Flight Testing of Fixed-Wing Aircraft

Flight Testing of Fixed-Wing Aircraft

Ralph D. Kimberlin
University of Tennessee
Knoxville, Tennessee

EDUCATION SERIES
Joseph A. Schetz
Series Editor-in-Chief
Virginia Polytechnic Institute and State University
Blacksburg, Virginia

Published by
American Institute of Aeronautics and Astronautics, Inc.
1801 Alexander Bell Drive, Reston, VA 20191-4344

American Institute of Aeronautics and Astronautics, Inc., Reston, Virginia

1 2 3 4 5

Library of Congress Cataloging-in-Publication Data

Kimberlin, Ralph D., 1940–
Flight testing of fixed-wing aircraft / Ralph D. Kimberlin.
p. cm. -- (AIAA education series)
Includes bibliographical references and index.
ISBN 1-56347-564-2
1. Airplanes--Flight testing. I. Title. II. Series.

TL671.7.K54 2003
629.134'53--dc21

2003008178

Foreword

Flight Testing of Fixed-Wing Aircraft by Ralph D. Kimberlin is an authoritative presentation of this important phase in the development of new types of aircraft. Kimberlin is particularly well qualified to write on this subject having spent his early career in the U.S. Air Force followed by assignments in industry as Chief Test Pilot at Rockwell General Aviation and Piper Aircraft Corporation. He served also as Chair of the AIAA Flight Test Technical Committee. He has been designated as a Federal Aviation Agency (FAA) Consultant Flight Test Pilot and Flight Analyst.

The structure of this text is tailored to the special needs of teaching design and therefore should contribute greatly to the learning of the design process that is the crucial requirement in any aeronautical engineering curricula. The 35 chapters of this book are divided into three parts. Part I: Performance Flight Testing, Part II: Stability and Control Flight Testing, and Part III: Hazardous Flight Tests. A brief listing of some of the topics of each part illustrates the broad scope of his text. Part I covers FAA requirements, propeller and jet engines, level performance, range and endurance, climb, turn, drag determination, and takeoff and landing. Part II covers stability and control, longitudinal static and dynamic stability, turning performance, lateral stability, directional control, and flying; and Part III covers stall and spin testing, and dive testing for flutter, vibration, and buffeting.

The AIAA Education Series of textbooks and monographs, inaugurated in 1984, embraces a broad spectrum of theory and application of different disciplines in aeronautics and astronautics, including aerospace design practice. The series also includes texts on defense science, engineering, and management. The books serve as teaching texts for students as well as reference materials for practicing engineers, scientists, and managers. The complete list of textbooks published in the series can be found on the end pages of this volume.

J. S. Przemieniecki
Editor-in-Chief (Retired)
AIAA Education Series

Table of Contents

Preface

This book was intended to fill two purposes. The first is as a text for those teaching courses in fixed wing flight testing. The second is as a reference for those involved in flight test on a daily basis or those who need knowledge of flight testing to manage those activities. The book is divided into three sections: Part I: Performance, Part II: Stability and Control, and Part III: Hazardous Flight Tests. The first two sections were arranged so that they might be taught as semester courses at the upper level undergraduate or graduate level. The third section on hazardous flight tests provides information based upon 30 plus years of experience in either performing or directing such tests. As such, it may serve as a reference for those about to become involved in such testing and provides some hard-learned safety information.

The measurement of performance during an airplane's flight testing is one of the more important tasks to be accomplished during its development as it impacts on both the airplane's safety and its marketability. Performance sells airplanes! If an airplane is a poor performer it will have a short production run. However, the measurement of performance, other than the calibration of the pitot-static system and the measurement of stalling speed, should be delayed until near the end of the development flight test program. The reason is that during development flight test the airplane configuration may change in order to have acceptable handling qualities, and changes in configuration will affect the airplane's drag. The test pilot and flight test engineer should be warned, however, that the politics of airplane development will exert pressures to measure performance early, which could result in an inefficient flight test program due to the requirement to repeat the performance tests after each developmental change in configuration.

This book discusses performance for both propeller-driven and jet aircraft. However, its emphasis is on propeller-driven aircraft since much of the methodology for testing of propeller-driven aircraft has been lost with time. The author recommends that the pilots and engineers of jet propelled aircraft also consult current publications of the U.S. military flight test organizations for additional information on the flight testing of jet aircraft.

The section on stability and control provides the classical flight test methods for measuring and evaluating these parameters and, as such, is aimed at the light aircraft industry and the new group of general aviation airplanes under development. Although more modern techniques exist, such as parameter identification, the cost and expertise required to use such methods prohibit their use on most small airplane programs. Also, the FAA Certification Regulations are still based upon the classical techniques.

Assuring that the airplane has adequate stability and control should be the first order of business in a new airplane development program, and in a book that was intended as only a reference for the flight test community these should also come first. However, from the standpoint of college courses in flight testing, it is better to address performance first as it is likely to appear in that position in most curriculum.

Ralph D. Kimberlin
May 2003

Part 1
Performance Flight Testing

1 Introduction

1.1 Introduction to Flight Testing

Modern aircraft are complex integrated systems with the propulsion, avionics, and aerodynamics blended together to achieve optimum performance, stability, and control.

The flight testing of such aircraft is an endeavor that involves a number of engineering disciplines in addition to the study of the man-machine interface that we know as human factors. Engineering disciplines such as aeronautical, mechanical, electrical and structural all come together at this phase of an aircraft's development. In addition, it is at this time that the human crew, which has both tremendous capabilities and finite limitations, must begin to interact with the aircraft and its systems. That interaction must be examined from the standpoint of human workload, which has a component of human psychology and industrial engineering. In addition, the management of a flight test program for a modern aircraft requires management skills that are not often included in most engineering curriculums. As a result of this mix of disciplines, flight testing can not be claimed by any one and is therefore a discipline unto itself.

The flight test comes at the end of the aircraft design process and is a unique part of it, as one of the purposes of the flight test is to validate and refine the design. This means that changes to the design will continue to be made to the aircraft during the flight testing, as a result of the testing. Such changes to the design may affect the results of testing previously done, requiring it to be reaccomplished. We can see then that unless our flight test is properly planned it could result in a never-ending series of tasks. Additionally, because the flight test comes at the end of the aircraft's development, it will face the pressures of schedule brought on by the economics of any product development. Since flight testing is also a hazardous endeavor with important requirements for the safety of the aircraft and its crew, we must have a well-managed program to avoid accidents while accomplishing the program in a timely manner. Therefore, the flight test team has a very difficult and exciting task.

The principal purpose of today's flight testing is to determine if this complex aircraft and its crew can safely accomplish its intended mission. Other purposes may include collection of aerodynamic, power plant or systems data, and research into these or other related fields.

1.2 Types of Flight Tests

There are a number of reasons to flight test. One reason has been the desire of man to push the frontiers of knowledge. This desire results in the research flight test. Examples are the Wright Brothers' research flights at Kitty Hawk and the tests of the X-series of aircraft during the 1940s, 1950s, and 1960s. Such flight tests continue today as man tries to expand his knowledge of aeronautics and space.

Another type of flight test is that for product development. Aircraft companies in their attempts to make profits for their stockholders must develop new products. These tests, unlike research flight tests, may not be to push the state of the art. Their purpose is to determine the characteristics of the new product and to determine and solve problems with the product.

A third and important reason is to determine if the new vehicle will accomplish its intended mission. A 10-place aircraft that can only carry 2 passengers is an example of an aircraft that does not meet its mission requirements. If the aircraft does not meet its mission, additional development may be required.

The final reason for flight testing is to comply with established requirements and regulations for safety of flight. These flight tests are the purview of government organizations such as the Federal Aviation Administration (FAA), which conducts or directs such flight tests in the United States. Other countries have similar organizations, and in most of the world it is not possible to sell an aircraft—other than a home-built kit—to the public without it having successfully completed these tests.

1.3 Sequence of Flight Testing

In general, the development of a new aircraft should follow a set sequence to have a safe and efficient flight test. First, the aircraft should receive a number of preliminary ground tests leading up to its first flight. If there is more than one aircraft in the flight test program, some testing may be conducted in parallel.

Certification testing usually comes at the end of the development program. The sequence of certification testing is somewhat different than developmental testing.

1.4 Planning the Test Program

The success or failure of the flight test program will hinge on how well it is planned. The planning process will allow the flight test team to think through the safest and most efficient way to administer the program. In conducting this planning the test team should first answer the following questions:

1) What are the purpose and objectives of this test program?
2) What specifications or regulations must be met?

All too often testing is started without a defined purpose and specific objectives. Such an approach always results in a larger expenditure of effort than does one in which the purpose and objectives are well defined.

Once the test objectives have been defined (and agreed upon by all concerned parties) one can set about to determine what regulations and/or requirements apply. In most cases these requirements may need to be negotiated with the aircraft certification agency (Federal Aviation Administration in the United States) who has responsibility for aircraft certification.

1.4.1 Test Program Plan

Once the purpose and objectives have been firmly established and the certification regulations that apply to the test defined, one is ready to start designing the tests.

First, we should decide what types of tests and test methods best satisfy our purpose and objectives. Next, we need to group the tests so as to reduce the total number of flights required. If the test program has more than one test aircraft, the planning needs to include what testing will be accomplished on each aircraft. Once this has been accomplished the instrumentation requirements for each aircraft can be established.

At this stage in the planning process the instrumentation engineer, if one exists, should be included to assist in planning the instrumentation requirements. This position can be of great assistance in the planning of the instrumentation required to meet the test objectives along with pointing out excessive requirements for instrumentation and system accuracy, which have a great impact upon costs. However, the instrumentation engineer should not be allowed to dominate this part of the planning process.

Although it may appear obvious, the project test pilot and other test crew members should be involved in every phase of the planning. They can be of particular assistance in planning the testing sequence and in establishing the safety limitations. Their understanding of the purpose of each test, and where it fits into the overall test program, will be invaluable as the testing proceeds.

If the test program requires off-site testing, such as high-altitude takeoff and landing tests, the planning should include any logistic, instrumentation, and data collection/reduction support required at the off-site.

If one has conducted previous test programs, one should review the lessons learned from those programs. In addition, one should ensure in the planning process that provisions are made for collecting and documenting, for future use, the lessons learned from the current test program. Smart testing does not "reinvent the wheel."

During the planning process, brainstorming sessions are well worth the time spent. During these sessions, attempt to cover all of the possibilities. If we have designed a good test plan, problems that arise during the testing will be less likely to have an adverse impact upon our test schedule. The shorter the test schedule the lower the costs.

1.4.2 Planning Individual Flights

Using the sequence developed in the test plan, individual flights should be planned so that they include more items than can be accomplished during a single flight. These items should be organized so that the most important items are accomplished first. In this manner, a flight will not be wasted if the instrumentation for the primary items fail because the back-up items can be accomplished. All test items should be thoroughly planned with test altitude ranges, power settings, trim airspeeds, airspeed limits, and other test-critical items considered. Airspeeds should be in the values that the pilot will observe on the instrument. Once the flight has been planned the flight data card should be prepared.

1.4.3 Flight Data Cards

Flight data cards can be of the 5 × 7-card variety if data must be taken on a kneeboard or of the 8.5 × 11-notebook type if data are taken on a clipboard. It is not a good idea to plan these cards out in detail more than a day in advance of the flight since the requirements of an individual flight may change as the flight test progresses. These data cards may be personalized, but they should all contain the following information.

General Data
1) date of the test
2) type of aircraft
3) aircraft N number or serial number
4) purpose of the test
5) aircraft takeoff gross weight and center of gravity or a loading number
6) time of day of takeoff and landing
7) total flight time
8) configuration or a configuration number
9) test technique

The data card may be so arranged as to have the above data as the heading of the card, and the cards may be standardized to your liking with a large number of blank boxes for test specific data.

For test-specific data the cards should be arranged for ease of data taking and should contain the following information.

Test Specific Data
1) trim conditions (airspeed, altitude, ambient temperature, power setting, and so forth) if required by the type of test
2) test limitations (airspeed, altitude, control force, and so on)
3) all required data arranged in the order in which it will be collected with the most likely to change data shown first
4) comments on test conditions (such as air turbulence), specific data points, or other relevant comments

1.5 Governing Requirements and Regulations

The federal regulations for the certification of fixed-wing, propeller-driven aircraft date back to the Air Commerce Act of 1926 and were codified in the U.S. Department of Commerce Bureau of Air Commerce Aeronautics Bulletin

No. 7, "Airworthiness Requirements for Aircraft." This bulletin was amended to Aeronautics Bulletin No. 7-A (Ref. 1) on 1 October 1934. Bulletin No. 7-A is 58 pages long, including figures, and applies to aircraft built during this time period. It still applies to certain antique aircraft. Special requirements for transport aircraft were covered by Aeronautics Bulletin No. 7-J.

In the late 1930s the Civil Aeronautics Authority was established and issued replacements for Bulletin No. 7. This regulation was Civil Air Regulation (CAR) Part 4. It recognized the need for separate regulations for small aircraft and larger transport aircraft with Part 4 being for small aircraft and Part 4T covering transport aircraft. Later when helicopters appeared on the scene certification regulations were added for these aircraft. The Civil Aeronautics Act of 1938 resulted in the issuance of new regulations: Civil Air Regulations 4A for small aircraft and 4B for transport aircraft. Later these were changed to Civil Air Regulation Part 3 for small aircraft (defined as aircraft not exceeding 12,500 lb takeoff gross weight [TOGW]) and CAR Part 4b for transport aircraft (defined as those aircraft with TOGW in excess of 12,500 lb). Corresponding Civil Aeronautics Manuals (CAM 3 (Ref. 2) and CAM 4b (Ref. 3)) gave the regulation along with accepted interpretations of certain regulations including acceptable methods of conducting the flight test. Aircraft in the restricted category, such as agricultural aircraft, were certified under CAR Part 8 and its corresponding CAM 8 (Ref. 4).

Since Civil and Federal Air Regulations set the Type Certification Regulatory Basis for an aircraft as the regulations that applied when the manufacturer applied for the *original* type of certificate, many of today's light aircraft and some transport aircraft still use the CARs as their certification regulations.

1.5.1 FAA Regulations

In 1958 the Civil Aeronautics Agency became the Federal Aviation Administration and new aircraft certification regulations were issued in 1965. They were FAR Part 23 (Ref. 5) for small airplanes and FAR Part 25 (Ref. 6) for transport airplanes. Originally they were nearly the same as the older corresponding CARs; however, politics, changes in technology, and aircraft accidents have caused them to be revised and changed considerably from the original CARs. Fig. 1.1 shows the development of the federal aviation certification regulations.

1.5.2 FAA Flight Testing Advisory Circulars

Since the technical details of how to perform a given flight test are not provided in the FARs as they were in the CAM material, the FAA, in the 1970s, issued "how-to" documents under FAA orders. These documents were FAA Order 8110.7 "Engineering Flight Test Guide for Normal, Utility and Aerobatic Category Aircraft"[7] and FAA Order 8110.8 "Engineering Flight Test Guide for Transport Category Aircraft."[8] However, FAA orders are only available to individuals with FAA responsibilities and are not available to the general public. As a result, the FAA rewrote these documents and issued them

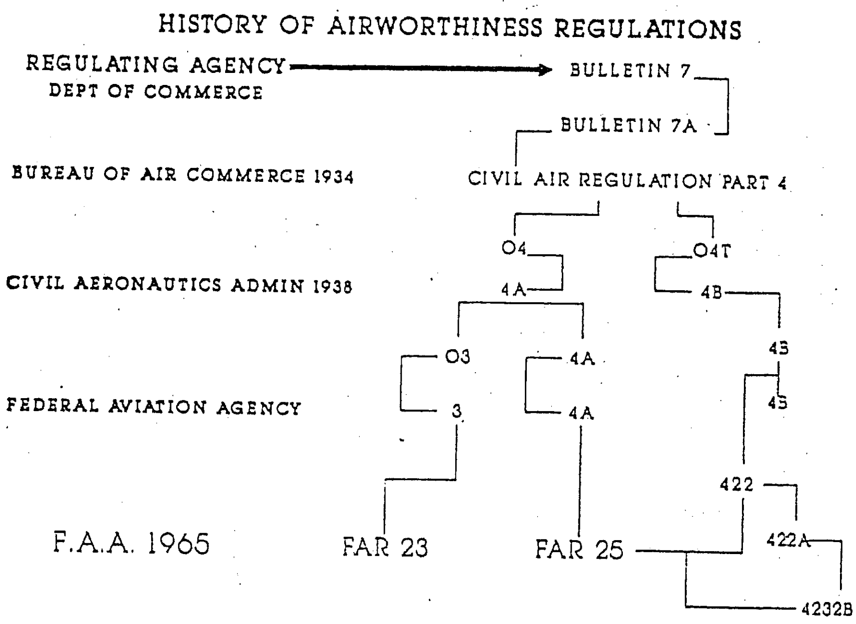


Fig. 1.1 History of U.S. airworthiness regulations.

as advisory circulars. Advisory Circular No. 23-8A is entitled "Flight Test Guide for Certification of Part 23 Airplanes."[9] This advisory circular is complete and replaces the FAA Order 8110.7. The replacement advisory circular applying to transport aircraft is AC No. 25-7 "Flight Test Guide for Certification of Transport Category Airplanes."[10]

One should understand that advisory circulars and FAA orders do not have the force of law as do FAA regulations, and the flight test methods and data reduction techniques given in these documents are not the only acceptable methods of performing the flight test. However, if an individual chooses to use a method other than one given in the advisory circular, that person is responsible for convincing the FAA pilots and engineers that it is a valid method.

