair the satisfactory completion of the tasks. High maneuverability airplanes may be particularly susceptible to the detrimental effects of roll coupling.

Flight testing for maximum rolling performance under conditions favorable for roll coupling, or flight testing specifically to determine if roll coupling may be encountered, require cautious, methodical approaches. The following general guidelines are offered for planning and conducting these test programs:

1. Thorough computer studies, based on known or assumed values of stability derivatives should be conducted prior to commencement of flight tests. The studies should be continued along with the flight tests, systematically being updated with flight test data.

2. Complete airplane instrumentation with automatic recording devices is mandatory if the results of these tests are to be analyzed properly. Telemetry and real-time data processing should be employed, if available. A qualified engineering observer, with communications to the test pilot, should monitor the flight test records continually at the telemetry station.

3. Thorough rolling performance tests should be conducted from unaccelerated flight conditions initially. (Violent roll coupling has been encountered during these tests on some airplanes.)

4. Assuming that no roll coupling is encountered during rolls from unaccelerated flight conditions, the test program may proceed into rolls initiated form accelerated flight conditions. The applied normal acceleration should be increased or decreased to maximum and minimum values consistent with operational piloting tasks. Generally, rolls from flight conditions where applied normal acceleration varies from 0 g to .8 N_L are considered adequate. However, airplane structural limits may restrict the scope to lower g levels. Initial normal acceleration should be increased or decreased from 1 g in small increments in a planned build-up program to maximum and minimum values. At each point (1.5g, 0.5g, 2.0g, 0.0g, etc.), partial lateral control deflection rolls should be utilized in a methodical build-up program to maximum performance rolls. These rolls may be initiated from steady turning flight, wings level pull-ups, or push-overs. The techniques utilized to establish the desired value of normal acceleration should be similar to those described earlier for longitudinal maneuvering stability tests.

5. If violent roll coupling is encountered, the pilot should make no attempt to control the yawing and pitching motion with rudder and elevator inputs. The pilot's attempts would probably be ill-timed and augment the violent excursions. The recommended procedure on encountering violent roll coupling is to stop the roll by use of lateral control movements and neutralize the longitudinal and directional controls.

6. Flight tests should be terminated when the test pilot, the observing engineer in telemetry, or the computer studies predict or indicate that the nest roll may exceed the "critical limit" of controllability. The only means of determining the exact critical limit, or course, is to exceed it, which is obviously not a required or desired approach.

5.8.3 Postflight Procedures

As soon as possible after returning from the flight, the test pilot should write a brief, rough qualitative report of the rolling performance and associated roll handling qualities exhibited during the mission tasks under evaluation. This report should be written while the events of the flight are fresh in the pilot's mind. Qualitative pilot opinion, appropriately related to the mission tasks under evaluation, will be the most important part of the final report.

Representative data should be selected to substantiate the pilot's opinion. Several suggested data presentation schemes will be introduced. No matter what method is used, it should be clear, concise, and complete.

5.8.3.1 MECHANICAL CHARACTERISTICS OF THE LATERAL CONTROL SYSTEM

Mechanical characteristics may be effectively presented in tabular form as previously discussed and illustrated in “Test Procedures and Techniques - Non-maneuvering Tasks” (Longitudinal Flying Qualities).

5.8.3.2 ROLLING PERFORMANCE

Rolling performance data may be presented as plots of steady state roll rate, and/or bank angle change in a given time versus airspeed or Mach number. Usually, only full lateral control deflection or maximum rolling performance is presented since full deflection is generally the only repeatable lateral control position utilized during the flight tests. However, partial control deflection data may be presented, if desired. Applicable minimum specification requirements are normally superimposed on the rolling performance plots. Typical data presentation is shown in Figure 5.87.

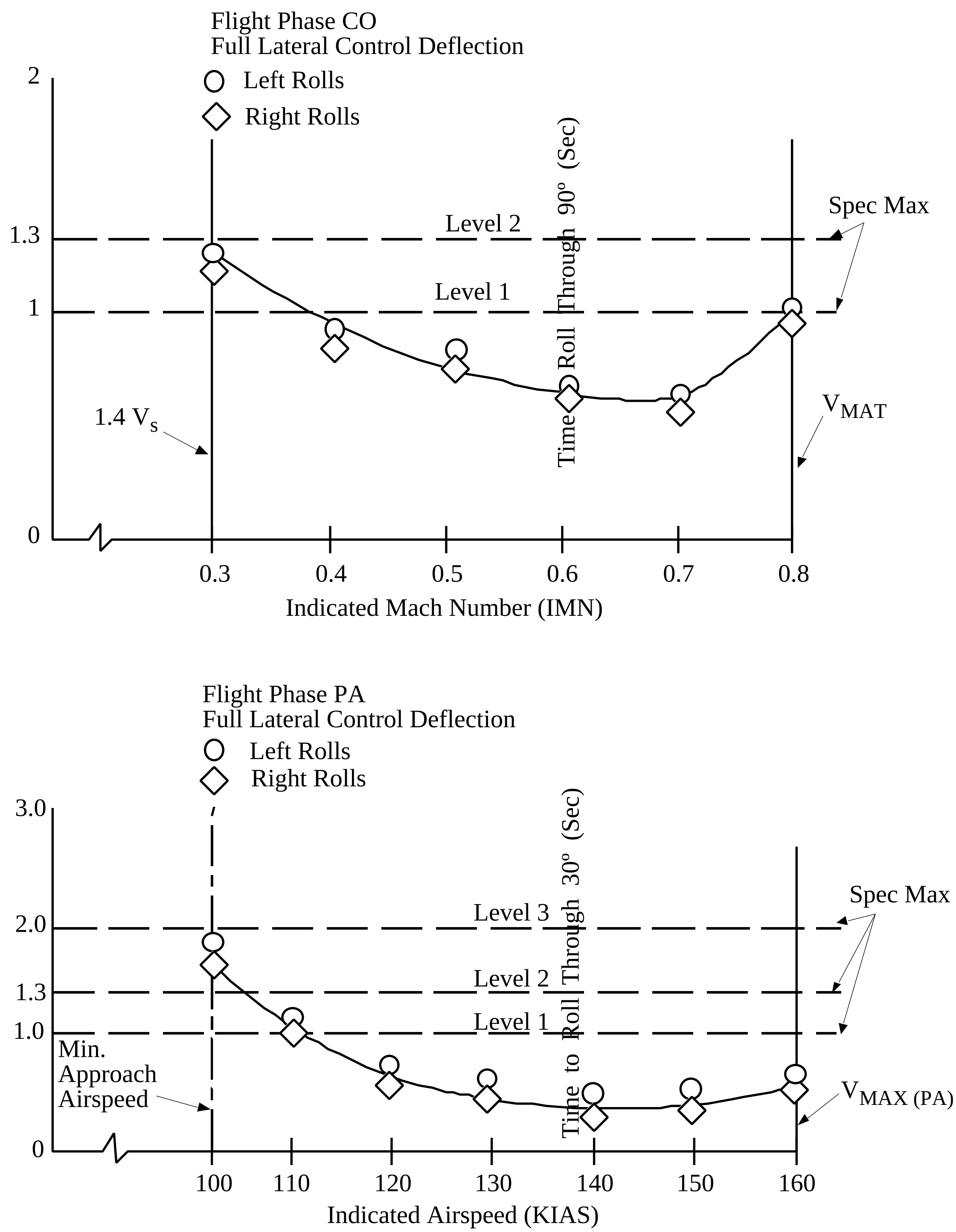


Figure 5.87
Rolling Performance in Configurations CO and PA

If lateral control force and sideslip data are available from the flight tests, these parameters may be presented with other rolling performance data (Figure 5.88). Lateral control forces may also be plotted versus lateral control deflection; this procedure is particularly appropriate for the irreversible lateral control system in which lateral control forces are merely a function of lateral control position.

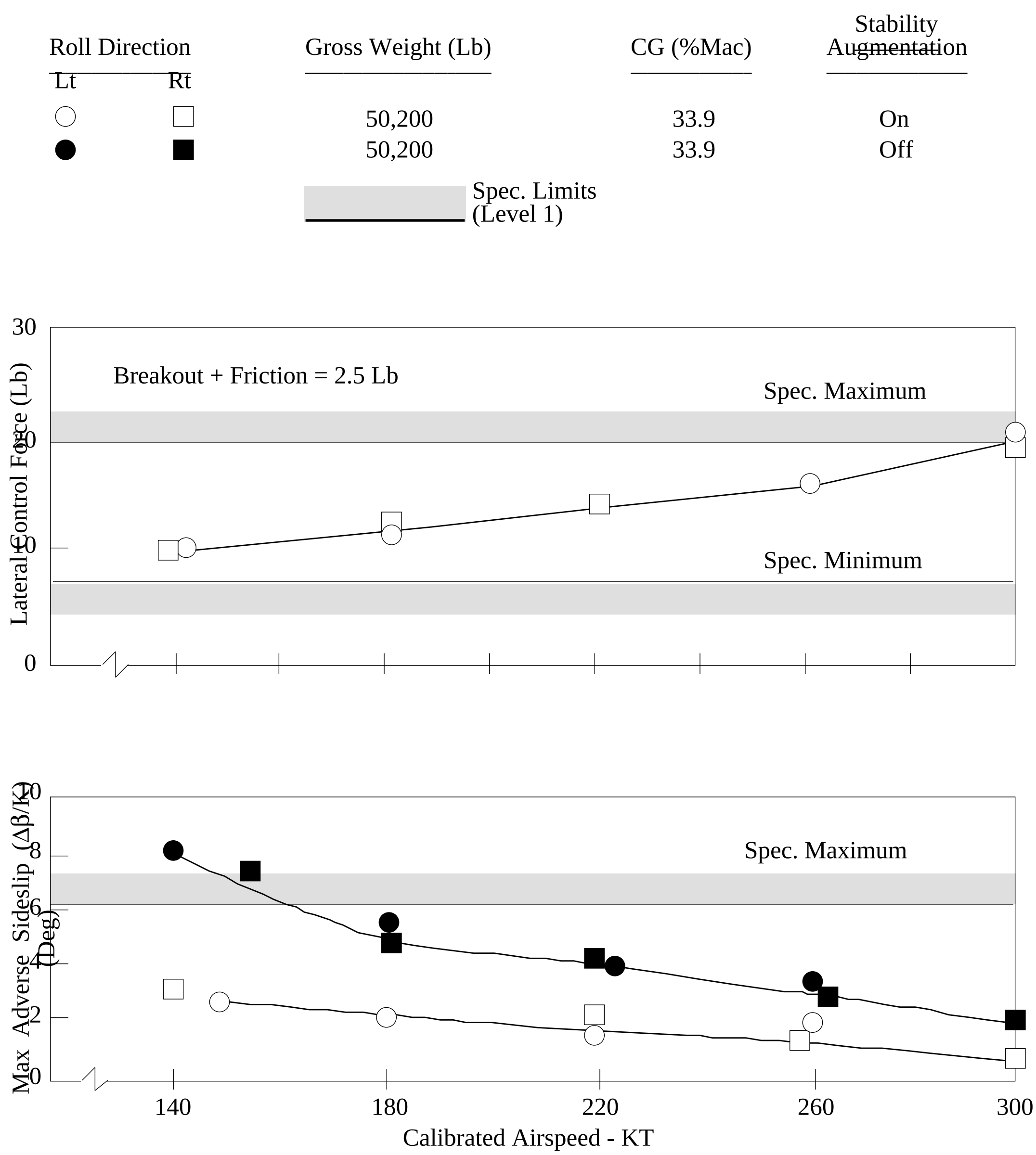


Figure 5.88

Rolling Performance in Configuration Power

If automatic recording devices have been available for the flight tests, more sophisticated data analysis and data presentation techniques may be employed. In particular, the flight test team may desire to present the roll mode time constant, τ_R, the degree of roll rate oscillation, expressed as $\frac{P_{osc}}{P_{ave}}$, and the time required for the pilot to apply the lateral control deflection (Figure 5.89). The latter parameter is a good indication of the quality of the data. Techniques for determining the roll mode time constant and $\frac{P_{osc}}{P_{ave}}$ will be discussed in subsequent paragraphs.

Full Abrupt Lateral Control Deflection Rudder Pedals Free

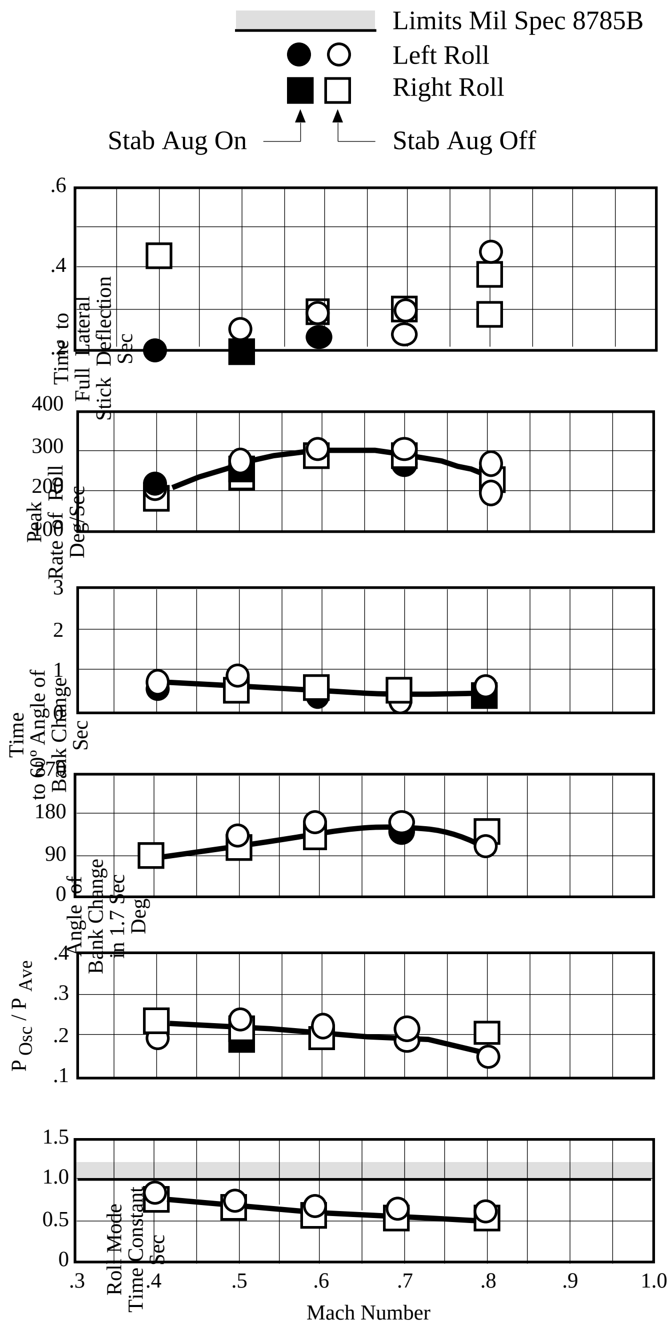


Figure 5.89
Rolling Performance in Configuration Cruise

Model __________ Airplane

BuNo __________________

5.8.3.3 ROLL COUPLING

The influence of roll coupling, if any, on mission accomplishment should be discussed in the technical report. The operational pilot rightly deserves sufficient information on the behavior of the airplane during rolling maneuvers to allow roll coupling to be avoided. Violent roll coupling, if encountered, may be illustrated with one or more time histories of the motion of the airplane during the maneuver. A time history showing violent roll coupling is present in Figure 5.90. Additional parameters may be shown if desired.

5.8.3.4 QUANTITATIVE INDICATIONS OF DUTCH ROLL INFLUENCE

If automatic recording traces are available for roll handing qualities analysis, the degree of Dutch roll excitation during rolling maneuvers may be determined in <u>quantitative terms</u>. Flying qualities investigations have revealed that requirements can and should be placed on the degree of Dutch roll excitation during moderate bank angle change maneuvers such as turn entries. These requirements, although designed to be <u>generally</u> applicable for <u>all</u> combinations of Dutch roll characteristics, are <u>specifically</u> aimed at <u>different</u> Dutch roll responses. This approach is justified on the basis that Dutch roll excitation is manifested in different ways depending on the <u>roll-to-yaw</u> ratio of the motion.

5.8.3.4.1 Low Roll-to-Yaw Ratio

For low roll-to-yaw ratios of the Dutch roll motion, sideslip excursions in themselves cause a degradation in roll handling qualities. Therefore, restrictions have been proposed on the maximum amount of sideslip experienced during large amplitude rolls (Figure 5.91).

5.8.3.4.2 Sideslip Excursions

Following a yaw-control-free step roll control command, the ratio of the sideslip increment, $\Delta\beta$, to the parameter k shall be less than the values specified herein. The roll command shall be held fixed until the bank angle has changed at least 90 degrees.

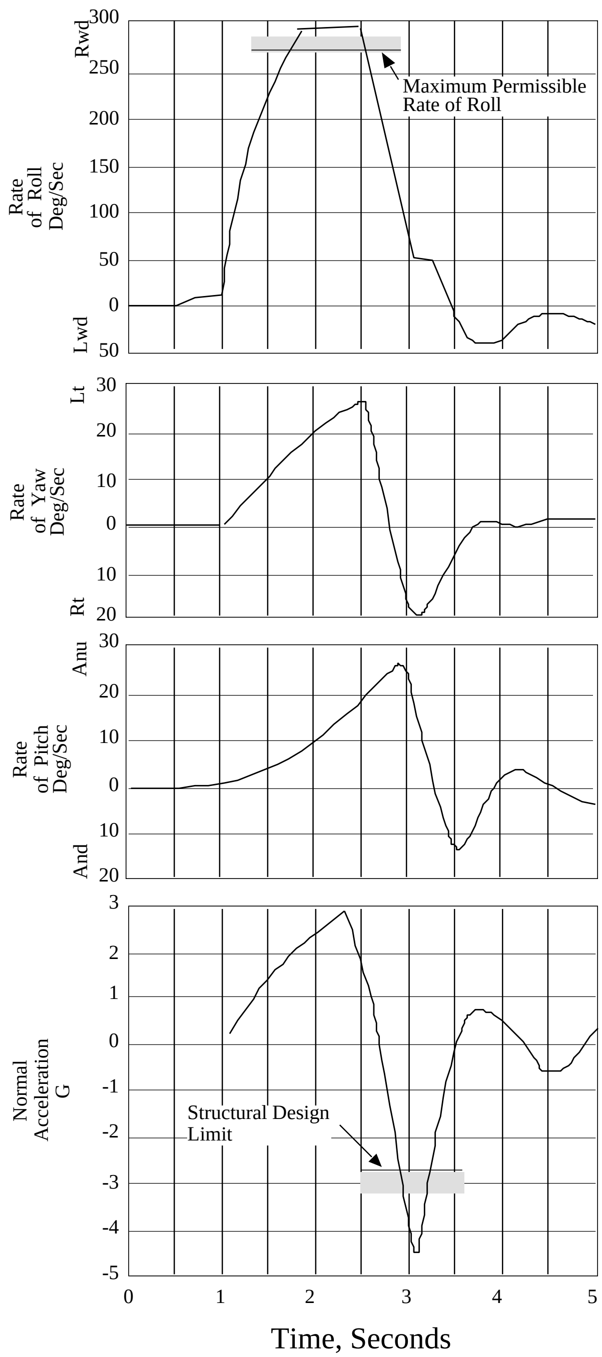


Figure 5.90
Roll Coupling During a Full Aileron Deflection Right Roll

Model_____________Airplane
BuNo__________________

Loading: A	Gross Weight: 12,650 Lbs.
Configuration: Dive	CG: 13.25% MAC
Trim: 350 KIAS, 10,000 Ft.	Stab Aug: On

Level	Flight Phase Category	Adverse Sideslip (Right roll command causes right sideslip)	Proverse Sideslip (Right roll command causes left sideslip
1	A	6 degrees	2 degrees
	B & C	10 degrees	3 degrees
2	A11	15 degrees	4 degrees

$\Delta\beta_{max}$ - maximum change in sideslip at the c.g., occurring within two seconds or one half-period of the Dutch roll, whichever is greater, for a step roll-control command

k - ratio of "commanded roll performance" to "applicable roll performance requirement" of Spec paragraphs 3.3.4 or 3.3.4.1 where:

(a) "Applicable roll performance requirement", (ϕ_t) requirement, is determined form 3.3.4 and 3.3.4.1 for the Class, Flight Phase Category and Level under consideration.

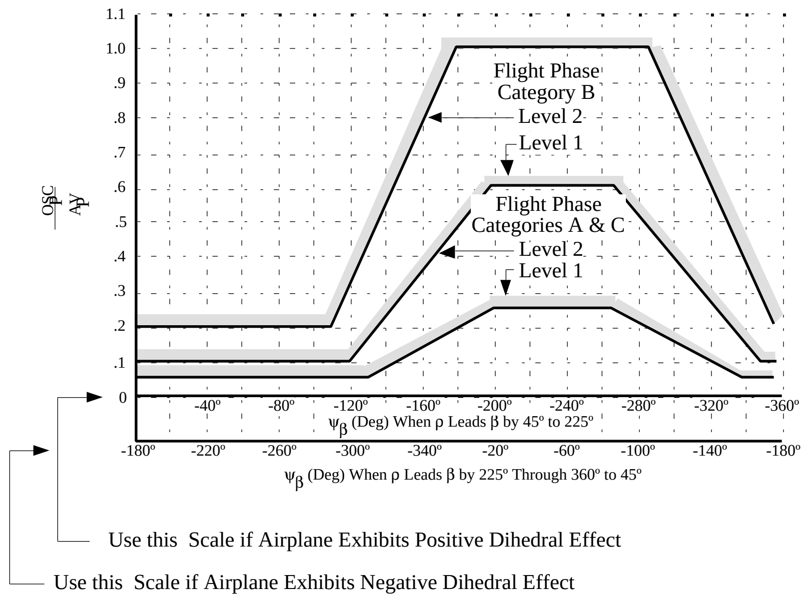
(b) "Commanded roll performance" (ϕ_t) command, is the bank angle attained in the stated time for a given step roll command with yaw control pedals employed as specified in 3.3.4 and 3.3.4.1

$$K = \frac{(\phi_t)_{command}}{(\phi_t)_{requirement}}$$

Figure 5.91
Sideslip Excursion Limitations

5.8.3.4.3 Moderate to High Roll-to-Yaw Ratios

For moderate to high roll-to yaw ratios of the Dutch roll motion, excessive Dutch roll excitation during rolls may generate oscillatory roll rate response. A degradation in roll handling qualities may result. Therefore, restrictions have been made on the amount of roll rate oscillation experienced during abrupt rolling maneuvers (Figure 5.97). (If the phase angle cannot be determined, a phase angle of -240 degrees should be assumed for airplanes exhibiting adverse sideslip, and a phase angle of -60 degrees should be assumed if the sideslip is proverse.)



Where:

$\frac{p_{osc}}{p_{AV}}$ - a measure of the ratio of the oscillatory component of roll rate to the average component of roll rate following a yaw-control-free step aileron control command:

$$\zeta d \leq 0.2: \frac{p_{OSC}}{p_{AV}} = \frac{p_1 + p_3 - 2p_2}{p_1 + p_3 + 2p_2}$$

$$\zeta d \leq 0.2: \frac{p_{OSC}}{p_{AV}} = \frac{p_1 - p_2}{p_1 + p_2}$$

where p_1, p_2, and p_3 are roll rates at the first, second, and third peaks, respectively.

ψ_β - phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in sideslip. (See Figure 5.56c)

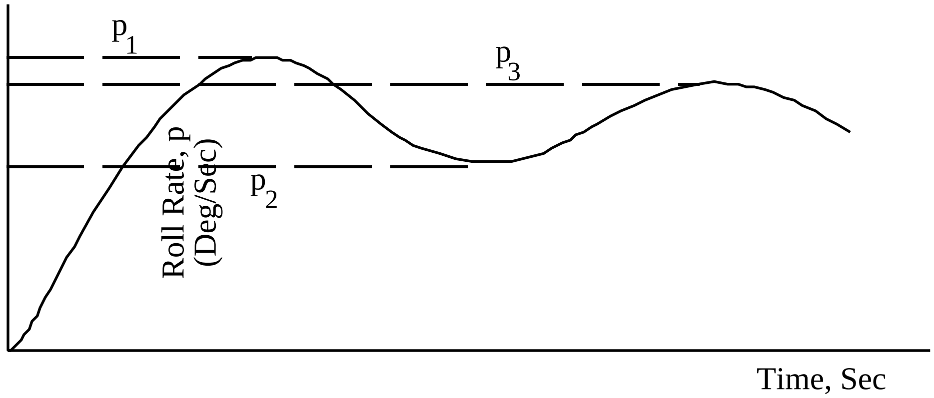


Figure 5.92
Roll Rate Oscillation Limitations

5.8.4 Control Force Coordination

Control forces required in normal maneuvering of the airplane should be of relative magnitudes which "feel" normal to the pilot. This is obviously a qualitative evaluation which the test pilot can perform while maneuvering the airplane through typical mission tasks. The elevator, aileron, and rudder forces and displacement sensitivities as well as breakout forces should be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to another. If the pilot is aware or conscious of a markedly different effort being applied to one control axis, control force coordination may be poor.

Certain specifications may state quantitative guidelines for the control forces required during coordinated maneuvers. The most common ratio utilized is 2:7:1 for longitudinal, directional, and lateral control forces, respectively. This ratio is applicable to all maneuvers normally required in the airplane's mission.

Quantitative requirements for control force coordination may be checked as follows:

1. Trim the airplane in level flight in the Power configuration at altitudes representative of the mission environment.

2. Maneuver the airplane to be at the trim airspeed and altitude in a dive.

3. As the trim airspeed is approached, perform a rolling pullout to simultaneously attain target normal acceleration and roll rate. Utilize rudder inputs, as necessary, to maintain coordinated flight.

4. As target normal acceleration and roll rate are attained, note longitudinal, lateral, and directional control forces.

Control force coordination should also be evaluated qualitatively in the Power Approach configuration during actual approaches.

5.8.4.1 ANALYTICAL DETERMINATION OF THE ROLL MODE TIME CONSTANT

The roll rate response to a step input of lateral control is characterized by an exponential increase in roll rate until a steady state value is attained. This response may be represented by the following relationship:

$$p(t) = p_{ss} \left\{ 1 - e^{-t/\tau_R} \right\} \qquad \textit{eq 5.113}$$

By appropriate manipulation of the last expression, a technique may be evolved by which the roll mode time constant, τ_R may be determined from flight test records. The roll rate response to a step lateral control input may be rewritten as:

$$p(t) = p_{ss} - p_{ss}\, e^{-t/\tau_R} \qquad \textit{eq 5.114}$$

This relationship indicated that p (t) can be graphically represented as the summation of the two parts of the expression (Figure 5.93).

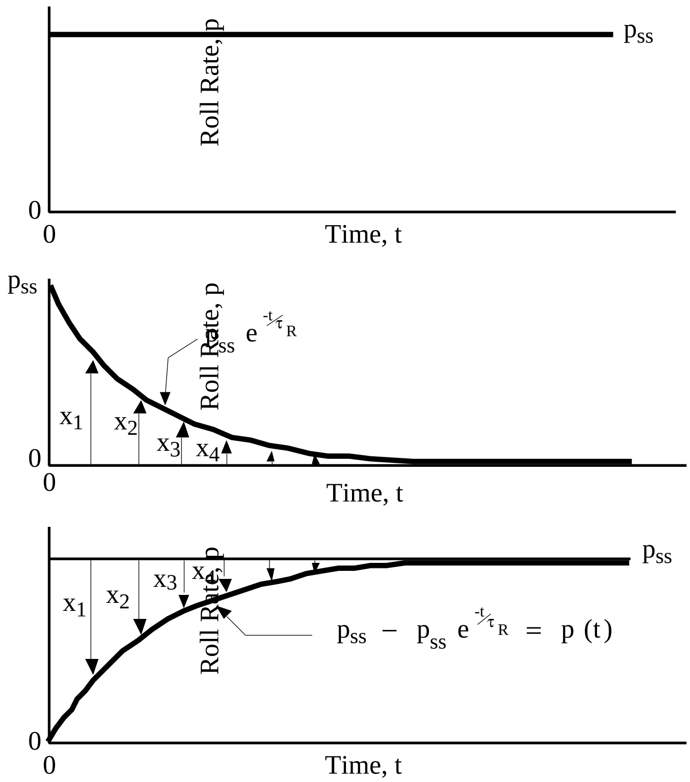


Figure 5.93
Roll Rate Response to Step Lateral Control Input

(Note that the "total" roll rate response curve, p (t), is merely a mirror image of the curve described by $p_{ss}\ e^{-t/\tau_R}$).

Consider now the part of the roll rate response represented by the middle plot of Figure 5.98 and the following expression:

$$X(t) = p_{ss} e^{-t/\tau_R} \qquad \textit{eq 5.115}$$

If the logarithm is taken of both sides of this equation, the following results:

$$\ln X(t) = \ln p_{ss} - \frac{t}{\tau_R} \qquad \textit{eq 5.116}$$

or:

$$\ln X(t) = K_1 - \frac{t}{K_2} \qquad \textit{eq 5.117}$$

Thus, there is a linear relationship between ln X(t) and t. Therefore, if the expression:

$$X(t) = p_{ss}\ e^{-t/\tau_R} \qquad \textit{eq 5.118}$$

is plotted on semilogarithmic graph paper, with X(t) as the logarithmic axis and t as the linear axis, a straight line is generated (Figure 5.99). This the crucial point which allows the determination of τ_R from actual flight record of rolling maneuvers. With the theoretical background presented above, the practical aspects of the manipulation of the flight test data should be easily comprehended.

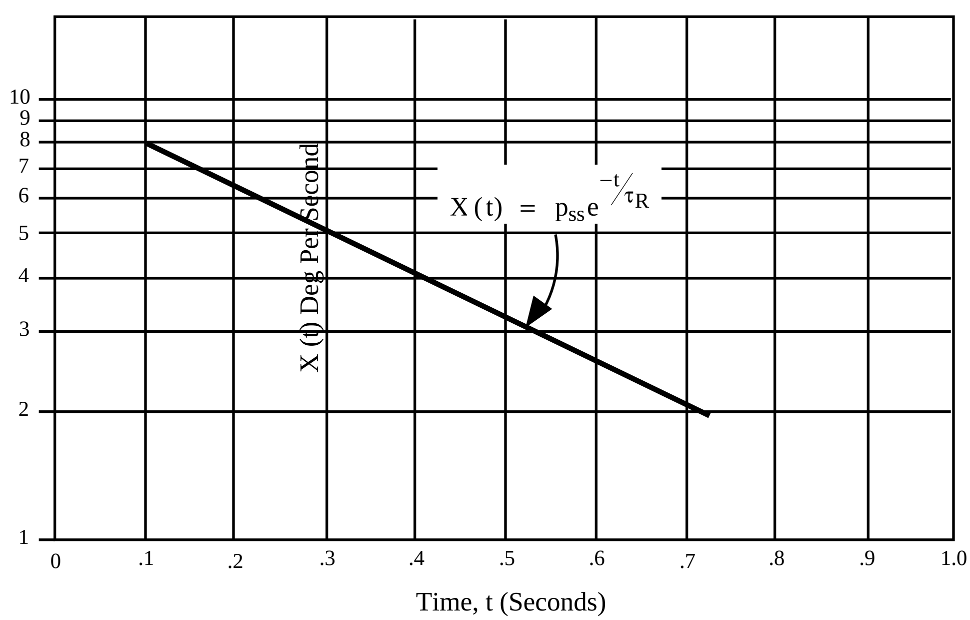


Figure 5.94
$X(t) = p_{ss}e^{-t/\tau_R}$ Plotted on Semi-Log Paper

The roll mode time constant, τ_R, may be determined as follows:

(A sample exercise is presented in Figure 5.95).