1.5.3 Other Requirements

To sell airplanes or aeronautical equipment anywhere other than the United States, one must ensure that the requirements of the foreign nation are met. Although many countries accept the FARs, others, such as European nations and Australia, do not. Many of these countries certify their aircraft under the Joint Airworthiness Regulations (JAR). Although considerable effort has been made by all parties to ensure that the current FARs and JARs are equivalent, if the aircraft was approved under an earlier regulation, such as CAR 3, the JARs should be consulted to ensure that all requirements are tested and comply.

1.6 The Atmosphere

To understand the performance and handling qualities of an aircraft the test pilot and flight test engineer must understand the medium in which it flies. The Earth's atmosphere is a mixture of several gases. Included (in approximate percent by volume) is nitrogen at 78%, oxygen at 21%, water vapor at 0 to 4%, and traces of argon and other rare gases.[11] Nearly all of the water vapor is contained within the first few thousand feet of the atmosphere. This concentration of water vapor sometimes presents problems for the flight tester since water vapor is much lighter than dry air.

The pressure and temperature of the atmosphere is constantly changing, which affects its density through the gas laws. This causes the performance of an airplane to vary with the time of day and the season of the year. As a result, a standard was needed upon which performance comparisons could be made. This resulted in the adoption of a standard atmosphere for flight testing.

The atmosphere does not have a well-defined upper limit since density of the air decreases slowly as altitude is increased. However, the atmosphere is divided into four major divisions that conform to certain physical characteristics. In fixed-wing flight testing we are only concerned with the first two of those divisions.

The part of the atmosphere closest to the Earth's surface is called the troposphere. Its upper limit is variable with location on the Earth and season of the year but is taken in standard atmospheres to be 36,089 ft. Temperature in this part of the atmosphere decreases with height. This temperature lapse rate has a significant effect on weather, most of which is contained within the troposphere. The division between the troposphere and the next layer of atmosphere is called the tropopause.

As we increase in distance from the Earth's surface, the next part of the atmosphere is the stratosphere. This part extends from the tropopause to an altitude of roughly 50 miles.[11] Originally it was thought that the temperature remained constant throughout this altitude band; however, more recent data show that the temperature only remains constant between the tropopause and 14 miles of the stratosphere.[11] Since propeller-driven aircraft can only operate in the lower regions of the stratosphere we will not concern ourselves with what goes on at those extreme altitudes.

The atmosphere changes constantly with variations in pressure, temperature, and quantity of water vapor. Therefore, observed flight test results obtained on one day will not be the same as those obtained on another day. This variance gives rise for the need to have a standard day, or atmosphere, to which we can correct flight test results obtained on different days.

1.6.1 Standard Atmosphere

Every few years the National Oceanographic and Atmospheric Administration (NOAA) tabulates a new standard atmosphere. This is because, in addition to seasonal and daily changes, the atmosphere's parameters change as the world grows older. However, the changes in each new standard atmosphere are small. As a result, the flight testing community and the FAA have settled upon the 1962 U.S. Standard Atmosphere.[5]

The 1962 U.S. Standard Atmosphere assumes the following:[11]

1) The air is a perfect gas.
2) The air is dry.
3) The standard sea level conditions are: $T_{SL} = 15°C = 288.15$ K, $P_{SL} =$ 29.92 in. Hg = 760 mm Hg.
4) The tropopause occurs at 36,089 ft.
5) The temperature decreases linearly with altitude to the tropopause.
6) The gravitational field decreases with altitude. At sea level the acceleration due to gravity is 32.1741 ft/s (Ref. 2).
7) The temperature at the tropopause is −56.5°C or 216.65 K.

This atmosphere is tabulated or can be defined by a set of equations given in the next section, which can be programed into a computer.

1.6.2 Atmospheric Variables

On a standard day below the tropopause, the temperature decreases linearly with altitude increase. This can be defined by the equation:[12]

$$T_a = T_{SL} - (\mathrm{d}t/\mathrm{d}H)H \tag{1.1}$$

where
$\mathrm{d}t/\mathrm{d}H$ = the temperature lapse rate
H = the geopotential altitude

This equation can be restated in a more useable form using the pressure altitude:[13]

$$T_a = T_{SL}(1 - 6.87535 \times 10^{-6} H_P) \tag{1.2}$$

where
H_P = the pressure altitude

The standard temperature ratio, θ, can then be defined as:[13]

$$\theta = T_a/T_{SL} = (1 - 6.87535 \times 10^{-6} H_P) \tag{1.3}$$

The pressure reduces with altitude by the relation:[12]

$$\mathrm{d}P/\mathrm{d}h = -g\rho \tag{1.4}$$

where
g = the acceleration due to gravity
ρ = the air density

Assuming the air to be a perfect gas as is assumed in the 1962 U.S. Standard Atmosphere, the density is then defined as:[12]

$$\rho = P/RT = P/R(T_{SL} - Lh) \tag{1.5}$$

where
R = the gas constant for air
L = the temperature lapse rate, dT/dh

This equation may also be written in terms of pressure altitude as[13]

$$\rho = \rho_{SL}(1 - 6.87535 \times 10^{-6} H_P)^{4.2561} \tag{1.6}$$

If we substitute Eq. (1.5) into Eq. (1.4) we can write:[12]

$$dP/P = -g\,dh/R(T_{SL} - Lh) \tag{1.7}$$

from which we can find the pressure ratio, δ (Ref. 12):

$$\delta = P/P_{SL} = (1 - Lh/T_{SL})^{g/RL} = (T/T_{SL})^{g/RL} \tag{1.8}$$

which can be written in terms of pressure altitude as:[13]

$$\delta = P/P_{SL} = (1 - 6.87535 \times 10^{-6} H_P)^{5.2561} \tag{1.9}$$

To determine true airspeed and density altitude we need to know the density ratio, σ (Ref. 12):

$$\sigma = \rho/\rho_{SL} = (T/T_{SL})^{g/(RL-1)} \tag{1.10}$$

Eqs. (1.3), (1.6), and (1.9) can be used to calculate the 1962 U.S. Standard Altitude Table in terms of the pressure altitude for altitudes below the tropopause. Equations for altitudes above the tropopause can be found in a number of references.[11,13]

However, during flight testing, when going from test conditions to standard day conditions, only Eq. (1.9) above is of value. Eq. (1.9) can only be used if you have set your test altimeter to pressure altitude (29.92 in. Hg in the Kollsman window) and not to the local altimeter setting. To determine other test day parameters such as temperature ratio θ and density ratio σ the following equations should be used:

Temperature Ratio

$$\theta = (T_a + 273.15)/288.15 \tag{1.11}$$

Density Ratio

$$\sigma = \delta/\theta \tag{1.12}$$

where the pressure ratio δ comes from Eq. (1.9) and the temperature ratio θ comes from Eq. (1.11). This conversion takes test day data to standard day conditions.

1.7 Aircraft Weight and Center of Gravity

The aircraft loading, or weight and balance as it is sometimes called, is one of the most important items in the flight test as both the aircraft's performance and its stability and control are functions of the loading. The most accurate way to determine the aircraft's loading is an actual weighing of the aircraft, with crew, ballasted to the test loading. However, this is not always possible since it is time consuming and costly. If an accurate empty weight and center of gravity (c.g.) have been determined through weighing then the loading can be calculated for noncritical tests. If the tests are considered critical to either flight safety or certification then the loading should be determined by weighing in the flight-ready configuration.

The weight and c.g. requirements are covered in CAR 3.71 through 3.76 (Ref. 2) and in FAR 23.21 through 23.31 (Ref. 5) and are discussed in AC 23-8A in paragraphs 6 through 10 (Ref. 9).

1.7.1 Weighing and Ballasting Techniques

In weighing an aircraft, the aircraft should be in a level attitude. Different designs use different methods for achieving a level attitude, but common methods are the use of leveling points on which a spirit level is placed or a point from which a plumb bob is suspended with a corresponding mark to show level.

The scales used in weighing the aircraft can be the platform type or the electronic type. One advantage to the electronic type, or load cell type, is that they are mounted on top of aircraft jacks and the aircraft is lifted from the surface with the jacks. If individual jack extension is adjusted the aircraft can also be leveled without compressing landing gear struts or flattening tires as is usually required to level the aircraft when platform scales are used. In most flight test programs of prototype aircraft a set of jack-mounted electronic scales will pay for themselves in time saved in aircraft weighing.

FAA Advisory Circular EA-AC 65-9A "Airframe and Powerplant Mechanics General Handbook"[14] Chapter 3, contains considerable detail on how to weigh an aircraft and additional weight and balance information may be found in AC 91-23A "Pilot's Weight and Balance Handbook."[15]

In ballasting the aircraft for flight tests several techniques and types of ballast may be used. The types of ballast include both solid and liquid.

Solid ballast can be bags of sand or cement, however, more compact solid ballast consist of lead shot as used in shotgun shells. This shot is available in 25-lb bags of various sizes from most gun shops, and the bags are small enough to be placed in nearly any location in the aircraft. Although lead bars can also be used, the lead shot is much safer in the advent of an accident as it is not as likely to become a missile as is a lead bar. Lead shot may also be placed in shot tanks that have dump valves allowing the shot to be dumped overboard in an emergency.

Liquid ballast in the form of water may also be used by securing water tanks in the fuselage of the aircraft. This form of ballast may also be dumped in an emergency.

Both solid and liquid ballast have their advantages and disadvantages which may be a function of the test to be performed. Whatever type of ballast is used, the FAA advises in AC 23-8A that attention be paid to both the longitudinal and vertical c.g. of the aircraft in ballasting. So if a solid ballast like lead shot is used, it should not be placed on the cockpit floor but in a location above the floor that would represent the vertical c.g. of the crew, passengers, and cargo.

1.7.2 Weight and c.g. Requirements for Testing

Both CAR 3 and FAR 23 require that the aircraft be loaded to certain weights and c.g. for certain tests. In general, the FAA requires that the aircraft be tested at the most forward c.g. at maximum TOGW for performance. For stability and control the aircraft is tested at both forward and aft centers of gravity at maximum TOGW and at the most forward c.g. regardless of weight for control. The tolerances on weight and c.g. for testing are provided in the applicable FAA regulation and in Sec. 1.8 of this chapter.

1.7.3 Determination and Use of the Mean Aerodynamic Chord

In flight testing it has been found that plotting the c.g. in percent of the mean aerodynamic chord (MAC) has more use than plotting it in inches aft of an arbitrary datum.

The aircraft's datum, from which all locations on the airplane are measured, is determined by the airplane's designer. As a result, the location of the datum is likely to be different for different airplane designs. An example of this difference is shown in Fig. 1.2 (Ref. 14). As a result it is nearly impossible to compare one design to another using the center of gravity stated in terms of inches aft of datum.

On the other hand, if the c.g. is referenced in terms of percent of the MAC it is easy to compare one aircraft to another. For instance, most tail aft aircraft have c.g. ranges of 10–30% when expressed in terms of MAC, while tail first aircraft will have c.g. ranges from -10 to $+5\%$ MAC. Therefore, if you were testing a tail aft aircraft that would still meet stability requirements at 35% MAC and control requirements at 5% MAC you would know that your aircraft had a wider c.g. range than most of that configuration (tail aft).

There are several methods of determining the MAC. Mathematically, the MAC can be defined as:[16]

$$MAC = 2\int_{-s}^{s} c^2 \mathrm{d}y/S \tag{1.13}$$

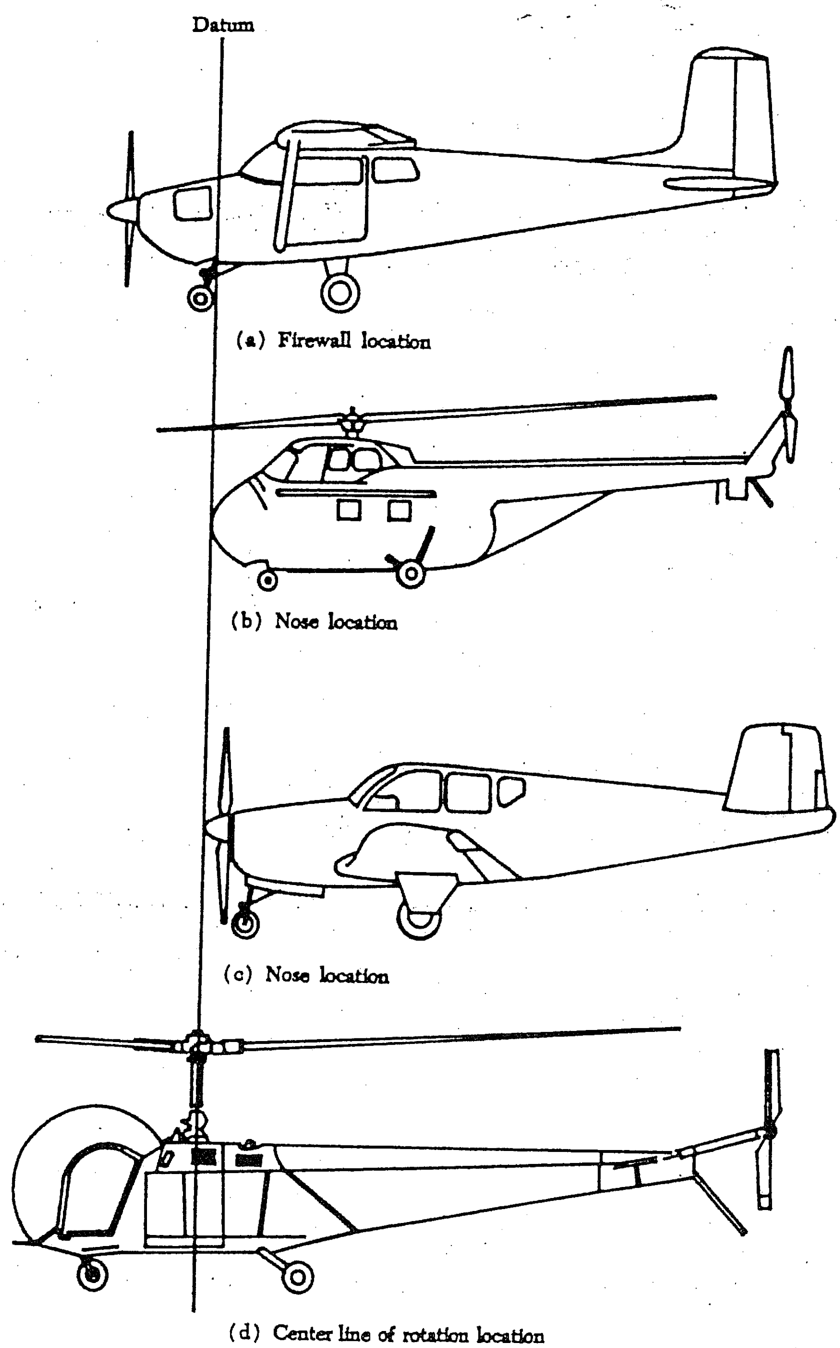


Fig. 1.2 Various locations for aircraft datum.[14]

where
s = the semi-span, $b/2$
c = the local chord length
y = the span wise location
S = the wing planform area

For straight tapered wings, the MAC can be written as:[16]

$$\mathrm{MAC} = C_R(2/3)(1 + \lambda + \lambda^2)/(1 + \lambda) \tag{1.14}$$

where
C_R = the root chord length
λ = the taper ratio, C_T/C_R

If the wing has several sections each with a different taper ratio, then the MAC of each section can be calculated and an equivalent straight-tapered wing determined.

In addition to the length of the MAC, the location of its leading edge with respect to the datum must be determined. The distance of the leading edge of MAC from the apex of a swept or tapered wing is expressed as:[16]

$$X_{LE} = C_R[(1 + 2\lambda)/12]A \tan \Lambda_0 \tag{1.15}$$

where
X_{LE} = the distance from the wing apex to the leading edge of the MAC
C_R = the root chord of the wing
λ = the wing taper ratio
A = the wing aspect ratio, b^2/S
Λ_0 = the sweep angle of the wing leading edge

The distance from the aircraft datum is then added to the value of X_{LE} obtained in Eq. (1.15) to determine the location of the leading edge of MAC in inches aft of datum as is shown in Fig. 1.3 (Ref. 14).

To determine the c.g. location in percent of MAC, the leading edge of MAC in inches aft of datum is subtracted from the c.g. in inches aft of datum and divided by the MAC in inches and multiplied by 100.

1.8 Flight Testing Tolerances

AC 23-8A (Ref. 9) states in paragraph 6(a)(4): "The purpose of the tolerances specified in FAR 23.21(b) is to allow for variations in flight test values from which data are acceptable for reduction to the value desired."

In AC 23-8A, the Federal Aviation Administration accepts the following tolerances for flight testing:

1) Airspeed: 3 kn or ±3%, whichever is greater
2) Power: ±5%
3) Wind (takeoff and landing tests): not to exceed 12% V_{S1} or 10 kn, whichever is lower

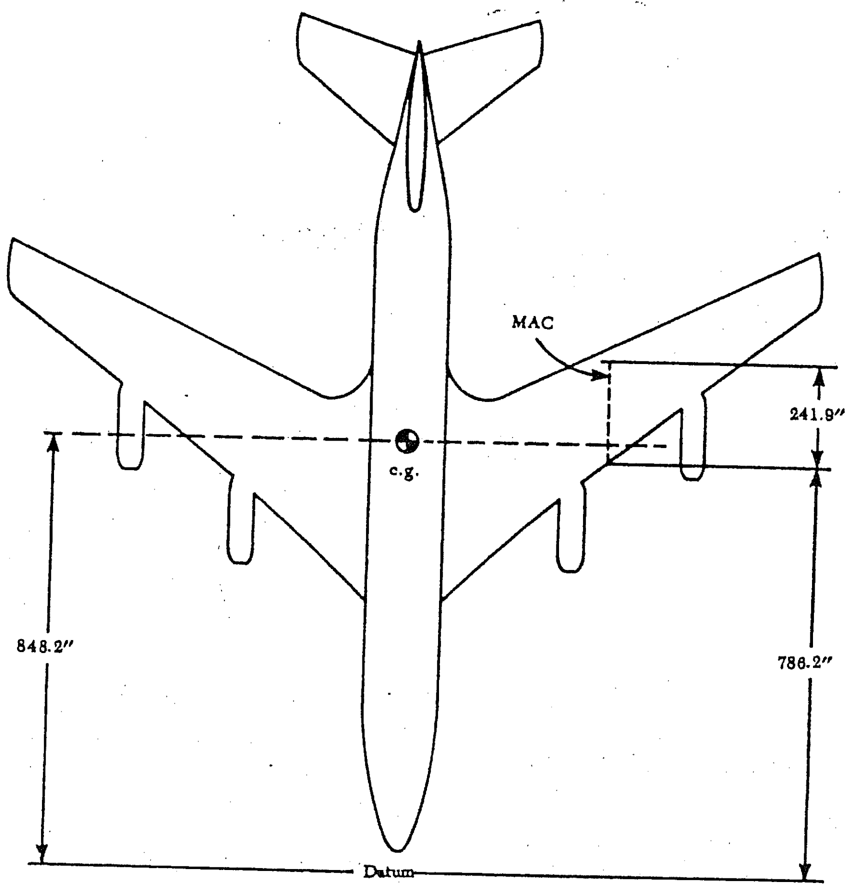


Fig. 1.3 The c.g. in relation to mean aerodynamic chord.[14]

4) Weight: +5% to −1% for weight critical items such as performance and +5% to −10% for nonweight critical items such as stability
5) Center of gravity: ±7% of total c.g. travel

References

[1]Aeronautics Bulletin No. 7-A, "Airworthiness Requirements for Aircraft," U.S. Department of Commerce Bureau of Air Commerce, Washington, D.C., 1934.

[2]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., March 1959.

[3]Civil Aeronautics Manual 4b, "Airplane Airworthiness: Transport Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1960.

[4]Civil Aeronautics Manual 8, "Aircraft Airworthiness; Restricted Category," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[5]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[6]Federal Aviation Regulation Part 25, "Airworthiness Standards: Transport Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[7]Federal Aviation Administration Order 8110.7, "Engineering Flight Test Guide for Small Aircraft," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1974.

[8]Federal Aviation Administration Order 8110.8, "Engineering Flight Test Guide for Transport Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1974.

[9]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

[10]Federal Aviation Administration Advisory Circular No. 25-7, "Flight Test Guide for Certification of Transport Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., March 1986.

[11]USAF Test Pilot School, FTC-TIH-70-1001, "Performance, Aerodynamic Theory, Vol. I," Edwards AFB, CA, 1973.

[12]Mair, W. Austyn and Birdsall, David L., *Aircraft Performance*, Cambridge Aerospace Series 5, Cambridge University Press, New York, 1992.

[13]Herrington, R. M., Maj., Shoemacher, P. E., Bartlett, E. P., and Dunlap, E.W., "USAF Flight Test Engineering Handbook," AFFTCTRN 6273, Air Force Flight Test Center, Edwards AFB, CA, May 1961, rev. June 1964, Jan. 1966.

[14]Federal Aviation Administration EA-AC 65-9A, "Airframe and Powerplant Mechanics, General Handbook," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1976.

[15]Federal Aviation Administration Advisory Circular No. 91-23A, "Pilot's Weight and Balance Handbook," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1977.

[16]Irving, F. G., *An Introduction to the Longitudinal Static Stability of Low-Speed Aircraft*, Pergamon Press, New York, 1966.

2
Methods for Reducing Data Uncertainty in Flight Test Data

2.1 Introduction

In most scientific endeavors, we must deal with errors or uncertainty in the data we collect. Flight testing is no exception. In fact, many of our error sources are unique to our field and require special attention. The purpose of this chapter is to examine the sources of error related to flight testing, to discuss some methods of eliminating or minimizing these errors, and to determine how to deal with the remaining uncertainty when we present our data. Flight testing is a complicated experimental technique. As a result, it combines most of the types of errors found in experimental methods. For instance, fixed, or systematic, errors are found in the instruments from which we collect our data. Random errors come from the atmosphere, from pilot technique, and from errors in taking readings. Gross blunder errors can come from poor test planning, pilot technique, and several other sources.

Although most flight tests can be performed as multisample experiments with considerable improvement in accuracy, the cost of performing the test normally limits us to a single sample experiment. This being the case, we must understand where the sources of our uncertainty lie, take steps to avoid or minimize that uncertainty, and then present the data in a manner that recognizes the remaining uncertainty.

2.2 Sources and Magnitudes of Errors

Before we can find the sources and magnitudes of errors we must first define errors. An error can be defined as the difference between the true value and the measured value. Although this definition seems simple enough, determining its magnitude from flight test data is difficult due to the number of potential error sources. Let us now examine some of these sources.

2.2.1 *Instrument Error*

Instrument error is one of the easiest errors to determine since it is determined in the laboratory. It consists of two types of error: hysteresis error and bias error.

Instrument hysteresis is the difference in instrument readings between increasing and decreasing values. Nearly all instruments have hysteresis (see Fig. 2.1). However, mechanical instruments are more prone to it than are other types. Instruments with large hysteresis should be discarded or repaired prior to use in flight tests.

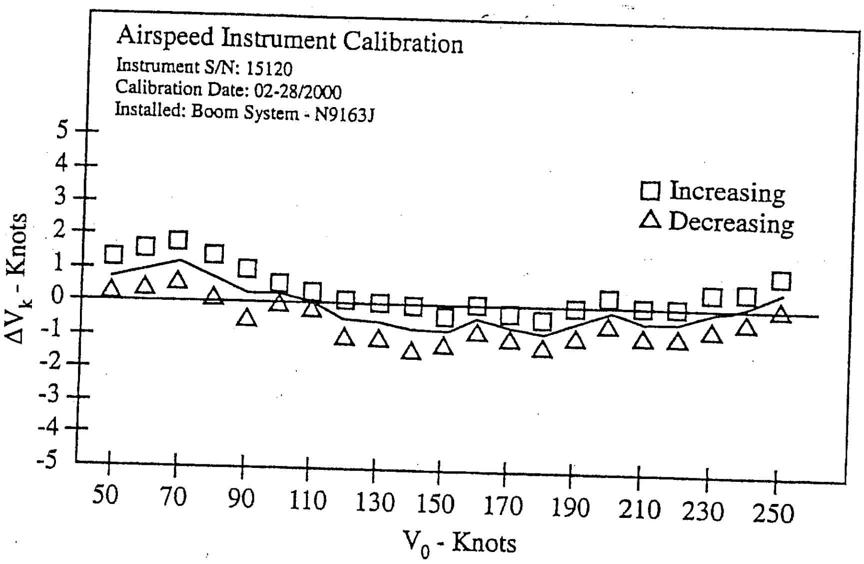


Fig. 2.1 Airspeed instrument calibration.

Bias error can be described as the difference between the correct reading and the average of the hysteresis band. For good instruments this error should not change significantly with time.

Flight test instruments should be calibrated for both bias and hysteresis on a regular basis. For normal testing, instruments should be calibrated every three to six months with the shorter interval being desirable. For critical testing, where flight safety is involved, or high accuracy is desired, instruments should be calibrated on monthly intervals or less.

The magnitude of instrument bias errors is not significant as long as it is a repeatable error and the instrument is not used as a pilot's flight instrument. The magnitude of hysteresis errors is significant and should not exceed $\pm 1\%$ of the scale reading.

Fig. 2.1 shows the method for plotting the instrument error for a mechanical instrument. Mechanical instrument error should be plotted with straight line variation between calibration steps since the mechanical friction in the instrument and friction bumps are more likely to make the instrument behave in this manner. Today, however, there is a tendency for new engineers to want to plot these data as a curve since curve fitting routines are available in most spread sheet programs. Curve fittings are acceptable for electrical transducers since their behavior follows a curve. However, mechanical instrument error should be plotted linearly as shown in Fig. 2.1.

One other item related to any data plotting: an old rule is that if one finds a data plot that has been dropped on the floor there should be enough information on the plot for the person who found the plot to find the owner and to have a good idea as to what the plot is about. Fig. 2.1 gives an example to follow.

2.2.2 Airspeed and Altitude Position Error

The error in airspeed and altitude readings caused by the location of the static pressure source on fixed-wing aircraft, and the static and total pressure source on helicopters, is determined by flight test. It is worth noting, however, that since the error is determined by flight test it is also subject to uncertainty. Since an accurate determination of airspeed and altitude is basic to all flight testing, having a high confidence level in the position correction is essential. Since the only way to obtain this confidence is through repeated sampling, this test is a definite candidate for a multisample experiment. In addition to multi-sampling, it is worthwhile to use several different flight test methods to obtain this data. If the data are repeatable using several methods, then the confidence in the position correction is greatly increased.

Since position correction data are subject to many of the causes of uncertainty to be discussed in the following sections, the use of curve fitting and statistical analysis techniques is necessary.

2.2.3 Reading or Discrimination Error

Reading error can occur in both hand- and automatic-data recording. For hand recording, errors can occur due to an indirect view of the instrument being read. This is sometimes called parallax error. Normally, these errors are small when compared to other hand-recording reading errors. Misreading errors are caused by powers of ten. If the instrument has not been consistently misread these errors are usually easy to identify.

Reading errors for automatic recording equipment are more correctly described as discrimination errors. In automatic equipment these errors are most pronounced in analog oscillograph and brush recorders where the width of the trace may provide the error magnitude.

2.2.4 Uncertainty Due to Atmospheric Conditions

The atmosphere can be the source of large random errors. Atmospheric temperature inversions and nonstandard temperature lapse rates can create data scatter that is difficult to explain or correct. Changes in wind velocity with altitude may also introduce error in data that are collected at several altitudes. Climb and descent data are particularly subject to temperature and wind effects.

Humidity or water vapor in the air can also create problems for the flight test engineer. Most water vapor in the atmosphere is found in the first 3000–4000 ft above sea level. This phenomena tends to make data collected in this region disagree with data collected at higher altitudes.

Atmospheric turbulence will always introduce errors in flight test data and should be avoided for all flying qualities and performance testing. The magnitude of atmospheric errors can be large and may not be correctable. Therefore, in flight testing, one should attempt to avoid atmospheric conditions that tend to introduce these errors.

2.2.5 *Errors Due to Pilot Technique*

Since most aircraft are flown by human pilots, we must be alert for errors caused by pilot technique. Errors due to pilot technique result from failures on the part of the pilot to properly trim the aircraft, to allow sufficient stabilization time on a data point, to properly configure the aircraft or engine, or to account for friction in a control system to name a few. The pilot can also be guilty of gross blunder errors by violating known principles of testing.

2.2.6 *Errors Due to Inaccurate Thrust or Power Determination*

For performance flight testing, knowing the correct value of thrust or power is essential. However, most aircraft engines have tolerances on the thrust or power that they develop for a given power setting. If it is not possible to calibrate the engine prior to the test, which is often the case, these tolerances become data uncertainty.

Typical values for the magnitude of these tolerances are +5 to −2% of the desired thrust or power value.

2.2.7 *Errors Due to Control System Friction or Hysteresis*

Both reversible and irreversible control systems suffer from problems of friction or hysteresis. These problems can introduce significant data scatter and error to control force data. In addition to error in flight test data, the friction—and breakout forces associated with it—have an adverse effect on the pilot's opinion of the airplane if they are large.

The magnitude of these errors is a function of control system design and maintenance. In general, the simpler the control system the smaller the friction or hysteresis.

2.3 Avoiding or Minimizing Errors

Since many of these errors just discussed are of unknown magnitude, they do not readily lend themselves to the error analysis technique used in other engineering disciplines. However, there have been techniques developed by the flight test community to minimize these errors.

2.3.1 *Instrument Calibrations*

To minimize instrument errors, some steps can be taken while applying the instrument correction. First, on tests such as climbs and accelerations the increasing side of the instrument hysteresis should be used. On descents and decelerations the decreasing side of the instrument hysteresis should be used. For level flight data the average of the increasing and decreasing hysteresis values should be used. A similar approach should be used for other types of tests.

2.3.2 Sample Size

As discussed previously the airspeed and altitude position error confidence level can be increased by increasing the sample size and applying the curve fitting techniques of statistical analysis. Nearly all flying qualities and performance flight test data can benefit from an increased sample size. However, increasing the sample size to increase confidence level in the data is a tradeoff with the increased cost of conducting the flight test. Before one can make the cost vs sample size decision, one should decide if the increased data confidence is worth the cost. In many cases it is not.

2.3.3 Methods to Avoid Reading Errors

There are several tricks to avoid reading errors on hand-recorded data.

Parallax errors may be avoided by the observer being positioned directly in front of the instrument to be read.

Misreading errors may be reduced by writing down the first one or two numbers prior to final stabilization on the data point. For example, the thousands and hundreds of feet in altitude may be recorded before reaching the data point, and upon reaching the data point the tens of feet are read. Similar approaches may be used on other parameters.

Additional data scatter may be avoided on hand-recorded data by determining which are the most important parameters and then reading those parameters first. Increased accuracy will also be obtained by reading parameters in the same sequence at each data point.

For automatic recording equipment, reading errors will be reduced by selecting larger rather than smaller scales for data traces and by reducing the amount of data to be collected on any one device. Digital rather than analog data recording may also offer advantages for certain types of data, however, this is not always true. For certain dynamic data analog recording is superior to digital.

2.3.4 Minimizing Atmospheric Errors

Errors caused by temperature inversions, nonstandard lapse rates, and wind shear may be avoided or reduced by using several techniques.

First, it has been found that collecting data through a temperature inversion will introduce uncorrectable errors in the data. Climb and descent data are particularly subject to these kinds of errors. One way of avoiding them is to have another aircraft climb through the altitude band where testing is to take place and record outside air temperature vs altitude. The test aircraft can then use this data to avoid temperature inversions and areas of turbulence.

The effects of wind shear may be reduced on climb and descents by performing these maneuvers at right angles to the prevailing wind and by repeating all climbs and descents on opposite headings. Opposite heading data is then averaged to come up with the corrected value of climb or descent for the given airspeed.

It is best to avoid high humidity days for flight testing. However, this is not always possible. In such cases, we may correct for humidity by taking wet and dry bulb temperature readings at each test altitude. With this data a psychometric chart may be used to determine the partial pressure e of the water vapor in the air at the time of test. Once this value is obtained the density ratio σ may be corrected using Eq. (2.1) (Ref. 1).

$$\sigma = 9.625(P - .375e)/(273 + t) \tag{2.1}$$

where
P = the atmospheric pressure in inches of mercury
t = observed air temperature in Celsius

Engine power will also be affected by humidity. There are two methods of correcting for humidity on reciprocating aircraft engines. The first method is to subtract the partial pressure of water vapor in the air from the instrument corrected manifold pressure prior to entering the engine power charts to determine horsepower. A second method that will work for a variety of engines is to use the following equation:[1]

$$HP(\text{dry air}) = HP(\text{moist air})\{P/(P - e)\} \tag{2.2}$$

2.3.5 *Minimizing Errors Due to Pilot Technique*

A good test pilot can trim and stabilize an aircraft in a minimum amount of time. However, all test pilots will not perform these acts in the same amount of time. Therefore, pilots should be given the amount of time they feel necessary to trim and stabilize on data points since most data errors due to pilot technique result from the pilot trying to rush through the procedure. It has been found that stabilization occurs faster if data points are approached from a higher airspeed rather than a lower airspeed.

Many gross blunder errors made by the pilot and crew can be avoided if test checklist are prepared and followed for each test and if the configurations and power settings are checked and rechecked prior to each data run.

2.3.6 *Minimizing Errors Due to Inaccurate Thrust and Power Measurement*

If possible prior to testing, the power plant should be calibrated while installed in the test aircraft. For jet aircraft this should be accomplished on a thrust stand and for propeller-driven aircraft using a torque meter. Although it is only possible to do the thrust calibration on the ground, these data may be used to adjust the engine manufacturers altitude data, thereby reducing some of the known error.

Other items for reducing installed thrust or power errors are accurate measurement of the engine inlet temperature and other engine parameters.

2.3.7 Minimizing Friction Errors

Prior to commencing flying qualities testing the control system friction and breakout forces should be measured and recorded. If these values are found to be excessive the control system should be inspected, cleaned, and lubricated, and any source of friction not inherent in the basic design corrected.

During data acquisition the pilot should attempt to work out the friction prior to taking readings or to obtain a friction hysteresis by obtaining increasing and decreasing readings. In addition, free return speeds should be obtained on longitudinal force data for aircraft with reversible control systems and breakout forces should be obtained for irreversible control systems.