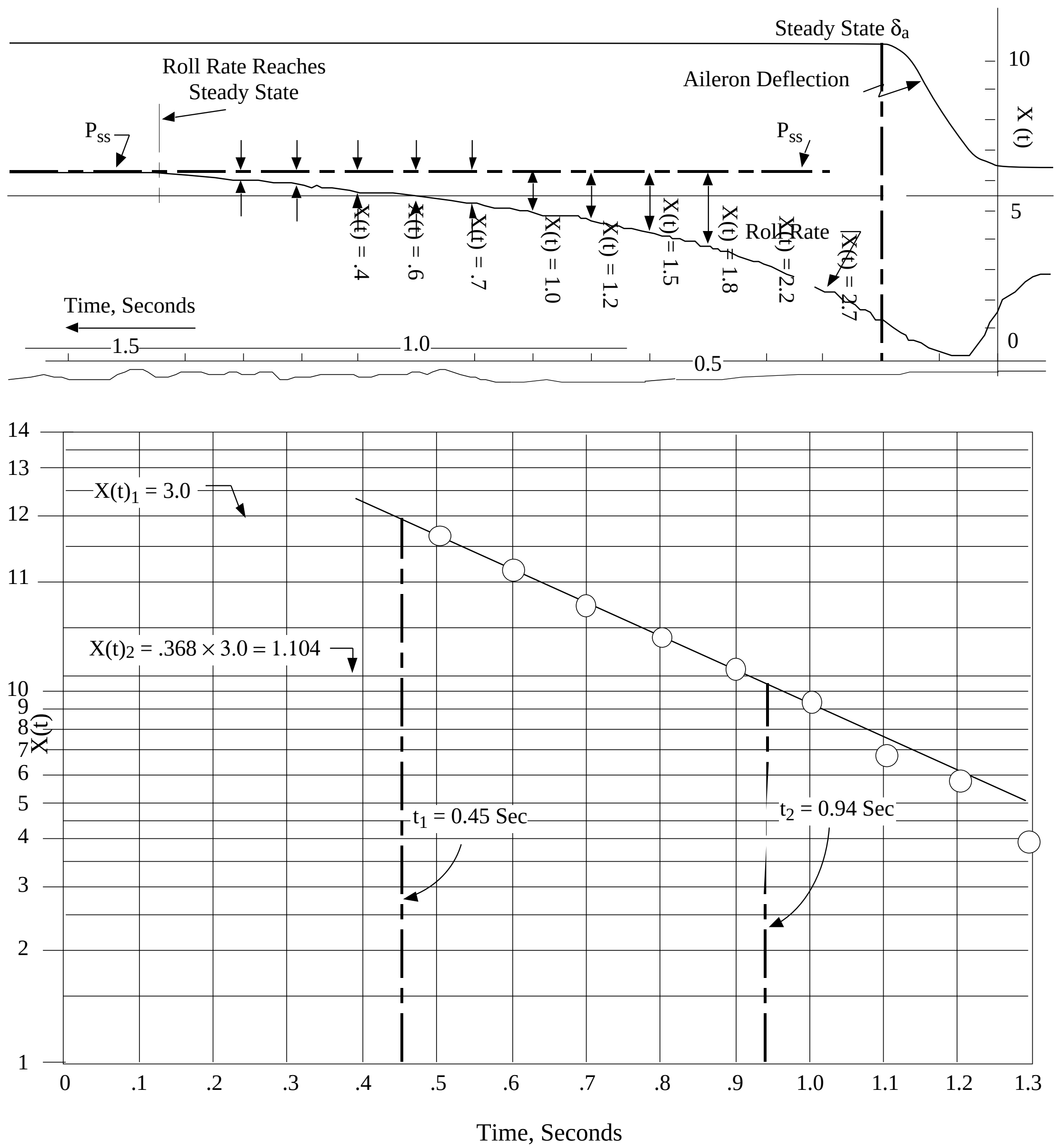


Figure 5.95
Determination of τ_R from Oscillograph Trace of a Rolling Maneuver

1. From the automatic recording trace of the roll maneuver, determine the steady state roll rate.

2. Using the portion of the roll rate trace past the point where the lateral control surface deflection reaches steady state, determine several X(t) and time values. The parameter, X(t), is the difference between the steady state roll rate and the roll rate at any given time. The portion of the roll rate trace prior to the point where the lateral control reaches steady state cannot be utilized since that portion is not characterized by an exponential increase in roll rate. It is not necessary to express the X(t) values (vertical scale) in actual units of degrees per second. Any convenient vertical scale may be utilized since the variation of the X(t) parameter with time is the only characteristic of concern. The horizontal scale must be actual time in seconds, although the starting, or zero, point's location on the trace is not critical.

3. Plot the X(t) and corresponding times on semilogarithmic graph paper. The vertical logarithmic scale must be the X(t) axis, while the horizontal, linear scale must be the time axis.

4. Fair a straight line through the points defined by the X(t) and t values on the semilog paper.

5. For a convenient $X(t)_1$ value on the semilog paper, determine a corresponding time, t_1. Compute 0.368 $X(t)_1$, which may be called $X(t)_2$. The value 0.368 is used as a multiplication factor (instead of .0632) because X(t) is measured from steady state roll rate not from zero roll rate. Determine a time, t_2, corresponding to $X(t)_2$ on the semilog paper.

6. Compute the roll mode time constant as follows:

$$\tau_R = t_2 - t_1 \qquad \textit{eq 5.119}$$

The results of this procedure will be rather accurate if the test pilot uses a good technique during the inflight test and if the Dutch roll motion is not evident in the roll response. The test pilot must make a very quick lateral control input and must hold the input constant during the roll rate increase to steady state. Full lateral control deflection rolls are not required; partial deflection rolls may be utilized if the partial input can be made quickly and held constant.

If Dutch roll excitation causes oscillatory roll rate response, data analysis will be more tedious and accuracy of the results will be derogated. When the roll rate is oscillatory, engineering judgement must be utilized to determine the steady state roll rate and to "fair" an approximate single degree of freedom roll rate response curve. The procedures shown in Figure 5.100 and described earlier may then be followed.

5.9 SPECIFICATION REQUIREMENTS

Requirements for rolling performance and associated roll handling qualities are contained in the following applicable paragraphs of Military Specification MIL-F-8785C of 5 November 1980, hereafter referred to as the Specification.

3.3.1.2 Roll mode

3.3.2 Lateral-directional dynamic response characteristics

3.3.4 Roll control effectiveness

3.4.3 Cross-axis coupling in roll maneuvers

3.4.4 Control harmony (including 3.4.4.1)

3.5.2 Mechanical characteristics of control systems

3.5.3 Dynamic characteristics

3.5.4 Augmentation systems

3.3.2.2.1 Additional Roll Rate Requirements for Small Inputs

The parameter $\frac{p_{osc}}{p_{av}}$ is used for determining the Dutch roll influence on roll performance during precise tracking tasks. Large lateral control deflections and roll rates should not be used to measure this parameter.

3.3.2.4 Sideslip Excursions

The table contained in this paragraph (page 24 of Spec) is used to determine adverse/proverse yaw Spec compliance during the tests listed in paragraph 3.3.4. The "adverse sideslip" and "proverse sideslip" referred to in the table are the parameter $\frac{\Delta\beta}{k}$.

3.3.2.4.1 Additional Sideslip Requirements for Small Inputs

Compliance with Figure 5.6 of the Spec should be made during precise tracking tasks. Large lateral control deflections and roll rates should not be used during these tests.

3.3.4 Roll Control Effectiveness

The airspeed and altitude requirements to determine compliance with Table IX are listed according to the flight phase in Table l of the Spec (page 7).

3.3.4.1.1 Air-to-Air Combat

3.3.4.1.2 Ground Attack with External Stores

The requirements of these paragraphs take precedence over Table IX of the Spec.

5.10 ROLLING PERFORMANCE - GLOSSARY

Single Degree of Freedom Roll	Rolling motion during which the airplane is allowed to roll but not allowed to yaw or pitch; pure roll response.
Roll Mode Time Constant, τ_R	Time required for the single degree of freedom roll rate to reach 63.2% of the steady state roll rate following a step lateral control input.
Steady State Roll Rate, p_{ss}	Roll rate attained when the roll damping contribution equals the roll control power contribution for a constant lateral control input.
Roll Helix Angle, $\frac{pb}{2V_T}$	Helix angle described by the wingtip of a rolling airplane; sometimes referred to as the non-dimensional roll rate.
Roll Coupling	Pitching and yawing motions induced by inertial and kinematic effects during high rate rolls.
Elastic Center	A point in the wing section about which torsional deflections occurs.
Aileron Reversal Airspeed, V_r	Airspeed at which the combined effects of wing twist and wing bending counteract the rolling moment generated by lateral control deflection.
Wingtip Vertical Velocity, $\frac{pb}{2}$	Vertical velocity of the wingtip of a rolling airplane; sometimes used as a measure of the rolling performance of large airplanes in the approach and landing phases of mission accomplishment.
Bank Angle Change In A Given Time, ϕ_t	The bank angle attained in a predetermined time interval following a step input of lateral control; time is measure from the initiation of the pilot's lateral control <u>force</u> application.

5.11 ROLLING PERFORMANCE-REFERENCES

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control,* by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability, and Control*, by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Analysis of Several Handling Quality Topics Pertinent to Advanced Manned Aircraft,* Systems Technology, Inc., AFFDL-TR-67-2, June, 1967.

4. *Criteria For Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training,* by F. D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F1, September, 1962.

5. *Determination of Lateral Control Requirements at Low Airspeeds,* Report No. 2, Final Report, by Lcdr. W. Spangenberg, USN and Mr. G. A. Patterson, U.S. Naval Air Test Center, Patuxent River, Maryland, Report No. FT212-222, 19 July 1963.

6. *Development of Satisfactory Lateral-Directional Handling Qualities in the Landing Approach*, by Robert L. Stapleford, Donald E. Johnston, Gary L. Teper, and David H. Weir, Systems Technology, Inc., Hawthorne, California, NASA CR-239, July, 1965.

7. *Dynamics of Flight, Stability, and Control*, by Bernard Etkins, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

8. *Evaluation of Lateral-Directional Handling Qualities of Piloted Reentry Vehicles Utilizing Fixed-Base and In-Flight Evaluations*, by J.I. Meeker, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, NASA CR-778, May 1967.

9. *Features of Large Transport Aircraft Affecting Control During Approach and Landing*, by W.J.G. Pinsker, Royal Aircraft Establishment, Bedford, England.

10. *Flight Testing of Aircraft*, by M.G. Kotik, et al., NASA Technical Translation F-442, April,1967.

11. *Flying Qualities Requirements for United States Navy and Air Force Aircraft*, by William Koven and Richard Wasicko, AGARD Specialists Meeting on Aircraft Stability and Control, Brussels, Belgium, April 1961.

12. *Handling and Performance Exercises Test Technique, Empire Test Pilot's School, Flying Department Notes.*

13. *In-Flight Evaluation of Lateral-Directional Handling Qualities for the Fighter Mission*, by James I. Meeker and G. Warren Hall, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-98, October 1967.

14. *In-Flight Simulation and Pilot Evaluation of Selected Landing Approach Handling Qualities of a Large Logistics Transport Airplane*, by Donald W. Rhodes, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-51, July, 1967.

15. *Lateral-Directional Flying Qualities for Power Approach: Influence of Dutch Roll Frequency,* by Edward Seckel, James A. Franklin, and George E. Miller, Princeton University, Princeton, New Jersey, Princeton University Report No. 797, September 1967.

16. *Lecture Notes on Lateral-Directional Stability,* by F.D. Newell and Michael Parrag, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. TB-1961-F-1, July, 1965.

17. *Modern Flight Dynamics*, by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

18. *Military Specification, Flying Qualities of Piloted Airplanes*, MIL-F-8785C, 5 November 1980.

19. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

20. *A Pilot Opinion Study of Lateral Control Requirements for Fighter-Type Aircraft*, by Brent Y. Creer, John D. Stewart, Robert B. Merrick, and Fred J. Drinkwater, III, Ames Research Center, Moffett Field, California, NASA Memorandum 1-29-59A, March 1959.

21. *The Provision of Adequate Lateral Control Power for Landing Approach Conditions*, by Mr. G.A. Patterson and Lcdr. W. Spangenberg, USN, U.S. Naval Air Test Center, Patuxent River, Maryland.

22. *A Review of Recent Research on Handing Qualities and Its Application to the Handling Problems of Large Aircraft*, by P.L. Bisgood, Ministry of Aviation, Aeronautical Research Council, London, R. & M. No. 3458, June 1964.

23. *Stability and Control, Empire Test Pilot's School Ground School Notes, Book "C"*.

24. *Stability and Control*, USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

25. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

26. *A Study of Conventional Airplane Handling Qualities Requirements - Part I Roll Handling Qualities*, by I.L. Ashkenas, Systems Technology, Inc., AFFDL-TR-65-138, Part I.

27. *Substantiation Information for Selected Lateral-Directional Requirements*, Cornell Aeronautical Laboratory Report No. BM-2238-F-1, May 1967.

5.12 MISCELLANEOUS LATERAL-DIRECTIONAL TESTS

5.12.1 Introduction

Certain miscellaneous lateral-directional tests, not previously introduced, are presented in this section. Crosswind takeoffs and landings are generally the most significant of the tests to be introduced.

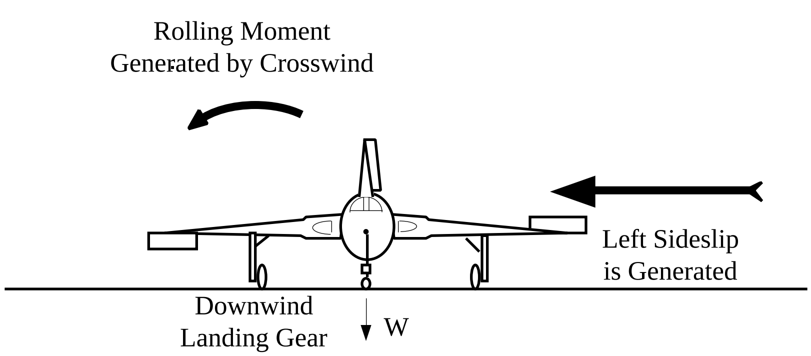
5.12.2 Lateral-Directional Control During Crosswind Takeoffs, Approaches, and Landings

A crosswind is defined as a wind blowing across the direction of movement of an airplane; i.e., a wind blowing across, as across a runway. The most important influence of a crosswind is to tend to drive the airplane sideways and change its direction of movement. Thus, the concern of the test pilot while performing crosswind evaluations is the directional controllability of the airplane under crosswind conditions. The operational pilot must be afforded sufficient lateral-directional control to consistently and safely perform takeoffs and landings in crosswind components representative of operational conditions.

5.12.2.1 DIRECTIONAL GROUND STABILITY

Directional ground stability is the capability of the airplane to maintain a straight ground path under representative wind conditions. Directional ground stability in a crosswind is actually degraded by strong directional stability, since the strong directional stability increases the tendency of the airplane to "weather-cock." However, the typical landing gear placement - two main gears located laterally in line and nose gear or a tail wheel - contributes to the maintenance of a straight ground path under crosswind conditions. Directional ground stability is increased by increasing the reaction of the nosewheel or tailwheel with the runway surface. From an airplane design viewpoint, increasing the lateral displacement of the main landing gear and increasing the longitudinal displacement of the third gear increases directional ground stability. Pilot technique also influences directional ground stability. Maximum resistance to "weather-cocking" is achieved by holding full nose-up longitudinal control in tailwheel airplanes and full nose-down longitudinal control in nosewheel configured airplanes. If the "weather-cocking" tendency of the airplane cannot be counteracted by all available means, the airplane will leave the runway on the side <u>from which</u> the wind is blowing (upwind side).

Directional ground stability is insidiously influenced by dihedral effect. Rolling moments are generated by the sideslip imposed on the airplane during crosswind takeoffs, and landings (Figure 5.96). These rolling moments tend to roll the airplane <u>about the downwind point of contact</u> (downwind landing gear). Thus, the weight of the airplane and the lateral placement of the main landing gear have a large effect on the "lateral ground stability" of the airplane. Widely spaced main landing gear obviously enhances the lateral ground stability. From a pilot technique viewpoint, maintenance of full nose-down longitudinal control provides maximum lateral ground stability by decreasing the lift coefficient, thus increasing the weight vector acting in opposition to the roll excursion (Figure 5.96).



The rolling moment about the downwind point of contact (Generated by the cross-wind) is opposed by moments generated by airplane weight and lateral control inputs.

Figure 5.96
Lateral Ground Stability

If the rolling moment generated by the crosswind cannot be overcome by all available means, the upwind wing will rise. The "lateral ground stability" problem can then quickly become a "directional ground stability" problem. The increased friction generated by the downwind landing gear generates a yawing moment toward the downwind side of the runway. Pilot brake applications may only complicate the situation, since upwind brake applications result in little or no braking action, while downwind brake applications are extremely effective. If the pilot is unable to maintain directional control of the airplane via all available means, the airplane will depart the runway on the downwind side.

5.12.2.2 CROSSWIND APPROACH TECHNIQUES

The pilot's major concern during the crosswind approach is to keep the airplane track superimposed over the runway centerline extended. This may be accomplished utilizing either of two well-known techniques or a combination of the two.

5.12.2.2.1The Sideslipping Approach

When executed correctly, the sideslipping approach technique results in the airplane heading and ground track being identical to the runway heading. The airplane is merely flown in a steady heading sideslip; the angle of sideslip is determined by the airspeed and the crosswind component (Figure 5.97). Bank angle, rudder, and lateral control inputs are generally required to compensate for sideforce, yawing, and rolling moments, respectively. The sideslip may be maintained through touchdown if the landing gear is sufficiently strong and if wingtip and/or external store clearance permits. Otherwise, the pilot may desire to level the wings just prior to touchdown and complete the landing.

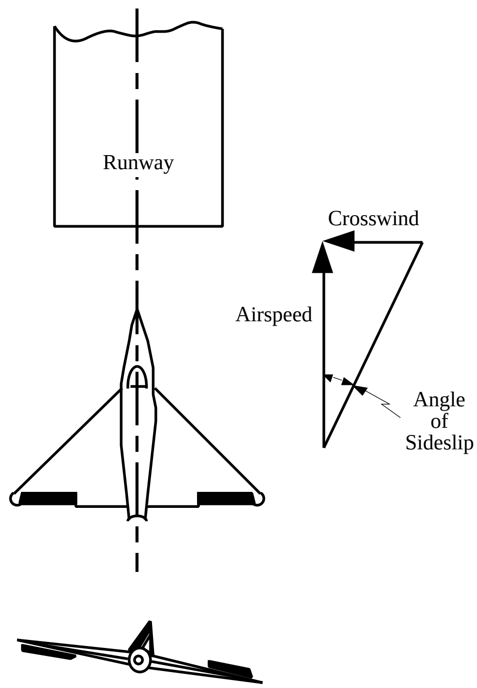


Figure 5.97
Sideslipping Technique for Cross-Wind Approaches

Although the sideslipping approach allows the pilot to easily determine if the track is along the extended runway centerline, displacements from the desired track are difficult to correct. A simple change in bank angle generates a lateral displacement. This maneuver is not easily performed, however, due to the "crossed controls" condition and the necessity for maintaining a precise flight path. The pilot generally resorts to continually adjusting the magnitude of the sideslip in order to make track corrections. As a consequence, trimming the control forces to zero in the sideslip is unrealistic. During a long approach, the lateral and directional control forces may become tiring, particularly in the large airplane which may require considerable forces.

Lateral and directional control inputs required during the sideslipping approach leave correspondingly less control deflections available to counteract gusts. The sideslipping technique is particularly unsuitable for the instrument approach; a steady heading sideslip is extremely difficult to establish and maintain without external references.

This significant advantage of the sideslipping technique is the easy transition from approach to landing. The pilot merely holds the sideslip through the touchdown or levels the wings just prior to touchdown.

5.12.2.2.2The Crabbing Approach

The crabbing approach again result in the airplane ground track lying along the runway centerline extended. However, the airplane is headed or "crabbed" into the crosswind so that the sideslip angle is zero (Figure 5.98). In the equilibrium condition, bank angle and lateral-directional control inputs are zero. In some airplanes, e.g., T-38, F-4, touchdown in the crabbed attitude is recommended but, in others, the pilot must align the airplane with the runway heading prior to touchdown unless the airplane is equipped with crosswind landing gear.

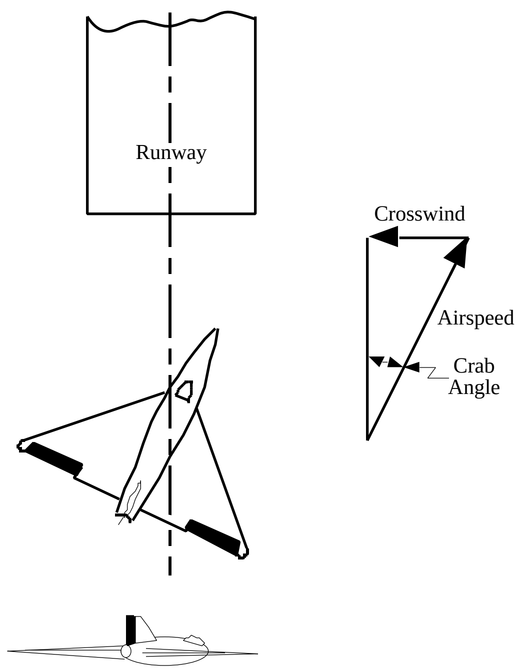


Figure 5.98
Crabbing Technique for Cross-Wind Approaches

The crabbing technique presents the pilot with few difficulties other than those already present during the approach. The airplane is flown in a rather normal manner with wings level; errors in airplane track are corrected with simple turning maneuvers. During visual approaches, the pilot's view of the runway may be slightly degraded in airplanes with side-by-side seating. However, the crabbing technique is consistent with instrument approach conditions; the airplane is merely flown in the confidential manner until visual contact with the runway is established.

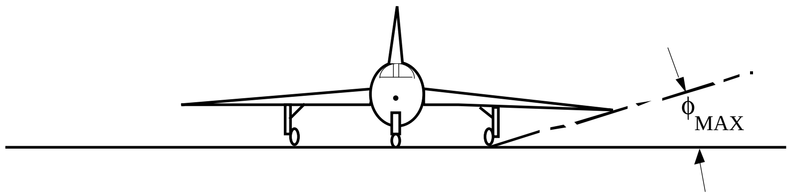
The major and important disadvantage of the crabbing technique is the transition required in landing if the crab angle must be removed before touchdown to align the airplane with the runway centerline. The control coordination may be quite difficult and the timing of the maneuver must be precise. The pilot workload is thus increased substantially during a critical phase of flight.

5.12.2.3 TEST PROCEDURES AND TECHNIQUES

Certain preliminary investigations should be conducted before the actual crosswind tests.

5.12.2.3.1Preliminary Investigations

The flight test team must determine the <u>maximum permissible bank angle</u> which will provide clearance between the airplane wingtip or external stores and the runway surface. This is a simple problem in geometry (Figure 5.99). The bank angle so determined may restrict the crosswind capabilities of the airplane or result in the utilization of a particular crosswind technique.



ϕ_{max} = *Maximum permissible bank angle which may safely be utilized while in close proximity to the ground. In this example, wingtip clearance is utilized. External store clearance should also be considered, if applicable.*

Figure 5.99
Geometric Maximum Permissible Bank Angle

The largest sideslip angle which can be generated in an intentional steady heading sideslip corresponds to the maximum crosswind component which can be counteracted during a sideslipping approach. This relationship is shown in Figure 5.100. Steady heading sideslip tests in Configurations Power Approach and Land at Representative Approach airspeeds will provide the largest obtainable sideslip angles. These sideslip angles may then be utilized to compute crosswind components. This procedure does not consider the additional control authority necessary to counteract turbulence and gusts during the approach. Further, it must be emphasized that these crosswind components are applicable only to the approach phase. Significantly less crosswind may conceivably cause intolerable control problems during the landing rollout.

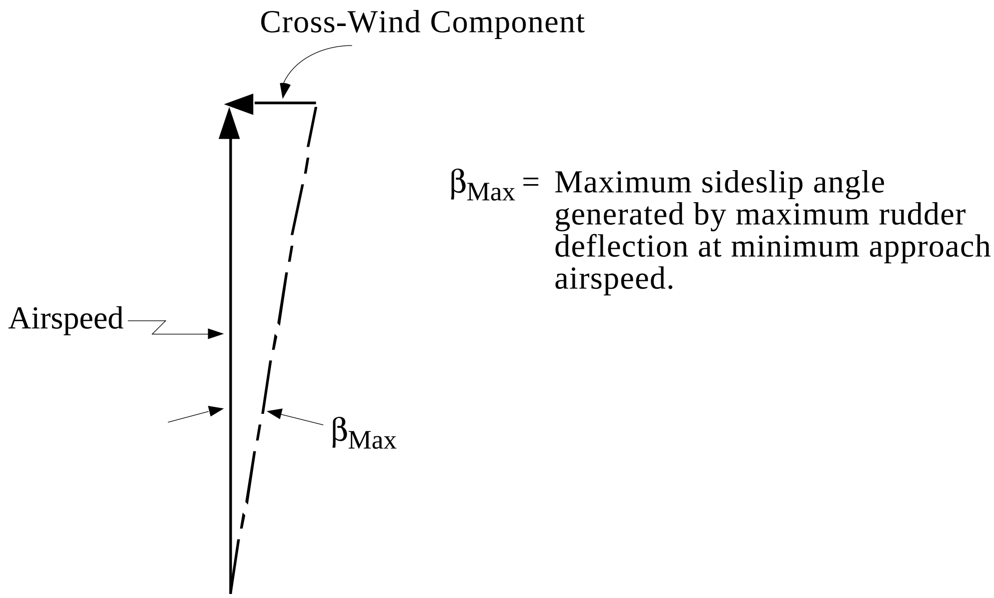


Figure 5.100
Absolute Maximum Cross-Wind for the Sideslipping Approach

The determination of minimum speeds at which the rudder and lateral control devices are sufficiently effective to provide directional and lateral control on the runway completes the preliminary tests. In general, a low airspeed at which the rudder is effective for directional control enhances crosswind handling qualities. The same generality cannot be applied to the aileron effectiveness minimum speed; for example, widely spaced main landing gear may result in the ailerons being effective for bank angle control only at high speeds on the runway, yet provide excellent lateral ground stability. The results of these tests must be analyzed logically in relation to airplane design and the availability of nose-wheel or tailwheel steering and wing spoiler or flaperon pop-up devices.

Rudder effectiveness minimum speed may be determined as follows:

1. Under essentially zero crosswind conditions, align the airplane with the runway heading in configuration Takeoff.

2. Apply full rudder in one direction and begin the take-off roll.

3. A speed will be attained at which the airplane begins to respond to the directional control input. This is the rudder effectiveness minimum speed in the take-off configuration. The rudder input should be quickly revered as the initial response is noted in order to verify that rudder effectiveness minimum speed has been attained. After the minimum speed is ascertained, continue with a normal takeoff.

4. If a light crosswind component is present, the rudder input should initially be made <u>opposite</u> to the direction from which the crosswind is blowing. Rudder effectiveness minimum speed will easily be recognized as the speed at which the rudder input counteracts the weather-cocking tendency of the airplane. Note that some airplanes with tricycle landing gear exhibit negative weather-cock stability on the ground. In this event, reverse the direction of rudder application.

5. The above test should also be performed during the landing rollout. In this case, the initial rudder input must be cautiously made and of a small magnitude. (Alternate applications will be helpful in keeping the airplane near the runway centerline.) As speed decreases, the rudder inputs must be increased in amplitude. The speed at which full rudder deflection is just barely effective for directional control is the rudder effectiveness minimum speed during landing.

Aileron effectiveness minimum speed may be determined as follows:

1. Under essentially zero crosswind conditions, align the airplane with the runway heading in configuration Takeoff.

2. Initiate the take-off roll; simultaneously begin smooth pulsing of the lateral control from full deflection to full deflection.

3. A speed will be attained at which the airplane begins to respond to the lateral control inputs. This is the aileron effectiveness minimum speed in the Takeoff configuration. After the minimum speed is ascertained, continue with a normal takeoff.

4. The above test should also be performed during the landing rollout. Small alternating lateral control inputs are smoothly increased in amplitude as speed decreases. The speed at which full lateral control deflection is just barely effective for roll control is the aileron effectiveness minimum speed during landing. (This speed may be quite high if pop-up spoilers or flaperon pop-up devices are installed.)

5.12.2.3.2Crosswind Tests

After completion of the preliminary tests and procedures, the test program can be expanded to include actual crosswind takeoffs and landings. Accurate analytical estimations of maximum crosswind components for a particular airplane and configuration are almost impossible; therefore, the crosswind flight tests must be conducted utilizing a build-up program. Initial tests should be performed with small crosswind components. As familiarity is gained, the components may be systematically increased in increments of 5 knots or less until the limits are determined.

The following guidelines and general information should aid in planning and conducting the crosswind tests:

1. Crosswind tests should be terminated when maximum allowable crosswind components are reached (if limits are published by higher authority) or if lateral or directional control becomes <u>marginal</u>. Initiating a takeoff or landing in a crosswind which exceeds the airplane capabilities may result in the airplane departing the runway with catastrophic consequences. An intolerable situation may be rectified if the pilot can abort the takeoff or convert a full stop landing into a touch-and-go. However, the test pilot <u>should not be required to resort to these measures</u>. An adequate build-up program will preclude the inadvertent entry into dangerous flight conditions.

2. Initial tests should be conducted with all relevant airplane systems operative. Degraded system operation may be investigated later, if appropriate. These tests might be aimed at the determination of maximum recommended crosswind components for engine-out condition, wing spoilers or flaperon pop-up inoperative conditions, or flight control system malfunctions.

3. One of the most important aspects of the crosswind evaluation is the formulation of optimum techniques for crosswind takeoffs, approaches, and landings. The optimum techniques should be explicitly stated in a technical report and published as recommended operational procedures in the pilot's handbook.

4. Applicable specification conformance or non-conformance should be ascertained during these tests.

5. One of the most frustrating aspects of crosswind testing is <u>the availability of useable crosswinds</u>. This factor may require offsite testing to obtain suitable crosswinds.

6. The <u>turbulence and gustiness</u> usually associated with high surface wind conditions tends to complicate the test pilot's task. These factors may, additionally, restrict the airplane's capabilities or influence optimum piloting technique.

7. The test pilot should be aware of possible complications generated by lateral ground stability problems. For example, if the downwind main landing gear is equipped with micro-switches which activate nosewheel steering, spoiler or flaperon pop-up, anti-skid braking, etc., these features may not be available because of insufficient weight on the gear.

8. The condition of the test airplane's tires should be checked frequently during crosswind tests.

9. The possibility of blowing tires during crosswind tests is high. The test pilot should plan an exact course of action to be followed in the event a tire fails. Availability, capacity, and location of arresting gear should be given due consideration.

5.12.3 Lateral-Directional Trim Changes

Lateral-directional trim changes are generally not a significant as the previously introduced longitudinal trim changes. Usually, no flight test time is specifically allocated for their determination. These trim changes can be evaluated during other tests, such as climbs and descents. However, the test pilot must be continually alert for excessive lateral and/or directional trim changes. "Short-term" lateral-directional trim changes might be associated with configuration changes, power changes, external stores separation, bomb bay door operation, rocket or missile firing, etc. "Long-term" trim changes, such as lateral and/or directional trim changes during level accelerations or decelerations, climbs, dives, etc. should not be seriously objectionable or impair tactical maneuverability or weapons delivery.

5.12.4 Irreversibility of Lateral-Directional Trim Systems

Lateral and directional trim systems should maintain given settings indefinitely unless intentionally changed by the pilot. This characteristic may be quickly evaluated by subjecting the systems momentarily to large hinge moments, then checking the lateral-directional trim of the airplane.

The lateral trim system may be checked for irreversibility as follows:

1. Stabilize and trim the airplane precisely at a high equivalent airspeed (low altitude, near maximum operational airspeed).

2. Perform an abrupt roll utilizing maximum allowable lateral control deflection. Continue the roll to attain a maximum allowable bank angle change.

3. Stabilize the airplane carefully at the original flight condition. If the airplane is still in trim laterally, the lateral trim system is irreversible.

4. Perform this test utilizing rolls in both directions.

The directional trim system may be checked for irreversibility as follows:

1. Stabilize and trim the airplane carefully at a medium airspeed, such as the airspeed for maximum range in configuration Cruise. A medium airspeed has been arbitrarily chosen; a larger sideslip angle may be generated at a medium airspeed than at a very high airspeed.

2. Perform a steady heading sideslip utilizing full rudder deflection or maximum allowable sideslip angle.

3. Return the airplane smoothly and slowly to the original stabilized condition. If balanced flight conditions still exist, as evidenced by the ball position in the needle-ball instrument, the directional trim system is irreversible.