2.4 Error Analysis

Even though we apply all of the steps and procedures previously mentioned, we will still find that our data have error or scatter. If we survey the engineering literature we find several methods for determining the uncertainty of a given experiment.[2–6] The problem with many of these methods, however, is that they require a knowledge of the magnitudes of the uncertainties for the individual parts that go to makeup the whole. In flight testing, we cannot define the magnitude of all of these contributing uncertainties. For instance, how do you define the magnitude of the uncertainty introduced by the pilot when it is mixed in with all of the other uncertainties? Although one may estimate the unknown magnitudes and perform the uncertainty analysis, the value of results from such analysis is questionable.

A common-sense approach may be a more effective way of dealing with uncertainty in flight test data. Several common-sense tests may be applied to the data to determine its validity. These tests can be described as: 1) test of consistency; 2) test against theory; and 3) correlation test.

2.4.1 Test of Consistency

As a general rule flight test results are smooth and consistent. If the data collected does not meet this test it is suspect. This does not mean that the data cannot have scatter. What it does mean is that there should not be large discontinuities in a smooth curve faired through the data.

2.4.2 Test Against Theory

If a theoretical method exists to generate the results being obtained by flight test, then the results from the flight test should generally agree with the theory. This is particularly true with respect to curve shapes if not absolute values. Most theories for aircraft performance, stability, and control are well established and flight test data should be in general agreement. However, one should be careful in applying this test since certain aerodynamic theory, such as that for stability and control, have simplifying assumptions, which may be violated in actual flight tests.

2.4.3 Correlation Test

This test may be used to answer many questions with regard to the accuracy of data from a given flight test. First, does the data correlate with other data of the same type? Does it correlate from pilot to pilot? If so, then the error due to pilot technique may be dismissed. Does it correlate from test method to test method. If so, then data confidence is greatly improved. If not, then it may be questionable.

Once data have been given and passed the above listed tests, then the final step is to apply a statistical analysis to the remaining scatter. One very commonly used statistical method is the least squares curve fit. This method is discussed in most probability and statistics books and will not be explained here. However, there are at least two items that should be discussed pertaining to applying a least squares fit to flight test data.

The first is that it is much easier to fit a straight line through scattered data than it is a curve. As a result, it is worthwhile to attempt to find some scheme that will normalize the data from a curve into a straight line prior to applying the curve fit. The normalization procedure of plotting PIW × VIW against VIW^4 to make PIW vs VIW data become a straight line is such an approach (Figs. 2.2 and 2.3).

Second, the use of the weighted least squares fit is preferable for flight test data since use of the weighting factor for individual data points allows one to give less weight to points that are obviously scattered. It also allows for pilot comment—on which were good or bad data points—to be considered in the data analysis.

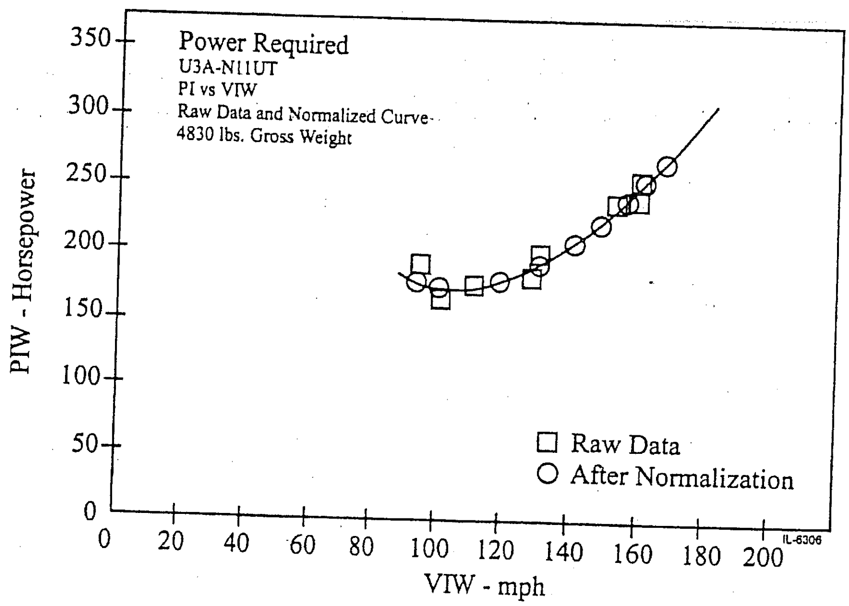


Fig. 2.2 Power required.

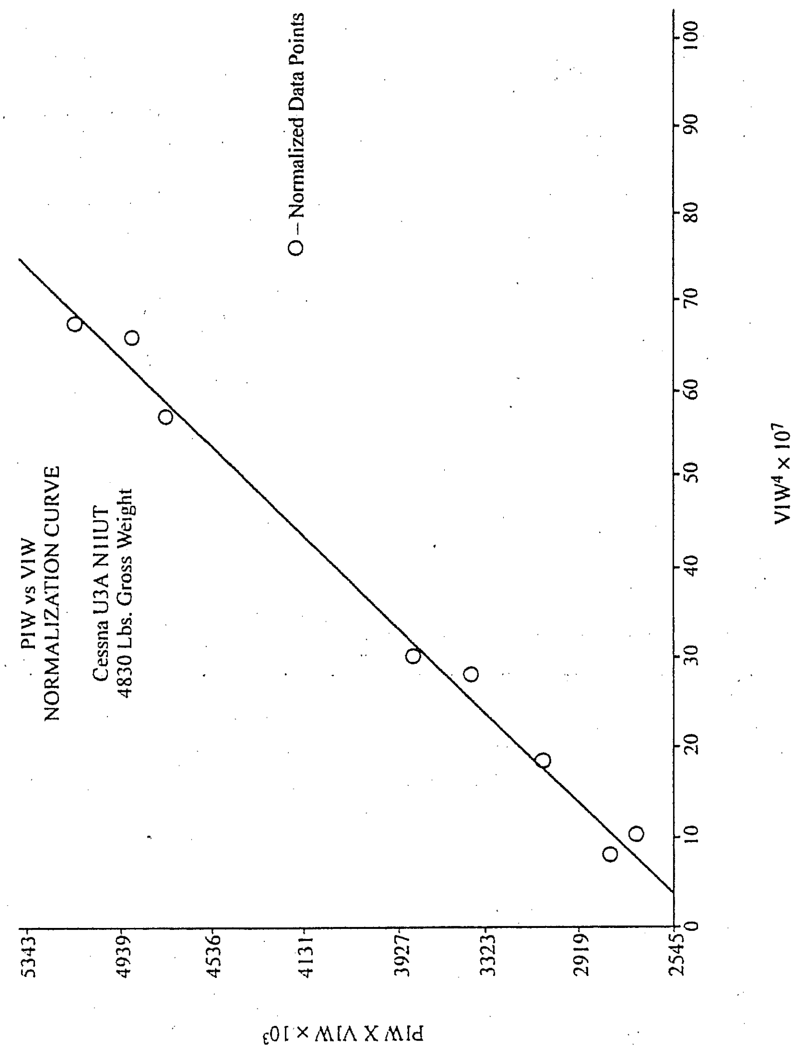


Fig. 2.3 PIW vs VIW normalization.

References

[1]Bilke, Paul F., Army Air Force Technical Report No. 5069, "Performance Flight Testing Methods in Use by the Flight Section," War Department Army Air Forces Material Command, Dayton, OH, 15 Jan. 1944.

[2]Beckwith, T. G., and Buck, N. Lewis, *Mechanical Measurements*, Addision-Wesley Publications, Reading, MA, 1969.

[3]Holman, J. P., and Gajda, W. J., Jr., *Experimental Methods for Engineers*, 3rd ed., McGraw-Hill Book Co., New York, 1978.

[4]Schenck, Hilbert, Jr., *Theories of Engineering Experimentation*, 2nd ed., McGraw-Hill Book Co., New York, 1968.

[5]Pool, A., and Bosman, D., "Basic Principles of Flight Test Instrumentation Engineering." AGARD-AG-160-Vol. 1, Technical Editing and Reproduction Ltd., Hartford House, London, April 1974.

[6]Tuve, G. L., and Domholdt, L. C., *Engineering Experimentation*, McGraw-Hill Book Co., New York, 1966.

3
Airspeed Systems Theory and Calibration

3.1 Introduction

An accurate measurement of airspeed and altitude is necessary for safe flying. This is especially true for flight testing. In addition to the error caused by the airplane's instruments, several other errors are associated with the pitot-static system. This chapter will discuss these errors, the methods for evaluating them, and the requirements for calibration and accuracy as defined in the Federal Aviation Regulations.

3.2 Federal Aviation Regulation Requirements

The FAA and its predecessors have always considered flight instrument accuracy to be important. Aeronautics Bulletin 7-A states: "The 'indicated' airspeed is defined as the speed which would be indicated by a perfect air-speed indicator, namely one which would indicate true air-speed at sea level under standard atmospheric conditions." It further refers the reader to Aeronautics Bulletin 26, section 6(A)(8), for further information on airspeed indicators. Current regulations require accuracies over specified ranges of airspeed and altitude.

3.2.1 Civil Aeronautics Requirements 3.663 Airspeed Indicating System

This regulation requires the airspeed indication system to be so installed as to indicate true airspeed at sea level (calibrated airspeed) under standard conditions within an allowable installation error of no more than $\pm 3\%$ of the calibrated airspeed or 5 statute mph, whichever is greater, between $1.3V_{S1}$ and V_C with the flaps up and at $1.3V_{S1}$ flaps down. The regulation requires the calibration to be made in flight.

3.2.2 Civil Aeronautics Requirement 3.665 Static Air Vent System

This regulation states that airplane speed, the opening and closing of windows, air-flow variation, moisture, or other foreign matter shall not seriously affect the accuracy of instruments that depend upon static pressure.

3.2.3 Federal Aviation Regulation 23.1323 Airspeed Indicating System

FAR 23.1323 is similar to CAR 3.663 except mph have been changed to kn and the ranges for the calibration have changed. In this regulation, the calibration range flaps up is from $1.3V_{S1}$ to V_{MO}/M_{MO} or V_{NE}, whichever is appropriate, and the flaps extended range is from $1.3V_{S1}$ to V_{FE}. The pitot-static system is required to have positive drainage for moisture and must have a heated pitot if IFR certification is sought.

Commuter category airplanes are required to have an additional calibration with the airplane on the ground between $0.8V_{1\min}$ and $1.2V_{1\max}$, which considers the approved ranges of altitude and weight. This calibration must be determined considering an engine failure at $V_{1\min}$. Duplicate systems are also required for commuter category aircraft.

3.2.4 Federal Aviation Regulation 23.1325 Static Pressure System

FAR 23.1325 is considerably more stringent than is CAR 3.665. In addition to requiring the items of CAR 3.665, this regulation contains requirements for drainage, chafing, and distortion of the tubing, and the materials used must be suited for the intended use and protected against corrosion.

In addition, a proof test must be conducted to demonstrate the integrity of the system. These tests are for both pressurized and unpressurized airplanes. For unpressurized airplanes, the test is to evacuate the static system to 1000 ft above the airplane's altitude and determine that the system does not leak down more than 100 ft in 1 min. For pressurized airplanes, the test is to evacuate the system until a pressure differential equal to the maximum pressure differential for which the airplane's cabin is approved is reached. Then, the system should not leak down more than 2% of the altitude equal to the cabin's differential pressure or 100 ft, whichever is greater.

Static pressure ports are also to be designed so that there will not be a significant change in the calibration when the airplane encounters icing.

If a system incorporates a primary and an alternate static source, it must be shown that when either source is selected the other is blocked off and that both can not be blocked off simultaneously.

The static pressure system must be calibrated in flight and the error at sea level on a standard day may not exceed ± 30 ft per 100 kn of airspeed in the speed range between $1.3V_{S0}$ and $1.8V_{S1}$. However the error need not be less than 30 ft.

3.2.5 Advisory Circular 23-8A

Advisory Circular 23-8A provides acceptable methods for calibrating the airspeed system plus precautions for calibration such as static sources located adjacent to the propeller. It also provides data reduction methods in an appendix and an equation to convert airspeed position correction into altimeter position correction.

3.3 Theory of Airspeed Systems

The conventional airspeed indicator is actually a differential pressure gauge calibrated according to the law of frictionless adiabatic flow, with the assumption that the ambient conditions are those of the standard atmosphere at sea level. For subsonic flow the calibration equation is derived from the isentropic flow relations. Then for subsonic speeds it can be said that:

$$V^2 = 2a^2/\gamma - 1[\{(P_t - P_s/P_s) + 1\}^{\gamma-1/\gamma} - 1] \qquad (3.1)$$

For supersonic flow the calibration equation assumes the existence of adiabatic flow with a normal shock located just forward of the total pressure pickup. In this case,

$$P_{t1} - P_s/P_s = \{(\gamma+1)V^2/2a^2\}^{\gamma/(\gamma-1)}\{\gamma - 1/1 - \gamma + (2\gamma V^2/a)\}^{1/(\gamma-1)} \qquad (3.2)$$

where
P_t = free stream total pressure
P_{t1} = total pressure aft of normal shock
P_s = free stream static pressure
γ = specific heat ratio, 1.4 for air
a = free stream sonic speed

Because the airspeed indicator only senses $P_t - P_s$ or $P_{t1} - P_s$ the indicator is calibrated according to the assumption that the sonic velocity has its sea level value of 1117 ft/s, and that the static pressure when it appears alone in the equation also has its sea level value of 2116 psf. By substituting these values into Eq. (3.1) we come up with an equation for calibrated airspeed.

$$V_C = 2497.7[\{(P_t - P_s/2116) + 1\}^{0.2857} - 1]^{.5} \qquad (3.3)$$

for subsonic flow with V_C in feet per second, and

$$P_{t1} - P_s = [0.2835V_C^2/\{7 - (1117/V_C)^2\}^{3.5} - 2116] \qquad (3.4)$$

for supersonic flow.

Another form of Eq. (3.3) can be obtained by letting

$$q_c = P_t - P_s \qquad (3.5)$$

Then by assuming that sonic speed and static pressure have their sea level values we can write:

$$q_C = P_t - P_s = P_{s0}[\{1 + \{(\gamma-1)/2\}(V_C^2/a_0)\}^{\gamma/(\gamma-1)} - 1] \qquad (3.6)$$

If we expand the bracketed quantity by a binomial expansion and retain only the first three terms we have:

$$q_C = P_{s0}\{1 + (\gamma/2)(V_C/a_0)^2 + (\gamma/8)(V_C/a_0)^4 - 1\} \qquad (3.7)$$

or

$$q_C = (\gamma P_{s0}/2)(V_C/a_0)^2\{1 + 1/4(V_C/a_0)^2\} \qquad (3.8)$$

then if we substitute $\rho_0 = \gamma P_{s0}/a_0^2$ we have:

$$q_C = (\rho_0 V_C^2/2)\{1 + 1/4(V_C/a_0)^2\} \tag{3.9}$$

3.4 Position Error

When we speak of the position error of an airspeed system, we are speaking about the airspeed error caused by the failure of the static and total pressure pickups to sense the actual free stream pressures. This is caused by the location of these pickups on the airframe, hence the name position error. Considerable study has shown that the total pressure source or pitot head is relatively insensitive to inflow angles and will have little error as long as it stays within 20 deg or so of the free stream flow direction. This says that most of the position error is due to location of the static source alone. If we were to plot the static pressure variations around an airplane in flight they would look somewhat like what is shown in Fig. 3.1.

In examining this figure we can see that as the airplane approaches, the pressure increases. In the vicinity of the wing it rapidly changes from maximum positive to maximum negative. About the middle of the aft fuselage it returns to zero and then increases with the oncoming tail. Again, in the vicinity of the horizontal tail it rapidly changes sign. Then somewhere aft of the airplane it returns to free stream value. From this figure we can see that there are very few places on or around the airplane where we can get an accurate reading of the free stream static pressure. This figure also shows why static ports on many airplanes are placed on the aft fuselage. We can also see that in order to do a proper job of flight testing we must have an accurate determination of position error.

To accomplish this by flight test we use several methods. One method is the pitot-static boom located on the nose or the wing tip. To minimize position error wing tip booms should have the static source located a minimum of one chord length ahead of the wing while nose booms should have their static source at least 1.5 fuselage diameters ahead of the fuselage.

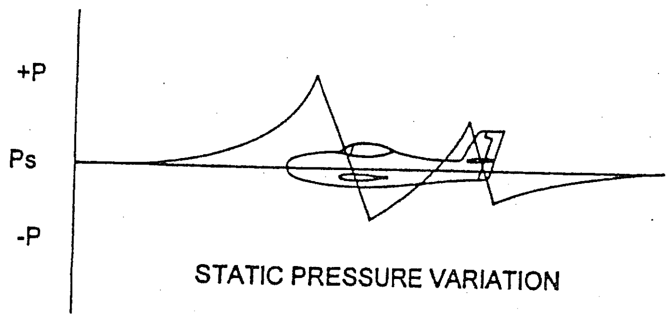


Fig. 3.1 Static pressure change with aircraft passage.

Another method is the airspeed bomb. In this method a bomblike pitot-static source is trailed far enough below and behind the aircraft to sense free stream pressures.

A third method is called the trailing cone method. In this method a trailing cone, with tubing to transmit static pressure, is trailed behind the aircraft at a distance sufficient to obtain free stream static pressure. The distance behind the aircraft of the static pressure ports should be at least 1.5 to 2 aircraft lengths behind the airplane.