4. Perform this test utilizing both left and right sideslips.

CHAPTER SIX

ASYMMETRIC POWER

CHAPTER SIX

FIGURES

CHAPTER SIX

EQUATIONS

$$\delta_{r_{Equilibrium}} = \frac{\begin{vmatrix} C_{y_\beta} & 0 \\ C_{n_\beta} & -\frac{F_N}{W} C_L \frac{y_p}{b} \end{vmatrix}}{\begin{vmatrix} C_{y_\beta} & C_{y_{\delta_r}} \\ C_{n_b} & C_{n_{\delta_r}} \end{vmatrix}} = \frac{-\frac{F_N}{W} \frac{y_p}{b} C_L C_{y_\beta}}{C_{n_{\delta_r}} C_{y_\beta} - C_{y_{\delta_r}} C_{n_\beta}}$$

eq 6.9 6.6

$$\delta_{r_{Equilibrium}} = \frac{-\frac{F_N}{W} \frac{y_p}{b} C_L}{C_{n_{\delta_r}} \left\{1 - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} \frac{C_{n_\beta}}{C_{y_\beta}}\right\}} \text{ (ZERO BANK ANGLE)}$$

eq 6.10 6.6

$$\beta = -\frac{C_{y_{\delta_r}} \delta_r}{C_{y_\beta}} \text{ (ZERO BANK ANGLE)}$$

eq 6.11 6.6

$$\beta = \frac{-\ {}^{F_N}\!/\!_W\ C_L\ {}^{y_p}\!/\!_b}{C_{n_\beta}} \text{ (ZERO RUDDER)}$$

eq 6.12 6.7

$$\phi = \frac{-C_{y_\beta} \beta}{C_L} \text{ (ZERO RUDDER)}$$

eq 6.13 6.8

$$\frac{L_0 - L_i}{W} C_L \frac{y_p}{b} + C_{\ell_\beta}\beta + C_{\ell_{\delta_a}} \delta_a = 0$$

eq 6.14 6.10

$$\delta_{a_{Equilibrium}} = -\frac{1}{C_{\ell_{\delta_a}}} \left\{\frac{L_0 - L_i}{W} C_L \frac{y_p}{b} + C_{\ell_\beta} \beta\right\}$$

eq 6.15 6.10

CHAPTER SIX

ASYMMETRIC POWER FLYING QUALITIES

6.1 THEORY

6.1.1 General

The asymmetric power flying qualities problem is invariably a lateral-directional control problem. Yawing and/or rolling moments generated by the asymmetric power condition must be counteracted by airplane stability and pilot control inputs. Although asymmetric power control problems are generally confined to the low airspeed flight regime, serious airplane departures from controlled flight may be encountered with asymmetric engine failures at very high airspeeds. Asymmetric flying qualities may also result from the asymmetric carriage of external or internal stores.

Basically, two aspects of flight on asymmetric power must be considered:

1. Regaining of control immediately following failure of one or more engines.

2. Maintaining control in steady flight with one or more engines inoperative.

The steady or equilibrium flight condition with asymmetric power will be considered first. (Note: Most of the stability derivatives and symbols utilized in this discussion have been introduced previously. Therefore, many of these derivatives and terms will not be redefined here.)

6.1.2 Steady Straight Flight on Asymmetric Power

6.1.2.1 THE DIRECTIONAL CONTROL PROBLEM

Flight on asymmetric power is characterized by a yawing moment generated by the asymmetric condition (Figure 6.1). It is important to consider the factors influencing the magnitude of this yawing moment since the degree of difficulty associated with asymmetric power flight is generally directly related to this parameter. If the inoperative engine is assumed to generate no thrust or drag, the yawing moment generated by the asymmetric condition, N_T, may be developed as follows (Figure 6.1):

$$N_T = F_N \; y_p \qquad \textit{eq 6.1}$$

Where:

F_N = thrust developed by operative engine, pounds.

y_p = distance from center of gravity to asymmetric thrust vector measured in wing plane, feet.

In non-dimensional form, the yawing moment coefficient, C_{n_T}, may be expressed:

$$C_{n_T} = \frac{N_T}{qSb} = \frac{F_N y_p}{qSb}$$

or for level flight $\left(\frac{W}{qS} = C_L\right)$:

$$C_{n_T} = \frac{F_N}{W} C_L \frac{y_p}{b}$$

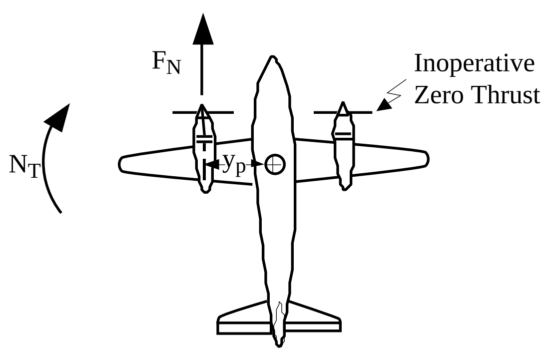


Figure 6.1
Yawing Moment Due to Asymmetric Power

Note that the asymmetric power yawing moment coefficient increases with increase in operative engine thrust, distance of operative engine from airplane center of gravity, and increase in lift coefficient (or decrease in airspeed).

Expressions for the thrust, F_N, developed by the operative engine will be different for jet and propeller-driven airplanes. For the jet, the thrust is simply F_N. However, for the propeller-driven airplane:

$$F_{N_{Prop}} = \frac{550\eta_p \ BHP}{V} \qquad \textit{eq 6.2}$$

Where:

550 = horsepower constant, foot-pounds/second.

η_p = propeller efficiency factor.

BHP = brake horsepower, HP.

V = airplane true airspeed, feet per second.

Thus, for the propeller-driven airplane:

$$C_{n_{T_{Prop}}} = \frac{550\eta_p \ BHP}{V \ W} \ \frac{C_L \ y_p}{b} = \frac{550\eta_p \ BHP \ y_p}{V \ qSb} \qquad \textit{eq 6.3}$$

The equilibrium equations for sideforce, yawing moment, and rolling moment may now be written as follows for the asymmetric power condition, ($C_{n_{\delta_a}}$ and $C_{\ell_{\delta_r}}$ are assumed to be zero for simplicity):

SIDEFORCE $$C_{y_\beta} \beta + C_{y_{\delta_r}} \delta_r + C_L \phi = 0 \qquad \textit{eq 6.4}$$

YAWING MOMENT $$\frac{F_N}{W} C_L \frac{y_p}{b} + C_{n_\beta} \beta + C_{n_{\delta_r}} \delta_r = 0 \qquad \textit{eq 6.5}$$

ROLLING MOMENT $$C_{\ell_\beta} \beta + C_{\ell_{\delta_a}} \delta_a = 0 \qquad \textit{eq 6.6}$$

(Note: Operative engine is assumed to be the port engine. If the starboard engine were operative, the asymmetric power yawing moment would be negative.)

Since the directional control problem with asymmetric power is of interest at present, expressions will be derived for the rudder required for steady heading, equilibrium flight under three flight conditions.

6.1.2.2.1 No Sideslip

If the pilot maintains zero sideslip, and expression for the rudder requirement may be obtained easily for the yawing moment equation:

$$\delta_{r_{Equilibrium}} = \frac{-\,{}^{F_N}\!/_{W}\; C_L \;\; {}^{y_p}\!/_{b}}{C_{n_{\delta_r}}} \text{ (ZERO SIDESLIP)} \qquad \textit{eq 6.7}$$

Several important relationships may be gathered from the last equation:

1. The rudder requirement increases with increasing asymmetric thrust, F_N .

2. The rudder requirement increases with increasing lift coefficient (decreasing airspeed).

3. The rudder requirement increases with lateral engine placement from the center of gravity.

4. The rudder required for equilibrium is inversely proportional to rudder control power.

Note that for zero sideslip, some bank angle must be used to balance the sideforce generated by the rudder input. From the sideforce equation:

$$\phi_{r_{Equilibrium}} = \frac{-\; C_{y_{\delta_r}} \;\; \delta_r}{C_L} \text{ (ZERO SIDESLIP)} \qquad \textit{eq 6.8}$$

For a positive asymmetric yawing moment (starboard engine inoperative), trailing edge left (positive) rudder deflection is required; thus, a negative (left) bank angle is necessary to maintain equilibrium flight (Figure 6.2). In most cases, the bank angle requirement is fairly small (approximately 5 degrees).

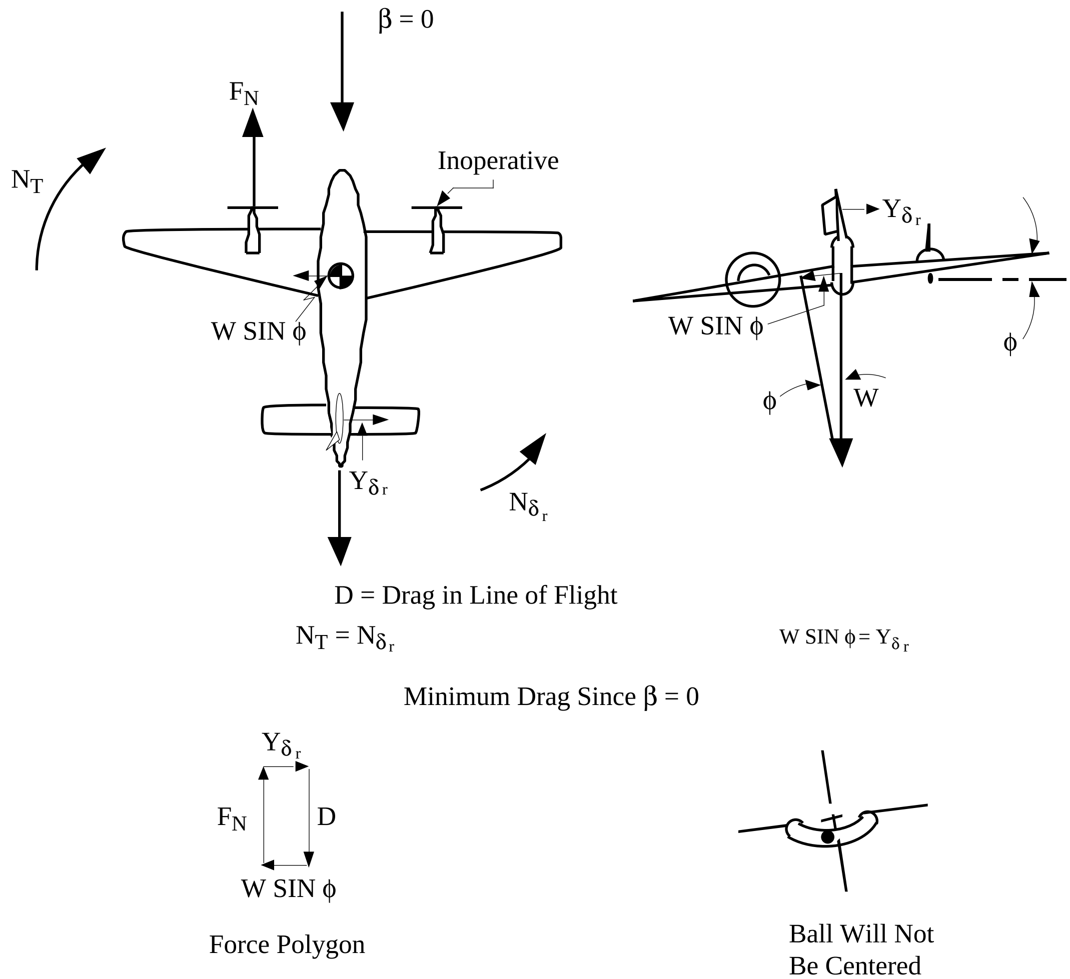


Figure 6.2
Equilibrium Asymmetric Power Condition with Zero Sideslip

6.1.2.2.2 No Bank Angle

If the pilot maintains zero bank angle, an expression for the rudder requirement for equilibrium, steady heading flight may be obtained via a determinant solution of the sideforce and yawing moment equations:

$$\delta_{r_{Equilibrium}} = \frac{\begin{vmatrix} C_{y_\beta} & 0 \\ C_{n_\beta} & -\frac{F_N}{W}\, C_L\, \frac{y_p}{b} \end{vmatrix}}{\begin{vmatrix} C_{y_\beta} & C_{y_{\delta_r}} \\ C_{n_b} & C_{n_{\delta_r}} \end{vmatrix}} = \frac{-\frac{F_N}{W}\,\frac{y_p}{b}\, C_L\, C_{y_\beta}}{C_{n_{\delta_r}}\, C_{y_\beta} - C_{y_{\delta_r}}\, C_{n_\beta}} \qquad \textit{eq 6.9}$$

$$\delta_{r_{Equilibrium}} = \frac{-\frac{F_N}{W}\,\frac{y_p}{b}\, C_L}{C_{n_{\delta_r}} \left\{ 1 - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\, \frac{C_{n_\beta}}{C_{y_\beta}} \right\}} \text{ (ZERO BANK ANGLE)} \qquad \textit{eq 6.10}$$

The only difference between the equation and the one derived for zero sideslip is the term in braces. This term can be rationalized as increasing the rudder requirement over the zero sideslip case; the increased rudder requirement will be necessary to balance the sideforce due to sideslip, C_{y_β}.

For zero bank angle, the sideslip required for equilibrium may be obtained from the sideforce equation:

$$\beta = -\frac{C_{y_{\delta_r}}\ \delta_r}{C_{y_\beta}} \text{ (ZERO BANK ANGLE)} \qquad \textit{eq 6.11}$$

If the asymmetric yawing moment is positive, the rudder requirement is positive, therefore, the sideslip angle must be positive (right sideslip). The balance of moments and forces is shown in Figure 6.3.

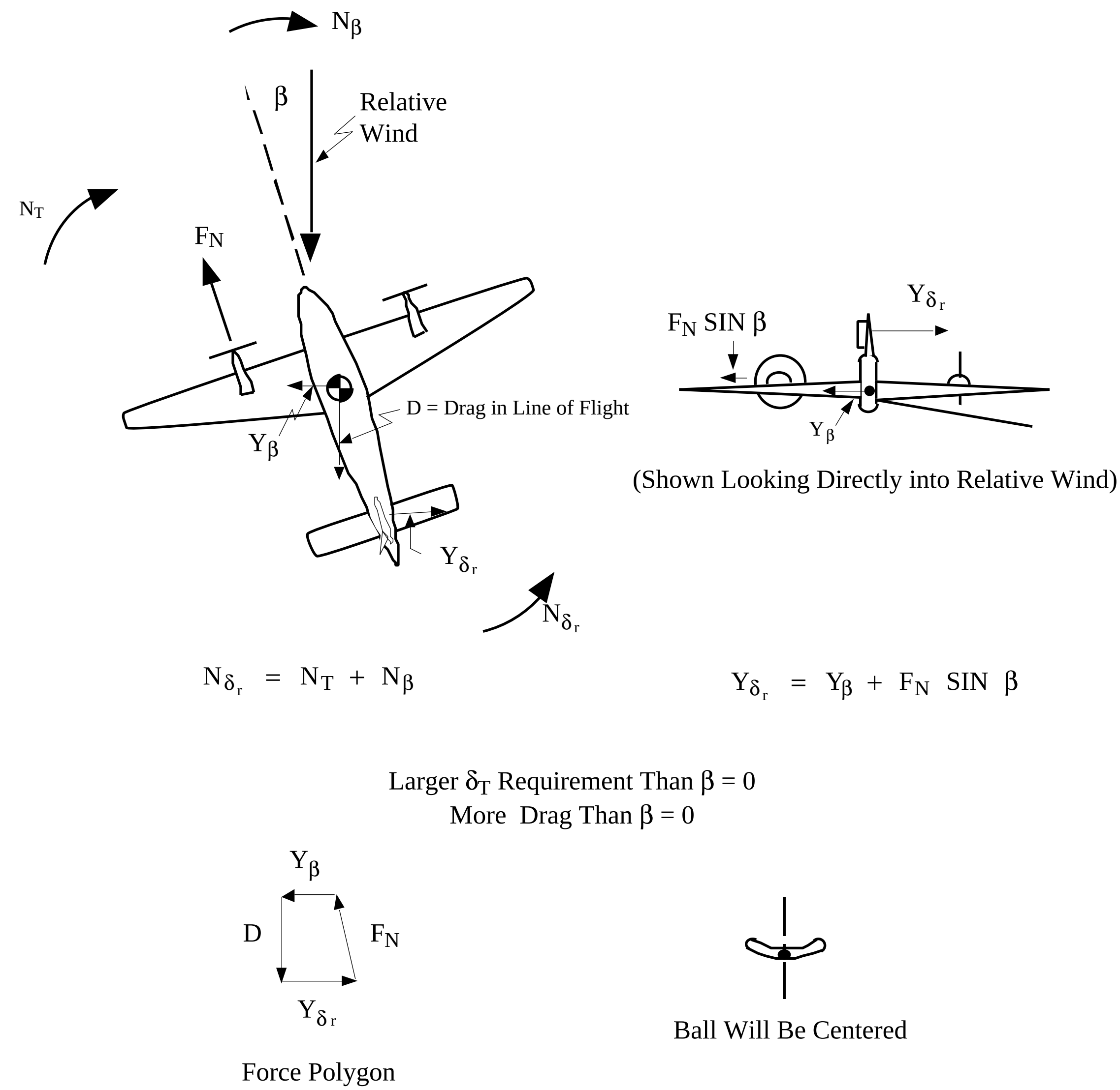


Figure 6.3
Equilibrium Asymmetric Power Condition with Zero Bank Angle

6.1.2.2.3 No Rudder Requirement

It is possible to balance the airplane in steady heading equilibrium flight under asymmetric power with zero rudder required. From the yawing moment equation, the sideslip required to balance the asymmetric yawing moment may be obtained:

$$\beta = \frac{-\ {}^{F_N}\!/_{W}\ C_L\ {}^{y_p}\!/_{b}}{C_{n_\beta}} \text{ (ZERO RUDDER)} \qquad \textit{eq 6.12}$$

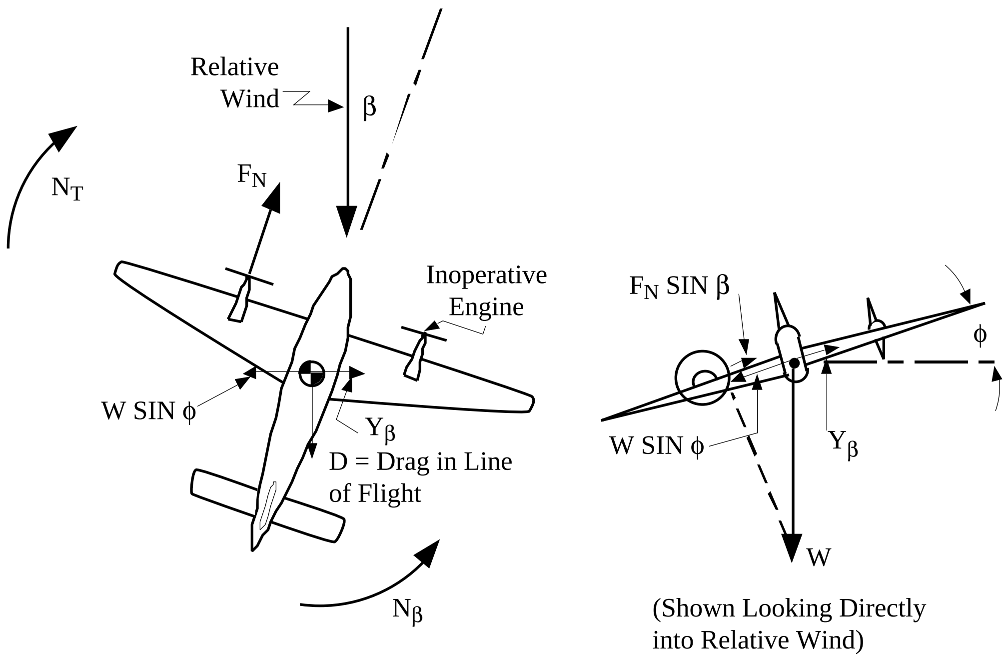
For a positive asymmetric yawing moment, the sideslip requirement is negative (left sideslip). The sideslip angle required for this condition is generally quite large, particularly at low airspeeds, high operative engine power, and with low directional stability.

The bank angle required to balance the sideforces for the zero rudder deflection condition may be obtained from the sideforce equation:

$$\phi = \frac{-C_{y_\beta}\ \beta}{C_L} \text{ (ZERO RUDDER)} \qquad \textit{eq 6.13}$$

For a positive asymmetric yawing moment, the sideslip angle must be negative; therefore, the bank angle must be negative (left bank angle). This bank angle is generally quite large (approximately 15 degrees) at low airspeeds. The balance of forces and moments is show in Figure 6.4.

At first glance, the equilibrium condition shown if Figure 6.4 might seem to be a desirable state of affairs since the pilot is required to hold no rudder input. However, the drag is high, there is a possibility of losing directional control due to vertical tail stalling, and the flight condition is uncomfortable because of the large side acceleration due to gravity. Usually, the pilot will achieve equilibrium in a flight condition somewhere between the conditions shown in Figures 6.2 and 6.3. (The operative engine will be banked down about 3 degrees and there will be a small sideslip from the inoperative engine side. If the directional trim system is sufficiently powerful, the rudder force requirement for steady heading flight can be trimmed to zero.)



$N_\beta = N_T$

W SIN $\phi = Y_\beta + F_N$ SIN β

Large β and φ required; more drag than β = 0 or φ = 0.

Potentially dangerous since vertical tail may stall due to large β; very uncomfortable for pilots and passengers because large φ generates large side acceleration due to gravity.

Force Polygon

Ball Will Not Be Centered

Figure 6.4
Equilibrium Asymmetric Power Condition with Zero Rudder Deflection

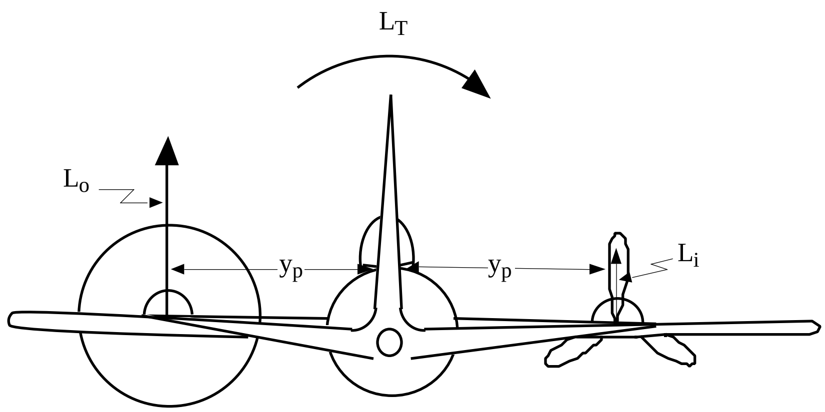
6.1.2.2.3 The Lateral Control Problem

Lateral controllability under equilibrium asymmetric power conditions is generally not as severe as the directional control problem. For pure-jet airplanes, minimum control speeds are almost always based on directional controllability. However, for propeller-driven airplanes under asymmetric power conditions, the differences in slipstream over the wings may generate large rolling moments (Figure 6.5). If the wings are almost completely immersed in propeller slipstream, the rolling moment generated by the asymmetric power condition may limit minimum airspeeds. Sideslip from the operative engine side coupled with positive dihedral effect (negative C_{ℓ_β}) complicates the lateral control problem. The lateral control requirement to counteract the rolling moments generated by asymmetric power and sideslip may be obtained from the equilibrium rolling moment equation:

$$\frac{L_0 - L_i}{W} C_L \frac{y_p}{b} + C_{\ell_\beta}\beta + C_{\ell_{\delta_a}} \delta_a = 0 \qquad \textit{eq 6.14}$$

(Note: If the asymmetric power rolling moment is in the left-wing-down direction, the first term of the equation will be preceded by a negative sign.)

$$\delta_{a_{\text{Equilibrium}}} = -\frac{1}{C_{\ell_{\delta_a}}}\left\{\frac{L_0 - L_i}{W} C_L \frac{y_p}{b} + C_{\ell_\beta} \beta\right\} \qquad \textit{eq 6.15}$$



Higher dynamic pressure over wing with operative engine generates unbalanced lift vectors and rolling moment, L_T, toward inoperative engine wing

$$L_T = L_0\ y_p - L_i\ y_p = y_p\ (L_0 - L_i)$$

$$C_{\ell_T} = \frac{L_T}{qSb} = \frac{y_p\ (L_0 - L_i)}{qSb} = \frac{L_0 - L_i}{W}\ C_L\ \frac{y_p}{b}$$

Figure 6.5
Rolling Moment Due to Asymmetric Power

6.1.3 Equilibrium Asymmetric Power Conditions

The previously discussed directional and the lateral control problems with asymmetric power will result in various equilibrium states. If an airplane displays conventional lateral-directional stability derivatives as shown on page V-33, the equilibrium flight conditions resulting from the failure of a right engine will be as shown in Figure 6.6.

Several things should be noted in Figure 6.6. Control of the airplane may be limited by either rudder or aileron. Although the rudder deflection required is reduced if the airplane is banked into the operating engine, high bank angles may be uncomfortable to the pilot and may be geometrically restricted in the take-off and landing environment. Furthermore, in order to maintain a constant vertical velocity with increasing bank angles, C_L must be increased with the resulting increase in induced drag and stall speed while increasing sideslip angles will result in higher form drag. These performance and control considerations will determine the optimum equilibrium flight condition and this optimum will probably be specified as a function of bank angle since that is the most obvious parameter to the operational pilot.

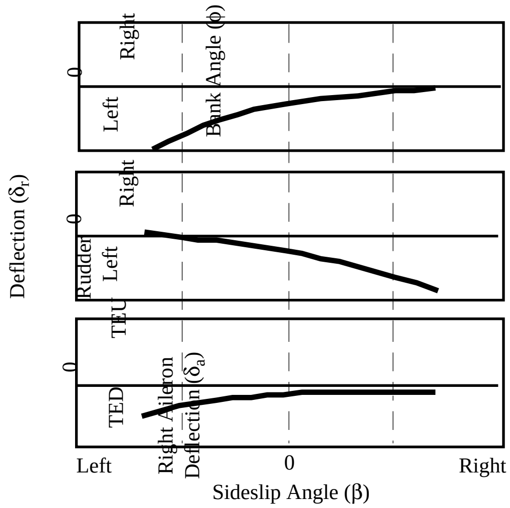


Figure 6.6
Asymmetric Power Equilibrium Flight Conditions Right Engine Failed

6.2 REGAINING CONTROL FOLLOWING SUDDEN ENGINE FAILURE

6.2.1 Engine Failure During Flight

When the pilot intentionally secures as engine in flight, the transient motions are generally mild and easily controlled if adequate control authority is available. However, sudden engine failures may occur under low altitude, low airspeed, high power flight conditions in a high lift or high drag configuration, such as during take-off or wave-off. The sudden engine failure in these cases may generate severe, potentially divergent rolling and/or yawing transients. The pilot may induce a similar situation by sudden application of asymmetric power to initiate a wave-off from an engine-out-approach.

The same factors which cause lateral-directional control problems in steady asymmetric flight conditions also are applicable to the sudden or dynamic engine failure. However, the control authorities required to arrest the motion following a sudden engine failure are usually larger than the control authorities necessary to maintain equilibrium flight. The severity of airplane response following a sudden engine failure is difficult to predict by theoretical analysis; the pilot delay time in recognizing the asymmetric power condition and applying appropriate control inputs influences the magnitude of the rolling and yawing motions. Actual flight test of critical conditions is the only means of establishing safe flight boundaries. The following hypothetical situation may aid in understanding some of the problems encountered with sudden engine failures (Figure 6.7).

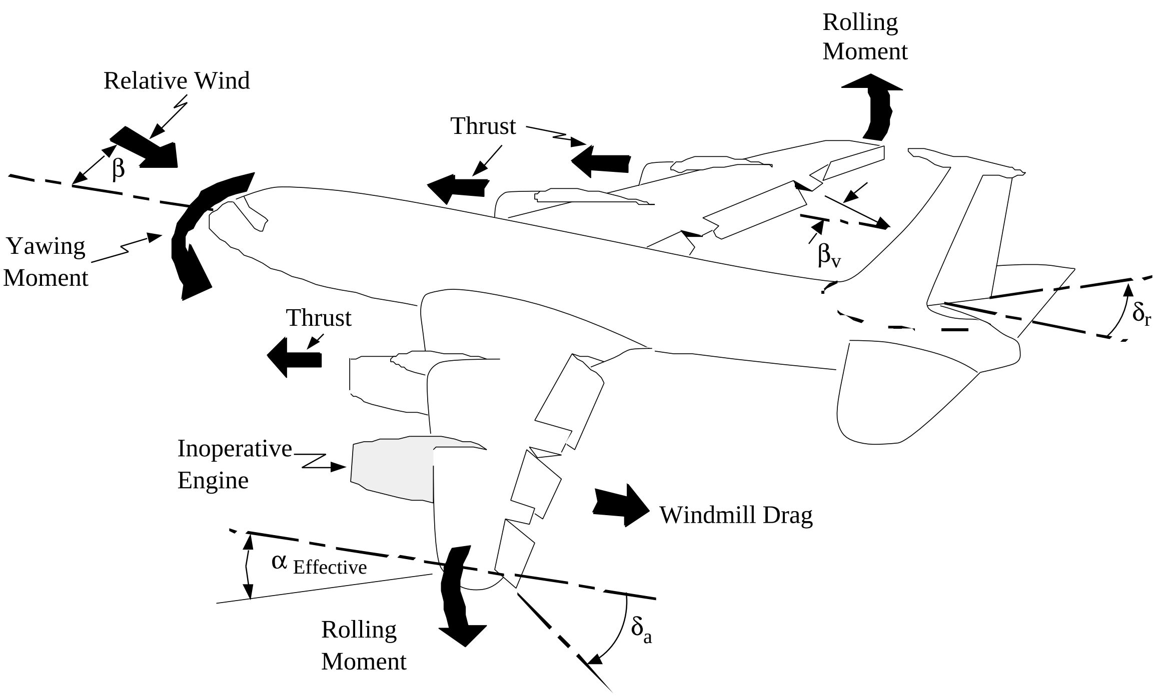


Figure 6.7
The Sudden Engine Failure

1. Assume the airplane is in a critical phase of flight, take-off configuration, take-off power on all engines, just after lifting off the runway.

2. The pilot experiences a sudden power failure on the left outboard engine. Because of the surprise factor, the pilot does not immediately react to the situation. The large yawing moment generated by the asymmetric power causes a large sideslip angle to develop <u>from the operative engine side</u>. If the sideslip angle reaches large enough proportions, the vertical tail may stall. An increase in drag accompanies the increase in sideslip, compounding an already (possibly) serious performance deficiency.

3. A rolling moment toward the inoperative engine will probably be generated by the yaw rate. This rolling moment will be increased if the airplane exhibits positive dihedral effect. Propeller-driven airplanes may rapidly diverge in roll due to slipstream effects, particularly if the wings are completely immersed in slipstream prior to the engine failure.

4. The pilot will likely apply large rudder and lateral control inputs to attempt to arrest the yawing and rolling motion. The large rudder input increases the tendency for the vertical tail to stall and may result in "rudder lock" if the control system is reversible. The lateral control input may generate an adverse yawing moment which increases the yawing moment toward the inoperative engine side. The large lateral control deflection, coupled with the rolling velocity, may cause the down-going wing to exceed stall angle of attack.

5. If the pilot is unable to achieve equilibrium flight with full lateral and directional control inputs, a <u>power reduction</u> on the operative engine side and/or an <u>increase in airspeed</u> will be required to prevent catastrophic consequences. Obviously, these measures may not be possible in a low altitude, marginal performance flight condition.

6.2.2 Engine Failure During Take-off

Engine failure on the ground during a take-off run is always a dynamic situation since the pilot must either abort his take-off or continue to accelerate to a lift-off airspeed. The ground minimum control speed will differ from the flight minimum control airspeed because of several things.

1. The inability to use bank angle and restrictions on the use of sideslip.

2. The moment arms for the vertical stabilizer and rudder are changed since they are taken from the airplane center of gravity in flight but generally act from the main landing gear while on the ground.

3. Additional yawing moments are produced on the ground by the landing gear and vary with the amount of side forces and differential longitudinal forces on the landing gear, the amount of steering used, and the runway condition.

4. Cross wind components essentially determine the take-off sideslip angle. Since the airplane must, in general, maintain the runway heading during take-off, the cross wind, in conjunction with the airplane's ground speed, will determine the magnitude of the sideslip and whether it is helping or hindering directional control of the airplane during a sudden engine failure.

6.3 ASYMMETRIC POWER PROBLEMS AT HIGH AIRSPEEDS

For the high performance, multiengined airplane, the failure of an engine or engines at high airspeed may be a more serious consequence than engine failure at low airspeeds. Asymmetric engine failure at high airspeeds may generate sideslip excursions large enough to exceed sideslip limitations and cause structural damage or catastrophic component failures.

For proper jet engine operation at very high Mach Numbers (over 2.0), the engine inlet shock wave pattern must be fashioned to provide the correct pressure in the engine for the given engine speed. If a disturbance (pressure of temperature fluctuation, abrupt power lever movement, etc.) upsets the shock pattern-pressure relationship, the shock wave may actually be expelled from the engine inlet. This phenomenon, known as "inlet unstart," can cause severe pressure fluctuations, compressor stalls, and engine failure. When engines are located in close proximity, one "inlet unstart" may trigger "inlet unstarts" on adjacent engines.

Asymmetric power problems at high airspeeds in high performance multiengined airplanes may be compounded by reduced directional stability at high supersonic Mach numbers and high altitude. These problems may result in limiting maximum airspeed or Mach numbers as functions of engine thrust settings. Another possible solution is to fail the corresponding engine on the opposite wing automatically in the event of engine failure in a flight condition where asymmetric thrust is catastrophic.

6.4 DEFINITIONS RELEVANT TO ASYMMETRIC POWER

Terminology used to describe airspeeds and conditions associated with asymmetric power flight is not standard throughout the aviation industry. The differences between civilian and military regimes are particularly noteworthy. When describing asymmetric power problems, the speaker or writer must be very careful to define the terminology of the presentation so that no misunderstanding is possible. The following definitions are generally considered to be standard by most flight test activities.

6.4.1 Critical Engine

The critical engine is that engine of a multiengined airplane, the failure of which produces the most critical condition to the pilot. The most critical condition will probably occur at high thrust and low airspeed (high C_L) as is the situation during take-off or wave-off. Under this condition, lateral or directional control cannot be regained and maintained following a sudden engine failure below a certain airspeed. The critical engine is the engine for which this minimum airspeed is higher than that associated with failure of any other engine. The critical engine may generally be predicted for a propeller airplane. Providing that the airfoil surfaces (wings, vertical, and horizontal stabilizers) are symmetrically attached to the fuselage and that the available control surface deflections are symmetric, the critical engine may be predicted from several factors: (1) as the angle of attack increases (high C_L), the down-going propeller blade sees a relatively higher local angle of attack than the up-going blade, which results in moving the thrust vector laterally on the propeller disk toward the down-going blade side, and (2) air flow swirl about the fuselage created by the rotating propeller(s) can affect the flow at the vertical tail so as to create a sideslip angle in one direction or the other, depending on the direction of the rotation of the propeller(s).

For clockwise rotation of the propeller(s) (as viewed from the rear), the above effects usually result in the left outboard engine being the critical one.

For jet-powered airplanes, the differences observed between flying qualities with left or right engine(s) inoperative are usually small enough to be attributed to differing maximum of idle thrust between the engines. Thus, the critical engine is not clearly defined by asymmetric flying qualities considerations. For these cases, other considerations, such as hydraulic or electrical power generated by individual engines and the consequences of loss of various airplane functions, may be used to determine the critical engine.

6.4.2 Minimum Control Ground Speed, V_{mcg}

The minimum control ground speed is the lowest speed at which directional control can be maintained on the ground when the critical engine fails during the take-off roll. The allowable deviation from the runway centerline and the pilot technique utilized influence the value of this speed.

6.4.3 Minimum Control Airspeed, V_{mca}

The minimum control airspeed is the lowest airspeed at which control of the airplane is possible with the critical engine inoperative. It may be defined by an equilibrium or static condition in which the critical engine has been failed prior to approaching the minimum conditions. It may also be defined by a sudden or dynamic condition in which the critical engine is failed at various airspeeds approaching the minimum conditions. For both cases, V_{mca} may be limited by lateral or directional control deflection available to counteract rolling or yawing moments and/or the control forces involved. At any rate, there will be a different static and dynamic minimum control airspeed for each:

1. Power setting utilized on the operative engine(s).

2. Configuration.

3. Condition of the inoperative engine(s) (feathered or wind-milling).

4. Bank angle utilized in the static condition.

5. Pilot if limited by control force requirements.

6.4.4 Safety Speed

Safety speed is defined as the lowest possible airspeed on a multiengine airplane at which the average pilot can maintain steady, straight flight without loss of altitude in the take-off configuration in the event of a sudden, complete failure of the critical engine. The pilot may make full use of all flight controls, may make configuration changes (retract landing gear, flaps, etc.), and the propeller of the failed engine may be manually feathered after allowing a suitable delay for an average pilot to regain steady, straight flight and identify the failed engine. Use of automatic feathering systems is permitted; however, the power on the operating engine(s) may not be reduced and no trim inputs may be utilized. Generally, it is the airspeed which should be attained after take-off before any attempt is made to climb (a pilot's handbook number). Safety speed may be established based on stability and control or performance characteristics, or both. The take-off safety speed for civil airplanes (transport category), commonly referred to as V_2, depends on both flying qualities and performance. Generally, there is a different safety speed, or V_2, for each flap setting used for takeoff; it may also vary with gross weight.

6.4.5 Refusal Speed

Refusal speed is defined as the maximum ground speed from which the airplane can be brought to a full stop in the remaining runway available after failure of the critical engine. This speed depends on stopping technique (maximum effort is normally utilized) as well as the length of the runway. Refusal speed is low for short runways and high for long runways (Figure 6.8). It is also sometimes called Accel/Stop speed, Emergency Distance speed, or V_{stop}.

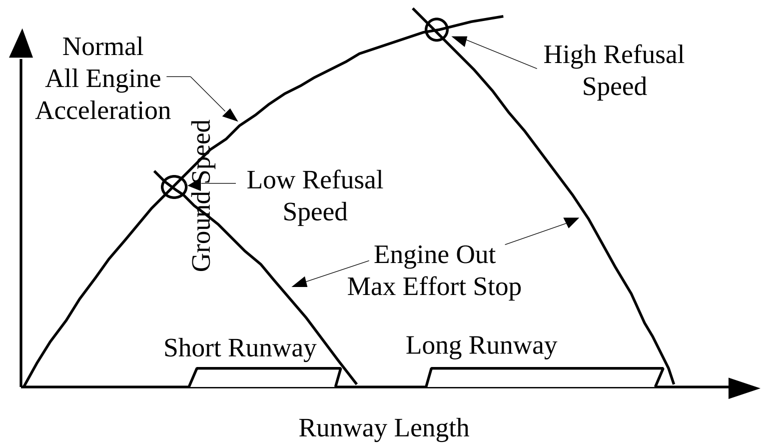


Figure 6.8
Refusal Speed Varies Directly with Length of Runway

6.4.6 Minimum Continue Speed

Minimum continue speed is the minimum ground speed to which an airplane can accelerate on the take-off roll, lose the critical engine, and continue the take-off with engine failed, becoming airborne just at the far end of the runway. This speed varies inversely with runway length; i.e., it is relatively low for long runways, etc. It is frequently referred to as Engine-Out Go Speed or V_{go} . If minimum continue speed is less than refusal speed, there is a "safe band" within which the pilot can either continue the take-off safely or abort the take-off safely (Figure 6.9). However, if refusal speed is less than minimum continue speed, there is a "dead man zone" within which the pilot can neither continue the take-off without running off the end of the runway nor abort the take-off without running off the end (Figure 6.10).

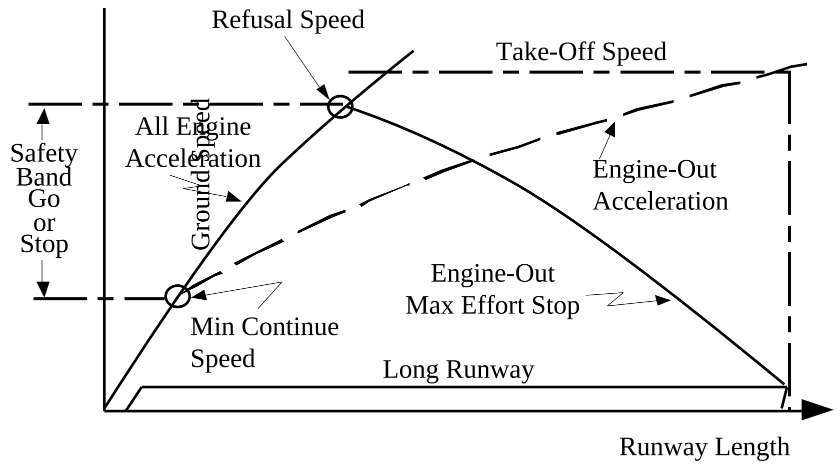


Figure 6.9
Refusal Speed Higher than Minimum Continued Speed

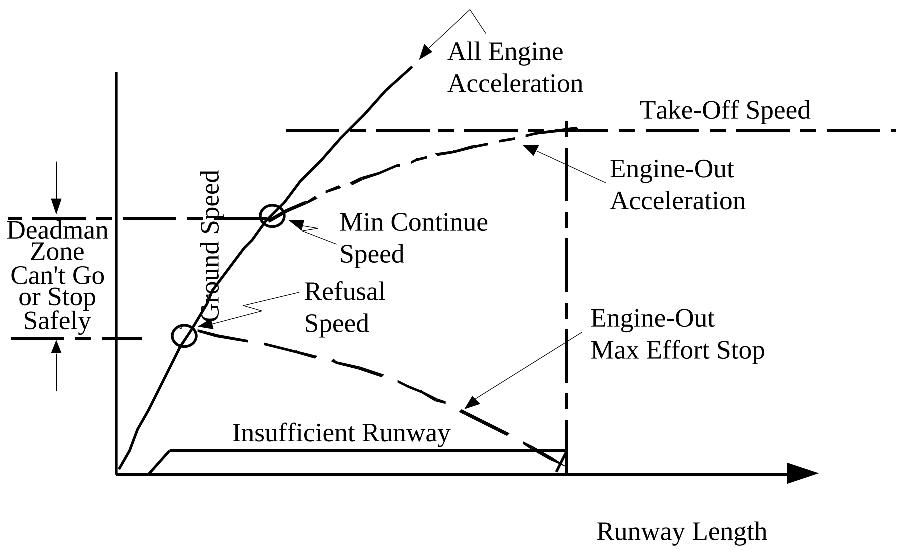


Figure 6.10
Minimum Continue Speed Higher than Refusal Speed

6.4.7 Critical Engine Failure Speed

If refusal speed and minimum continue speed are equal, the runway distance required to complete the take-off is equal to the distance required to stop. This speed is sometimes referred to as the Critical Engine Failure Speed or Decision Speed (V_1). The total runway distance required to accelerate to this speed, then stop or go after the engine failure, is called the Critical Field Length (Figure 6.11).

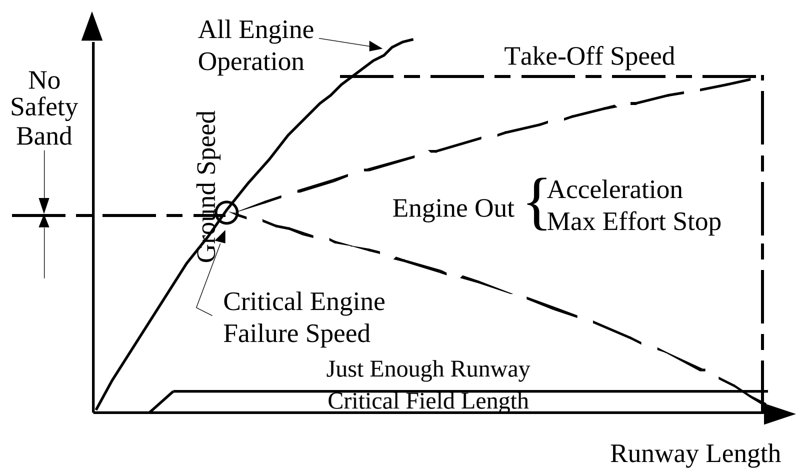


Figure 6.11
Critical Engine Failure Speed and Critical Field Length

6.4.8 Minimum Trim Airspeed

Minimum trim airspeed is the minimum airspeed at which steady heading flight can be maintained without pilot control force inputs with the critical engine inoperative. A different minimum trim airspeed exists for each configuration, power setting on operative engine(s), condition of inoperative engine, and bank angle (if limiting factor is directional trim). Minimum trim airspeed is most appropriately applied to an engine-out cruise condition with power for level flight or engine-out climb with normal rated power on the operative engine(s) and the inoperative engine feathered. These conditions relate to the problem of operation for relatively long periods during climb or cruise with an engine out.

6.5 TEST PROCEDURES AND TECHNIQUES ASYMMETRIC POWER

6.5.1 Preflight Procedures

A safe, yet rigorous, investigation of asymmetric power flying qualities must be conducted on all airplanes which may be expected to encounter asymmetric power. Thorough preflight planning is mandatory for these tests. The purpose and scope of the investigation must be clearly defined, then a plan of attack or method of test can be formulated.

Preflight planning must start with research. The airplane must be studied carefully - the flight test team can probably predict (roughly) the airplane's reaction to asymmetric power merely by looking at the airplane. Of course, a study of the lateral-directional control system is essential. All available information on normal lateral-directional flying qualities, rolling performance, and asymmetric power flying qualities and performance should be reviewed. Much useful information may be obtained by conversations with pilots and engineers familiar with the airplane. Additionally, the following points should be considered:

1. The function and influence of various flap settings; i.e., the airplane lift-to-drag ratio for various flap settings.

2. The consequences of engine-out operation of electrical, hydraulic, or pneumatic systems. Emergency electrical and/or hydraulic units may be required in the test airplane, particularly if a situation may arise where no normal electrical or hydraulic power is available.

3. The ability of the electrical system to carry the combined load of propeller feathering and landing gear or flap retraction.

Although the basic considerations of flight with asymmetric power are the same for both military and civil airplanes, the emphasis attached to the asymmetric power flying qualities varies. In the civil case, safety is the most important factor; the airplane must meet rigid minimum requirements before it is placed in operational commercial use. The same emphasis on safety will probably be applicable to large military transport airplanes as well.

For the commercial transport category airplanes, extensive ground and flight tests must be performed to determine safety speeds (V_2), refusal speeds (V_{stop}), minimum continue speeds (V_{go}), critical engine failure speeds (V_1), and critical field lengths. Based on these tests, the commercial transport airplane is certificated to operate from various runway lengths with various combinations of gross weights and center of gravity positions. The military transport airplane may be required to satisfy similar requirements. In many cases, the military transport has already been certificated as a commercial transport, and the minimum requirements for military operations already satisfied.

For the majority of military multiengine airplanes, operational effectiveness, vice safety, may be the most important design factor, particularly for combat aircraft. This fact lays a heavy burden on the test pilot designated to conduct asymmetric power tests on military airplanes. The test pilot must attempt to safely evaluate all asymmetric power conditions which may confront the operational pilot. If possible, the flight test team should attempt to extrapolate the results of the test to future service modifications, such as increased gross weight, increased engine output, etc.

Testing the engine-out characteristics of a military, multiengine airplane should include at least the following:

1. Determine the critical engine in the most critical configuration (probably take-off or wave-off).

2. Determine normal take-off acceleration (ground speed versus runway distance).

3. Determine take-off acceleration with the critical engine failed.

4. Determine abort deceleration with the critical engine failed.

5. Determine the minimum control ground speed with the critical engine failed.

6. Determine the minimum control airspeed, both static and dynamic.

7. Determine the minimum trim airspeed in pertinent configurations (probably

Engine Out Power and Engine Out Cruise (Power of maximum range)).

8. Evaluate approach, landing, and wave-off characteristics with asymmetric power.

The test conditions - altitude, configurations, center of gravity, and trim airspeeds - must be determined. Test conditions should commensurate, as much as possible, with the mission environment of the airplane. However, safety considerations dictate the investigations of asymmetric power flying qualities be performed in such a manner that the most critical conditions are approached with a reasonable build-up program. Altitude for conducting initial tests wherein engines are actually secured should never be less than 4000 feet above ground level. After adequate build up and with written permission from higher authority, the altitude restrictions may be relaxed so that engines may be secured in the very low altitude environment. Generally, simulated failures yield equally valid results at very low altitudes with much less risk. The airplane gross weight utilized for asymmetric power flying qualities investigations should be the lightest normal service loading for the configuration of interest. The light weight provides the best asymmetric power performance characteristics as well as allowing the maximum airplane response to a sudden engine failure. Additionally, for tests involving banking to balance the airplane under asymmetric power conditions in level flight, the bank angle required is inversely proportional to airplane gross weight. Center of gravity positions are not particularly critical for asymmetric power investigations; however, the most aft operational center of gravity positions should be utilized if feasible.

The amount and sophistication of instrumentation will depend on the purpose and scope of the investigation. A good, meaningful qualitative investigation can be performed with only cockpit and hand-held instruments. If accurate quantitative information is needed, automatic recording devices should be utilized. For initial tests on a new airplane, or for test on airplanes which may exhibit severe characteristics, telemetering pertinent parameters to a ground station may be required. A qualified engineering observer, with communications to the test pilot, should continually monitor the flight test records.

The final step in preflight planning is the preparation of pilot data cards. These data cards are best constructed from blank cards for each particular test. The cards should list all quantitative information desired and should be easy to interpret in flight. Adequate space should be provided for pilot comments.

6.5.2 Flight Test Techniques

When assessing the asymmetric power flying qualities of multiengine airplanes, the mission of the airplane and the influence of engine failure(s) on that mission must be considered. The failure of one or more engines asymmetrically generally results in an emergency condition. The primary mission of the airplane can usually not be accomplished in this situation; therefore, the mission reverts to regaining control of the airplane, cruising to a suitable landing spot, and accomplishing a safe carrier or filed landing. The pilot cannot expect flying qualities under asymmetric power conditions to be particularly pleasant; however, the pilot rightly expects acceptable characteristics which permit the airplane to be at least controllable. Some airplanes may, however, be designed to carry stores asymmetrically or to shut down engines asymmetrically for increased endurance and should therefore retain pleasant flying qualities even in these configurations.

6.5.2.1 PRELIMINARY TESTS

Certain preparatory tests are necessary before the asymmetric power tests are performed.

6.5.2.1.1 Check-Stalls

The airplane should not be stalled with asymmetric power until the stall characteristics and asymmetric power characteristics have been determined. A stall speed should, therefore, be determined for each test configuration with all engines at idle power. In subsequent asymmetric power tests, the stall speed should be regarded as minimum speeds; if the stall speed is reached prior to reaching minimum control speed, the asymmetric power investigation should be terminated. (This rule may be unduly restrictive for airplanes with the wings immersed in propeller slipstream. If so, additional check-stalls should be performed in theses airplanes with symmetric power representative of the test configuration.)

6.5.2.1.2 Sideslip

It is most important to determine if the airplane is prone to vertical tail stall or rudder lock prior to embarking on asymmetric power tests. Therefore, for each test configuration, steady heading sideslips, up to maximum permissible or obtainable sideslip angle, should be performed with symmetric power. The airspeed generally used for this test is approximately 1.4 times the stall speed previously determined for the configuration. The variation of indicated airspeed error and angle of attack with sideslip angle should also be noted because of the obvious impact of these characteristics of safety of flight with asymmetric power.

6.5.2.1.3 Failure Simulation

Power settings should be determined which simulate the drag characteristics of a failed jet engine or the drag characteristics of both a windmilling and a feathered propeller. These simulated power settings are convenient and relatively safe means of conducting asymmetric power testing and will be used by operational pilots for engine-out training. Since these drag characteristics will obviously vary with airspeed, a representative airspeed and configuration should be used to determine the simulation. This should be done both with propeller windmilling and with the propeller feathered for propeller airplanes. Determination of the simulation power setting will in general be an iterative process as follows:

1. Determine a static minimum control airspeed in the representative configuration (probably Power Approach) using idle thrust for jets and throttle closed, propeller windmilling for propeller airplanes.

2. Increase airspeed to a safe margin (at least 1.4 times the minimum control airspeed just determined) and secure the critical engine.

3. Carefully slow to approximately 1.2 times the previously determined minimum control speed and stabilize in level flight using power from the operative engine(s). This airspeed should be representative of take-off and approach airspeeds.

4. Without changing power on the operative engine(s), restart the secured engine and vary its power so as to re-stabilize in level flight at the airspeed determined in paragraph 3. This power setting should then be a good failure simulation.

5. Continue asymmetric power testing using the failure simulation to more precisely determine the minimum control airspeeds, etc.

6.5.2.1.4 Critical Engine

Several assumptions may generally be made in determining the critical engine. If we assume that the take-off configuration is representative of the most critical configuration, that idle thrust or idle power is approximately the same as that from a failed engine, and that the engine with the highest minimum control speed in a dynamic failure will also have the highest minimum control speed in a static case, then the critical engine may be determined as follows:

1. Determine a static minimum control airspeed, wings level, in the take-off configuration using idle power on the left outboard engine and maximum power on the remaining engines. Trim should probably remain at the take-off setting as this is the most representative case.

2. Alternate the test with the right outboard engine. The idle and maximum power settings used above should be adjusted slightly to ensure that the exact same power asymmetry exits in each direction so that thrust differences caused by engine trim at maximum power do not affect the test.

3. The engine with the <u>highest</u> minimum control airspeed may then be assumed to be the critical engine.

6.5.2.2 CONTROL IN STEADY FLIGHT WITH ASYMMETRIC POWER: MINIMUM TRIM AND MINIMUM CONTROL AIRSPEEDS IN EQUILIBRIUM FLIGHT

The pilot will generally be able to cope with sudden failures under normal cruise flight conditions with little difficulty. Therefore, the primary purposes of asymmetric power flying qualities investigations under cruise conditions are:

1. To determine the degree of difficulty the pilot will encounter in a long-range cruise task with asymmetric power.

2. To provide a "build-up" to more demanding and critical tests in take-off and wave-off flight conditions.

Obviously, an infinite number of minimum trim and minimum control airspeeds could be determined as the result of variations in configuration, power setting, and bank angle. The test pilot should determine appropriate conditions, in which to evaluate these minimum trim airspeeds. For minimum trim airspeed determinations, several obvious conditions would include engine-out climb and engine-out cruise. Engine-out climb initial conditions would be: critical engine simulated failed (and feathered for propeller airplane), maximum continuous (normal rated) power on the operating engine(s), and zero bank angle. Engine-out cruise would require: critical engine simulated failed (simulated feathered for a propeller airplane), power set on operating engine(s) to provide level flight at engine-out maximum range airspeed, and zero bank angle. Since the drag due to sideslip may be reduced by flying in a slight bank, it may be advisable to determine the minimum trim airspeed in the above configuration using the bank angle for minimum drag. It is obviously desirable to be able to climb, hands off, at the optimum maximum range engine-out climb airspeed and to cruise, hands off, at the maximum range engine-out cruise airspeed.

6.5.2.2.1 Minimum Trim Airspeed

The minimum trim airspeed may be determined as follows:

1. Stabilize in the desired configuration at approximately twice the stall speed determined in the preliminary tests.

2. Establish the critical engine in a simulated feathered condition and the other engine(s) at the desired power setting.

3. Trim all control forces to zero in steady heading flight, initially maintaining zero bank angle.

4. Smoothly and slowly reduce airspeed by means of longitudinal control inputs while maintaining steady heading flight. Continue to trim all control forces to zero as the airspeed decreases.

5. Eventually an airspeed will be reached where one or the other of the lateral or directional trimmers is at its limit. Below this airspeed, the pilot cannot trim all control forces to zero in steady straight flight. This is the <u>minimum trim airspeed</u> for the test conditions and the <u>limiting trim axis</u> (lateral or directional) should be noted.

6. If the limiting trim axis was directional, the test may be continued by applying a small bank angle (usually 5 degrees) towards the good engine.

7. Care must be exercised to obtain data only when the airplane is stabilized in unaccelerated flight conditions. Primarily, outside visual references should be used to maintain bank angle as desired and zero yaw rate; cockpit instruments should be cross-checked frequently. The ball of the needle-ball instrument should be <u>perfectly centered</u> in its race during wings level tests. It is an excellent indicator of lateral accelerations resulting from unbalanced lateral forces.

8. Altitude variance during the determination of minimum trim airspeeds should not exceed ±1000 feet from the test altitude.

6.5.2.2.2 Static Minimum Control Airspeed

The test pilot must carefully define both configuration and trim settings for static minimum control airspeed testing. In general, the primary interest should be in critical flight evolutions such as take-off and wave-off. Trim controls may be left at some specified setting during static minimum control testing or may be used to their full range as required depending on what the test pilot determines is most representative. The most critical condition will usually be the take-off case. When testing for this condition, the trim settings should be those normally recommended for take-off. Static minimum control airspeed may be determined as follows.

1. For the initial determination of V_{mc} (static), stabilize at approximately twice the stall airspeed in the desired configuration and set the desired asymmetric power (simulate the failed engine using the previously determined power setting). Subsequent investigation of V_{mc} (static) may be made by stabilizing initially at approximately 1.4 times the V_{mc} previously determined.

2. Smoothly and slowly reduce airspeed by means of longitudinal inputs while using lateral and directional controls to maintain steady, straight flight with zero bank angle. If testing for the engine failure after take-off case, trimmers must remain at the settings recommended for take-off with symmetric thrust. If desired, stabilize at predetermined airspeed intervals (3-5 KIAS increments) and record estimated or measured control forces and deflections; otherwise continue to decelerate at a rate which should not exceed 0.5 KIAS/second.

3. Eventually, an airspeed will be reached where either full directional or full lateral control surface deflection is required to maintain steady heading, wings level flight. In some cases, the strength capacity of the pilot will be reached prior to full control deflection. This airspeed, below which steady heading, wings level flight cannot be maintained, is the <u>minimum control airspeed for the test conditions</u>. This airspeed and the limiting factor (usually directional or lateral control deflection or force) should be noted.

4. If the limiting factor is directional control deflection or rudder force requirements, minimum control airspeed can be <u>reduced</u> by banking toward the operating engine(s). (Obviously, different minimum control airspeeds could be determined for each bank angle utilized. Empirically, <u>5 degrees</u> of bank has generally been used as an approximation to the optimum bank angle considering both performance and flying qualities.) If appropriate, minimum control airspeed and limiting factor with 5 degrees of bank should be determined.

5. If at any point during the minimum control airspeed tests, the pilot loses lateral or directional control of the airplane, control may be regained by increasing airspeed and reducing power on the operative engine(s) (or increasing power on the simulated inoperative engine).

6. After determining minimum control airspeeds with the critical engine in a simulated feathered condition, the airspeeds may be checked with the critical engine actually secured and the propeller actually feathered.

7. The static minimum control airspeed does not imply that the airplane is unsafe to fly at a lower airspeed either by slightly reducing the power asymmetry or by accepting a resulting yaw rate. If the power asymmetry is maintained, the airplane may or may not be safe to fly below the minimum control airspeed depending upon whether the departure from controlled flight is a mild (but steady) yaw rate or it is more violent or radical.

6.2.5.2.3 Qualitative Investigation

After the quantitative tests described above, the test pilot should conduct a qualitative investigation of the flying qualities exhibited at representative cruise airspeeds with asymmetric power. The pilot should be able to trim all control forces to zero at these airspeeds without undue effort. Turns and heading changes, representative of maneuvers required in instrument or visual cruise conditions, should be performed to determine if excessive pilot coordination, control forces, or control movements are required. Generally, bank angle changes of up to 30 degrees from wings level are considered adequate for most maneuvering on asymmetric power.

6.5.2.3 CONTROL IMMEDIATELY FOLLOWING AN ENGINE FAILURE: MINIMUM CONTROL AIRSPEEDS WITH SUDDEN ENGINE FAILURES

The difficulty the pilot experiences in maintaining control of the airplane following sudden, asymmetric power failures increases with the following factors:

1. Increase in the operating engine power output. For a constant throttle or power lever position (assume full or maximum), engine thrust usually increases as altitude decreases.

2. Decrease in airspeed.

3. Decrease in excess power available for climb and acceleration.

Thus, the take-off and wave-off flight conditions, characterized by high power settings, low airspeeds, low altitude, and high drag configurations, are generally the most critical for the investigation of sudden engine failures. The asymmetric power flying qualities in these conditions should allow the average operational pilot to regain and maintain control of the airplane at all airspeeds representative of operational procedures.

Minimum dynamic control airspeed for the average pilot experiencing a sudden failure of the critical engine may be determined as follows:

1. Stabilize at approximately twice the stall speed (or 1.4 times the static minimum control speed) determined in the preliminary tests in the desired configuration at a safe test altitude. Power should be maximum obtainable or allowable on all engines and trim should be set for a symmetric power take-off (take-off configuration) or for a normal symmetric power approach (wave-off configuration).

2. Smartly reduce the power on the critical engine to minimum power, simulating a sudden failure. The test pilot should pause a reasonable time interval to account for the surprise factor of a sudden engine failure under operational conditions. Engine failure cues should be determined (yaw, roll, audio, or cockpit instrument) and a suitable reaction delay time should then be specified and used for continued testing. In no case should recovery control inputs be applied until

1 second has elapsed, a 20 degree bank angle change has occurred, or the sideslip limit is reached (whichever occurs first). After the time delay, steady, straight flight conditions should be regained at the original stabilized airspeed. Longitudinal, lateral, and directional control inputs may be used as required to effect the recovery to controlled flight.

3. The test pilot should note control forces and positions required while regaining control and to maintain steady, straight flight with less than 5 degrees of bank. If automatic recording devices are available, the entire maneuver, from "power chop" to steady, controlled flight, should be recorded.

4. Reduce the airspeed at which engine failure is simulated by small increments (5-10 KIAS) and repeat steps 2 and 3. An alternative method of build-up would be to make several power chops at each stabilized airspeed starting with a very slow power reduction (approximately a static condition), resetting symmetric power, and incrementally increasing the speed of the power reduction until it becomes a true power chop.

5. Eventually an airspeed will be reached where control can barely be regained or where, in the test pilot's opinion, the aircraft motions following the engine failure and while control is begin regained become unacceptable. Full control defection requirements may not be a good indication that limiting conditions have been reached since the pilot may elect to use full deflections at speeds higher than V_{mc} (dyn) to quicken the recovery; however, excessive control forces or excessive pilot skill and coordination requirements may well define a limit. The limiting factor must be specifically defined by the test pilot. It must also be noted that by definition, V_{mc} (dyn) cannot be lower than V_{mc} (static) for the same conditions.

6. Based on the test results, the minimum dynamic control airspeed must be decided upon. Such factors as ease of regaining and maintaining control, control forces and deflections required, and reaction time allowed must be taken into account. The airspeed recommended must allow an adequate safety margin for average pilot skill and proficiency.

7. If control of the airplane is lost during these tests, the pilot may regain control

by increasing airspeed and reducing power on the operative engine(s) (or increasing power on the simulated inoperative engine). Particular caution should be exercised at slower airspeed test points since exaggerated nose-up pitch attitudes will be required to stabilize with symmetrical power at the slow airspeeds. Therefore, airspeed decrease may be quite rapid after power reduction on the critical engine.

8. After the minimum dynamic control airspeed is decided upon, the airspeed may be checked by actually failing the critical engine and feathering the propeller at the minimum airspeed.

The static and dynamic minimum control airspeeds determined at altitude may be extrapolated to sea level as shown in Figure 6.12.