All of the methods mentioned have their own set of problems. The fixed pitot-static boom becomes inaccurate at low speed when the angle to the relative wind allows total pressure to enter the static port. A free swiveling boom cures this problem but is not good at high speed due to flutter.

The trailing bomb method is only good for low-speed work since at higher speeds it tends to become unstable.

The trailing cone is good at higher speeds but has a problem with low speed since its weight will cause it to sag, introducing total pressure into the static port.

As you can see, measurement of static pressure during flight test is a sticky problem.

3.5 Lag Errors

Pitot-static systems also suffer from an error called lag error. This particular error shows up on large aircraft where the pitot-static lines are long and on high-performance aircraft where things change rapidly. This error will also show up when either the pitot or static line is longer than the other. To correct and minimize lag errors two steps may be taken. First, attempts should be made to only obtain data during stabilized flight conditions so pressures in the pitot-static system will have time to stabilize. Second, proper attention should be paid to pitot-static system design to insure that both sides have equal volume.

3.6 Altimeter Position Error

Since the altimeter is connected to the static system, it suffers from the same position errors as the airspeed system. The altimeter is calibrated using the equation of balance of the atmosphere:

$$dp = \rho g \, dh \tag{3.10}$$

where

dp = the static pressure change
ρ = air density
g = acceleration due to gravity
dh = change in height

Then for small errors such as position error we can say:

$$\Delta P_s = -\rho g \Delta hp \tag{3.11}$$

where
ΔP_s = the static position error
Δhp = the altimeter position error in feet of pressure altitude

It is worth noting that the altimeter position is a function of altitude since the atmospheric density appears in the equation.

If the airspeed position correction is known, the altimeter position correction can be found using the equation:[7]

$$dH = 0.08865(dV_C)\left\{1 + 0.2\left(\frac{V_C}{661.5}\right)^2\right\}^{2.5}\left(\frac{V_C}{\sigma}\right) \tag{3.12}$$

where
dH = the altimeter position correction in feet
V_C = the calibrated airspeed in knots
dV_C = the airspeed position correction in knots
σ = the ambient air density ratio

3.7 In-Flight Calibration Methods

Several methods are currently in use for airspeed calibration. They are:

1) speed course method
2) tower fly-by method, or altimeter depression method
3) pace method
4) radar method
5) onboard reference method
6) global positioning system (GPS) method

3.7.1 Speed Course Method

In this method the aircraft is flown over a measured course on the ground at low altitude so an accurate measure of ground speed can be made. The course is flown in both directions and the ground speeds averaged to minimize wind errors. The average ground speed is then compared to the true flight speed and the position error is arrived at. In using this method, attempts are made to fly the course during crosswind conditions and allow the aircraft to drift. This also helps to minimize wind errors. Problems with the method are that it requires a measured course in a remote area and must be flown in very stable air so that airspeed can be maintained accurately. A data reduction sequence for the speed course method is shown in Table 3.1.

3.7.2 Aneroid or Tower Fly-by Method

Since both the airspeed system and altimeter are hooked to the same static system on most aircraft, it is possible to relate altimeter position error directly to airspeed position error using Eq. (3.12).

In this method the aircraft flies at a constant airspeed and altitude by a tower of known height that has an observer and sensitive barometer on top.

Table 3.1 Data reduction for measured speed course method for pitot-static position correction calibration

Step	Item	How to obtain	Units	1	1	2	2	3	3
1	direction of flight	from course layout[a]	deg	090	270	090	270	090	270
2	elapsed time	flight data from stopwatch	min						
3	V_o	flight data	kn	100	100	90	90	80	80
4	H_{Po}	flight data	ft						
5	T_o	flight data	°C						
6	rpm_o	flight data[b]	rpm						
7	$M. P._o$	flight data[b]	in. Hg						
8	V_I	instrument calibration	kn						
9	H_{PI}	instrument calibration	ft						
10	T_I	instrument calibration	°C						
11	δ_I	tables or Eq. (1.9)	—						
12	θ_I	$273.16 + \#10 \div 288.16$	—						
13	σ	$\#11 \div \#12$	—						
14	$\sqrt{\sigma}$	$\sqrt{\#13}$	—						
15	GS	course distance $\times\ 60 \div \#2$	kn						
16	$GS\sqrt{\sigma}$	$\#15 \times \#14$	kn						
17	Ave. #16	average #16 for opposite headings	kn	—		—		—	
18	Ave. V_I	average #8 for opposite headings	kn	—		—		—	
19	ΔV_{PC}	#17 − #18	kn	—		—		—	

[a] Courses should be laid out so as to be flown cross wind and should be measured in nautical miles.
[b] These flight data only used to help set up the repeat run at the same airspeed.

The observer can determine the aircraft height with respect to the tower using a theodolite. This height may then be compared with the height shown on the aircraft altimeter and the position error determined using Eq. (3.11). This method is almost the exclusive method used by the military and by some aerospace companies. Its problems are:

1) It has speed limitations.
2) It also requires very stable air.
3) It assumes that pitot error is zero.

3.7.3 Pace or Calibrated Aircraft Method

In this method another aircraft is calibrated by one of the above methods and then used as the standard for calibrating the test aircraft. This method has the advantage of safety since the calibration can be done at altitude. It also saves test time since the wait for smooth air at low altitude is not required. Its disadvantage is the requirement to maintain one aircraft in a calibrated state as the standard. The method is performed by the pace or standard aircraft maintaining a steady speed while the test aircraft flies a tight formation on the pace. When there is no relative movement in the formation the test aircraft calls "read data," and both airspeed indicators are read simultaneously. This step is repeated for several speeds through the speed range of the test aircraft.

Correction of data is straightforward. First the pace aircraft's indicated speeds are corrected to calibrated airspeed, then the test aircraft's indicated speeds are corrected for instrument error. The difference between the V_I of the test aircraft and the V_C of the pace aircraft is the ΔV_{PC} for the test aircraft.

3.7.4 Radar Method

The radar method is similar to the speed course method except that the times to traverse the known distance are obtained from the radar. This method has the advantage of being able to be used at altitude and it does not have the speed limitations of other methods. It does, however, require smooth air and low wind conditions. This method is flown in the same manner as the speed course method.

3.7.5 Onboard Reference Method

The onboard reference method consists of using an airspeed bomb or trailing cone as a reference system to calibrate the test aircraft's system. The method is performed by flying the aircraft at a number of speeds throughout its speed range and reading the indications of the test system and the reference system. Data are then reduced in the same manner as the pace method.

The advantages of this method are that it only requires one aircraft and that the reference system may be calibrated on one aircraft and then used on several other aircraft prior to requiring recalibration.

The disadvantages are that the reference system does require calibration and that a special rig or fixture is required on the test aircraft to handle the reference system.

3.7.6 Global Positioning System Method

The global positioning system (GPS) method developed by the author replaces the ground speed obtained from the measured course in the speed course method with the ground speed along the aircraft's heading obtained from GPS ground speed and aircraft track over the Earth. By knowing the GPS ground speed, aircraft track over the ground, and the aircraft heading one can obtain the component of ground speed along the aircraft heading. Like the speed course method, the GPS method uses the ground speed measured along the heading on reciprocal headings to cancel the effects of winds and then follows the speed course procedure for data reduction to determine the pitot-static systems position correction.

The principal advantage of this method is that it can be flown at any altitude as long as the air is smooth. In addition, the aircraft only needs to be stable on altitude and airspeed long enough for the GPS receiver to update—usually a matter of seconds—before data can be read and the opposite heading established.

It should be noted that it is not necessary to have a differential GPS receiver to use this method since its accuracy is quite high with conventional receivers. This is because the ground speed taken from the GPS receiver is the first derivative of the position so a fixed error in the location of the aircraft will not affect the ground speed.

3.8 Temperature Probe Calibration

Since it is customary during the initial stage of flight testing to calibrate all test instruments, we will also discuss calibration of temperature probes. Determination of free air temperature in flight is also subject to correction because of probe design and location. With a well-designed probe the major error is caused by compressing the air passing the probe. If the air passing the probe is brought to a complete stop adiabatically and the probe senses the resulting temperature, then the instrument corrected temperature T_I may be stated by the equation:

$$T_I = T_C + V_T^2/7592 \tag{3.13}$$

where
T_I and T_C are in degrees Kelvin
V_T is in knots
7592 is a constant from the physical properties of air

Since in actuality the air does not come to a complete stop and may exhibit nonadiabatic recovery, a correction factor is added to Eq. (3.13), and the resulting equation is written:

$$T_I = T_C + KV_T^2/7592 \tag{3.14}$$

The correction factor K will vary with installation and probe design from 0.7 to 1.0 but will usually fall between .95 and 1.0. For subsonic speeds K

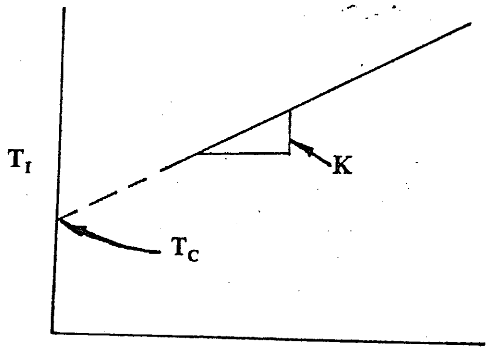


Fig. 3.2 Plot of temperature calibration.

does not vary significantly with Mach number and altitude, but at supersonic speeds temperature rises are much larger and variations may exist.

The constant K may be determined by taking the slope of the plot of T_I vs $V_T^2/7592$ (Fig. 3.2).

The free air temperature T_C may also be determined from this plot by extending the plot to where $V_T^2/7592$ equals zero.

References

[1]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Theory," FTC-TIH 64-2005, AFFTC, USAF Test Pilot School, Edwards AFB, CA, 1965.

[2]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH 64-2006, AFFTC, USAF Test Pilot School, Edwards AFB, CA, 1965.

[3]Federal Aviation Administration Order 8110.7, "Engineering Flight Test Guide for Small Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 20 June 1972.

[4]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[5]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. I—Performance*, Pergamon Press, New York, 1959.

[6]Hamlin, Benson, *Flight Testing*, The MacMillan Company, New York, 1946.

[7]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Printing Office, Washington, D.C., Feb. 1989.

4
Stall Speed Measurement

4.1 Introduction

The airplane's stalling speed is one of the most important parameters obtained in flight test, since most other criteria are based upon some multiple of stalling speed. Therefore, stalling speed should be determined early in the flight test program. It is the next item to be determined after the pitot-static system has been calibrated. Stalling speed is difficult to determine because the position error is hard to define at high angles of attack and changes rapidly in that region.

One of the difficulties in determining stalling speed is defining when the stall occurs. Airplane designers and aerodynamicists define the stalling speed as the speed at which the maximum lift coefficient, $C_{L\max}$, occurs. However, the various FAA regulations and the military specifications have different definitions. This difference in definition of the stalling speed has led to some controversies during "off-the-shelf" buys of civil certified aircraft by the U.S. military.

4.2 Federal Aviation Administration Requirements

The FAA has several definitions for stalling speed depending upon which certification regulation applies. This also has led to some controversies even within the FAA organization.

4.2.1 Civil Aeronautics Manual 3 Requirements (Ref. 1)

CAR 3.82 Definition of Stalling Speeds states:

(a) V_{S0} denotes the true indicated stalling speed, if obtainable, or the minimum steady flight speed at which the airplane is controllable, in mph, with:

1) engines idling, throttles closed (or no more than sufficient power for zero thrust),
2) propellers in a position normally used for takeoff,
3) landing gear extended,
4) wing flaps in the landing position,
5) cowl flaps closed,
6) center of gravity in the most unfavorable position within the allowable landing range,

7) the weight of the airplane equal to the weight in connection with which V_{S0} is being used as a factor to determine a required performance.

(*b*) V_{S1} denotes the true indicated stalling speed, if obtainable, otherwise the calculated value in mph with:

1) engines idling, throttles closed (or not more than sufficient power for zero thrust);
2) propellers in a position normally used for takeoff, the airplane in all other respects (flaps, landing gear, etc.) in the particular condition existing in the particular test in connection with which V_{S1} is being used;
3) the weight of the airplane equal to the weight in connection with which V_{S1} is being used as a factor to determine a required performance.

(c) These speeds shall be determined by flight tests using the procedure outlined in section 3.120.

Several things should be noted about this regulation. First, the term "true indicated stalling speed" means the calibrated stalling speed. Second, the term "zero thrust" is interpreted by the FAA (then CAA) to permit "zero thrust at a speed not greater than 110% of the stalling speed." The third item that should be noted is that the stalling speed shall be determined by the procedure that is outlined in section 3.120 of CAR 3. This regulation says that: "the elevator control shall be pulled back at a rate such that the airplane speed reduction does not exceed 1 mph per s until a stall is produced as evidenced by an uncontrollable downward pitching motion of the airplane, or until the control reaches the stop." Of all the airplanes upon which the author has measured stalling speed, (50+), the stall has always been defined by when the elevator control reaches the up stop. This condition may be well beyond the maximum lift coefficient if the aircraft has ample elevator power. Even though the airplane may be well beyond the $C_{L\max}$ the elevator is powerful enough to keep the nose from pitching downward uncontrollably until the up stop is reached. If the airplane is elevator power limited the aircraft may reach a steady airspeed without being aerodynamically stalled. In this case the stalling speed is still defined as the speed when the elevator reaches the up stop. It should be noted that the definition of the stalling speed is different than that given in CAR 4b of the CARs and the one given in the military specifications. Fig. 4.1 taken from reference 3 shows these definitions in graphical form.

CAR 3.83 Stalling Speed also addresses the stalling speed. However, in this regulation the maximum value of the stalling speed is addressed. CAR 3.83 says: "V_{S0} at maximum weight shall not exceed 70 mph for 1) single-engine airplanes, and 2) multiengine airplanes which do not have the rate of climb with critical engine inoperative specified in section 3.85(b)." The purpose in this regulation is to lower the stalling speed to a point that in the event of an accident during landing the occupants of the aircraft would have a better chance of survival. Research has shown that the accident rate during landing approach goes up as the square of the approach speed. Therefore, if the stalling speed is low the approach speed will also be low since it is a function of stalling speed.

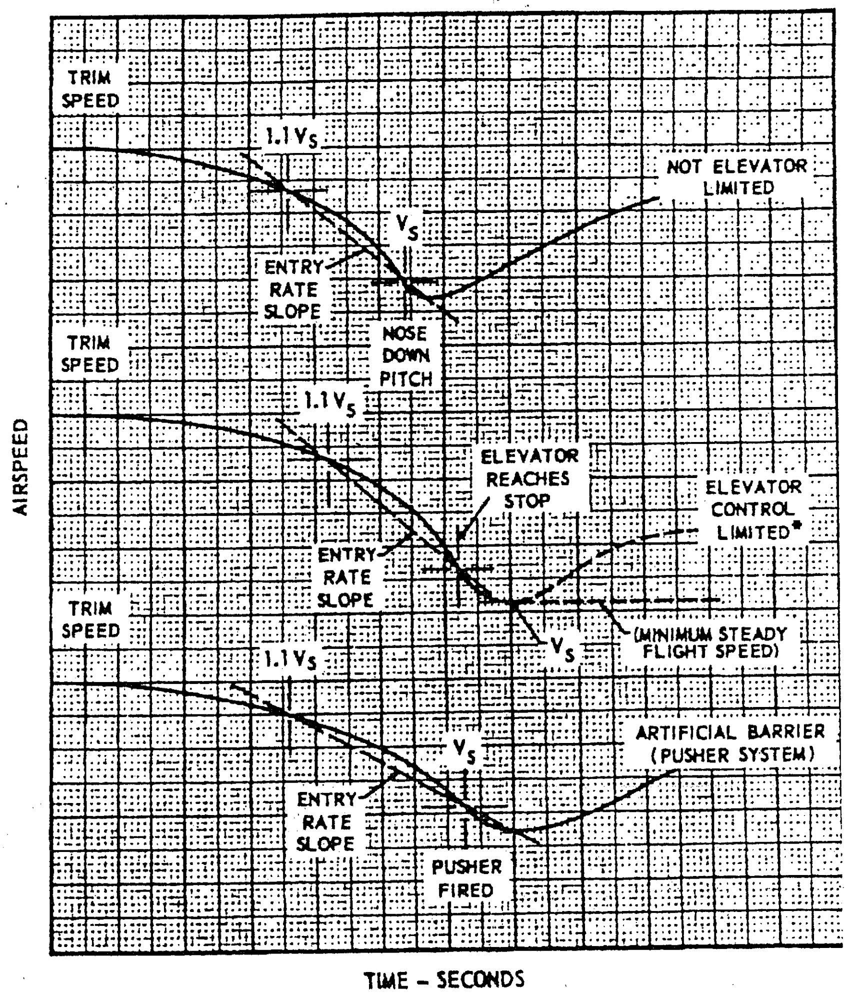


Fig. 4.1 FAA stalling speed definitions.[3]

It is also worthy of note from this regulation that multiengine airplanes that have a stalling speed lower than 70 mph do not have to have a rate of climb when the critical engine is inoperative.

4.2.2 *Federal Aviation Regulation Part 23 (Ref. 2)*

The Federal Aviation Regulations Part 23 reads quite similar to the Civil Aeronautics Regulations. One difference is that the airspeeds have been converted from miles per hour to knots. FAR 23.49 Stalling Speed combines the CAR 3.82 and 3.83 regulations and adds in what had previously been policy regarding the use of zero thrust at 110% of the stalling speed. FAR 23.49 also refers the reader to FAR 23.201 for test methodology. FAR 23.201 is essentially the same as CAR 3.120 regarding the methodology.