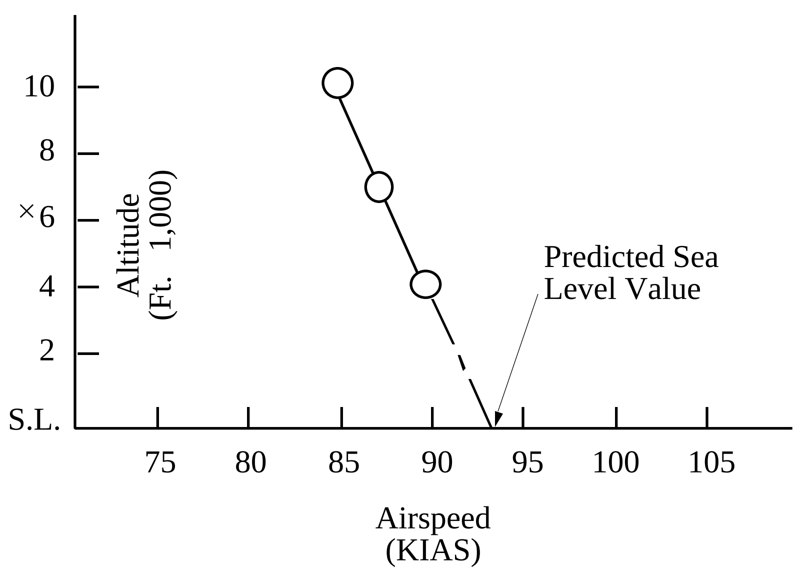


Figure 6.12
Extrapolation of Minimum Control Airspeed to Sea Level

The maximum power available at each altitude during the flight tests should be carefully noted and engine thrust or power available curves as a function of altitude should be consulted to ensure that no asymmetric thrust anomalies occur in the extrapolation altitude band.

6.5.2.4 MINIMUM CONTROL GROUND SPEEDS WITH SUDDEN ENGINE FAILURES

Minimum control ground speed testing is <u>extremely critical</u> and should generally be conducted after the test pilot is familiar with asymmetric power flying qualities in flight. The build-up to the minimum control ground speed must be slow and deliberate since there is no altitude and only limited area in which to recover control of an airplane following engine failure. Consideration must be given to runway length and width, arresting gear, brake temperatures, air crew escape system parameters and airfield crash and rescue equipment. Various methods may be used for minimum control ground speed testing depending upon the engine type, flight controls, and mission of the airplane; however, the following method may be used as a guide.

1. In the take-off configuration set the critical engine at a simulated failed power. Slowly accelerate with full rudder into the operating engines by adding power on the operating engine(s). Maintain directional control and accelerate down the runway by modulating the operating engine(s) until an airspeed is reached where full asymmetric power is controllable. This will be the minimum control ground speed. During the acceleration the ailerons should be neutral, asymmetric braking should not be used, and nose wheel steering should be used only if its use is recommended for normal take-offs. During initial tests, the crosswind should be zero or slightly into the operating engine(s). As the test pilot becomes more familiar with asymmetric power on the ground, the minimum control speed tests should be conducted with increasing crosswind components into the critical engine.

2. An alternate method would be to initially accelerate using symmetric power. The power on the critical engine would then be slowly reduced, while slowly applying rudder up to full rudder into the operating engines until an airspeed was reached where the airplane could be controlled with the power on the critical engine reduced to its failed simulation setting. This speed would be the minimum control ground speed.

3. Once the minimum control ground speed has been determined it should be verified by conducting power chops of the critical engine from a symmetric power take-off configuration. A safe build-up in airspeed and power chop quickness should be utilized just as in dynamic minimum control airspeed testing.

6.5.2.4.1 Safety Speed

Safety speed allows for failure of the critical engine in configuration take-off followed by configuration changes to reduce drag and conversion to a climb without loss of altitude. Thus, safety speed will be the higher airspeed of:

1. Minimum control ground speed.

2. Minimum dynamic control airspeed in configuration take-off.

3. That airspeed from which a climb can be initiated with the critical engine failed, after allowing for any deceleration, which the average pilot might experience during engine failure and subsequent propeller feathering and configuration change, without loss of altitude.

6.5.2.5 APPROACH AND LANDING CHARACTERISTICS WITH ASYMMETRIC POWER

The final phase of the asymmetric power investigation involves the determination of approach and landing characteristics. From these tests, the acceptability of asymmetric power flying qualities during VFR and IFR approaches and filed and carrier landings is determined. Additionally, optimum techniques for these evolutions may be derived and/or recommended techniques may be evaluated. The following points should be kept in mind while evaluating approach and landing characteristics with asymmetric power.

1. The critical engine may be placed in a simulated feathered condition vice actually secured. (For propeller-driven airplanes, the propeller control should be placed to full increase or maximum RPM on final approach in case a symmetric power wave-off is necessary.)

2. Standard traffic patterns and altitudes need not be adhered to; steep turns should be avoided.

3. Excessive crosswinds and turbulence unduly increase pilot workload for initial tests. Close attention to the crosswind must be given during each separate portion of the evaluation.

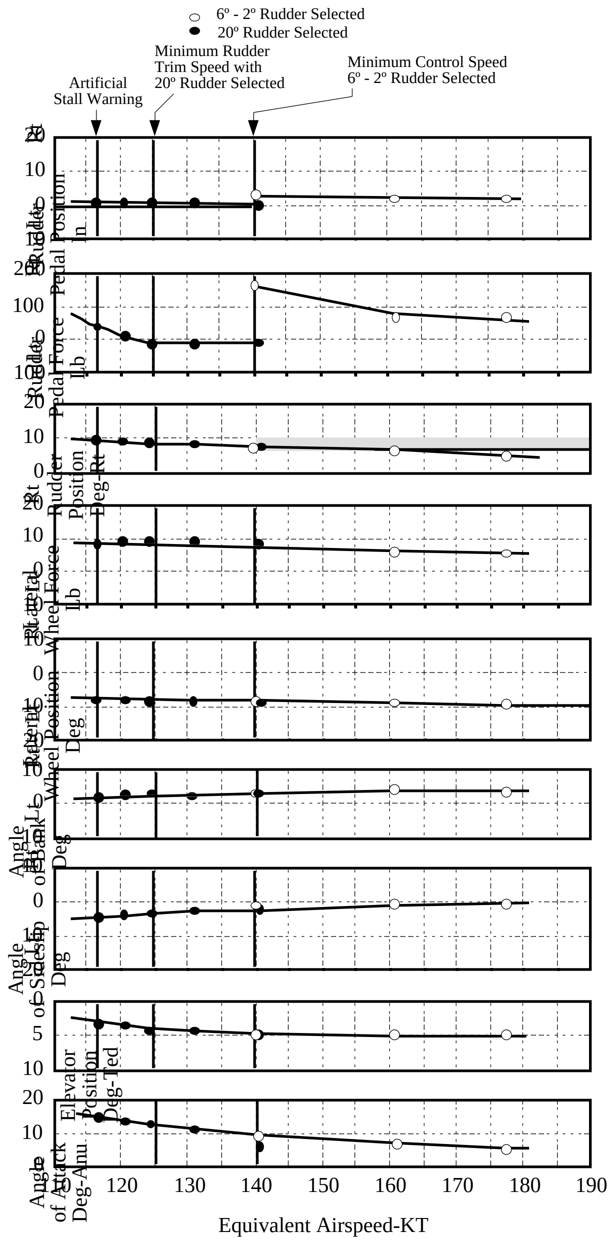
4. The tendency for the airplane to swerve toward the operative engine(s) with power reduction at field touchdown may be diminished by smooth power reduction, rudder inputs, braking, and nosewheel steering (if available). This swerve tendency may be particularly pronounced on twin-engine turboprop airplanes. For this type, initial power reduction at field touchdown should be only to FLIGHT IDLE. After counteracting initial swerve, the operative engine may be brought to GROUND IDLE. Use of reverse thrust <u>asymmetrically</u> may result in loss of directional control. (However, <u>symmetrical</u> reversing may be employed by using the symmetric operative engines on four-engine airplanes.)

6.5.3 POSTFLIGHT PROCEDURES

As soon as possible after returning from the flight, the test pilot should write a brief, qualitative report of the asymmetric power flying qualities. This report should be written while the events of the flight are fresh in the pilot's mind. The test pilot's qualitative opinion will be the most important portion of the final report of the asymmetric power flying qualities.

Asymmetric power characteristics in steady, equilibrium flight conditions are effectively presented as plots of pertinent control forces and positions versus airspeed (Figure 6.13). For dynamic characteristics (sudden engine failures), time histories should be presented if automatic recording devices were utilized (Figure 6.14).

The terminology used in the technical report regarding minimum speeds and conditions must be explicitly defined. Expressions utilized to describe airspeeds and conditions associated with asymmetric power are not standard throughout the aviation industry. Thus, when describing the test results, the writer must be extremely careful to precisely define each expression which possibly could be misinterpreted.



Model ____________ Airplane

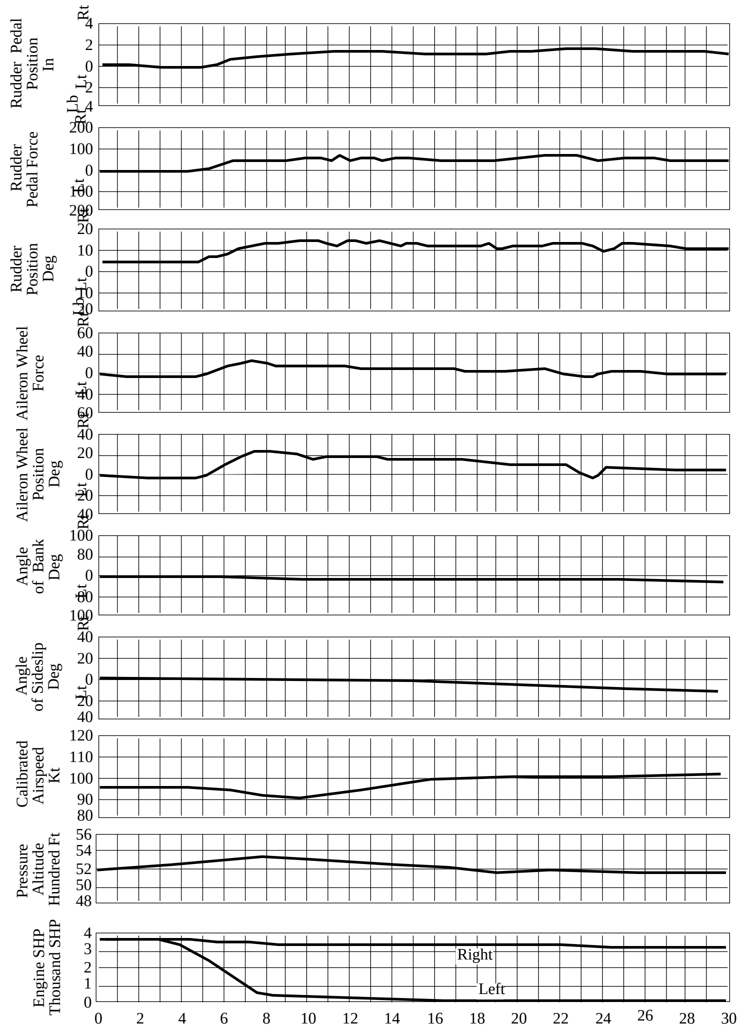
BuNo ______________

Pilot: ________________ Date: 2 December 1966

Configuration: P (Single Engine) Loading: Normal Transport

Figure 6.13
Static Asymmetric Power Characteristics
(Left Engine Secured, Propeller Feathered)

Gross Weight - 42,350 Lb
CG Position - 33.2 MAC
Stability Augmentation - On
Power Chopped to Flight Idle



Model_____________Airplane

BuNo ________________

Pilot: __________________ Date: 2 December 1966

Configuration: Takeoff Loading: Normal Transport

Figure 6.14
Time History of Simulated Left Engine Failure in Configuration Take-Off

6.6 SPECIFICATION REQUIREMENTS

Requirements for asymmetric power flying qualities are contained in the following applicable paragraphs of Military Specification MIL-F-8785C of 5 November 1980, hereafter referred to as the Specification.

3.3.9 Lateral-directional control with asymmetric thrust.

3.3.9.1 Thrust loss during take-off run.

3.3.9.2 Thrust loss after take-off.

3.3.9.3 Transient effects.

3.3.9.4 Asymmetric thrust- rudder pedals free.

3.4.2.1.3.1 One-engine-out stalls.

3.4.2.2 Post-stall gyrations and spins.

3.4.8 Transients following failures.

3.4.9 Failures.

3.4.10 Control margin.

3.6.1.1 Trim for asymmetric thrust.

The requirements of the Specification may be modified by the applicable airplane Detail Specification. Some comments to assist in interpretation of the requirements in the paragraphs listed above may be helpful and are presented below.

3.3.9 Lateral-directional control with asymmetric thrust - This is a general paragraph which contains the all-important sentence, "following sudden asymmetric loss of thrust from any factor the airplane shall be safely controllable." Any dangerous characteristic exhibited under any representative operational flight condition is a violation of the requirement. Additionally, the requirements stated in 3.3.9.1 through 3.3.9.5 must be met.

3.3.9.1 Thrust loss during take-off run - Normally, no asymmetric tests will be made on the take-off run at TPS.

3.3.9.2 Thrust loss after take-off - This paragraph refers to a sudden failure of the critical engine (worst case) in the take-off configuration. The pilot must be able to achieve and maintain straight flight following the sudden failure at all airspeeds above V_{stop} + 10 knots. No configuration change is permitted other than operation of automatic devices, such as autofeather. The bank angle used in the steady equilibrium condition must not exceed 5 degrees and rudder and aileron forces are to be within the stated limits with trim set for symmetric power take-off.

3.3.9.3 Transient effects - Note that no response to the simulated engine failure is permitted for at least 1 second.

3.3.9.4 Asymmetric thrust - rudder pedals free. This paragraph describes a maneuver utilized as a indication of the static directional stability in the worst asymmetric condition. Trim is set for wings level steady heading flight at a speed of $1.4V_{min}$ with symmetric normal rated power. After failure of the critical engine (a propeller may only be feathered if the automatic feathering system normally operates in the configuration under test) the pilot must be able to maintain straight flight at this and all higher speeds by banking without making any rudder inputs and allowing the airplane to sideslip. For most airplanes as speed is increased above $1.4V_{min}$ the test will become progressively less demanding. However, for certain airplanes the problem may become more acute at very high speeds.

3.4.2.1.3.1 One-engine-out stalls - This paragraph requires that in the event of a stall occurring at or above V_{mc} (as might be the case, for example, with a heavy airplane) the resulting stall shall be recoverable. Power may be reduced on the good engine(s) during recovery if required.

3.4.2.2 Post-stall gyrations and spins- - This paragraph effectively specifies that no concessions will be permitted for airplanes with asymmetric thrust in the entry to and recovery from post-stall gyrations and spins, though power may be reduced on the good engine(s) as required during recovery. These test will not normally be conducted at TPS.

3.4.8 Transients following failures - This paragraph refers to airplane motions following any airplane system or component failure. Failures resulting in asymmetric thrust are adequately covered under 3.3.9 and no additional requirements are stated here.

3.4.9 Failures - The relevant requirement in this paragraph is that the pilot shall be provided with immediate and easily interpreted indications of a failure resulting in asymmetric thrust. The requirement related to dangerous flying qualities is covered in 3.3.9.

3.4.10 Control Margin - With regard to the reference to "transients from failures in the propulsion ... and other relevant systems," this paragraph really says the same things as paragraphs 3.3.9 to 3.3.9.5.

3.6.1.1 Trim for asymmetric thrust - This paragraph requires that in the worst asymmetric case it shall be possible to trim elevator, aileron, and rudder forces to zero at all level flight cruise speeds from best range speed for the engine-out configuration to the maximum speed obtainable with normal rated thrust on the functioning engine(s). Or, in other words, minimum trim speed should be less than maximum range speed for the engine-out configuration.

6.7 LIST OF REFERENCES

1. *Advisory Group For Aeronautical Research and Development,* Flight Test Manual, Volume II, Stability and Control, by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control,* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University, and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Background Information and User Guide for MIL-F-8785B(ASG);* "Military Specification - Flying Qualities of Piloted Airplanes," by C.R. Chalk, T.P. Neal, T.M. Harris, F.E. Prichard, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Technical Report AFFDL-TR-69-72, August 1969.

4. *Federal Aviation Regulations, Part 25, Airworthiness Standards*: *Transport Category Airplanes*, Federal Aviation Agency, Washington, D.C.

5. *Flight Testing of Aircraft* by M. G. Kotick, et al., NASA Technical Report TT F-442, April, 1967.

6. *Handling and Performance Exercises Test Technique*, Empire Test Pilots' School, Flying Department Notes.

7. *Military Specification, Flying Qualities of Piloted Airplanes* MIL-F-8785C, of 5 November 1980.

8. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

9. *Stability and Control, USAF Aerospace Research Pilot School,* Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

10. *Stability and Control, Empire Test Pilot's School Ground School Notes,* Book "C."

11. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

CHAPTER SEVEN

TRANSONIC-SUPERSONIC FLYING QUALITIES

CHAPTER SEVEN

FIGURES

CHAPTER 7

TRANSONIC-SUPERSONIC FLYING QUALITIES

7.1 INTRODUCTION

Airplanes designed to fly operationally at transonic and supersonic airspeeds generally experience few of the instabilities which were encountered in the first experimental supersonic types. Knowledge of the transonic-supersonic regimes has been utilized to develop airplane conformations suited to the high Mach number operating environment. Thus, these regimes are no longer ones to be entered only inadvertently or with apprehension. Rather, the airplane capable of operating transonically and supersonically with satisfactory stability and control characteristics has obvious tactical, strategic, and logistics advantages.

The same standard test techniques presented previously may be utilized in the transonic-supersonic testing environment. However, several peculiarities of the stability and control characteristics of the supersonic airplane must be understood in order to effectively conduct a test program in this environment. Additionally, the approach to transonic-supersonic flight testing may be somewhat different than the approach to subsonic flight testing.

7.2 THEORY

7.2.1 General

7.2.1.1 TRANSITION FROM SUBSONIC TO SUPERSONIC FLIGHT

Various components of the airplane structure are subject to local velocities which may be lower or higher than the airplane velocity. As the airplane's airspeed increases, a Mach number will be attained at which some component of the airplane is subjected to local sonic velocity. This free-stream Mach number at which a local Mach number of 1.0 is attained at any point on the airplane is called the critical Mach number (Figure 7.1). There exists a Mach number band within which regions of both subsonic and supersonic flow are present on the airplane. This Mach number band is called the transonic region or transonic regime. The transonic region for a particular airplane obviously depends on its design characteristics.

The flying qualities of airplanes operating in the transonic-supersonic flight regime are generally characterized, to some degree, by various peculiarities not normally encountered in subsonic flight. These peculiarities may be attributable to one or more of the following factors:

1. Air compressibility effects.

2. Elastic deformation of the airplane structure.

3. Airplane conformation and mass distribution.

4. Reduction of air density at high altitudes.

Each of these factors, and its possible influence on flying qualities will be discussed in turn.

7.2.2 Air Compressibility Effects

Compressibility is defined as that property of a substance which causes its density to increase with increase in pressure. In aerodynamics, this property of air is particularly manifested at high airspeeds. Compressing the air about an airplane may generate buffeting, control surface buzz, trim changes, and other phenomena not ordinarily encountered at low airspeeds. These phenomena are commonly know as compressibility effects and may be attributed to various peculiarities encountered at transonic and supersonic Mach numbers.

7.2.2.1 AERODYNAMIC CENTER MOVEMENT

In the transonic region, the formation of shock waves on the wing surface and the resulting separated flow (Figure 7.1) causes movement of the wing aerodynamic center. The magnitude of the aerodynamic center movement depends on the design parameters of the wing (Figure 7.2).

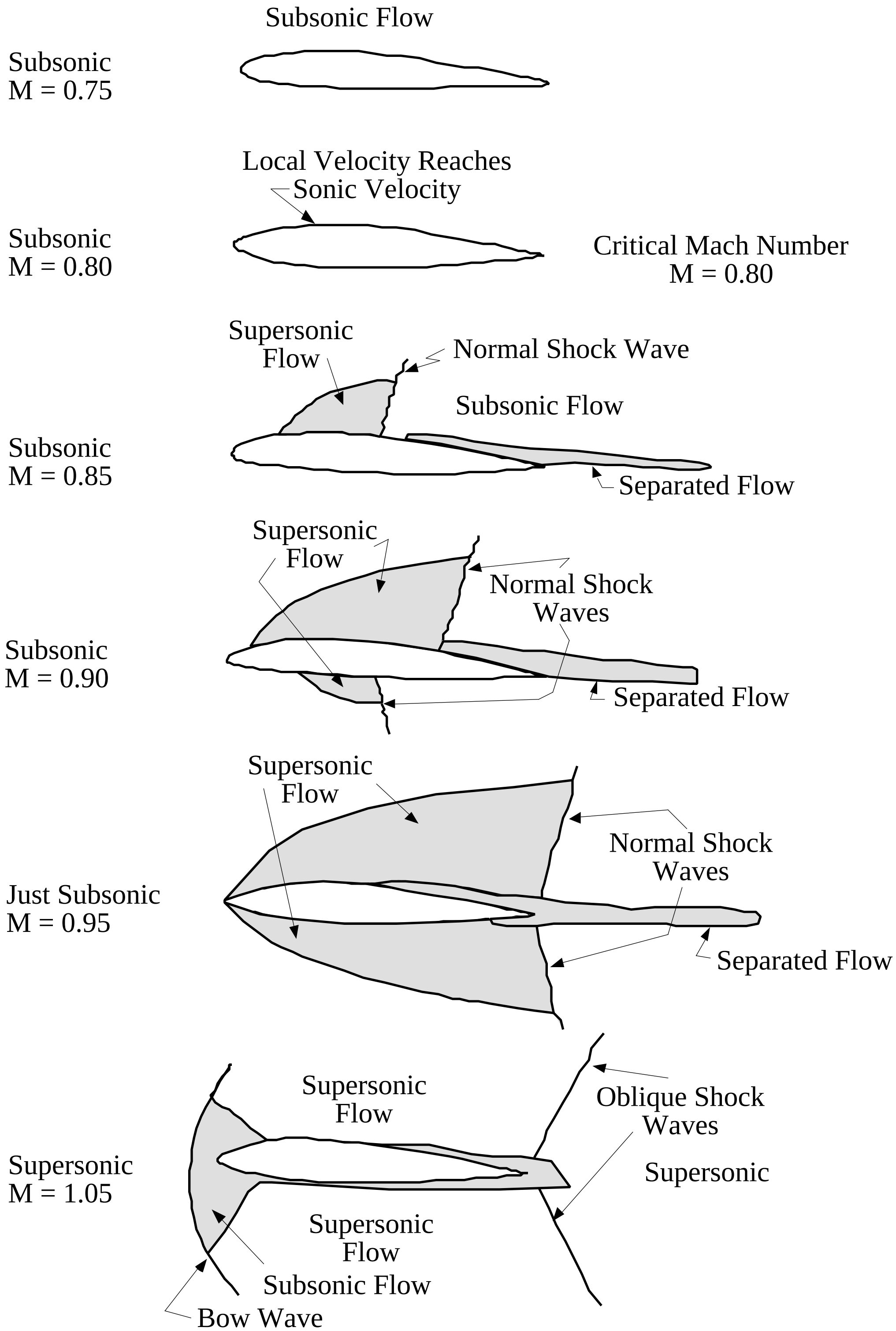


Figure 7.1
Typical Flow About a Wing Section in Transitioning from Subsonic to Supersonic Flow

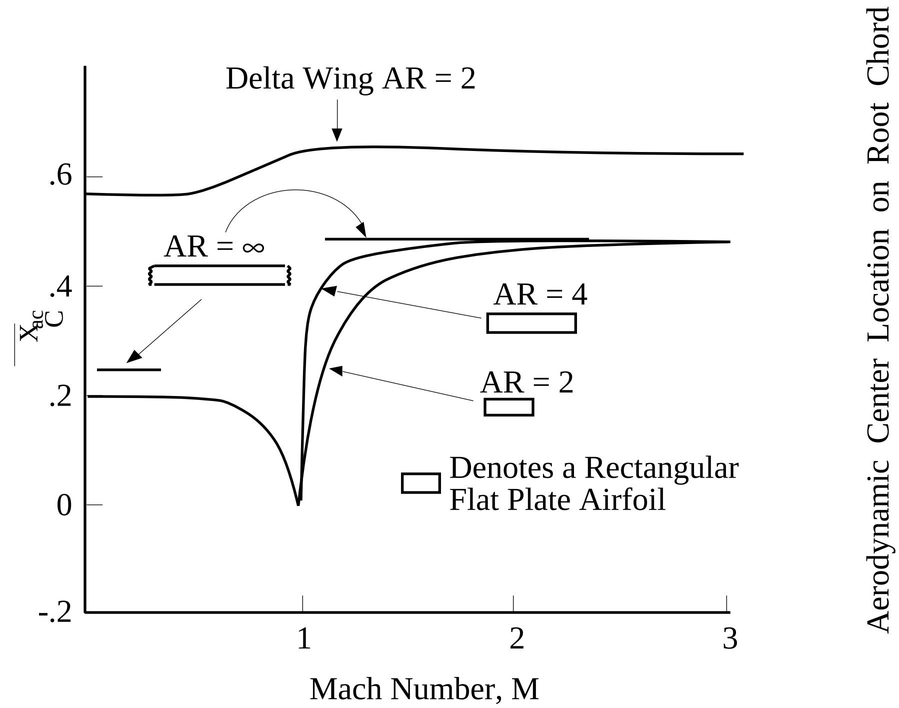


Figure 7.2
Aerodynamic Center Variation with Mach Number

The aft shift of the aerodynamic center results in an increase in longitudinal stability at a constant Mach number or an increase in angle of attack stability. (The aft shift of the aerodynamic center is analogous to moving the airplane center-of-gravity forward.) This change in stability can contribute to several characteristics which influence the flying qualities associated with transonic and supersonic flight regimes. The most well-known of these characteristics is the transonic longitudinal trim change. The transonic longitudinal trim change is manifested by increasing longitudinal pull forces and increasing trailing edge up elevator deflection required to stabilize at higher transonic Mach numbers with a constant longitudinal trim setting. This trim change also contributes to the common increase in normal acceleration for a fixed longitudinal control pull force and elevator deflection while decelerating through the transonic regime at a constant longitudinal trim setting, generally know as transonic pitch-up.

In the pure supersonic flight regime, the primary effect of the aerodynamic center shift is simply the increase in longitudinal stability. This can contribute to high longitudinal maneuvering control forces and/or the inability to obtain maximum useable load factors (design limit normal acceleration) in supersonic flight.

7.2.2.2 REDUCTION IN LIFT AND CHANGE IN DOWNWASH

During accelerations through the transonic regime, a reduction of lift from the wing occurs because of the shock wave formation and subsequent airflow separation (Figure 7.3). Since downwash behind the wing is a direct function of wing lift, a reduction in downwash also occurs while accelerating through the transonic regime. The reduction in downwash generates an increased angle of attack at the horizontal tail, which requires the pilot to apply additional trailing edge up elevator deflection to maintain altitude. Therefore, this factor also contributes to the transonic longitudinal trim change.

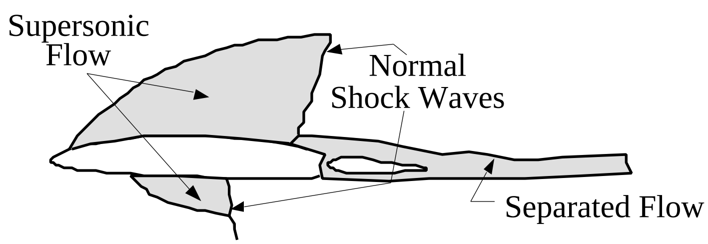


Figure 7.3
Trailing Edge Control Surface in Transonic Flow

7.2.2.3 CONTROL EFFECTIVENESS AND HINGE MOMENTS

The effectiveness or control power of the conventional trailing edge control surface (rudder, aileron, elevator) is particularly susceptible to transonic and supersonic influence. In the transonic flight regime, the trailing edge surface may be operating in a region of separated flow behind the normal shock wave, since the flow forward of the shock is supersonic. Thus, the control power, or the change in pitching, yawing, or rolling moment created per incremental change in control deflection, may be decreased.

The control power of the conventional trailing edge control surface decreases also at supersonic airspeeds. Because the flow over the wing, horizontal stabilizer, or vertical stabilizer is supersonic, the deflection of a trailing edge control surface generates little change in the aerodynamic loading over the wing or stabilizer (Figure 7.4). These phenomena have resulted in the use of the all-moving stabilizer, wing spoiler control surfaces, and irreversible power control systems on most modern supersonic airplanes. The all-moving stabilizer control power also decreases supersonically; however, the reduction is generally not as severe as for the trailing edge control.

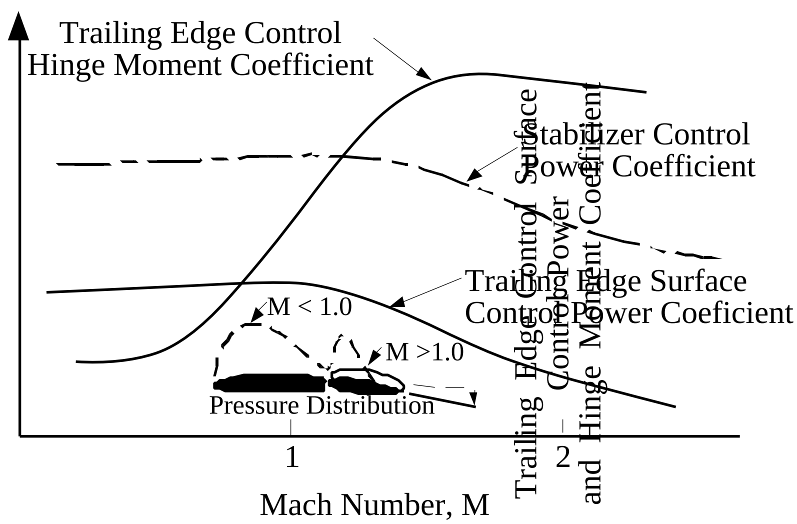


Figure 7.4
Typical Variations of Control Power and Hinge Moment Coefficients with Mach Number

7.2.2.4 CHANGES IN LIFT CURVE SLOPE AND STABILITY DERIVATIVES

Differences in subsonic and supersonic pressure distributions over wing, horizontal tail, and vertical tail surfaces lead to significant changes in the effectiveness of the surfaces. These changes are the result of the typical reduction in lift cure slope at supersonic airspeeds (Figure 7.5). Essentially, the variation in lift curve slope indicates that the aerodynamic surface becomes less sensitive to angle of attack or sideslip changes at high Mach numbers. The implications are several and significant:

1. The damping of the longitudinal short period and lateral-directional Dutch roll motions may be decreased.

2. Control power of the all-moving stabilizer may be reduced.

3. Directional stability, C_{n_β} may be reduced drastically.

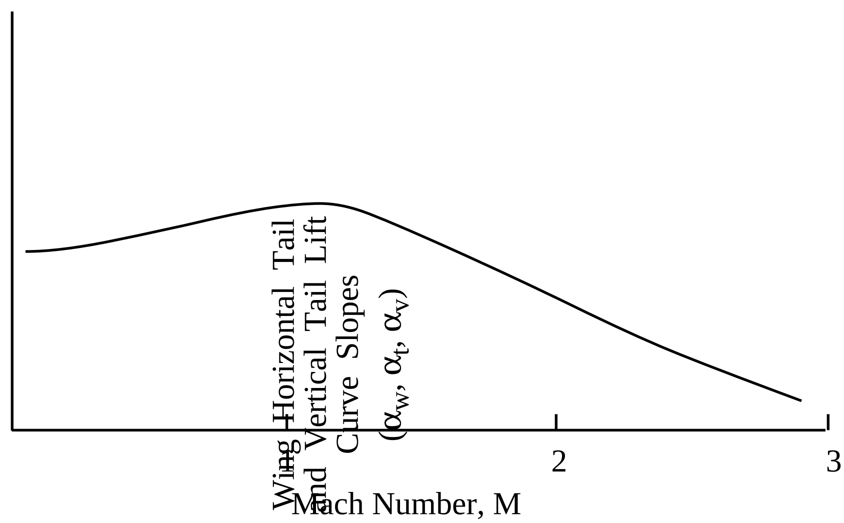


Figure 7.5
Lift Curve Slope Variation with Mach Number

The reduction of directional stability at high Mach numbers can be a particularly serious phenomenon. Because directional control power, $C_{n_{\delta_r}}$, generally decays with Mach number as does directional stability, C_{n_β}, the rudder position variation with sideslip angle, $\frac{d\delta r}{d\beta}$ is not a good indication of the decay in directional stability. The pilot may feel perfectly secure with the $\frac{d\delta r}{d\beta}$ gradient exhibited by the airplane, although directional stability may be dangerously low (Figure 7.6). The directional stability in supersonic flight decreased further with increasing lift coefficient, angle of attack, or normal acceleration (Figure 7.7). This is caused by:

1. A reduction in dynamic pressure at the vertical tail, due to the turbulent wing and fuselage wake at high angles of attack.

2. Strong vortex shedding from sharp-nosed fuselages at high angles of attack.

The decrease in lift curve slope and associated changes in stability derivatives have contributed to the requirement for stability augmentation systems in most modern supersonic airplanes.

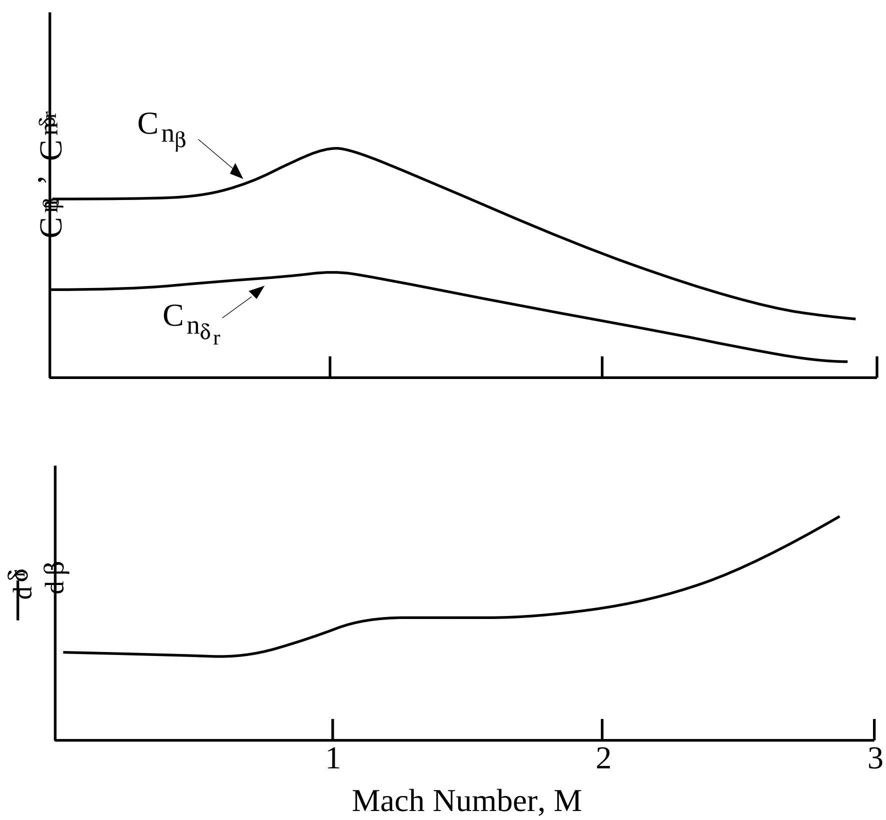


Figure 7.6
Possible Variation of Directional Stability and Control Power with Mach Number

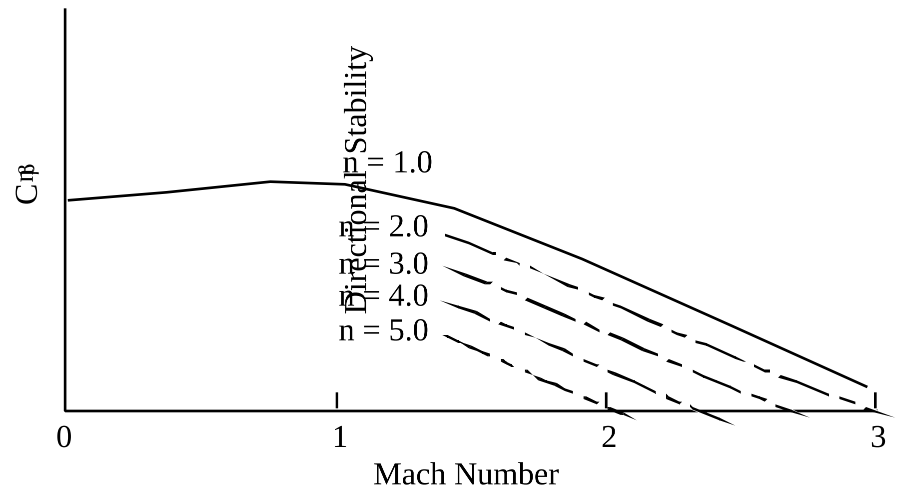


Figure 7.7
Typical Influence of Increasing Normal Acceleration on Directional Stability

7.2.2.5 MISCELLANEOUS COMPRESSIBILITY PHENOMENA

Miscellaneous phenomena which may be observed in the transonic flight regime are:

1. Buffeting of fuselage, wings, and empennage due to separation induced by shock wave formation.

2. Abrupt lateral-directional trim changes due to asymmetric shock wave formation on wings and aft fuselage.

3. Unusual roll response to rudder inputs. The airplane may roll opposite to the direction of rudder input because the forward moving wing will be subjected to a slightly higher Mach number and, possibly, more shock-induced separation. On a swept-wing airplane, the change of effective sweep angles in sideslipping flight contributes to negative dihedral effect in the transonic regime (Figure 7.8).

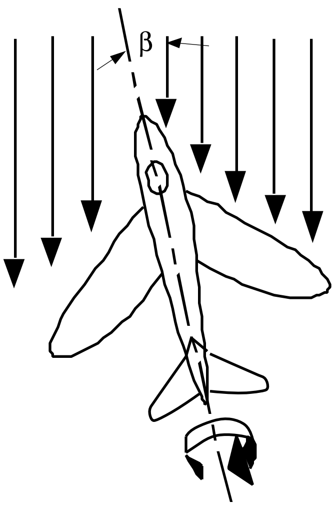


Figure 7.8
Possible Rolling Moment Generated by Sideslipping a Swept-Wing Airplane at Transonic Mach Numbers

4. Unusual roll response to small lateral control inputs. The airplane may roll opposite to the direction of lateral control input (for small lateral inputs). If the airplane is equipped with spoiler-type lateral controls, this effect may be attributed to boundary layer regeneration (vortex generator effect) for very small spoiler deflection. The energizing of the boundary layer can reduce the magnitude of shock-induced separation, actually increasing the lift on the wing.

5. Very high frequency control surface oscillations. These oscillations, generally called control surface "buzz," may be caused by control surface immersion in separated flow behind the shock wave or by shock wave formation and movement on the control surface. Prolonged, large magnitude, control surface buzzing could conceivably result in structural failure of the control surface.

7.2.3 Elastic Deformation of the Airplane Structure

Elastic deformation of the airplane structure, or aero-elastic effects, are likely to have some influence on transonic and supersonic flying qualities. The flexible structure of the airplane will bend or deform in response to applied moments so as to tend to reduce the moments or "unload itself." Thus, aero-elastic effects may be manifested by a decrease in stabilizing influence of the horizontal and vertical tail. This results in a reduction in damping of all perturbed motions, as well as a reduction in static stability. An example of how elastic deformation can reduce static directional stability is shown in Figure 7.9.

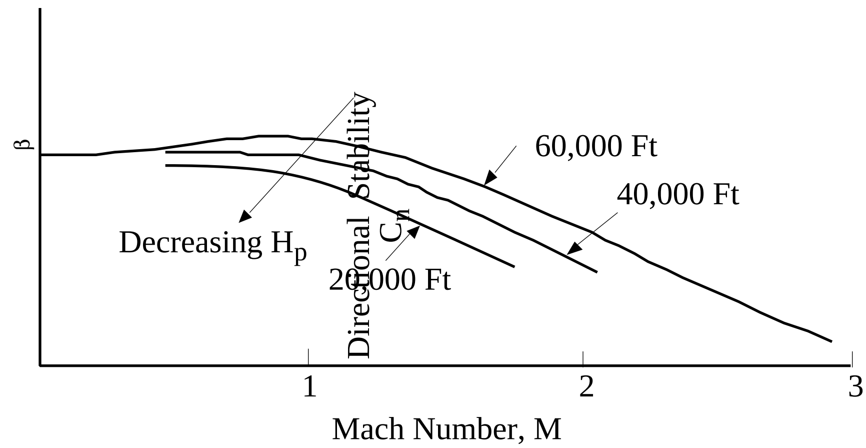


Figure 7.9
Possible Influence of Elastic Deformation Directional Stability

The most significant influence of aero-elasticity may be in the area of rolling performance. Wing torsional deflections which occur with aileron usage may be considerable. The result may be a drastic reduction in lateral control effectiveness at high Mach numbers. This subject is more thoroughly discussed in the Rolling Performance Theory.

It should be remembered that aero-elastic effects occur as a function of dynamic pressure for a given Mach number. Therefore, high altitude flight at a given Mach number will be relatively free of aero-elastic effects when compared to low altitude flight.

7.2.4 Airplane Conformation and Mass Distribution

Peculiarities of the aerodynamic form and mass distribution of the supersonic shape can generate some unusual stability and control characteristics. In particular, longitudinal flying qualities may be significantly influenced in many flight regimes by the location of the horizontal tail. The low-mounted horizontal tail generally generates more supersonic drag and operates in a region of disturbed flow from the wing and fuselage at angles-of-attack below stall. Thus, its effectiveness may be reduced and a more abrupt transonic longitudinal trim change may be encountered. However, the high-mounted horizontal tail, which may yield better longitudinal flying qualities over much of the total flight envelope of the airplane, may precipitate severe "pitch-up" characteristics at high angles-of-attack. A thorough discussion of the influence of horizontal tail location on stall characteristics may be found in the Stalls Theory section.

The supersonic airplane is characterized by a concentration of mass within a relatively long, slender fuselage; this results in large inequities between yawing and pitching; moments of inertia, and the rolling moments of inertia. Thus, the influence of inertia on airplane motion is generally more pronounced. The supersonic airplane may be plagued by roll coupling tendencies which are compounded by reduced longitudinal and directional stability at high Mach numbers. Roll coupling is discussed more thoroughly in the Rolling Performance Theory section.

7.2.5 Reduction of Air Density at High Altitudes

The reduction of air density at high altitudes leads to a marked reduction of the damping moments generated by the airplane in opposition to perturbations. In comparison, inertial moments are not directly related to air density and therefore remain fairly constant with altitude increase. Thus, in supersonic, high altitude flight regimes, the airplane may exhibit lower damping of longitudinal and lateral-directional oscillations as well as lower roll damping. In order to counteract these adverse tendencies, supersonic airplanes are normally equipped with stability augmentation systems.

7.3 TEST PROCEDURES AND TECHNIQUES TRANSONIC-SUPERSONIC FLYING QUALITIES

7.3.1 Preflight Procedures

Transonic and supersonic investigations must begin with extensive preflight preparation. The purpose and scope of the investigation must be clearly defined, then a plan of attack or method of test can be formulated.

Preflight planning must start with research. The step-by-step process by which Mach number and maneuvering envelopes of high performance airplanes are enlarged during prototype flight testing is a joint effort on the part of analytical and flight test engineers. Abrupt, nonlinear effects often encountered may preclude increasing Mach number or maneuvers by simple interpolation and extrapolation based on previously attained conditions. Predictions of flying qualities should be based on wind-tunnel and theoretical data corrected as necessary by the results of inflight tests. For experimental or prototype flight testing programs, large scale analog or digital computing equipment is probably essential to allow the necessary speed and flexibility of computation. For flight test programs within established flight envelopes which have been demonstrated structurally and aerodynamically safe, the sophisticated computing equipment may not be required. However, preflight research should be no less thorough. The design of the airplane should be studied in relation to its influence on stability and control characteristics. The flight control system, including stability and control augmentation, should be closely scrutinized. All available flight test, wind tunnel, and theoretical data should be reviewed. Much useful information may be gained from conferences with pilots and engineers familiar with the airplane.

The particular tasks to be investigated must be determined and clearly understood by the flight test team. These tasks, of course, depend on the mission of the airplane. The transonic region may be of relatively minor concern for airplanes designed to operate exclusively at supersonic speeds; accelerations and decelerations through the transonic regime would obviously require careful study, however. For airplanes designed to operate for prolonged periods in the transonic regime and for highly maneuverable airplanes capable of tactical transonic operations, the transonic flying qualities should be of major concern. The tactical feasibility of maneuvering in the transonic region in a close air-to-air engagement environment should be established, if applicable. Satisfactory transonic flying qualities for vigorous maneuvering tasks can provide a tactical advantage for defensive or offensive purposes; the advantage may revert to the enemy combatant airplane if unacceptable transonic characteristics exist.

The particular mission tasks to be investigated dictate the test conditions - configurations, centers-of-gravity, altitudes, gross weights, and trim airspeeds. Test conditions should be commensurate as much as possible with the mission environment of the airplane. However, the nature of transonic and supersonic stability and control characteristics requires that the most critical conditions be approached with due caution. Forward center-of-gravity positions should be used for initial tests unless adverse longitudinal control problems for recovery from high speed dives, etc., are predicted. The maximum practical altitude should be used for initial tests, to permit the attainment of high Mach numbers at the lowest dynamic pressure. The amount and sophistication of instrumentation will depend on the purpose and scope of the evaluation. Automatic recording devices - oscillograph, magnetic tape, and telemetry - are essential in a long test program of quantitative nature. Telemetering appropriate parameters to a ground station is almost mandatory for tests on prototype or experimental models. Qualified engineers, with communications to the test pilot, should continually monitor the flight test records.

Consideration must also be given to the airspace to be utilized for transonic and supersonic tests. Only designated areas may be used and the location and size of the area can have a large influence on the flight test approach utilized.

The final step in preflight planning is the preparation of pilot data cards. Example data cards presented previously for each area of investigation may be utilized, or data cards may be constructed to meet the requirements of the individual test program. The data cards should list all quantitative information desired, should be easy to interpret in flight, and should allow adequate space for appropriate qualitative pilot comments.

7.3.2 Flight Test Techniques

7.3.2.1 GENERAL

The same test techniques described earlier in the manual may be employed in the transonic and supersonic environment. However, the approach to this testing environment will be somewhat modified because of various peculiarities of transonic and supersonic flight.

Classically, the test pilot should desire the same satisfactory flying qualities at transonic or supersonic speeds as are expected in the subsonic environment. Of course, various characteristics not normally encountered at subsonic speeds may be anticipated at transonic and supersonic speeds. However, these peculiarities should not degrade flying qualities to a degree inconsistent with satisfactory mission accomplishment.

Due regard must be given to the following considerations in determining the acceptability of various transonic and supersonic characteristics:

1. Whether the mission tasks will be performed in VFR and IFR conditions or strictly VFR weather.

2. The availability of an autopilot or automatic flight control system for pilot relief.

3. If stability and control augmentation are installed, the consequences of their failure.

The pace of the transonic and supersonic investigations depends upon the degree of familiarity with the flight environment of interest. The pace if initial tests must be very slow and methodical. As familiarity increases, the pace may be quickened.

7.3.2.2 PRELIMINARY DECELERATION TEST

The test pilot must ascertain the ability to decelerate from transonic and supersonic airspeeds prior to penetrating into these regions. Therefore, a deceleration check should be performed at a Mach number just below the Mach number at which compressibility effects are noted. The effectiveness of all means of deceleration-speed brake extension, power reduction, etc., should be determined. Both individual and simultaneous actuation of deceleration means or devices should be performed. Large trim changes associated with actuation of deceleration devices may generate adverse flying qualities during transonic and supersonic decelerations; therefore, these trim changes should be noted during this preliminary test.

7.3.2.3 TYPICAL TRANSONIC CHARACTERISTICS

Stability and control characteristics which may be encountered exclusively in the transonic flight regime include abrupt changes in longitudinal, lateral, and directional trim as well as high frequency control surface "buzz". The abrupt change of longitudinal trim in a narrow range of transonic airspeed is manifested as the "transonic longitudinal trim change" (Figure 7.10).

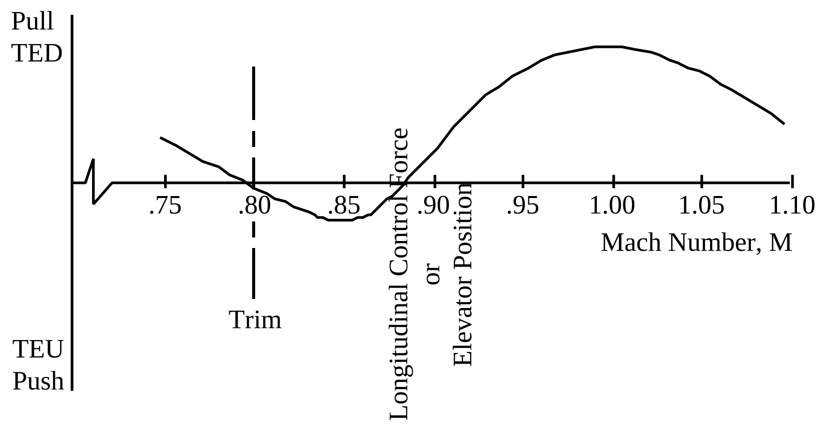
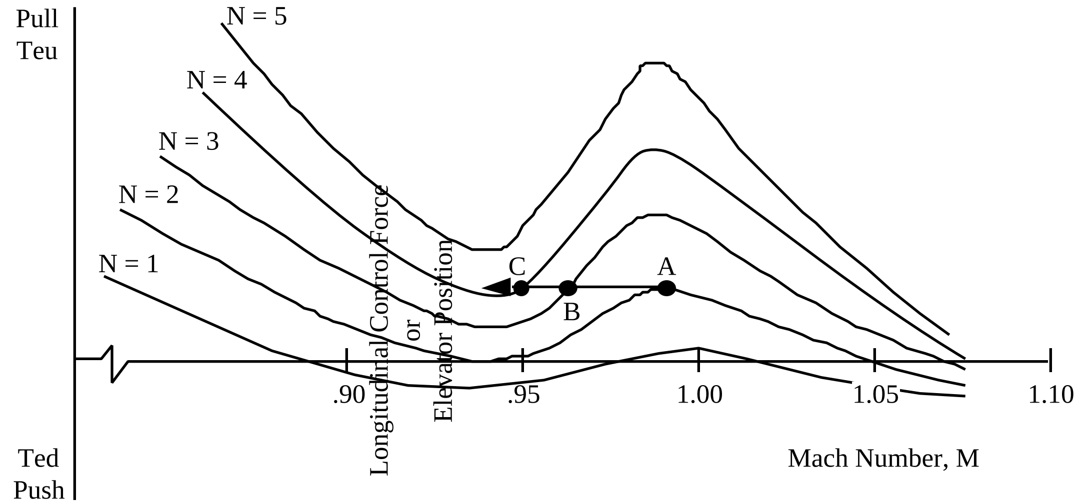


Figure 7.10
Typical Transonic Longitudinal Trim Change

The severity of the longitudinal trim change increases with increasing angle of attack, lift coefficient, or normal acceleration. This can create a sudden increase in normal acceleration for a constant longitudinal control force and elevator position while decelerating through the transonic region (Figure 7.11).

Changes in airplane lateral trim transonically may be manifested in small abrupt roll excursions not generated by lateral control inputs. These excursions may be triggered by directional trim changes which result in small sideslip excursions. Roll response to sideslip angles may indicate negative dihedral effect; this phenomenon may be particularly noticeable on swept-wing designs.

High frequency control surface oscillations or control surface "buzz" are sometimes encountered in the transonic regime.



Considering the static curves if the pilot decelerated from point "A" (2g) with a fixed control force and position, the normal acceleration would tend to increase to 3g at point "B" and 4g at point "C".

Figure 7.11
Abrupt Increases in Normal Acceleration may be Encountered During Transonic Deceleration

7.3.2.3 TRANSONIC INVESTIGATIONS (INITIAL)

If little or nothing is known about the transonic behavior of the airplane, the pace of the transonic investigation will be <u>very slow</u>. After performing the preliminary deceleration test, the transonic region may be penetrated as follows:

1. At the highest practical altitude, establish the power or combat configuration with military or maximum power at a Mach number just below transonic influences. Full power is used so that altitude loss during the test may be minimized and maximum initial deceleration may be realized with power reduction.

2. Increase Mach number in steps of approximately 0.01 IMN until transonic characteristics are first detected.

3. After encountering the initial transonic characteristics, increase Mach number in steps of approximately 0.005 IMN. Longitudinal trim settings should not be altered after penetrating the transonic region unless trimming is required to reach higher Mach numbers. ("Trimming into" the transonic region may result in excessive normal acceleration response during deceleration or a lack of longitudinal response when initiating deceleration.)

4. Stabilize at each incremental Mach number and note appropriate airplane characteristics. The automatic recording devices are effectively utilized to quickly gather the airplane response or characteristics in the following manner:

    a. Steady, straight stabilized flight.

    b. Steady turns at small increments of normal acceleration.

    c. Small longitudinal doublet or pulse inputs.

    d. Steady sideslips at small increments in sideslip angles.

    e. Small directional pulls inputs.

    f. Small lateral control deflections.

5. As familiarity with the transonic characteristics of the airplane is gained, the maneuvers listed above may be methodically increased in magnitude.

6. At predetermined Mach number increments, the test pilot should check deceleration characteristics. Slow decelerations using only slight power reduction should be used initially; as familiarity is gained, faster declarations may be evaluated with simultaneous actuation of all deceleration devices.

7. If control is lost or becomes marginal during the transonic investigation or if unexpected characteristics are encountered, the test pilot should decelerate to a subsonic condition and analyze the situation.

7.3.2.5 TRANSONIC INVESTIGATIONS (PROVEN AIRPLANE)

After initial investigations of transonic behavior or for transonic investigations on proven airplanes, a faster pace and more vigorous approach may be utilized. After performing the preliminary deceleration test, the following procedures may be employed:

1. Perform an initial acceleration through the transonic region to obtain an overall picture of the characteristics exhibited. If longitudinal trim is maintained constant at the setting for the subsonic starting point, the transonic longitudinal trim change data may be obtained. The acceleration may be performed by diving with full power or by a level acceleration with full power for airplanes with the performance capability. If automatic recording devices are available, the entire acceleration should be recorded for later analysis.

2. Perform a deceleration through the transonic region using only power reduction. Note the Mach number at which an increase in normal acceleration may be expected for a constant longitudinal control force and elevator position. Record the deceleration with the automatic recording devices, if appropriate.

3. Begin a detailed assessment by stabilizing at predetermined Mach numbers in the transonic region (0.02 to 0.05 IMN increments). At each stabilized trim point, briefly investigate the following characteristics:

    a. Longitudinal, lateral, and directional trimmability.

    b. Longitudinal short period characteristics (small doublet or pulse inputs). Note any tendencies toward pilot-induced oscillations.

    c. Longitudinal maneuvering stability in steady turning flight. Use approximately the same normal acceleration increment at each test point (for example, 1-3 g), so that the variation of longitudinal control force per g with Mach number may be easily noted and presented.

d. Steady heading sideslips (small).

e. Lateral-directional oscillations (small rudder pulses).

f. Rolling performance with small lateral control inputs and small bank angle changes.

If automatic recording devices are available, these maneuvers are easily and quickly recorded for accurate quantitative information.

4. If desired, the maneuvers listed above may be gradually increased in magnitude.

5. Evaluate deceleration characteristics through the transonic region with various combinations of deceleration means - speed brake extension, power reduction, etc. In addition, the flying qualities exhibited during decelerations at increasing values of normal acceleration should be carefully determined. Caution should be exercised during this phase of the test in order to avoid exceeding structural limit normal acceleration.

6. For highly maneuverable airplanes capable of tactical transonic operation, a qualitative investigation should be performed to determine the feasibility of utilizing the transonic region during typical mission tasks.

7. If control is lost or becomes marginal during the transonic investigation, or if unexpected characteristics are encountered, the test pilot should decelerate to a subsonic airspeed and analyze the situation.

7.3.2.6 TYPICAL SUPERSONIC CHARACTERISTICS

At supersonic flight speeds, the following general peculiarities in airplane stability and control characteristics may be encountered:

1. Increased longitudinal, lateral, and directional control deflections required in all maneuvering, particularly if control surfaces are trailing edge type.

2. Increased longitudinal maneuvering control force gradient; reduction in available normal accelerations with full trailing edge up longitudinal control surface deflection.

3. Deterioration of transient behavior, i.e., reduced damping of longitudinal and lateral-directional motions; particularly noticeable at high altitude for airplanes without stability augmentation.

4. Change in magnitude of the rolling motion in the Dutch roll oscillation (roll-to-yaw ratio); particularly noticeable in airplanes with no stability augmentation.

5. Weak static directional stability, particularly at high normal acceleration values.

6. Reduction of rolling performance.

7. Tendencies toward roll coupling.

7.3.2.7 SUPERSONIC INVESTIGATIONS

Supersonic investigations may be performed according to the same procedures as presented for transonic investigations. For initial investigations, the pace must be very slow and methodical. The operating envelope should be expanded with a step-by-step process. Computer programs, systematically updated with actual flight test data, should accompany the test program.

After initial investigations, or for supersonic investigations on proven airplanes, a faster pace and more vigorous approach may be utilized. An initial acceleration to a Mach number near the maximum operational Mach number should provide valuable overall impressions which may be helpful in dividing the remainder of the test into particular phases. Test techniques presented earlier in the manual are still applicable to the supersonic flight regime; however, an adequate build-up program must be utilized since exceeding limits of controllability at supersonic speeds could precipitate catastrophic results.

7.3.3 Postflight Procedures

As soon as possible after returning from the flight, the pilot should write a brief, rough qualitative report of the transonic and/or supersonic flying qualities. This report should be written while the events of the flight are fresh in the pilot's mind. Qualitative pilot opinion, appropriately related to the mission tasks under evaluation, will be the most important part of the final report.

Appropriate data should be selected to substantiate the pilot's opinion. Data presentation methods introduced earlier may be utilized for various characteristics of interest. The transonic longitudinal control force gradients will probably be one of the most relevant transonic characteristics to be illustrated (Figure 7.12).

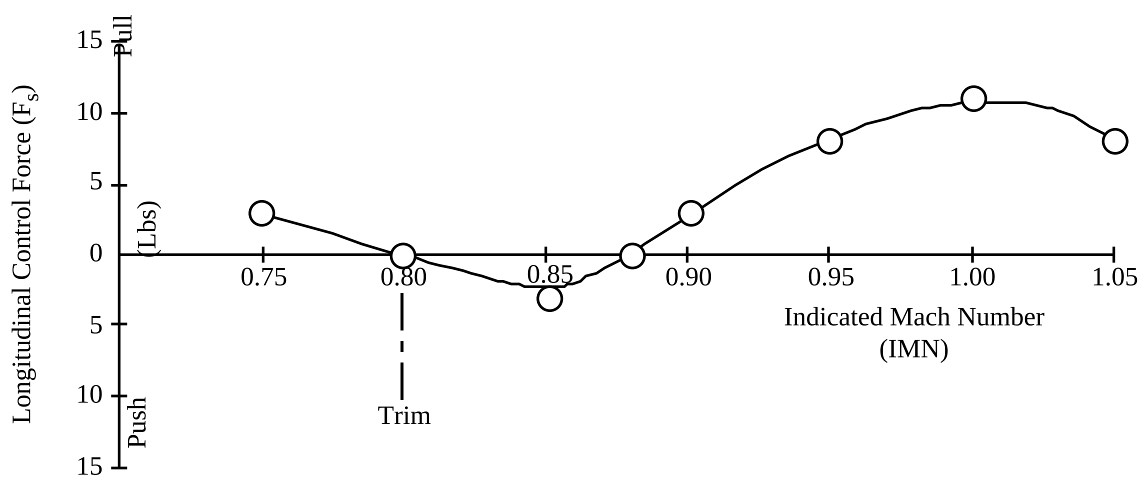


Figure 7.12
Transonic Characteristics

Model _________ Airplane

BuNo ______________

Pilot: ______________________	Gross Weight: 16,700 Lbs
Configuration: Power	CG: 23.0 % MAC
Loading: Alpha	Altitude: 40,000 - 30,000 Ft
Date: 18 April 1966	Stab Aug: On

7.4 SPECIFICATION REQUIREMENTS

Requirements for transonic and supersonic flying qualities are contained in applicable paragraphs of Military Specification, MIL-F-8785C, of 5 November 1980, hereafter referred to as the Specification. Paragraph 3.1.7, Operational Flight Envelopes, should be used as a guide in determining the flight envelope within which the Specification is considered applicable.

Relevant exceptions for longitudinal flying qualities in the transonic flight regime are stated in Specification paragraph 3.2.1.1.1, Relaxation in Transonic Flight, and paragraph 3.2.1.1.2, Pitch Control Force Variations During Rapid Speed Changes.

7.5 GLOSSARY

Critical Mach Number The free-stream Mach number at which a local Mach number of 1.0 is attained at any point on the body under consideration.

Transonic Speed Flow in which regions of both subsonic and supersonic velocities are present.

Supersonic Of, pertaining to, or dealing with, speeds greater than the speed of sound.

Compressibility The property of a substance, such as air, by virtue of which its density increases with increase in pressure.

Compressibility Effects Phenomena encountered by airplanes operating at high Mach numbers which are attributable to air compressibility.

Aero-Elastic Effects Effect of aerodynamic forces acting upon an elastic body, such as an airplane.

7.6 REFERENCES

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control,* by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Aerodynamics of Supersonic Flight*, by Alan Pope, Head, Aerodynamics Department, Sandia Corporation, Publisher: Pitman Publishing Corporation, New York, Toronto, London.

3. *Airplane Performance, Stability and Control,* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

4. *Dynamics of Flight, Stability and Control*, by Bernard Etkin, Professor of Aeronautical Engineering, University of Toronto, Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

5. *Flight Testing of Aircraft*, by M. G. Kotik, et al., NASA Technical Translation F-422, April, 1967.

6. *Handling and Performance Exercises Test Techniques,* Empire Test Pilot's School, Flying Department Notes.

7. *High Speed Stability and Control Problems as They Affect Flight Testing*, by Malcolm J. Agzug, Advisory Group For Aeronautical Research and Development, Report 120, May, 1957.

8. *Military Specification, Flying Qualities of Piloted Airplanes MIL-F-8785C,* of 5 November 1980.

9. *Modern Flight Dynamics,* by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

10. *National Aeronautics and Space Administration Aeronautical Dictionary,* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

11. *Stability and Control,* Empire Test Pilot's School Ground School Notes, Book "C."

12. *Stability and Control,* USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

13. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

APPENDIX I

GLOSSARY

APPENDIX I

GLOSSARY

Adverse Yaw	Yawing moments created act so as to rotate the nose of the airplane opposite to the direction of roll. The term "adverse" does not, in itself, denote unfavorable flying qualities.
Aerodynamic Balancing	Methods of controlling the magnitude of the hinge moment parameters.
Aero-Elastic Effects	Effect of aerodynamic forces acting upon an elastic body, such as an airplane.
Aileron Reversal Airspeed, V_r	Airspeed at which the combined effects of wing twist and wing bending counteract the rolling moment generated by lateral control deflection.
Aperiodic; Deadbeat	A motion which does not exhibit periodic oscillations.
Aspect Ratio	The ratio of the span of the wing to the mean chord.
Autorotation	Uncontrolled rolling or rotating, as in a spin.
Average Stick Force Per G Gradient	Slope of a line drawn from the 1g point where breakout, including friction is overcome to the point under consideration on the curve of longitudinal control force versus normal acceleration.
Bank Angle Change In A Given Time, ϕ_t	The bank angle attained in a predetermined time interval following a step input of lateral control; time is measure from the initiation of the pilot's lateral control <u>force</u> application.

Camber The curvature of the mean line of an airfoil section from leading edge to trailing edge.

Compressibility The property of a substance, such as air, by virtue of which its density increases with increase in pressure.

Compressibility Effects Phenomena encountered by airplanes operating at high Mach numbers which are attributable to air compressibility.

Critical Mach Number The free-stream Mach number at which a local Mach number of 1.0 is attained at any point on the body under consideration.

Damping Progressive diminishing in amplitude. A measure of the subsidence of the motion when excited.

Damping Ratio Ratio of the damping exhibited to the critical damping.

Deep Stall A flight condition in which the airplane has attained an angle of attack far higher than the angle of maximum lift coefficient.

Elastic Center A point in the wing section about which torsional deflections occurs.

Endplate A plate or surface at the end of an airfoil attached in a plane normal to the airfoil that inhibits the formation of tip vortex, thus producing an effect similar to that of increased aspect ratio.

Float As applied to the control surface of a reversible control system: to ride in the airstream.