Advisory Circular 23-8A (Ref. 3) has several notes relating to stalling speed. It states that: "the 61 kn (70 mph) stalling speed applies to the maximum takeoff weight for which the airplane is to be certificated." It also points out that most standard airplane pitot-static systems are not acceptable for determining stall speeds and provides some acceptable systems. It also provides acceptable methods for determining zero thrust for reciprocating engine powered airplanes and turbopropeller aircraft. The test crew should consult both the applicable regulations and this advisory circular prior to commencing testing.

4.3 Stall Theory

The stall occurs due to an adverse pressure gradient developing over the upper surface of the wing as the angle of attack is increased causing the flow to separate from the upper surface of the wing. This flow separation is affected by both two-dimensional and three-dimensional effects, in addition to the effects of Reynolds number.

4.3.1 Two-Dimensional Effects

Two-dimensional effects upon the stall and the angle of attack at which it occurs include:

1) Reynolds number
2) wing camber
3) wing thickness
4) size of the leading edge radius
5) surface roughness
6) leading and trailing edge devices such as slots and slats and flaps

4.3.1.1 Reynolds number two-dimensional effects. The larger the Reynolds number, the higher the maximum lift coefficient and the angle of attack where it occurs. Reynolds number has little effect upon the lift curve slope as shown in Fig. 4.2 (Ref. 4).

4.3.1.2 Wing camber effects. Wing camber tends to reduce the angle of attack where stall occurs while increasing the maximum lift coefficient. This effect is similar to that of trailing-edge flaps as shown in Fig. 4.3. Positive camber also moves the angle of zero lift to the left as is also shown in the figure. It is worth noting that for most airfoil families the lift curve slope stays nearly constant with an increase in camber.

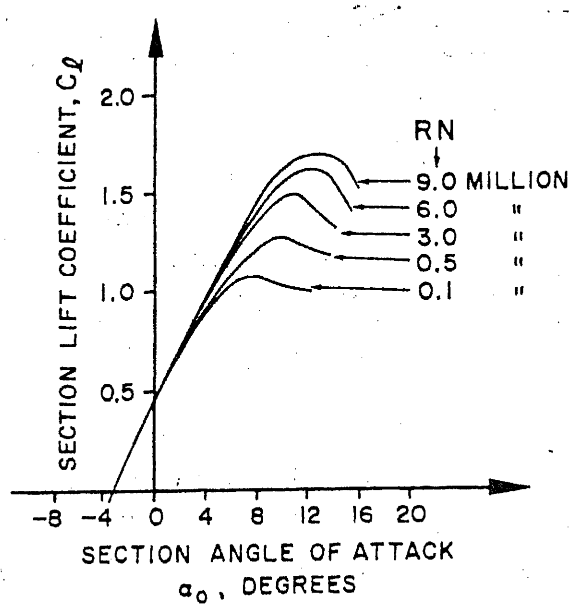


Fig. 4.2 Effect of Reynolds number—NACA 4412 (Ref. 4).

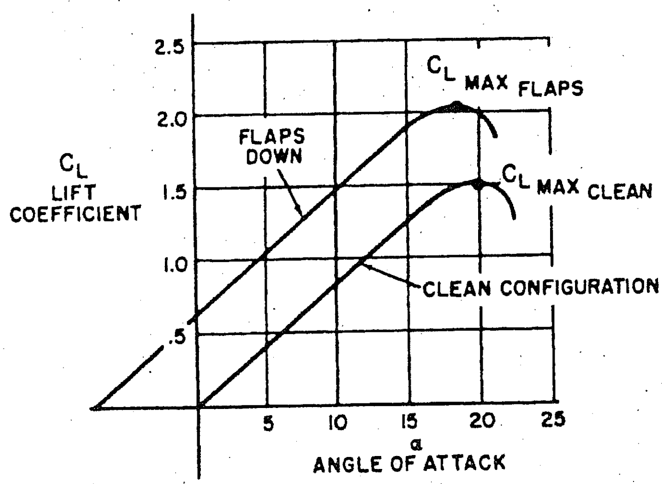


Fig. 4.3 Effects of flaps on the lift curve.[4]

4.3.1.3 Wing thickness effects. Wing thickness has the effect of increasing the maximum lift coefficient and the angle of attack at which it occurs.

4.3.1.4 Leading-edge radius effects. Increasing the size of the leading-edge radius has the effect of increasing the maximum lift coefficient and the angle of attack at which it occurs, thereby delaying the stall.

4.3.1.5 Surface roughness effects. Surface roughness may have varying effects. If the flow over the wing is laminar, an increase in surface roughness will cause the boundary layer to transition to a turbulent boundary layer that contains more energy and will not separate as fast as a laminar boundary layer and the stall will be delayed. However, if the surface roughness is large it may trip the boundary layer causing early separation. Roughness the size of frost or ice will cause the boundary layer to trip resulting in an earlier than normal stall.

4.3.1.6 Leading- and trailing-edge devices effects. Leading- and trailing-edge devices both increase the maximum lift coefficient that an airfoil can produce. However, trailing-edge flaps, of all types, tend to lower the angle of attack at which maximum lift occurs as shown in Fig. 4.3 (Ref. 4). Leading-edge devices, on the other hand, tend to increase the angle of attack where maximum lift occurs. These effects are shown in Fig. 4.4 (Ref. 4).

4.3.2 Three-Dimensional Effects

Three-dimensional effects upon the stall and the angle of attack at which it occurs include:

1) Reynolds number
2) wing planform
3) wing sweep
4) wing aspect ratio
5) effects of aircraft weight
6) effects of c.g. location

4.3.2.1 Reynolds number three-dimensional effects. The Reynolds number role in the three-dimensional effects upon the stall occur as a result of wing planform. If the wing is tapered, the local chord length at the tip is smaller than the chord length at the root, which results in a lower Reynolds number at the tip. As discussed previously, this results in a lower $C_{L\max}$ at the tip and a lower stalling angle of attack. Wing tip stalls are not preferred for stall characteristics since they may produce a roll-off at the stall. Therefore, on many high aspect ratio wings, the chord near the tip is held constant to avoid a reduction in Reynolds number.

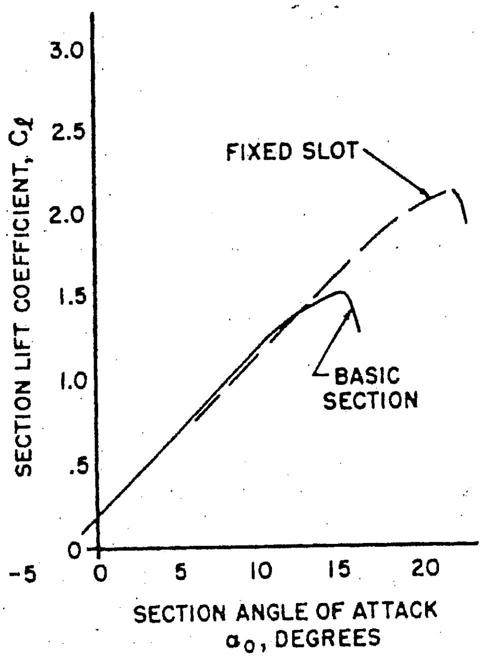


Fig. 4.4 Effects of slots or slats on the lift curve.[4]

4.3.2.2 Wing planform effects. Wing planform has a major effect upon stall. Fig. 4.5 (Ref. 4) shows the stall patterns for several wing planforms. It is desirable from the stall characteristics standpoint to have the wing stall at the root first. From Fig. 4.5, we can see that the rectangular planform is best from the standpoint of stalling at the root first. This good stall behavior is the reason that it is found on many trainer aircraft. However, it is not good from the standpoint of induced drag since it generates large wing tip vortices. The elliptical planform is the best from the standpoint of induced drag, but not good from the standpoint of the stall because it stalls equally along the span. This causes a loss in aileron control early in the stall. Fig. 4.5 also shows the effects of other planforms.

4.3.2.3 Wing sweep effects. Fig. 4.5 also shows the effects of aft sweep. From the figure we can see that aft sweep tends to cause tip stall due to spanwise flow. Forward sweep, on the other hand, uses the spanwise flow to cause the root to stall first, which has been one of the reasons for interest in forward sweep for high-performance aircraft.

4.3.2.4 Effects of aspect ratio. The effects of wing aspect ratio on the stall are shown in Fig. 4.6 (Ref. 4). High aspect ratio wings tend to have higher maximum lift coefficients that occur at lower angles of attack than do low aspect ratio wings. A reduction in aspect ratio can sometimes be used to correct

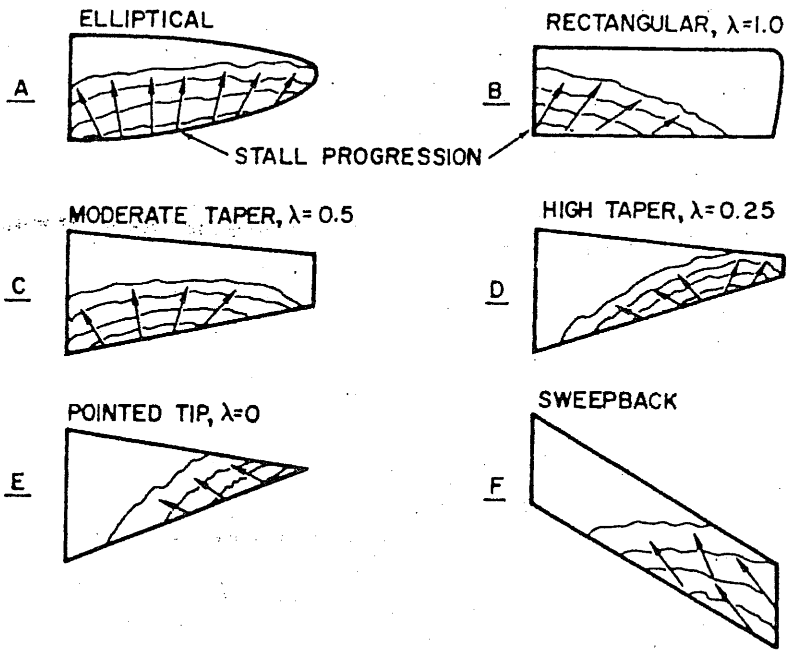


Fig. 4.5 Effects of wing planform on stall patterns.[4]

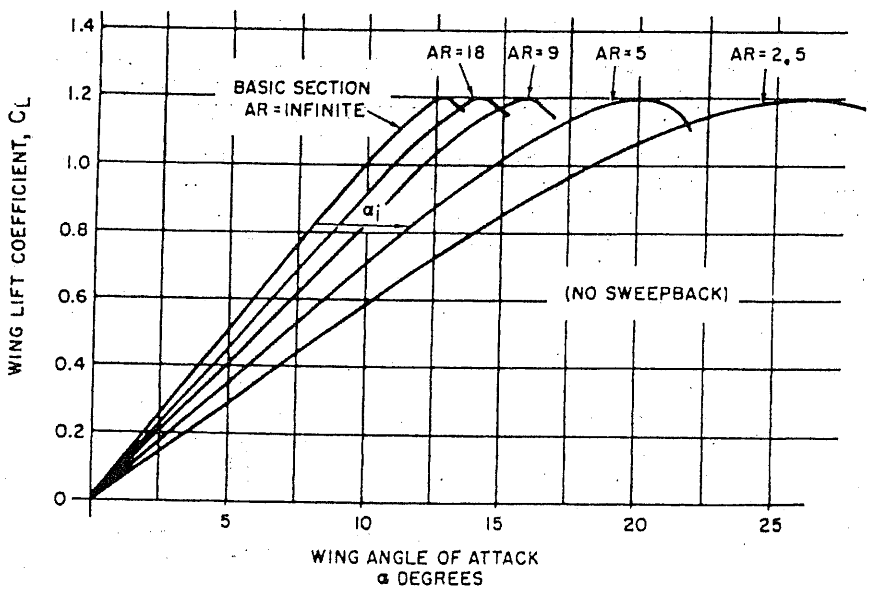


Fig. 4.6 Effects of aspect ratio on three-dimensional wing stall.[4]

problems that occur due to a low stall angle of attack. One such fix is the use of the dorsal fin to correct problems of rudder lock. Although low aspect ratio wings tend to have a higher stall angle of attack, the stall is not as well defined as it is with higher aspect ratios. This can lead to a failure by the pilot to recognize the stall with the resultant development of a very high sink rate. With the introduction of the delta wing F-102 fighter, a number of landing accidents occurred due to this effect.

4.3.2.5 Aircraft weight effects. Aircraft weight has a direct effect upon stalling speed. Stalling speed will increase as weight increases since the maximum lift coefficient is a fixed value.

4.3.2.6 Center of gravity location effects. The location of the aircraft c.g. also has an effect upon stalling speed as well as stall characteristics. Movement of the c.g. aft results in a reduction in the balancing tail down load. This means that the total force the wing must generate is lower, which results in a lower stalling speed. It is for this reason that the FAA regulations require the stalling speed to be measured at the most forward c.g. at maximum TOGW. The c.g. also has an effect upon stalling characteristics. As the c.g. moves aft the longitudinal control becomes more effective allowing the pilot to force the aircraft deeper into the stall prior to reaching the up stop on the elevator. This can result in problems from longitudinal instability (pitch up) to the generation of rolling moments resulting in roll-off.

4.4 Aircraft Loading

As discussed in the previous section, the FAA requires that the stalling speed be measured at the maximum takeoff gross weight at the most forward center of gravity, abbreviated (forward gross). This loading normally results in the highest stalling speed and is the speed published in the "Pilot's Operating Handbook" (POH) or "Airplane Flight Manual" (AFM). Other stalling speeds such as those at aft c.g. at maximum TOGW (aft gross) and the most forward c.g. regardless of weight (forward regardless) may be measured for determining certain flight characteristics, but they will not be used for any performance determination.

4.4.1 Weight and c.g. Tolerances

The weight tolerances as specified by the FAA in both CAM 3 and FAR Part 23 are +5% to −1% of the gross weight at the c.g. where the stalling speed is being determined. It is generally best to overload the aircraft to near the +5% tolerance so that as the flight progresses and fuel is burned the weight does not decrease below the −1% limit.

The c.g. tolerance is ±7% of the total allowable c.g. travel. However, unless it is not possible to do so, it is better to load the aircraft forward of the desired c.g. location particularly if fuel burn off results in aft c.g. movement.

4.5 Safety Considerations

As a rule, measurement of the stalling speed at the forward c.g. is not a hazardous test. However, any time stalls are done for the first time on a new aircraft design care should be taken. Quick release doors should be installed and tested and the aircrew should be attired in helmet, flying suits, parachutes, and jump boots. At forward c.g. it is not necessary to have the aircraft equipped with a spin recovery parachute, but that should be considered before any stall testing is done at c.g. aft of the forward limit.

The altitude selected for initial stall tests is somewhat dependent upon the size of the aircraft. However, in no case should stalls be attempted at altitudes below 5000 ft above ground level (AGL).

Initial stalls should be approached incrementally by recovering at first indication, then at the initial nose pitch, and finally to a complete stall when the elevator hits the up stop. Once it has been determined that the aircraft has reasonable characteristics at this loading, more aggressive testing can be accomplished.

4.6 Flight Test Method

Both CAM 3 and FAR Part 23 provide that stalling speeds are measured at the forward gross loading. In addition, the following other conditions are specified:

1) aircraft trimmed at $1.5V_{S1}$ or $1.5V_{S0}$ depending upon configuration
2) power idle or zero thrust, cowl flaps closed
3) deceleration rate to the stall at 1 kn/s

4.6.1 Trim Airspeed

The trim airspeed for stall speed measurement is 1.5 times the stalling speed in the configuration being tested. One might ask how one knows the value of the stalling speed since it has not been measured. For the first few stalls in a given configuration one can use the calculated stalling speed based upon airfoil data and aircraft takeoff gross weight. Once a more valid test number is obtained from the first few stalls, the test stalling speed can be used and a few stalls repeated at the new trim speed to confirm that stalling speed value.

4.6.2 Power Settings

Both CAM 3 and FAR Part 23 specify either idle power with the propeller set at the maximum rpm setting, or zero thrust. In nearly all cases zero thrust will provide a lower stalling speed, but it is more difficult to obtain since more attention must be paid to power setting during the test. Advisory Circular 23-8A paragraph 23.49(a)(5) provides an acceptable method for determining zero thrust for a propeller-driven aircraft. In essence, the thrust generated by a propeller will be zero when the propeller thrust coefficient, C_T, is zero. Using the propeller manufacturer's coefficient curves one can determine the propeller advance ratio J at which the thrust coefficient is zero. Knowing this value allows calculation of the propeller rpm necessary for zero thrust by use of the equation:

$$\text{rpm} = 101.27V/JD \tag{4.1}$$

where
rpm = the desired propeller speed setting
V = the airspeed in kn for zero thrust, usually taken at $1.1V_{S1}$
J = the propeller advance ratio at $C_T = 0$
D = the propeller diameter in feet

The test pilot would then assure by adjusting the throttle that this value of rpm was set as the airspeed reached a value 10% in excess of the stalling speed.

4.6.3 Deceleration Rate

The deceleration rate to the stall is 1 kn/s, which should be achieved by the time the speed of $1.1V_{S1}$ is reached during the deceleration. A method for determining this rate is shown in Fig. 4.1 (Ref. 3).