Frequency	Number of cycles per unit time. A measure of the “quickness” of the motion.
Incidence	The acute angle between a chord of an airfoil and the longitudinal axis of the airplane.
Local Stick Force per g Gradient	Slope of the tangent to the curve of longitudinal control force versus normal acceleration at any point.
Longitudinal Control Power	A measure of the pitching moment coefficient change per degree deflection of the longitudinal control surface.
Maneuvering Tasks	Those tasks which result in accelerated flight conditions; during these tasks, transitions for one equilibrium flight condition to another are made quickly, and possibly, somewhat roughly.
Mode of Motion	Manner of doing, method. In this case, a method of changing flight conditions in the airplane's plane of symmetry.
Neutral Point	The location of the center of gravity of an aircraft for which static longitudinal stability would be neutral. The neutral point may be described as “stick-fixed,” “stick-free,” “elevator-fixed,” “elevator-free,” “elevator position,” or “longitudinal control force” depending on the manner in which it was determined.
Nonmaneuvering Tasks	Those tasks during which the transition from one equilibrium flight condition to another is accomplished smoothly and gradually; results in essentially unaccelerated flight conditions.
Oscillatory	Characterized by periodic motion.

Period	Time required per cycle. Inversely proportional to frequency.
Pilot-Induced-Oscillation (PIO)	A divergent oscillation of the pilot - airplane combination where the airplane alone exhibits at least some degree of dynamic stability.
Pitch Rate Damping	Pitching moment created because of the angular rotation of the airplane in pitch during curvilinear flight. Sometime called "damping in pitch" or "viscous damping in pitch."
Post-Stall Gyrations	Random oscillations of the airplane about all axes following departure from controlled flight.
Proverse Yaw	Yawing moments generated act so as to rotate the nose of the airplane toward the direction of roll. The term "proverse" does not necessarily indicate favorable flying qualities.
Reynolds Number	A nondimensional parameter representing the ratio of the momentum forces to the viscous forces about a body in motion. Reynolds number decreases with increase in altitude and increases with increase in true velocity, if the dimensions of the body remain constant.
Roll Coupling	Pitching and yawing motions induced by inertial and kinematic effects during high rate rolls.
Roll Helix Angle, $\frac{pb}{2V_T}$	Helix angle described by the wingtip of a rolling airplane; sometimes referred to as the non-dimensional roll rate.
Roll Mode Time Constant	Time required for the roll rate to reach 63.2 percent of the steady state roll rate following a step input of lateral control.

Roll Mode Time Constant, τ_R	Time required for the single degree of freedom roll rate to reach 63.2% of the steady state roll rate following a step lateral control input.
Roll-To-Yaw Ratio	Ratio of bank angle envelope to sideslip angle envelope during Dutch roll oscillation.
Shock Stall	A stall brought on by compressibility burble; i.e., by separation aft of a shock wave.
Single Degree of Freedom Roll	Rolling motion during which the airplane is allowed to roll but not allowed to yaw or pitch; pure roll response.
Slat	Any of certain long narrow vanes or auxiliary airfoils. The vane used in an automatic slot.
Slot	A long and narrow opening, as between a wing and a deflected Fowler flap. A long and narrow spanwise passage in a wing, usually near the leading edge, for improvement of airflow conditions at high angles of attack.
Spring Constant	As applied to a dynamic system, a measure of the static restoring tendency.
Static Margin	The distance between the actual center of gravity and the neutral point of the airplane usually expressed as a percentage of the mean aerodynamic chord.
Steady State Roll Rate, p_{ss}	Roll rate attained when the roll damping contribution equals the roll control power contribution for a constant lateral control input.

Stick-Fixed Maneuvering Neutral Point The location of the center of gravity of an aircraft for which the gradient of elevator position versus normal acceleration at constant airspeed would be zero. Sometimes called the "elevator position maneuvering neutral point."

Stick-Free Maneuvering Neutral Point The location of the center of gravity of an aircraft for which the gradient of longitudinal; control force versus normal acceleration at a constant airspeed would be zero. Sometime called the "longitudinal control force maneuvering neutral point."

Supersonic Of, pertaining to, or dealing with, speeds greater than the speed of sound.

Tail Efficiency Factor A measure of the modification in energy level of the airflow between the point where the airflow first encounters the airplane until it reaches the horizontal tail.

Tail Volume Coefficient A measure of the size and location of the horizontal tail in relation to the size of the wing and the airplane center of gravity, respectively.

Taper A gradual reduction in chord length from wing root to wingtip.

Thickness Ratio The ratio of the maximum thickness of an airfoil section to its chord length.

Transonic Speed Flow in which regions of both subsonic and supersonic velocities are present.

Undamped Natural Frequency The frequency of a dynamic system if zero damping is exhibited.

Wingtip Vertical Velocity, $\frac{pb}{2}$ — Vertical velocity of the wingtip of a rolling airplane; sometimes used as a measure of the rolling performance of large airplanes in the approach and landing phases of mission accomplishment.

APPENDIX II

REFERENCES

APPENDIX II

REFERENCES

CHAPTER 2

1. *Advisory Group for Aeronautical Research and Development*, Conference Proceedings No. 17, Stability and Control, September 1966.

2. *Advisory Group for Aeronautical Research and Development*, Flight Test Manual, Volume II, Stability and Control, by Courtland D. Perfins, Professor, Aeronautical Engineering Department, Princeton University.

3. *Flight Testing of Aircraft*, by M.G. Kotick, et al., NASA Technical Translation F-442, April 1967.

4. *Handling and Performance Exercises Test Techniques*, Empire Test Pilots' School, Flying Department Notes.

5. *Military Specification, Flying Qualities of Piloted Airplanes*, MIL-F-8785C, of 5 November 1980.

6. *National Aeronautics and Space Administration Aeronautical Dictionary*, by Frank D. Adams, United States Government Printing Office, Washington, 1959.

7. *A Piloted Simulation Study of Operational Aspects of the Stall Pitch-Up*, by Maurice D. White and George E. Cooper, Ames Research Center, Moffett Field, California, NASA Technical Note D-4071, July, 1967.

8. *Principles of Aerodynamics*, by James H. Dwinnell, Boeing Airplane Company, Seattle, Washington. Publisher: McGraw-Hill Book Company, Inc., New York, New York, 1949.

9. *A Review of the Stall Characteristics of Swept Wings*, by Charles W. Harper and Ralph L. Maki, Ames Research Center, Moffett Field, California, NASA Technical Note D-2373, July 1964.

10. *Stability and Control*, Empire Test Pilots' School Ground School Notes, Book “C”.

11. *Stability and Control*, USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

12. *Supersonic and Subsonic Airplane Design*, by Gerald Corning, Professor, Aeronautical Engineering Department, University of Maryland. Publisher: Edwards Brothers, inc., Ann Arbor, Michigan, 1953.

13. *“Zero Rate of Climb Speed” as a Low Speed Limitation for the Stall-Free Aircraft*, by W.J.G. Pinsker, British Ministry of Technology Current Paper No. 931, May 1966.

CHAPTER 3

1. *AGARD Flight Test Manual*, Volume II, Chapter 8, Parts I and II.

2. MIL-S-25015(ASG), *Military Specification; Spinning Requirements for Airplanes.*

3. NACA TN 1045, *Tail-Design Requirements for Satisfactory Spin Recovery*, of April 1946.

4. NASA Langley Research Center Paper *“Airplane Spinning”* by James S. Bowman.

5. NASA TR R-57, 1960, *Status of Spin Research for Recent Airplane Designs.*

6. NASA TT F 422, *Flight Testing of Aircraft*, Chapter 9.

7. *"Spin Testing United States Navy High Performance Airplanes"* by John A. Nial, NAWCAD Patuxent River, Maryland.

8. *U. S. Air Force Flight Test Center Stability and Control Techniques*, Section 4.

CHAPTER 4 - NON-MANUEVERING TASKS

1. *Advisory Group for Aeronautical Research and Development,* Flight Test Manual, Volume II, Stability and Control, by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control*, by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training,* by F.D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F-1, September, 1962.

4. *Dynamics of Flight, Stability, and Control,* by Bernard Etkin, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

5. *Flight Evaluations of the Effect of Variable Phugoid Damping in a KTB-26B Airplane,* by Fred Newell and Donald W. Rhoads, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, WADC TR 56-223, Report No. TB-969-F-1, October 1956.

6. *Flight Evaluation of Various Phugoid Dynamics and* $1/T_{h_1}$ *Values for the Landing Approach Tasks;* by Charles R. Chalk, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. TC-1921-F-4, February, 1966.

7. *Flight Testing of Aircraft,* by M.G. Kotik, et al., NASA Technical Translation F-442, April, 1967.

8. *Handling and Performance Exercises Test Techniques,* Empire Test Pilots' School, Flying Department Notes.

9. *Investigation of Phugoid Effects on GCA Landings,* by Fred Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, WADC TR 57-650, Report No. TB-1142-F-1, December, 1957.

10. *Military Specification, Flying Qualities of Piloted Airplanes,* MIL-F-8785B(ASG), 7 August 1969.

11. *Minimum Flyable Longitudinal Handling Qualities of Airplanes,* by Gifford Bull, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Report No. TB-1313-F-1, December, 1959.

12. *Modern Flight Dynamics,* by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

13. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

14. *Recommendations For Revision of MIL-F-8785(ASG),* “Military Specification - Flying Qualities of Piloted Airplanes,” by C. R. Chalk, T. P. Neal, T. M. Harris, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Technical Report AFFDL-TR-67, March, 1967.

15. *Stability and Control,* USAF Aerospace Research Pilot School, Edwards Air Force Base, California, FTC-TIH-64-2004, revised November, 1964.

16. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

17. *Stability and Control,* Empire Test Pilots' School Ground School Notes, Book “C”.

18. *Substantiation Information For Selected Longitudinal Requirements,* Cornell Aeronautical Laboratory Report No. BM-2238-F-1, May, 1967.

CHAPTER 4 - MANUEVERING TASKS

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control*, by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Analysis of Frequency and Damping Characteristics of Airplane Longitudinal Short Period Oscillations*, by William B. Rhodes, Flight Test Division, Naval Air Test Center, Patuxent River, Maryland, FT-TM-1-64, 28 December 1964.

4. *Calculation of the Response of a Fighter Aircraft to Turbulent Air and a Comparison with Flight Measurements,* by C.G.B. Mitchell, Royal Aircraft Establishment, Ministry of Aviation, Farnborough, Hants, RAE Technical Report 66275, August 1966.

5. *Computer Programmers to Calculate the Response of Flexible Aircraft to Gusts and Control Movements*, by C.G.B. Mitchell, Ministry of Technology, Aeronautical Research Council, London, Current Paper No. 957, October 1966.

6. *Considerations in the Determination of Stability and Control Derivatives and Dynamic Characteristics from Flight Data*, by Chester H. Wolowicz, NASA Flight Research Center, Edwards Air Force Base, California, AGARD Report 549 - Part 1, 1966.

7. *Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training*, by F. D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F-1, September, 1962.

8. *Dynamics of Flight, Stability and Control*, by Bernard Etkins, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

9. *Experience With Shinbrot's Method of Transient Response*, by B.R.A. Burns, British Aircraft Corporation, Preston Division, Warton Aerodrome, Preston, Lancashire, AGARD Report 549 - Part II, 1966.

10. *Flight Investigation of Longitudinal Short Period Frequency Requirements and PIO Tendencies*, by Dante A. Difranco, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-66-163, June 1967.

11. *Flight Test Evaluations of Airplane Dynamic Longitudinal Stability in the Low Altitude High Speed Region*, Mr. Thomas M. McPherson and Captain Kenneth W. Weir, USMC, Flight Test Division, Naval Air Test Center, Patuxent River, Maryland.

12. *Flight Testing of Aircraft*, by M. G. Kotik, et al., NASA Technical Translation F-442, April, 1967.

13. *Frequency Response Method of Determining Aircraft Longitudinal Short-Period Stability and Control System Characteristics in Flight*, by Henry A. Klung, Jr., Captain, USAF, Aerospace Engineer, AFFTC-TR-66-24, August 1966.

14. *Handling and Performance Exercises Test Techniques*, Empire Test Pilots' School, Flying Department Notes.

15. *Investigation of Low Level, High Speed Flight Problems*, by Richard C. Adams, Commander, USN, Flying Qualities and Performance Branch, Flight Test Division, Naval Air Test Center, Patuxent River, Maryland, Society of Experimental Test Pilots 8th Annual Symposium, 25-26 September 1964.

16. *Low Altitude, High Speed Handling and Riding Qualities*, by Ralph C. A'Harrah, AGARD Flight Mechanics Meeting, Turin, Italy, April 1963.

17. *Maneuverability and Gust Response Problems Associated with Low Altitude, High Speed Flight*, By R.C. A'Harrah, AGARD Consultant, AGARD Report 556, October 1967.

18. *Military Specifications, Flying Qualities of Piloted Airplanes*, MIL-F-8785C, of 5 November 1980.

19. *Minimum Flyable Longitudinal Handling Qualities of Airplanes*, by Gifford Bull, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Report No. TB-13130-F-1, December, 1959.

20. *Modern Flight Dynamics*, by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

21. *National Aeronautics and Space Administration Aeronautical Dictionary by Frank D. Adams*, United States Government Printing Office, Washington, 1959.

22. *Stability and Control*, Empire Test Pilots' School Ground School Notes, Book "C".

23. *Stability and Control*, USAF Aerospace Research Pilot School, Edwards Air Force Base, California, FTC-TIJ-64-2004, revised November, 1964.

24. *Stability and Control of Airplanes and Helicopters*, by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

25. *Study of Some Hand-Computing Techniques to Determine the Approximate Short Period Mode From Airplane Response*, by Benjamin H. Dolbin, Jr., Cornell Aeronautical Library, Inc., Buffalo, New York, CA FSDM-371, March 1966.

26. *Substantiation Information for Selected Longitudinal Requirements*, Cornell Aeronautical Laboratory Report No. BM-2238-F-1, May 1967.

CHAPTER 5 - LATERAL-DIRECTIONAL FLYING QUALITIES

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control*, by Courland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control*, by Courland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Analysis of Several Handling Quality Topic Pertinent to Advanced Manned Aircraft,* Systems Technology, Inc., AFFDL-TR-67-2, June, 1967.

4. *Calculation of the Response of a Fighter Aircraft to Turbulent Air and a Comparison with Flight Measurements*, by C. G. B. Mitchell, Royal Aircraft Establishment, Ministry of Aviation, Farnborough, Hants, RAE Technical Report 66275, August, 1966.

5. *Computer Programmes to Calculate the Response of Flexible Aircraft to Gusts and Control Movements*, by C. G. B. Mitchell, Ministry of Technology, Aeronautical Research Council, London, Current Paper No. 957, October, 1966.

REFERENCES

6. *Considerations in the Determination of Stability and Control Derivatives and Dynamic Characteristics From Flight Data*, by Chester H. Wolowicz, NASA Flight Research Center, Edwards Air Force Base, California, AGARD Report 549 - Part 1, 1966.

7. *Criteria for Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training,* by F. D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F-1, September, 1962.

8. *Development of Satisfactory Lateral-Directional Handling Qualities in the Landing Approach,* by Robert L. Stapleford, Donald E. Johnston, Gary L. Teper, and David H. Weir, Systems Technology, Inc., Hawthorne, California, NASA CR-239, July 1965.

9. *Dynamics of Flight, Stability, and Control,* by Bernard Etkin, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

10. *Evaluation of Lateral-Directional Handling Qualities of Piloted Reentry Vehicles Utilizing Fixed-Base and Inflight Evaluations*, by J.I. Meeker, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, NASA CR-778, May, 1967.

11. *Experience with Shinbrot's Method of Transient Response*, by B. R. A. Burns, British Aircraft Corporation, Preston Division, Warton Aerodrome, Preston, Lacashire, AGARD Report 549 - Part II, 1966.

12. *Features of Large Transport Aircraft Affecting Control During Approach and Landing,* by W. J. G. Pinsker, Royal Aircraft Establishment, Bedford, England.

13. *Flight Testing of Aircraft*, by M. G. Kotik, et al., NASA Technical Translation F-442, April, 1967.

14. *Flying Qualities Requirements for United States Navy and Air Force Aircraft,* by William Koven and Richard Wasicko, AGARD Specialist Meeting on Aircraft Stability and Control, Brussels, Belguim, April, 1961.

15. *Handling and Performance Exercises Test Techniques,* Empire Test Pilots' School, Flying Department Notes.

16. *Inflight Evaluation of Lateral-Directional Handling Qualities for the Fighter Mission,* by James I. Meeker and G. Warren Hall, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-98, October 1967.

17. *Inflight Simulation and Pilot Evaluation of Selected Landing Approach Handling Qualities of a Large Logistics Transport Airplane*, by Donald W. Phoads, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-51, July, 1967.

18. *Lateral-Directional Flying Qualities for Power Approach: Influence of Dutch Roll Frequency,* by Edward Seckel, James A. Franklin, and George E. Miller, Princeton University, Princeton, New Jersey, Princeton University Report No. 797, September, 1967.

19. *Lecture Notes on Lateral-Directional Stability,* by F. D. Newell and Michael Parrog, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. TB-1961-F-1, July 1965.

20. *Military Specification, Flying Qualities of Pilot Airplanes MIL-F-8785C,* of 5 November 1980.

21. *Modern Flight Dynamics,* by W. Richard Kokl, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

22. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

23. *Proposal for a Revised Military Specification, "Flying Qualities of Piloted Airplanes" (MIL-F-8785 ASG), With Substantiating Text,* by C. J. Mazza, William Becker, Marshall Cohen, and Alvin Spector, U. S. Naval Air Development Center, Johnsville, Pennsylvania, NADC-ED-6282, January, 1963.

24. *A Review of Recent Research on Handling Qualities, and its Application to the Handling Problems of Large Aircraft*, by P. L. Bisgood, Ministry of Aviation, Aeronautical Research Council, London, R & M. No. 3458, June 1964.

25. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

26. *Stability and Control, Empire Test Pilots' School Ground School Notes, Book "C".*

27. *Stability and Control,* USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

28. *Substantiation Information for Selected Lateral-Directional Requirements,* Cornell Aeronautical Laboratory, Report No. BM-2238-F-1, May, 1967.

CHAPTER 5 - ROLLING PERFORMANCE

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control,* by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability, and Control*, by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Analysis of Several Handling Quality Topics Pertinent to Advanced Manned Aircraft,* Systems Technology, Inc., AFFDL-TR-67-2, June, 1967.

4. *Criteria For Acceptable Representation of Airplane Dynamic Responses in Simulators Used for Pilot Training,* by F. D. Newell, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. BM-1642-F1, September, 1962.

5. *Determination of Lateral Control Requirements at Low Airspeeds,* Report No. 2, Final Report, by Lcdr. W. Spangenberg, USN and Mr. G. A. Patterson, U.S. Naval Air Test Center, Patuxent River, Maryland, Report No. FT212-222, 19 July 1963.

6. *Development of Satisfactory Lateral-Directional Handling Qualities in the Landing Approach*, by Robert L. Stapleford, Donald E. Johnston, Gary L. Teper, and David H. Weir, Systems Technology, Inc., Hawthorne, California, NASA CR-239, July, 1965.

7. *Dynamics of Flight, Stability, and Control*, by Bernard Etkins, Professor of Aeronautical Engineering, University of Toronto. Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

8. *Evaluation of Lateral-Directional Handling Qualities of Piloted Reentry Vehicles Utilizing Fixed-Base and In-Flight Evaluations*, by J.I. Meeker, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, NASA CR-778, May 1967.

9. *Features of Large Transport Aircraft Affecting Control During Approach and Landing,* by W.J.G. Pinsker, Royal Aircraft Establishment, Bedford, England.

10. *Flight Testing of Aircraft*, by M.G. Kotik, et al., NASA Technical Translation F-442, April,1967.

11. *Flying Qualities Requirements for United States Navy and Air Force Aircraft,* by William Koven and Richard Wasicko, AGARD Specialists Meeting on Aircraft Stability and Control, Brussels, Belgium, April 1961.

12. *Handling and Performance Exercises Test Technique, Empire Test Pilot's School, Flying Department Notes.*

13. *In-Flight Evaluation of Lateral-Directional Handling Qualities for the Fighter Mission*, by James I. Meeker and G. Warren Hall, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-98, October 1967.

14. *In-Flight Simulation and Pilot Evaluation of Selected Landing Approach Handling Qualities of a Large Logistics Transport Airplane*, by Donald W. Rhodes, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, AFFDL-TR-67-51, July, 1967.

15. *Lateral-Directional Flying Qualities for Power Approach: Influence of Dutch Roll Frequency,* by Edward Seckel, James A. Franklin, and George E. Miller, Princeton University, Princeton, New Jersey, Princeton University Report No. 797, September 1967.

16. *Lecture Notes on Lateral-Directional Stability*, by F.D. Newell and Michael Parrag, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, CAL Report No. TB-1961-F-1, July, 1965.

17. *Modern Flight Dynamics*, by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

18. *Military Specification, Flying Qualities of Piloted Airplanes*, MIL-F-8785C, 5 November 1980.

19. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

20. *A Pilot Opinion Study of Lateral Control Requirements for Fighter-Type Aircraft,* by Brent Y. Creer, John D. Stewart, Robert B. Merrick, and Fred J. Drinkwater, III, Ames Research Center, Moffett Field, California, NASA Memorandum 1-29-59A, March 1959.

21. *The Provision of Adequate Lateral Control Power for Landing Approach Conditions*, by Mr. G.A. Patterson and Lcdr. W. Spangenberg, USN, U.S. Naval Air Test Center, Patuxent River, Maryland.

22. *A Review of Recent Research on Handing Qualities and Its Application to the Handling Problems of Large Aircraft*, by P.L. Bisgood, Ministry of Aviation, Aeronautical Research Council, London, R. & M. No. 3458, June 1964.

23. *Stability and Control, Empire Test Pilot's School Ground School Notes, Book "C"*.

24. *Stability and Control*, USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

25. *Stability and Control of Airplanes and Helicopters*, by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

26. *A Study of Conventional Airplane Handling Qualities Requirements - Part I Roll Handling Qualities*, by I.L. Ashkenas, Systems Technology, Inc., AFFDL-TR-65-138, Part I.

27. *Substantiation Information for Selected Lateral-Directional Requirements*, Cornell Aeronautical Laboratory Report No. BM-2238-F-1, May 1967.

CHAPTER 6

1. *Advisory Group For Aeronautical Research and Development,* Flight Test Manual, Volume II, Stability and Control, by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Airplane Performance, Stability and Control,* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University, and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

3. *Background Information and User Guide for MIL-F-8785B(ASG);* "Military Specification - Flying Qualities of Piloted Airplanes," by C.R. Chalk, T.P. Neal, T.M. Harris, F.E. Prichard, Cornell Aeronautical Laboratory, Inc., Buffalo, New York, Technical Report AFFDL-TR-69-72, August 1969.

4. *Federal Aviation Regulations, Part 25, Airworthiness Standards*: *Transport Category Airplanes,* Federal Aviation Agency, Washington, D.C.

5. *Flight Testing of Aircraft* by M. G. Kotick, et al., NASA Technical Report TT F-442, April, 1967.

6. *Handling and Performance Exercises Test Technique,* Empire Test Pilots' School, Flying Department Notes.

7. *Military Specification, Flying Qualities of Piloted Airplanes* MIL-F-8785C, of 5 November 1980.

8. *National Aeronautics and Space Administration Aeronautical Dictionary* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

9. *Stability and Control, USAF Aerospace Research Pilot School,* Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

10. *Stability and Control, Empire Test Pilot's School Ground School Notes,* Book "C."

11. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

CHAPTER 7

1. *Advisory Group for Aeronautical Research and Development, Flight Test Manual, Volume II, Stability and Control,* by Courtland D. Perkins, Professor, Aeronautical Engineering Department, Princeton University.

2. *Aerodynamics of Supersonic Flight,* by Alan Pope, Head, Aerodynamics Department, Sandia Corporation, Publisher: Pitman Publishing Corporation, New York, Toronto, London.

3. *Airplane Performance, Stability and Control,* by Courtland D. Perkins, Professor and Chairman, Aeronautical Engineering Department, Princeton University and Robert E. Hage, Senior Engineer, Preliminary Design Department, Boeing Airplane Company. Publisher: John Wiley and Sons, Inc., New York, New York, 1949.

4. *Dynamics of Flight, Stability and Control,* by Bernard Etkin, Professor of Aeronautical Engineering, University of Toronto, Publisher: John Wiley and Sons, Inc., New York, New York, 1959.

5. *Flight Testing of Aircraft,* by M. G. Kotik, et al., NASA Technical Translation F-422, April, 1967.

6. *Handling and Performance Exercises Test Techniques,* Empire Test Pilot's School, Flying Department Notes.

7. *High Speed Stability and Control Problems as They Affect Flight Testing,* by Malcolm J. Agzug, Advisory Group For Aeronautical Research and Development, Report 120, May, 1957.

8. *Military Specification, Flying Qualities of Piloted Airplanes MIL-F-8785C,* of 5 November 1980.

9. *Modern Flight Dynamics,* by W. Richard Kolk, Systems Engineer, United Aircraft Corporation. Publisher: Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1961.

10. *National Aeronautics and Space Administration Aeronautical Dictionary,* by Frank D. Adams, United States Government Printing Office, Washington, 1959.

11. *Stability and Control,* Empire Test Pilot's School Ground School Notes, Book "C."

12. *Stability and Control,* USAF Aerospace Research Pilot School, Edwards Air Force Base, California, AFFTC-TIH-77-1, revised February 1977.

13. *Stability and Control of Airplanes and Helicopters,* by Edward Seckel, Aeronautical Engineering Department, Princeton University. Publisher: The Academic Press, New York, New York, 1964.

APPENDIX III

FIGURES

APPENDIX III

FIGURES

PAGE

CHAPTER 1

CHAPTER 2

CHAPTER 3

CHAPTER 4

FIGURES

CHAPTER 5

CHAPTER 6

CHAPTER 7

FIGURES

APPENDIX IV

EQUATIONS

APPENDIX IV

EQUATIONS

CHAPTER 2

$$V_S = \sqrt{\frac{2nW}{\rho C_{L_{max}} S}}$$ *eq 2.1* 2.14

CHAPTER 4

$$\bar{V} = \text{tail volume coefficient} = \frac{S_t}{S_w}\frac{\ell_t}{\bar{c}}$$ *eq 4.1* 4.5

$$\eta_t = \text{tail efficiency factor} = \frac{q_t}{q}$$ *eq 4.2* 4.6

$$\frac{dC_{L_t}}{d\alpha_t} = a_t$$ *eq 4.3* 4.6

$$\frac{dC_{m_{CG}}}{dC_L}_{\text{Airplane}} = \frac{X_a}{\bar{c}} + \frac{dC_m}{dC_L}_{\substack{\text{Fuselage}\\ \text{Nacelle}}} - \frac{a_t}{a_w}\bar{V}\eta_t\left(1 - \frac{d\varepsilon}{d\alpha}\right)$$ *eq 4.4* 4.8

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{\text{Airplane}} = \frac{X_{CG}}{\bar{c}} - N_0$$ *eq 4.5* 4.10

$$C_{m_{CG}} = C_{m_{ac}} + \frac{X_a}{\bar{c}} C_L + C_{m_{CG}}{}_{\substack{\text{Fus}\\ \text{Nac}}} - a_t\ \alpha_t\ \eta_t\ \bar{V}$$ *eq 4.6* 4.12

<u>"Slab Tail"</u> $$C_{m_{i_t}} = \frac{dC_{m_{CG}}}{di_t} = -\frac{dC_L}{d\alpha_t}\eta_t\bar{V} = -a_t\ \eta_t\ \bar{V}$$ *eq. 4.7* 4.13

<u>Elevator</u> $$C_{m_{\delta_e}} = \frac{dC_{m_{CG}}}{d\delta_e} = -\frac{dC_L}{d\alpha_t}\frac{d\alpha_{t_{EFF}}}{d\delta_e}\bar{V}\eta_t = -a_t\ \tau\ \eta_t\ \bar{V}$$

eq 4.8 4.13

$$\alpha_t = \alpha_w - \varepsilon - i_w + i_t + \tau\delta_e$$ *eq 4.9* 4.14

$$C_{m_{CG}} = C_{m_{ac}} + \frac{X_a}{\bar{c}} C_L + C_{m_{CG_{\substack{Fus\\Nac}}}} - a_t \left(\alpha_w - \varepsilon - i_w + i_t + \tau\delta_e\right) \bar{V}\eta_t$$

eq 4.10 4.14

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

eq 4.11 4.15

$$\frac{d\delta_e}{dC_L} = \frac{-\left(\frac{dC_m}{dC_L}\right)_x}{C_{m_{\delta_e}}}$$

eq 4.12 4.16

$$C_{h_e} = C_{h_{\alpha_t}} \alpha_t + C_{h_{\delta_e}} \delta_e$$

eq 4.13 4.20

$$\delta_{e_{Float}} = -\frac{C_{h_{\alpha t}}}{C_{h_{\delta_e}}} \alpha_t$$

eq 4.14 4.20

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Free} = \left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Fixed} + C_{m_{\delta_e}} \frac{d\delta_{e_{Float}}}{dC_L}$$

eq 4.15 4.20

$$\left.\frac{dC_{m_{CG}}}{dC_L}\right|_{Free} = X_{CG} - N_0'$$

eq 4.16 4.21

$$\frac{dF_s}{dV_e} = 2K \frac{W}{S} \frac{Ch_{\delta_e}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{Free} \frac{V_e}{V_{e_{Trim}}^2}$$

eq 4.17 4.24

$$\delta_{e_{Float}} = -\frac{C_{h_{\alpha t}}}{C_{h_{\delta_e}}} \alpha_t$$

eq 4.18 4.26

$\frac{dC_m}{dC_L} = \frac{\partial C_m / \partial \alpha}{\partial C_L / \partial \alpha}$	*eq 4.19*	4.35
$\begin{vmatrix} S + D_u & g \\ L_u / u_0 & -S \end{vmatrix} = 0$	*eq 4.20*	4.36
$S^2 + D_u S + g \frac{L_u}{u_0} = 0$	*eq 4.21*	4.37
ω_{n_p} = undamped phugoid frequency $= \sqrt{2} \frac{g}{u_0}$	*eq 4.22*	4.37
ζ_p = phugoid damping ratio $= \frac{1}{\sqrt{2}} \frac{C_D}{C_L}$	*eq 4.23*	4.37
ω_p = damped natural frequency $\approx \sqrt{2} \frac{g}{u_0}$	*eq 4.24*	4.37
p_p (sec) = .138 u_0 (where u_0 is in feet per sec.)	*eq 4.25*	4.37
$\zeta_p \approx \frac{.707}{L/D}$	*eq 4.26*	4.37
$\dot{\theta}_{pull-up} = \frac{g\ (n-1)}{V}$	*eq 4.27*	4.87
$\dot{\theta}_{\text{steady level turn}} = \frac{g}{V}\left(n - \frac{1}{n}\right)$	*eq 4.28*	4.87
$M_{CG_{\text{Due to } \dot{\theta}}} = -a_t \frac{\ell_t^2\ \dot{\theta}}{V} q_t S_t$	*eq 4.29*	4.88

$$C_{m_{\dot\theta}} = \frac{\partial C_{m_{CG}}}{\partial\left(\frac{\dot\theta \bar c}{2V}\right)} = -2a_t \ \eta_t \bar V \ \frac{\ell_t}{\bar c}$$ *eq 4.30* 4.89

$$\delta_{e_{\text{Pull-Ups}}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} n + \frac{C_{m_{\dot\theta}}\rho g \bar c}{4 \ W/S} (n-1)\right\}$$

eq 4.31 4.89

$$\left(\frac{d\delta_e}{dn}\right)_{\text{Pull-Ups}} = -\frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{\rho g \bar c}{4 \ W/S} C_{m_{\dot\theta}}\right\}$$

eq 4.32 4.90

$$\delta_{e_{\text{Steady Turns}}} = \delta_{e_0} - \frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot\theta}}\rho g \bar c}{4 \ W/S} \left(n - \frac{1}{n}\right)\right\}$$

eq 4.33 4.90

$$\left(\frac{d\delta_e}{dn}\right)_{\text{Steady Turns}} = -\frac{1}{C_{m_{\delta_e}}} \frac{W/S}{\frac{1}{2}\rho_{SSL}V_e^2} \left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot\theta}}\rho g \bar c}{4 \ W/S} \left(n + \frac{1}{n^2}\right)\right\}$$

eq 4.34 4.90

$$F_{s_{\text{Pull-Up}}} = K \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} \left\{\frac{V_e^2}{V_{e_{\text{Trim}}}^2} - n\right\} + K\tfrac{1}{2}\rho \ell_t g(n-1)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.35 4.97

$$F_{s_{\text{SteadyTurns}}} = K \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} \left\{\frac{V_e^2}{V_{e_{\text{Trim}}}^2} - n\right\} + K\tfrac{1}{2}\rho \ell_t g\left(n - \frac{1}{n}\right)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.36 4.98

$$\left(\frac{dF_s}{dn}\right)_{\text{Pull-Up}} = -K \frac{W}{S} \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \left(\frac{dC_m}{dC_L}\right)_{\text{Free}} + K \ \tfrac{1}{2} \ \rho \ell_t g \left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.37 4.98