4.6.4 Defining the Stall

The point when the nose pitches uncontrollably is the definition given in CAM 3.120 and FAR 23.201 as the point when the stalling speed is to be measured. Again, it should be stressed that this point occurs for most airplanes of this category when the elevator reaches the up stop. It should also be noted that it must be possible to keep the wings level and sideslip to zero by normal control usage up to this point. Normal control usage means the correct sense of control movement. It does not mean the speed with which the controls are moved. On most airplanes the controls must be moved much faster to correct roll-off after the flow starts to separate.

4.6.5 Statistical Sample

One stall is not adequate to define the stalling speed. At least five stalls should be measured to ensure that a large enough statistical sample is obtained, for a given configuration, to ensure valid data. After each stall has been corrected to standard conditions, the values for the five stalls should be averaged to determine the stalling speed.

4.7 Data Reduction Method

The FAA requires that only a correction for nonstandard weight be made to stall speed data. For small aircraft that is all that is necessary. However, in certain instances it is necessary to correct for deceleration rates that are greater than 1 kn/s or for c.g. positions that are not at forward gross. Therefore, methods for making these corrections are given. However it should be noted that the FAA does not accept these last two corrections as valid for FAA testing.

4.7.1 Weight Correction

The test weight for each stall accomplished should be determined by knowing the takeoff aircraft loading and the fuel consumed at the time the stall was performed. This can be accomplished by dividing the total fuel consumed during the flight by the total flight time to determine the pounds of fuel used per minute. Then by taking the elapsed time into the flight that the stall occurred, the airplane weight at the time of the stall can be calculated. A second, more accurate and more expensive method to determine the weight at the stall is to use a fuel counter such as is now being installed in many production airplanes and record the pounds of fuel used at the time of the stall.

Once the aircraft weight has been determined, the calibrated stalling speed determined at each data point is corrected for weight using the following equation:

$$V_S = V_{ST}\sqrt{\frac{W_S}{W_T}} \tag{4.2}$$

where

V_S = the weight corrected calibrated stalling speed
V_{ST} = the observed stalling speed corrected for instrument and position error
W_T = the aircraft weight at the stall
W_S = the standard weight, or the weight to which certification is sought; normally the maximum TOGW

Each of the stalls for a specified configuration should be corrected and then averaged to determine the stalling speed to be published.

4.7.2 Deceleration Rate Correction

For most light aircraft testing obtaining the deceleration rate of 1 kn/s by a speed of $1.1V_S$ is not difficult.

4.7.2.1 Correction by the graphical method. Advisory Circular 23-8A provides a graphical method for correcting deceleration rates of greater or less than 1 kn/s. This method may also be used to insure that the stalling speed is measured at a deceleration rate of 1 kn/s. Fig. 4.1 taken from AC 23-8A shows this method and its use.

4.7.2.2 Equation method of correction. Data taken at rates of deceleration greater than 1 kn/s may also be corrected using the following approach:

1) Correct the observed stalling airspeed for instrument and position errors.
2) Apply the nonstandard deceleration correction as follows:

a. Determine the nonsteady flow correction factor, R.

$$R = \frac{V_{Sc}}{(C/2)\dot{V}} \tag{4.3}$$

where
V_{Sc} = the observed calibrated stalling speed
C = the mean aerodynamic chord
$\dot{V}$ = the deceleration rate in knots per second

b. Determine the nonstandard rate corrected stalling speed, V_{ST}:

$$V_{ST} = \sqrt{\frac{R+2}{R+1}}(V_{Sc}) \tag{4.4}$$

4.7.3 Center of Gravity Correction

Although not allowed by the FAA regulations, it is possible to make a correction to the stalling speed for an incorrect c.g. position. To accomplish this, one needs to correct the speed for the change in tail down load due to an incorrect c.g. To accomplish this, one must first correct the weight corrected stalling speed to a value of lift coefficient. One then corrects this lift coefficient using the following equation:

$$C_{L_{REF}} = C_{L_{TEST}}\left[1 + \frac{\bar{C}}{l_t}\left(\frac{CG_{REF} - CG_{TEST}}{100}\right)\right] \tag{4.5}$$

where
C_{LREF} = the maximum lift coefficient at the desired c.g.
C_{LTEST} = the maximum lift coefficient at the test c.g.
$\bar{C}$ = the MAC
l_t = the length of the tail (assumed to be from 1/4 chord of the wing to 1/4 chord of the horizontal tail)

The new lift coefficient is then converted back to stalling speed.

4.7.4 Averaging Corrected Data

Once the corrections have been made to each of the test stalling speeds, they are then averaged to determine the value to be used in pilots' handbooks and for complying with FAA regulations. It is normally a good idea to have at least five stall speeds in a given configuration to average. This insures that the value for the stalling speed that is obtained is a reasonable value and not a scatter point.

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airplane Airworthiness; Normal, Utility, and Acrobatic Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1965.

[3]Federal Aviation Administration Advisory Circular No: 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

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

5
Determination of Engine Power in Flight

5.1 Introduction

In flight testing the determining of engine power is performed for essentially two purposes:

1) To determine the engine power as installed in the airframe.
2) To measure the drag of the airplane with the propulsive system operating.

Because the aircraft's induction, exhaust, and cooling system affect engine power, and engine accessories (such as generators, alternators, or hydraulic pumps) also require power, the manufacturer's tests on a test stand will not give the power of the engine installed in the airframe. In flight testing, then, we must devise methods to measure the actual installed power so we can in turn use this number to define airplane drag. Because the airplane and engine function as a unit we cannot divorce the effects of one on the other. Even though these interaction effects may be small, they can cause data errors when they are ignored. They may also be the cause of poor performance and may be an area to examine when performance does not meet predictions. Some of these interaction effects for propeller-driven airplanes are as follows:

1) effects of cowling shape and blockage on propeller efficiency
2) propeller slipstream effects on airplane drag
3) effects of engine cooling drag on airplane performance
4) effects of engine cooling and fuel distribution on range, cruise, and climb performance
5) effects of engine exhaust thrust on apparent power required
6) effects of induction and exhaust system on power available

Some flight test methods, at least in part, take care of some of these interaction effects and we will discuss these in a later section. The measurement of brake horsepower of the piston engine is fairly straightforward. On the other hand, measurement of thrust horsepower is much more difficult because the interaction effects begin to enter the picture. Also, propeller efficiency charts seldom represent the efficiency of the propeller in its actual operating conditions. As a result most flight test performance data is based on brake horsepower (bhp) rather than on thrust horsepower.

5.2 Power Measurement of an Internal Combustion Engine in Flight

In propeller engine combinations there are several ways of determining engine power. The common methods in use today are: 1) torque meter method; 2) engine power charts; and 3) fuel flow method. Each of these methods has its strengths and weaknesses.

5.2.1 Torque Meter Method

Some flight test organizations believe that engine power is best determined by use of a torque meter attached to the crankshaft to measure torque output. In this method the brake horsepower is given by the equation:

$$BHP_T = KNQ \tag{5.1}$$

where
BHP_T = the test brake horsepower
K = a constant for the torque meter based on dynamometer tests
N = the engine speed (rpm)
Q = the torque meter reading

In some installations the torque is expressed as a function of brake mean effective pressure. In this case Eq. (5.1) can be expressed as:

$$BHP_T = P_b NK \tag{5.2}$$

where
P_b = brake mean effective pressure
K = a constant based on the calibration of P_b and different from the K of Eq. (5.1).

The torque meter method is a good method in the instances where it can be used and where there is a good calibration on the torque meter. It does have some limitations in that most of these devices are large and must be mounted between the propeller and the engine. This moves the propeller farther away from the cowling and may change the propeller efficiency. In addition, the torque meter will change the drag characteristics of the airplane due to its location and considerable care must be exercised in using it when measuring performance. It may be best to use the torque meter to develop an installed power chart by recording manifold pressure (M.P.) and rpm at the same time torque meter readings are taken. These M.P. and rpm values may then be corrected and plotted vs the bhp derived from the torque meter readings and the installed power chart developed. Since in-flight horsepower will vary with airspeed due to ram effects there would be some error depending on how the torque meter affected the overall drag of the airplane. It is, however, still one of the more accurate methods for use in performance testing of propeller-driven airplanes.

5.2.2 Engine Chart Method

In cases where a torque meter cannot be mounted or may be too expensive for the results desired, the engine manufacturer's power charts are an acceptable method. These charts solve for brake horsepower when M.P., rpm, carburetor air temperature, and pressure altitude are known. They are partly derived from actual engine test and partly from theory. They assume that variables other than those mentioned above, which affect engine power, are either constant or vary in a predictable manner. In addition, they assume that fuel flows and fuel distribution to the cylinders are as found on the engine test stand. As you would suspect, all of these items do not exist in an actual installation, so the power charts are not exact. However, for cases where a better method is not available these charts do provide reasonably accurate values of engine power.

The engine performance charts consist of two plots, a sea level chart and an altitude chart. The sea level plot is developed by connecting the engine to a dynamometer with runs being made at constant rpm while varying manifold pressure. The power is measured at each point and a plot such as the left-hand side of Fig. 5.1 (Ref. 3) developed. These lines of constant rpm are straight and converge to a point at zero manifold pressure which is the friction horsepower of the engine.

The altitude side (right-hand side of Fig. 5.1) of the chart converges to zero horsepower at 55,000 ft for normally aspirated engines. It can be developed mathematically from the following equation:

$$BHP_{ALT} = BHP_{SL}\{\sigma - (1 - \sigma)/7.55\} \tag{5.3}$$

or be obtained by flight tests or altitude capable wind tunnels. The test method is preferable, but the equation is based on data collected over many years on different makes of reciprocating engines and is accurate. The altitude side of the chart for turbocharged engines is determined by the engine manufacturer and provided as shown in Fig. 5.2 (Ref. 4). To determine the altitude where the installed engine will no longer develop sea level horsepower (called the critical altitude), one must conduct a flight test discussed later in this chapter.

If the sea level power calibration is available from the engine manufacturer for the test engine, then a power chart may be developed for that specific engine. This provides a more accurate method of power determination for flight testing than using the standard power chart for that type engine. However, if an engine calibration was not obtained for the test engine then the standard manufacturer's chart must be used.

5.2.3 Fuel Flow Method

A third method of determining engine power in flight is the fuel flow method. This method was developed by Textron Lycoming and is based on data accumulated over a period of 30 years. It has been known for many years that the fuel flow going to an engine is a function of the horsepower being developed by that engine or vice versa. In knowing that fact and observing the

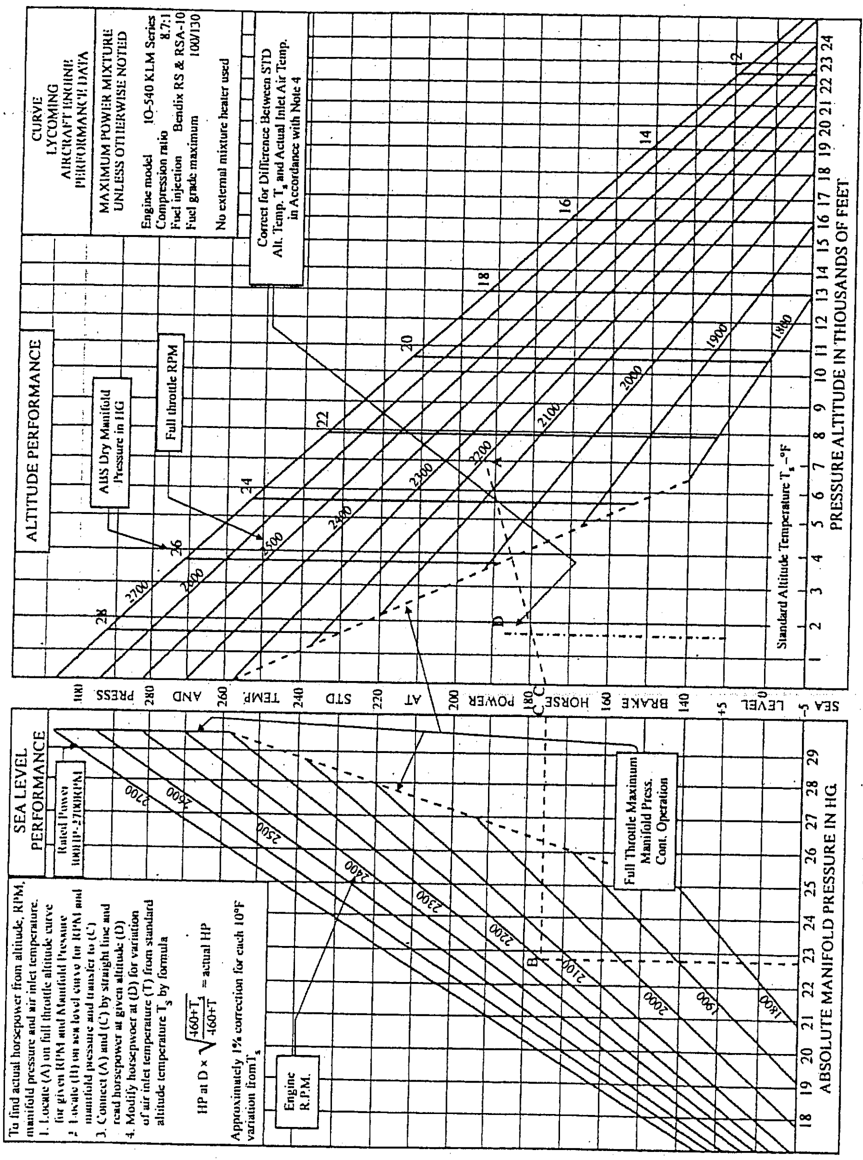


Fig. 5.1 Engine power chart—Lycoming IO-540 engine.[3]

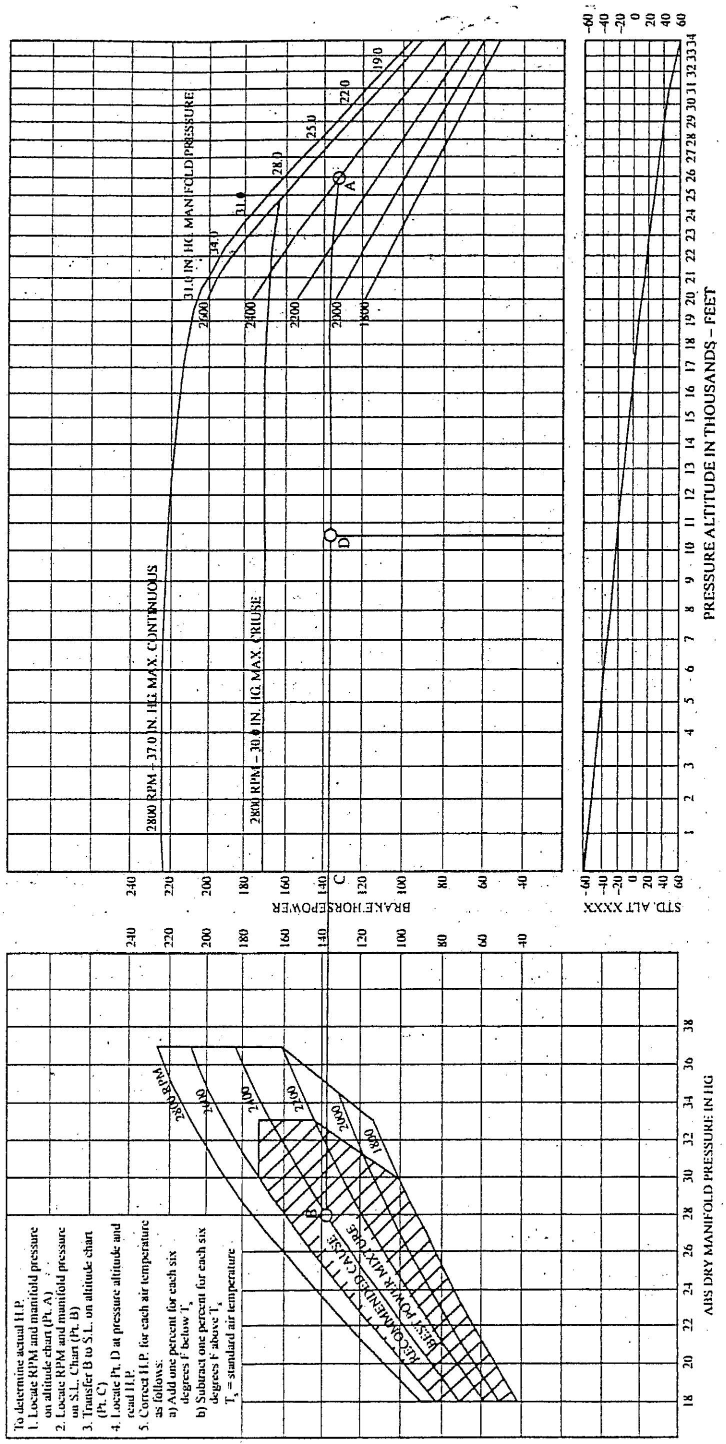


Fig. 5.2 Engine power chart—Continental TSIO-360-C/CB.

data from their engines over a period of time Textron Lycoming reached the following conclusions:

1) At peak exhaust gas temperature (EGT) the engine is developing 96.8% of its best indicated horsepower for the particular throttle setting.
2) At peak EGT the engine has a fuel-air ratio of .0648, which is 84.9% of the best power indicated specific fuel consumption.
3) 100% power occurs between 0.0765 and 0.0840 fuel-air ratio and that for calculations .078 has proved to be an accurate value.
4) The fuel flow varies with the same relationship as the fuel-air ratio.
5) A definite value for indicated specific fuel consumption can be established for each engine as long as the compression ratio is known.