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turn}} = -K\ \frac{W}{S}\ \frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{\text{Free}} + K\ \frac{1}{2}\ \rho \ell_t g\left(1 + \frac{1}{n^2}\right)\left\{C_{h_{\alpha_t}} - \frac{C_{h_{\delta_e}}}{\tau}\right\}$$

eq 4.38 4.98

$$F_s = K_1 \Delta\delta e$$

eq 4.39 4.105

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turns}} = -\frac{K_1}{C_{m_{\delta_e}}}\ \frac{W/S}{\frac{1}{2}\rho_{SSL}\ V_{e_{Trim}}^2}\left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot\theta}}\rho g\bar{c}}{4\ W/S}\left(1 - \frac{1}{n^2}\right)\right\}$$

eq 4.40 4.106

$$Fs = K_2 q \Delta\delta_e$$

eq 4.41 4.107

$$\left(\frac{dF_s}{dn}\right)_{\text{Steady Turns}} = -\frac{K_2\ W/S}{C_{m_{\delta_e}}}\left\{\left(\frac{dC_m}{dC_L}\right)_{\text{Fixed}} + \frac{C_{m_{\dot\theta}}\rho g\bar{c}}{4\,W/S}\left(1 - \frac{1}{n^2}\right)\right\}$$

eq 4.42 4.107

$$\begin{vmatrix} S + \frac{L_\alpha}{\mu_0} & -1 \\ -M_{\dot\alpha} S - M_\alpha & S - M_{\dot\theta} \end{vmatrix} = 0$$

eq 4.43 4.118

$$S^2 + \left(\frac{L_\alpha}{u_0} - M_{\dot\theta} - M_{\dot\alpha}\right) S - \left(M_\alpha + \frac{L_\alpha}{u_0}\ M_{\dot\theta}\right) = 0$$

eq 4.44 4.118

$\omega_{n_{sp}}$ = undamped short period frequency

$$= \sqrt{\frac{\frac{1}{2}\ P_a\ M^2}{I_{yy}}\ S\bar{c}\ C_{L_\alpha}\left(\frac{X_{CG}}{\bar{c}} - N_M\right)}$$

eq 4.45 4.119

$$M_{\dot{\alpha}} \doteq 0$$

$$\frac{L_\alpha}{u_0} \doteq -M_{\dot{\theta}}$$ *eq 4.46* 4.119

$$\omega_{sp} = \sqrt{-M_\alpha}$$ *eq 4.47* 4.120

$$\zeta_{sp} = \frac{\sqrt{\frac{\rho S}{2}}}{2\sqrt{-\frac{\bar{c}}{I_{yy}} C_{L_\alpha} \left(\frac{X_{CG}}{\bar{c}} - N_M\right)}} \left\{\frac{C_{L_\alpha}}{w/g} - \frac{C_{m_{\dot{\theta}}} \bar{c}^2}{2I_{yy}} - \frac{C_{m_{\dot{\alpha}}} \bar{c}^2}{2I_{yy}}\right\}$$

eq 4.48 4.121

$$\frac{F_s}{\alpha} = \left(\frac{F_s}{n}\right)\left(\frac{n}{\alpha}\right)$$ *eq 4.49* 4.135

CHAPTER 5

$$Y_{\beta_v} = -a_v (\beta - \sigma) q_v S_v$$ *eq 5.1* 5.8

$$C_{y_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w}$$ *eq 5.2* 5.9

$$Y_{\delta_r} = a_v \tau_v \delta_r q_v S_v$$ *eq 5.3* 5.11

$$C_{y_{\delta_r}} = a_v \tau_v \eta_v \frac{S_v}{S_w}$$ *eq 5.4* 5.12

$$C_{\ell_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} \frac{Z_v}{b}$$ *eq 5.5* 5.17

$$L_{\delta_r} = Y_{\delta_r} Z_v$$ *eq 5.6* 5.19

$$C_{\ell_{\delta_r}} = C_{y_{\delta_r}} \frac{Z_v}{b}$$ *eq 5.7* 5.19

$$L_{\delta_a} = 2a_w \ \tau_a \ q_a \int_{y_1}^{y_2} c\, y\, dy$$ *eq 5.8* 5.20

$$C_{\ell_{\delta_a}} = \frac{-2a_w \ \tau_a \ \eta_a}{S_w \ b} \int_{y_1}^{y_2} c\, y dy$$ *eq 5.9* 5.21

$$N_{\beta_v} = -Y_{\beta_v} \ \ell_v = a_v \left(1 - \frac{d\sigma}{d\beta}\right) q_v S_v \ell_v$$ *eq 5.10* 5.24

$$C_{n_{\beta_V}} = -C_{y_{\beta_v}} \ \frac{\ell_v}{b} = a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \ \frac{S_v}{S_w} \ \frac{\ell_v}{b}$$ *eq 5.11* 5.24

$$C_{n_\beta} = C_{n_{\beta_{w,\,F,\,N}}} + C_{n_{\beta_v}}$$ *eq 5.12* 5.25

$$N_{r_v} = -a_v \ S_v \ q_v \ \frac{{\ell_v}^2}{V}$$ *eq 5.13* 5.26

$$C_{n_{r_v}} = \frac{\partial C_n}{\partial\left(\frac{rb}{2V}\right)} = -2a_v \ \frac{S_V}{S_w} \ \eta_v \ \frac{{\ell_v}^2}{b^2}$$ *eq 5.14* 5.26

$$N_{\delta_r} = -a_v \ \tau_v \ q_v \ S_v \ \ell_v$$ *eq 5.15* 5.27

$$C_{n_{\delta_r}} = -a_v \ \tau_v \ n_v \ \frac{S_v}{S_w} \ \frac{\ell_v}{b}$$ *eq 5.16* 5.27

<u>SIDEFORCE</u> $C_{y_0} + C_{y_\beta} \ \beta + C_{y_{\delta_r}} \ \delta_r + C_L \ \phi = 0$

eq 5.17 5.34

<u>YAWING MOMENT</u> $C_{n_0} + C_{n_\beta} \ \beta + C_{n_{\delta_r}} \ \delta_r + C_{n_{\delta_a}} \ \delta_a = 0$

eq 5.18 5.34

<u>ROLLING MOMENT</u> $C_{\ell_0} + C_{\ell_\beta} \ \beta + C_{\ell_{\delta_r}} \ \delta_r + C_{n_{\delta_a}} \ \delta_a = 0$

eq 5.19 5.34

SIDEFORCE $$C_{y_\beta} + C_{y_{\delta_r}} \frac{d\delta_r}{d\beta} + C_L \frac{d\phi}{d\beta} = 0$$

eq 5.20 5.34

YAWING MOMENT $$C_{n_\beta} + C_{n_{\delta_r}} \frac{d\delta_r}{d\beta} + C_{n_{\delta_a}} \frac{d\delta_a}{d\beta} = 0$$

eq 5.21 5.34

ROLLING MOMENT $$C_{\ell_\beta} + C_{\ell_{\delta_r}} \frac{d\delta_r}{d\beta} + C_{\ell_{\delta_a}} \frac{d\delta_a}{d\beta} = 0$$

eq 5.22 5.34

$$\frac{d\delta_r}{d\beta} = \frac{-\dfrac{C_{n_\beta}}{C_{n_{\delta_r}}}\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\dfrac{C_{\ell_\beta}}{C_{n_\beta}}\right\}}{\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}}$$

eq 5.23 5.35

$$\frac{d\delta_a}{d\beta} = \frac{-\dfrac{C_{\ell_\beta}}{C_{\ell_{\delta_r}}}\left\{1 - \dfrac{C_{\ell_{\delta_a}}}{C_{n_{\delta_a}}}\dfrac{C_{n_\beta}}{C_{\ell_\beta}}\right\}}{\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}}$$

eq 5.24 5.35

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} + C_{y_{\delta_r}}\frac{d\delta_r}{d\beta}\right\}$$

eq 5.25 5.35

$$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}}$$

eq 5.26 5.35

$$\left\{1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\frac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}$$

eq 5.27 5.36

$$1 - \frac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\frac{C_{\ell_\beta}}{C_{n_\beta}}$$

eq 5.28 5.36

$F_r = K_1 \Delta\delta_r$ (linear feel spring system) *eq 5.29* 5.37

$F_r = K_2 q \Delta\delta_r$ ("q – feel" system) *eq 5.30* 5.37

$\delta_{r_{Float}} = -\dfrac{C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\beta_v$	*eq 5.31*	5.38
$F_r = -K\,C_{h_{\delta_r}}\,q_v S_r \bar{c}_r \left\{\delta_{r_{Equilibrium}} - \delta_{r_{Float}}\right\}$	*eq 5.32*	5.39
$\dfrac{dF_r}{d\beta} = -K\,C_{h_{\delta_r}}\,q_v S_r \bar{c}_r \left\{\dfrac{d\delta_r}{d\beta} - \dfrac{d\delta_{r_{Float}}}{d\beta}\right\}$	*eq 5.33*	5.39
$\dfrac{d\delta_r}{d\beta} = -\dfrac{C_{n_\beta}}{C_{n_{\delta_r}}}$ (if the assumption $C_{n_{\delta_a}} = 0$ is valid)	*eq 5.34*	5.39
$\dfrac{d\delta_{r_{Float}}}{d\beta} = -\dfrac{C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\left(1 - \dfrac{d\sigma}{d\beta}\right)$	*eq 5.35*	5.39
$\dfrac{dF_r}{d\beta} = -K\dfrac{C_{h_{\delta_r}}}{C_{n_{\delta_r}}}\,q_v S_r \bar{c}_r \left\{-C_{n_\beta} + \dfrac{C_{n_{\delta_r}}\,C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\left(1 - \dfrac{d\sigma}{d\beta}\right)\right\}$	*eq 5.36*	5.40
$\dfrac{C_{n_{\delta_r}}\,C_{h_{\beta_v}}}{C_{h_{\delta_r}}}\left(1 - \dfrac{d\sigma}{d\beta}\right)$	*eq 5.37*	5.40
$\dfrac{d\delta_a}{d\beta} = \dfrac{-\dfrac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}}\left\{1 - \dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\,\dfrac{C_{n_\beta}}{C_{\ell_\beta}}\right\}}{\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_r}}}\,\dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}}$	*eq 5.38*	5.43
$\dfrac{d\delta_a}{d\beta} = -\dfrac{C_{\ell_\beta}}{C_{\ell_{\delta_a}}}$	*eq 5.39*	5.44
$\left\{1 - \dfrac{C_{n_{\delta_a}}}{C_{\ell_{\delta_a}}}\,\dfrac{C_{\ell_{\delta_r}}}{C_{n_{\delta_r}}}\right\}$	*eq 5.40*	5.44
$\left\{1 - \dfrac{C_{\ell_{\delta_a}}}{C_{n_{\delta_a}}}\,\dfrac{C_{n_\beta}}{C_{\ell_\beta}}\right\}$	*eq 5.41*	5.44

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} + C_{y_{\delta_r}}\frac{d\delta_r}{d\beta}\right\}$$ *eq 5.42* 5.47

$$\frac{d\delta_r}{d\beta} = -\frac{C_{n_\beta}}{C_{n_{\delta_r}}}$$ *eq 5.43* 5.47

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} C_{n_\beta}\right\}$$ *eq 5.44* 5.47

$$C_{y_\beta} = C_{y_{\beta_{\text{Wing, Fuselage, Nacelles}}}} + C_{y_{\beta_{\text{Vertical Tail}}}}$$ *eq 5.45* 5.47

$$C_{n_\beta} = C_{n_{\beta_{\text{Wing, Fuselage, Nacelles}}}} + C_{n_{\beta_{\text{Vertical Tail}}}}$$

$$C_{n_\beta} = C_{n_{\beta_{W,F,N}}} - C_{y_{\beta_v}}\frac{\ell_v}{b}$$ *eq 5.46* 5.47

$$\frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} = \frac{C_{y_{\delta_r}}}{-C_{y_{\delta_r}}\frac{\ell_v}{b}} = -\frac{b}{\ell_v}$$ *eq 5.47* 5.47

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_{\beta_{W,F,N}}} + C_{y_{\beta_v}} + \frac{b}{\ell_v}\left(C_{n_{\beta_{W,F,N}}} - C_{y_{\beta_v}}\frac{\ell_v}{b}\right)\right\}$$

$$\frac{d\phi}{d\beta} = -\frac{1}{C_L}\left\{C_{y_{\beta_{W,F,N}}} + \frac{b}{\ell_v} C_{n_{\beta_{W,F,N}}}\right\}$$

eq 5.48 5.48

$$L_\beta = 0 \quad L_r = 0$$ *eq 5.49* 5.51

$$S\left(S - L_p\right)\begin{vmatrix} S - Y_\beta & 1 \\ -N_\beta & S - N_r \end{vmatrix} = 0$$ *eq 5.50* 5.53

$$S\left(S - L_p\right)\left\{S^2 + \left(-Y_\beta - N_r\right)S + \left(N_\beta + Y_\beta N_r\right)\right\} = 0$$ *eq 5.51* 5.53

$$\frac{g}{u_0}\left\{L_\beta N_r - N_\beta L_r\right\}$$ *eq 5.52* 5.54

$$p_{SS} = -\frac{L_{\delta_a}}{L_p}\delta_a = -\frac{C_{\ell_{\delta_a}}}{C_{\ell_p}}\frac{2V}{b}\delta_a$$ *eq 5.53* 5.55

$$L_{\delta_a} = \frac{\partial L / \partial \delta_a}{I_{XX}} = C_{\ell_{\delta_a}} \frac{qSb}{I_{XX}}$$ = rolling moment due to lateral control deflection (lateral control power) term.

eq 5.54 5.55

$$\tau_R = -\frac{1}{L_p}$$

eq 5.55 5.56

$$\left\{S^2 + \left(-Y_\beta - N_r\right) S + \left(N_\beta + Y_\beta N_r\right)\right\} = 0$$

eq 5.56 5.57

$$\omega_{n_{DR}} \doteq M\sqrt{C_{n_\beta} \frac{\gamma P_a Sb}{2 I_{ZZ}}}$$

eq 5.57 5.57

$$\zeta_{DR} = C_{n_r}\sqrt{\frac{\rho Sb^3}{8 C_{n_\beta} I_{ZZ}}}$$

eq 5.58 5.59

$$\frac{\phi}{\delta_a}(S) = \frac{L_{\delta_a}\left[S^2 + 2\zeta_\phi \omega_{n_\phi} S + \omega_{n_\phi}^2\right]}{\left(S + \frac{1}{T_s}\right)\left(S + \frac{1}{T_R}\right)\left[S^2 + 2\zeta_d \omega_{n_d} S + \omega_{n_d}^2\right]}$$

eq 5.59 5.67

$$g \sin \phi = Vr$$

eq 5.60 5.75

SIDEFORCE $$C_{y_\beta} \beta + C_{y_r}\left(\frac{rb}{2V}\right) + \frac{W}{qS} \sin \phi = \frac{W}{g} \frac{Vr}{qS}$$

eq 5.61 5.76

YAWING MOMENT $$C_{n_\beta} \beta + C_{n_r}\left(\frac{rb}{2V}\right) = 0$$

eq 5.62 5.76

ROLLING MOMENT $$C_{\ell_\beta} \beta + C_{\ell_{\delta_a}} \delta_a + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0$$

eq 5.63 5.76

$$\delta_{a_{\text{Equilibrium}}} = -\frac{1}{C_{\ell_{\delta_a}}}\left\{C_{\ell_r}\left(\frac{rb}{2V}\right) + C_{\ell_\beta} \beta\right\}$$

eq 5.64 5.76

$$\beta = -\frac{C_{n_r}}{C_{n_\beta}}\left(\frac{rb}{2V}\right)$$ *eq 5.65* 5.76

$$\frac{d\delta_{a_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = -\frac{1}{C_{\ell_{\delta_a}}\ C_{n_\beta}}\left\{C_{\ell_r}\ C_{n_\beta} - C_{\ell_\beta}\ C_{n_r}\right\}$$ *eq 5.66* 5.77

SIDEFORCE $$C_{y_\beta}\ \beta + C_{y_{\delta_r}}\ \delta_r + C_{y_r}\ \frac{rb}{2V} + \frac{W}{qs}\sin\phi = \frac{W}{g}\ Vr$$ *eq 5.67* 5.81

YAWING MOMENT $$C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_r}\ \frac{rb}{2V} = 0$$ *eq 5 .68* 5.81

ROLLING MOMENT $$C_{\ell_\beta}\ \beta + C_{\ell_r}\ \frac{rb}{2V} = 0$$ *eq 5.69* 5.81

$$\delta_{r_{Equilibrium}} = -\frac{1}{C_{n_{\delta_r}}}\left\{C_{n_r}\left(\frac{rb}{2V}\right) + C_{n_\beta}\ \beta\right\}$$ *eq 5.70* 5.81

$$\beta = -\frac{C_{\ell_r}}{C_{\ell_\beta}}\ \frac{rb}{2V}$$ *eq 5.71* 5.81

$$\frac{d\delta_{r_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = \frac{1}{C_{n_{\delta_r}}\ C_{\ell_\beta}}\left\{C_{\ell_r}\ C_{n_\beta} - C_{\ell_\beta}\ C_{n_r}\right\}$$ *eq 5.72* 5.81

SIDEFORCE $$C_{y_\beta}\ \beta + C_{y_{\delta_r}}\ \delta_r + C_{y_r}\left(\frac{rb}{2V}\right) = 0$$
$$\left(\text{Since } \frac{W}{qS}\sin\phi = \frac{W}{g}\ \frac{Vr}{qS}\right)$$ *eq 5.73* 5.82

YAWING MOMENT $$C_{n_\beta}\ \beta + C_{n_{\delta_r}}\ \delta_r + C_{n_{\delta_a}}\ \delta_a + C_{n_r}\left(\frac{rb}{2V}\right) = 0$$ *eq 5.74* 5.82

ROLLING MOMENT $$C_{\ell_\beta}\ \beta + C_{\ell_{\delta_r}}\ \delta_r + C_{\ell_{\delta_a}}\ \delta_a + C_{\ell_r}\left(\frac{rb}{2V}\right) = 0$$ *eq 5.75* 5.82

$$\delta_{r_{Equilibrium}} = \frac{\begin{vmatrix} C_{y_\beta} & -C_{y_r} \\ C_{n_\beta} & -C_{n_r} \end{vmatrix}}{\begin{vmatrix} C_{y_\beta} & C_{y_{\delta_r}} \\ C_{n_\beta} & C_{n_{\delta_r}} \end{vmatrix}} \left(\frac{rb}{2V}\right)$$ *eq 5.76* 5.82

$$\frac{d\delta_{r_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = -\frac{C_{n_r}}{C_{n_{\delta_r}}} \left\{ \frac{C_{y_\beta} - \frac{C_{y_r}}{C_{n_r}} C_{n_\beta}}{C_{y_\beta} - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}} C_{n_\beta}} \right\}$$ *eq 5.77* 5.82

$$\delta_{r_{Equilibrium}} = \frac{C_{n_r}}{C_{n_{\delta_r}}} \left(\frac{rb}{2V}\right)$$

$$C_{n_{\delta_a}} = 0$$ *eq 5.78* 5.83

$$C_{n_\beta}\ \beta + C_{n_{\delta_r}} \left\{ -\frac{C_{n_r}}{C_{n_{\delta_r}}} \left(\frac{rb}{2V}\right) \right\} + C_{n_r} \left(\frac{rb}{2V}\right) = 0$$

$$\beta = \frac{\left(C_{n_r} - C_{n_r}\right)\left(\frac{rb}{2V}\right)}{C_{n_\beta}}$$

$$\beta = 0$$ *eq 5.79* 5.83

$$\delta_{r_{Equilibrium}} = -\frac{C_{n_r}}{C_{n_{\delta_r}}} \left(\frac{rb}{2V}\right) \qquad \beta_{Equilibrium} = 0$$ *eq 5.80* 5.83

$$C_{\ell_{\delta_r}} \left\{ -\frac{C_{n_r}}{C_{n_{\delta_r}}} \left(\frac{rb}{2V}\right) \right\} + C_{\ell_{\delta_a}}\ \delta_{a_{Equilibrium}} + C_{\ell_r} \left(\frac{rb}{2V}\right) = 0$$

$$\frac{d\delta_{a_{Equilibrium}}}{d\left(\frac{rb}{2V}\right)} = -\frac{C_{\ell_r}}{C_{\ell_{\delta_a}}} \left\{ 1 - \frac{C_{\ell_{\delta_r}} C_{n_r}}{C_{\ell_r} C_{n_{\delta_r}}} \right\}$$ *eq 5.81* 5.84

Level 1 $-\Delta\zeta_d\ \omega_{n_d} = .014 \left(\omega_{nd}^2\ |\phi/\beta|_d - 20\right)$

Level 2 $-\Delta\zeta_d\ \omega_{n_d} = .009 \left(\omega_{nd}^2\ |\phi/\beta|_d - 20\right)$

Level 3 $-\Delta\zeta_d\ \omega_{n_d} = .005 \left(\omega_{nd}^2\ |\phi/\beta|_d - 20\right)$

with ω_{n_d} in rad/sec. *eq 5.82* 5.109

$$\psi_\beta = \frac{-360}{T_d} t_{n_\beta} + (n-1)\,360 = \frac{-360}{3.5}(2.95) = -303^\circ$$ *eq 5.83* 5.116

SIDEFORCE $$C_{y_\beta}\beta + C_{y_{\delta_r}}\delta_r + C_L \sin\phi = \frac{mu_0\dot{\beta} + mu_0\, r}{qS}$$ *eq 5.84* 5.131

YAWING MOMENT $$C_{n_\beta}\beta + C_{n_{\delta_r}}\delta_r + C_{n_{\delta_a}}\delta_a + C_{n_r}\frac{rb}{2V} + C_{n_p}\frac{pb}{2V} = \frac{1}{qSb} I_{yy}\dot{r}$$ *eq 5.85* 5.131

ROLLING MOMENT $$C_{\ell_\beta}\beta + C_{\ell_{\delta_r}}\delta_r + C_{\ell_{\delta_a}}\delta_a + C_{\ell_r}\frac{rb}{2V} + C_{\ell_p}\frac{pb}{2V} = \frac{1}{qSb} I_{yy}\dot{p}$$ *eq 5.86* 5.131

$$C_{\ell_\beta}\beta\ ;\ C_{\ell_{\delta_r}}\delta_r\ ;\ C_{\ell_r}\frac{rb}{2V}$$ *eq 5.87* 5.131

$$\dot{p}\frac{I_{XX}}{qSb} - C_{\ell_p}\frac{pb}{2V_T} - C_{\ell_{\delta_a}}\delta_a = 0$$ *eq 5.88* 5.132

$$\dot{p} - \frac{C_{\ell_p} qSb}{I_{XX}}\frac{b}{2V_T}\, p - \frac{C_{\ell_{\delta_a}} qSb}{I_{XX}}\delta_a = 0$$ *eq 5.89* 5.132

$$\dot{p} - L_p\, p - L_{\delta_a}\delta_a = 0$$ *eq 5.90* 5.132

$$p(t) = \frac{L_{\delta_a}\delta_a}{L_p}\left\{e^{L_p t} - 1\right\}$$ *eq 5.91* 5.132

$$p(t) = p_{ss}\left\{1 - e^{-t/\tau_R}\right\}$$ *eq 5.92* 5.133

$$\dot{p} - L_p\, p - L_{\delta_a}\delta_a = 0$$ *eq 5.93* 5.133

$$\dot{p}_t = 0 = L_{\delta_a} \delta_a$$ *eq 5.94* 5.133

$$\dot{p} - L_p p - L_{\delta_a} \delta_a = 0$$

however, when $|L_p p| = |L_{\delta_a} \delta_a|$, $\dot{p} = 0$, and $p = p_{ss}$, thus:

$$p_{ss} = -\frac{L_{\delta_a} \delta_a}{L_p}$$ *eq 5.95* 5.135

$$\left(\frac{pb}{2V_T}\right)_{Max} = -\frac{C_{\ell_{\delta_a}}}{C_{\ell_p}} \delta_{a_{Max}}$$ *eq 5.96* 5.136

$$\tau_R = -\frac{1}{L_p}$$ *eq 5.97* 5.137

$$p_{ss} = -\frac{L_{\delta_a}}{L_p} \delta_a$$ *eq 5.98* 5.138

$$\tau_R = -\frac{1}{L_p} \text{ or } \tau_R = \frac{4 I_{xx}}{C_{\ell_p} \rho V_T S b^2} \text{ or } \tau_R = \frac{4 I_{xx}}{C_{\ell_p} \sqrt{\sigma \rho_{ssl}} V_e S b^2}$$ *eq 5.99* 5.138

$$P_{CRIT_1} \sim \sqrt{\frac{\frac{C_{m_\alpha} qS\bar{c}}{I_{yy}}}{\frac{I_{xx} - I_{zz}}{I_{yy}}}}$$ *eq 5.100* 5.148

$$P_{CRIT_2} \sim \sqrt{\frac{\frac{C_{n_\beta} qSb}{I_{zz}}}{\frac{I_{yy} - I_{xx}}{I_{zz}}}}$$ *eq 5.101* 5.148

$F_a = K_1 \Delta\delta_a$ (linear feel spring system) *eq 5.102* 5.156

$F_a = K_2 q\Delta\delta_a$ ("q – feel" system) *eq 5.103* 5.156

$$C_{h_\alpha} = C_{h_{\delta_\alpha}} \delta_\alpha + C_{h_\alpha} \Delta\alpha_{Ave}$$ *eq 5.104* 5.157

$$\Delta\alpha_{Ave} = \frac{py'}{V}$$ *eq 5.105* 5.157

$$\delta_{a_{Float}} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}} \frac{py'}{V} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}} \left(\frac{2y'}{b}\right)\left(\frac{pb}{2V}\right)$$ *eq 5.106* 5.157

$$F_a = -K\, C_{h_{\delta_a}}\, q\, S_a\, \bar{c}_a \left\{\delta_{a_{Equilibrium}} - \delta_{a_{Float}}\right\}$$ *eq 5.107* 5.158

$$\delta_{a_{Equilibrium}} = -\frac{C_{\ell_p}}{C_{\ell_{\delta_a}}} \frac{pb}{2V} \text{ (steady state roll)}$$ *eq 5.108* 5.159

$$\delta_{a_{Float}} = -\frac{C_{h_\alpha}}{C_{h_{\delta_a}}} \left(\frac{2y'}{b}\right)\left(\frac{pb}{2V}\right)$$ *eq 5.109* 5.159

$$F_a = V_p \left\{-\frac{K}{4}\, C_{h_{\delta_a}}\, \rho\, S_a \bar{c}_a\, b\right\}\left\{-\frac{C_{\ell_p}}{C_{\ell_{\delta_a}}} + \frac{C_{h_\alpha}}{C_{h_{\delta_a}}} \frac{2y}{b}\right\}$$ *eq 5.110* 5.159

$$F_a = K_1\, V_p$$ *eq 5.111* 5.159

$$pV = K_2$$ *eq 5.112* 5.159

$$p(t) = p_{ss}\left\{1 - e^{-t/\tau_R}\right\}$$ *eq 5.113* 5.190

$$p(t) = p_{ss} - p_{ss}\, e^{-t/\tau_R}$$ *eq 5.114* 5.190

$$X(t) = p_{ss} e^{-t/\tau_R}$$ *eq 5.115* 5.191

$$\ln X(t) = \ln p_{ss} - \frac{t}{\tau_R}$$ *eq 5.116* 5.192

$$\ln X(t) = K_1 - \frac{t}{K_2}$$ *eq 5.117* 5.192

$$X(t) = p_{ss}\, e^{-t/\tau_R}$$ *eq 5.118* 5.192

$$\tau_R = t_2 - t_1$$ *eq 5.119* 5.194

CHAPTER 6

$$N_T = F_N \; y_p$$ *eq 6.1* 6.2

$$F_{N_{Prop}} = \frac{550 \eta_p \; BHP}{V}$$ *eq 6.2* 6.3

$$C_{n_{T_{Prop}}} = \frac{550 \eta_p \; BHP}{V \;\; W} \; \frac{C_L \; y_p}{b} = \frac{550 \eta_p \; BHP \; y_p}{V \; qSb}$$ *eq 6.3* 6.3

<u>SIDEFORCE</u> $$C_{y_\beta} \; \beta + C_{y_{\delta_r}} \; \delta_r + C_L \; \phi = 0$$ *eq 6.4* 6.3

<u>YAWING MOMENT</u> $$\frac{F_N}{W} \; C_L \; \frac{y_p}{b} + C_{n_\beta} \; \beta + C_{n_{\delta_r}} \; \delta_r = 0$$ *eq 6.5* 6.3

<u>ROLLING MOMENT</u> $$C_{\ell_\beta} \; \beta + C_{\ell_{\delta_a}} \; \delta_a = 0$$ *eq 6.6* 6.3

$$\delta_{r_{Equilibrium}} = \frac{- {}^{F_N}\!/_{W} \; C_L \; {}^{y_p}\!/_{b}}{C_{n_{\delta_r}}} \text{ (ZERO SIDESLIP)}$$ *eq 6.7* 6.4

$$\delta_{r_{Equilibrium}} = \frac{- \; C_{y_{\delta_r}} \; \delta_r}{C_L} \text{ (ZERO SIDESLIP)}$$ *eq 6.8* 6.4

$$\delta_{r_{EQUILIBRIUM}} = \frac{\begin{vmatrix} C_{y_\beta} & 0 \\ C_{n_\beta} & - \frac{F_N}{W} \; C_L \; \frac{y_p}{b} \end{vmatrix}}{\begin{vmatrix} C_{y_\beta} & C_{y_{\delta_r}} \\ C_{n_b} & C_{n_{\delta_r}} \end{vmatrix}} = \frac{- \frac{F_N}{W} \; \frac{y_p}{b} \; C_L \; C_{y_\beta}}{C_{n_{\delta_r}} \; C_{y_\beta} - C_{y_{\delta_r}} \; C_{n_\beta}}$$

eq 6.9 6.6

$$\delta_{r_{\text{Equilibrium}}} = \frac{-\frac{F_N}{W}\,\frac{y_p}{b}\,C_L}{C_{n_{\delta_r}}\left\{1 - \frac{C_{y_{\delta_r}}}{C_{n_{\delta_r}}}\,\frac{C_{n_\beta}}{C_{y_\beta}}\right\}} \text{ (ZERO BANK ANGLE)}$$

eq 6.10 6.6

$$\beta = -\frac{C_{y_{\delta_r}}\,\delta_r}{C_{y_\beta}} \text{ (ZERO BANK ANGLE)}$$

eq 6.11 6.6

$$\beta = \frac{-\,{}^{F_N}\!/_{W}\,C_L\,{}^{y_p}\!/_{b}}{C_{n_\beta}} \text{ (ZERO RUDDER)}$$

eq 6.12 6.7

$$\phi = \frac{-\,C_{y_\beta}\,\beta}{C_L} \text{ (ZERO RUDDER)}$$

eq 6.13 6.8

$$\frac{L_0 - L_i}{W}\,C_L\,\frac{y_p}{b} + C_{\ell_\beta}\beta + C_{\ell_{\delta_a}}\,\delta_a = 0$$

eq 6.14 6.10

$$\delta_{a\,\text{Equilibrium}} = -\frac{1}{C_{\ell_{\delta_a}}}\left\{\frac{L_0 - L_i}{W}\,C_L\,\frac{y_p}{b} + C_{\ell_\beta}\,\beta\right\}$$

eq 6.15 6.10