Using the above facts and a series of curves developed on a dynamometer for the engine in question, the actual installed horsepower of the engine can be determined if the fuel flow is accurately measured and the peak EGT carefully established.

Although this method is probably the most accurate method for determining in-flight power, it requires very accurate measurements of fuel flow and EGT which may be beyond the means of some flight test organizations.

5.3 Installed Horsepower Losses and How They Affect Power Measurement

Installed horsepower losses come from four general areas of the engine installation and can be classified as follows:

1) induction system losses
2) exhaust system losses
3) accessory losses
4) improper cooling losses

5.3.1 Induction System Losses

Induction system losses may be further broken down into three items:

1) poor pressure recovery
2) heat rise in the induction system
3) poor fuel distribution due to turbulence in the induction system

Poor pressure recovery in the induction system can be caused by blockage of the inlet, an improper air filter, too many sharp turns in the inlet, or too small an inlet for the engine size. It will normally show up at full throttle as a lower manifold pressure than would be expected for the given conditions.

Heat rise in the induction system will manifest itself as a carburetor air temperature in excess of the free air temperature. It is caused by the induction system being routed through a hot area of the engine compartment, or by leaks of hot air from the carburetor heat or alternate air system.

Poor fuel distribution due to turbulence in the induction system can be caused by any turbulence-producing device in the induction system. If severe enough it may cause rough running of the engine at some power settings. If EGT probes are installed on each cylinder it will show up as EGT for one or more cylinders being much hotter or cooler than the other cylinders.

5.3.2 Exhaust System Losses

Exhaust system losses are caused by poor exhaust system design and by poor selection or design of mufflers. Along with induction system losses, exhaust system losses are two of the largest producers of controllable losses. The exhaust system should be so designed that the pressure pulses from the cylinders do not oppose each other, causing back pressure in the exhaust. The muffler should also be designed with sufficient expansion to prevent excessive back pressure.

5.3.3 Accessory Losses

Accessory losses are fixed losses caused by the power requirements of the various accessories. These losses may be determined by having the engine tested on the dynamometer with and without the accessories. They are variable, however, since the load on such accessories as alternators, hydraulic pumps, or air conditioning compressors may vary. There is little that can be done about these losses other than select the lowest power requirement accessories in design.

5.3.4 Improved Cooling Losses

Improper cooling losses are caused when parts of the engine are not cooled evenly. This can cause increased friction, excessive valve leakage, or piston blow-by losses. To minimize these losses the cooling system should be so designed that the engine and oil temperatures are maintained within the ranges specified by the engine manufacturer. It is essential that differences between individual cylinders and cylinder heads be kept to a minimum.

5.4 Power Corrections

Power corrections may be divided into two general categories: 1) normally aspirated engine corrections; and 2) turbocharged engine corrections.

For normally aspirated engines we can make up to four power corrections:

1) altitude corrections
2) nonstandard temperature corrections
3) humidity corrections
4) full throttle corrections

5.4.1 Altitude Corrections

If we set our altimeter to 29.92 in.Hg and read pressure altitude we do not need to make any correction for altitude. Since this reduces workload it is the route taken by flight test organizations.

5.4.2 Nonstandard Temperature Corrections

Corrections for nonstandard temperature are made because temperature affects air density which affects power. These corrections are made using the following equation:

$$BHP_T = BHP_C\{T_S/(C.A.T._I + 288.16)\}^{0.5} \tag{5.4}$$

where

BHP_T = the corrected test brake horsepower
BHP_C = the brake horsepower obtained from the engine power chart using test conditions of rpm and M.P.
T_S = the standard temperature at the test pressure altitude in degrees Kelvin
$C.A.T._I$ = the instrument corrected carburetor air temperature in degrees Celsius

5.4.3 Humidity Corrections

Since an increase in water vapor in the air causes the air charge entering the engine cylinders to be less dense, it is necessary to make a correction for this where high humidity exists. The correction can be made directly by subtracting the vapor pressure of water, for the given relative humidity, from the instrument corrected observed M.P. If the wet and dry bulb temperatures are read at the flight data point then the vapor pressure of water for that flight data point can be obtained from a psychrometric chart.

5.4.4 Full Throttle Corrections

When operating at full throttle on a normally aspirated engine, or above the critical altitude for a turbocharged engine, we have a condition where any deviation from a standard day results in both a correction for temperature and a correction for manifold pressure. The manifold correction is needed because we cannot increase the manifold pressure due to the throttle being fixed at full open. The full throttle horsepower can then be expressed by the relation:

$$BHP_T = BHP_C + \Delta BHP_{\Delta C.A.T.} + \Delta BHP_{\Delta M.P.} \tag{5.5}$$

The ΔBHP for $\Delta M.P.$ consists of two parts: 1) the $\Delta M.P.$ due to nonstandard temperature; and 2) the $\Delta M.P.$ due to ram effect from increased airspeed. Both of the previously mentioned corrections require considerable instrumentation and, as a result, the full throttle correction is only rarely applied since certain performance methods will account for this correction by simpler means.

For turbocharged engines there are two additional corrections that may need to be considered. They are: 1) power corrections for turbocharger rpm and back pressure variation at a constant manifold pressure; and 2) corrections for determining critical altitude.

5.4.5 Power Corrections for Turbocharger rpm and Back Pressure Variation at Constant Manifold Pressure

On a turbocharged engine, if a given engine rpm and manifold pressure are set, the turbocharger, through its sensing devices, will maintain that power setting by varying its rpm. To vary the turbocharger rpm one must vary the engine back pressure. So, even though the engine power setting remains the same the power output will vary due to the change in engine back pressure and the change in C.A.T. caused by the change in turbo rpm and the free air temperature change.

The power variation due to the change in C.A.T. can be expressed in the same way as Eq. (5.4). The change in power due to change in back pressure has been empirically determined to be one percent increase in power for each 2-in. Hg decrease in back pressure. Then the total correction can be stated as:

$$BHP_T = BHP_C[\{T_S/(C.A.T. + 288.16)\}^{.5} + 0.005(EBP_T - EBP_S)] \qquad (5.6)$$

where
EBP_T = exhaust back pressure for the test conditions
EBP_S = exhaust back pressure for standard day conditions

5.5 Critical Altitude Determination

On a turbocharged engine we reach an altitude where the turbocharger can no longer extract enough energy from the exhaust gases to compress the inlet air enough to maintain the rated manifold pressure. This altitude is defined as the critical altitude. In order to determine this altitude by test we must correct the manifold pressures we observe at each pressure altitude to standard conditions. This is accomplished by the equation:

$$M.P._C = M.P._{OBS}\{T_S/(C.A.T._C + 288.16)\}^{.5} \qquad (5.7)$$

where
$M.P._C$ = standard day manifold pressure
$M.P._{OBS}$ = instrument corrected observed manifold pressure

By plotting the standard day manifold pressure vs pressure altitude the critical altitude may be determined.

References

[1]Herrington, R. M., Maj., Shoemacher, P. E., Bartlett, E. P., and Dunlap, E. W., "USAF Flight Test Engineering Handbook," AFFTCTRN 6273, Air Force Flight Test Center, Edwards AFB, CA, May 1961, rev. June 1964, Jan. 1966.

[2]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. I—Performance*, Pergamon Press, New York, 1959.

[3]Textron Lycoming, "Engine Model Specification," Lycoming IO-540-M Engine, Williamsport, PA, 1967.

[4]Teledyne-Continental Motors, "Owners Handbook," Continental TSIO360-C Engine, Mobile, AL, 1972.

6
Propeller Theory

6.1 Introduction

For certain missions and speed ranges the propeller-driven aircraft is the most efficient. Some examples are:

1) speed range of 0–450 kn
2) certain STOL missions
3) long endurance maritime reconnaissance missions

Recent NASA research has shown it possible to extend the cruise speed of propeller-driven aircraft to 0.8 Mach while maintaining a propeller efficiency in excess of 80%. Therefore for certain applications the propeller will be around indefinitely, and may be reintroduced in such areas as air transport.

6.2 Propeller Theory

There are three theories now used in the design of propellers. They are:

1) momentum theory
2) blade element theory
3) vortex theory

Until recently only the first two theories were in use. This is primarily due to the mathematical complexity of the vortex theory.

Most of today's propellers were designed using blade element theory. The problem for the designer is to find an existing propeller design that can be adapted to their requirements.

Since few propellers have been designed using vortex theory we will limit our discussion to momentum theory and blade element theory. Those interested in vortex theory should read Barnes W. McCormick's *Aerodynamics for V/STOL Flight*, Chapter 4 (Ref. 1).

6.2.1 Momentum Theory

Derived from Newton's second law:

$$F = m(\mathrm{d}V/\mathrm{d}t) \tag{6.1}$$

Assumptions:

1) Propeller does not add rotation to the air.
2) No profile losses.
3) Air is inviscid and incompressible.
4) Propeller has an infinite number of blades.

Since most of these assumptions are unrealistic, this theory is only useful in predicting ideal or maximum propeller efficiencies.

The mass of air passing through propeller (per unit time) is:

$$M = \rho A_1(V + v) \tag{6.2}$$

where
M = air mass
ρ = air density
A_1 = propeller disc area
V = true airspeed
v = velocity change through propeller

The thrust generated by the propeller is the mass per unit time multiplied times the total change in velocity per unit time through the control volume:

$$F = \rho A_1(V + v)2v \tag{6.3}$$

The power input to the propeller is:

$$P_i = F(V + v) \tag{6.4}$$

or the thrust times the velocity change the propeller. The power output by the propeller is:

$$P_o = FV \tag{6.5}$$

The ideal propeller efficiency is defined by:

$$\eta_{Pi} = P_o/P_i = FV/F(V + v) = V/(V + v) \tag{6.6}$$

6.2.2 Blade Element Theory

This theory considers the propeller blade to be a highly twisted wing. Most propellers in use today were designed using this theory. Since this theory considers the propeller to be a highly twisted wing, we will relate, where possible, the propeller parameters to more familiar wing parameters.

r = radius from hub to blade element
V = airplane velocity
β = geometric angle of attack of blade element
γ = reduction in blade angle of attack due to forward velocity

α = actual angle of attack
ω = rotational speed of propeller $\omega = 2\pi n$ where n = rps
u = propeller velocity due to rotation $u = r\omega = 2\pi rn$

$$\tan\gamma = V/u = V/r\omega = V/2\pi rn \tag{6.7}$$

Since β is fixed for a given blade element,

$$\alpha = \beta - \gamma \tag{6.8}$$

For a propeller α is different for each blade element and is not very useful. A better method is to evaluate the propeller at the tip since the twist is fixed. At the tip:

$$\tan\gamma_{tip} = 2V/\omega d \doteq V/\pi nd \tag{6.9}$$

where
d = propeller diameter

The dimension less quantity $V/nd = \pi \tan\gamma_{tip}$ is called the advance ratio J.

$$J = V/nd \tag{6.10}$$

This quantity in propeller theory is used to replace angle of attack in airfoil theory. Using this concept we know that the resultant velocity at a given blade section is proportional to nd, and that the aircraft velocity V is equal to the product of J and nd. We also know that the rotational velocity $u = (2\pi r/d)nd$. We can now arrive at the equation for the thrust generated by the propeller by using the laws of similarity: The propeller diameter as the reference length in determining Reynolds number; The thrust is then the product of the reference area d^2, the dynamic pressure, which corresponds to nd, and a thrust coefficient C_T, which is only a function of J, and the Reynolds number.

$$F = \rho(nd)^2 d^2 C_T = \rho n^2 d^4 C_T \tag{6.11}$$

This equation can be compared to the lift equation of airfoil theory where the constant $1/2$ is absorbed into C_T.

In propeller theory we plot C_T vs J like we plot C_L versus α in airfoil theory. In momentum theory we did not consider the drag of the propeller. In blade element theory we do.

The drag goes to make up a part of the moment required to drive the propeller. This moment is the propeller torque Q:

$$Q = \rho n^2 d^5 C_Q \tag{6.12}$$

Where the torque coefficient C_Q is dependent upon J, R_e, and propeller shape.

The power into the propeller is a function of the torque Q and the rate of rotation ω:

$$P_i = \omega Q = 2\pi n Q = \rho n^3 d^5 2\pi C_Q \tag{6.13}$$

or

$$P_i = \rho n^3 d^5 C_p \tag{6.14}$$

where

$$C_P = 2\pi C_Q$$

Since C_Q was dependent upon J, R_e, and propeller shape, so is C_P. The power coefficient is similar to the drag coefficient of airfoil theory and is usually also plotted vs J. The two curves then resemble lift and drag coefficient plots.

To determine efficiency we consider the power out FV divided by the power in P_i:

$$\begin{aligned} \eta_P &= (TV)/P = (\rho n^2 d^4 C_T V)/(\rho n^3 d^5 C_P) \\ &= (C_T/C_P)(V/nd) = (C_T/C_F)(J) \end{aligned} \tag{6.15}$$

Figure 6.1 would be typical of a chart for fixed-pitch propellers. For this kind of propeller there is only one value of J where the efficiency is at a maximum, or since $J = V/nd$ only one flight condition where η_P is at a maximum. So the propeller for the fixed-pitch, propeller-driven airplane is optimized for

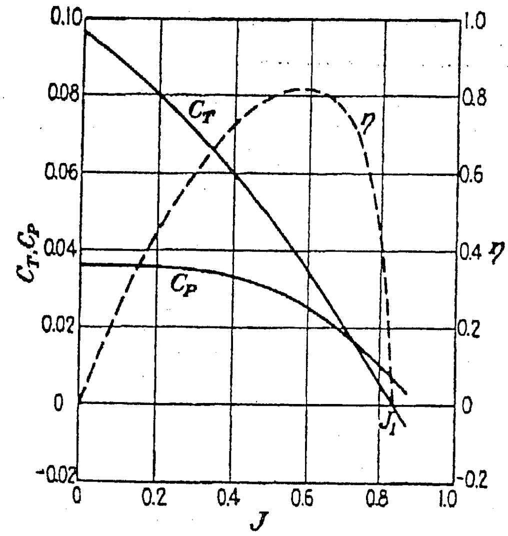


Fig. 6.1 Characteristic curves of a propeller at 15.5 deg blade pitch.[2]

max efficiency in either the climb or cruise condition depending upon its mission.

6.3 Propeller Polar Diagram

For certain flight tests of airplanes it is useful to define the propeller forces to airspeed squared rather than to $(nd)^2$. To do this C_P/J^2 is plotted versus C_T/J^2 such as is shown in Fig. 6.2 for the propeller of Fig. 6.1. From this figure we can see that in this form the propeller data is a straight line which is much easier to use. This curve can be obtained experimentally from flight test data.

6.4 Constant Speed or Controllable Propellers

The efficiency problem with fixed-pitch propellers led to the development of controllable and constant speed propellers. For these propellers there is a family of curves, one set of curves for each blade angle β. This allows us to maintain optimum efficiency over a wide range of J and corresponding flight conditions.

Although a propeller polar such as the one shown in Fig. 6.2 cannot be obtained for a constant speed propeller, Lowry[4] provides a generalized chart for constant speed propellers for light aircraft and Perkins and Hage[5] provide a normalized chart for larger aircraft that was developed by the Boeing Company.

6.5 Activity Factor

The ability of a propeller to absorb power is a function of the blade area as a ratio of the area of the propeller disk. This is expressed as a term called the "activity factor," which represents the integrated power absorption capability

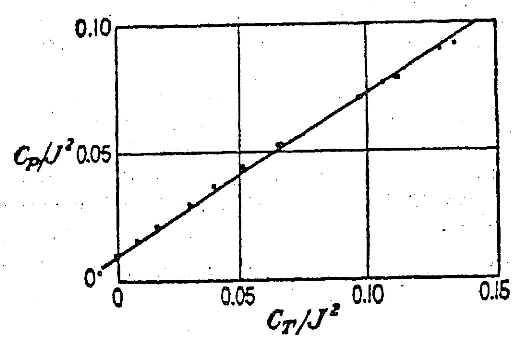


Fig. 6.2 Propeller polar for the propeller of Fig. 6.1 (Ref. 3).

of all the blade elements. The activity factor for a single blade can be found from:

$$A.F. = (100{,}000/16)\int_{.15}^{1.0} (C/D)x^3\,\mathrm{d}x \tag{6.16}$$

where $x = (r/R)$ or

$$A.F. = (100{,}000/16)[(C/D)(x^4/4)]_{.15}^{1.0} \tag{6.17}$$

To determine the total activity factor for the propeller one should multiply the activity factor for a single blade by the total number of blades.

6.6 Propeller Noise

The efficiency of a propeller and the noise made by it are also functions of the tip Mach number. In order for a propeller to be efficient and quiet the tip Mach number should be well below 0.9 Mach. In addition, noise may be reduced by increasing the number of blades. However, as the number of blades increase the efficiency of the propeller decreases. A propeller with only one blade is the most efficient propeller.

References

[1]McCormick, Barnes W., *Aerodynamics of V/STOL Flight*, John Wiley and Sons, Inc., New York, 1967.

[2]Weick, Fred E., "Full-Scale Wind-Tunnel Tests with a Series of Propellers of Different Diameters on a Single Fuselage," NACA Technical Report 339, National Advisory Committee on Aeronautics, Langley Field, VA, Jan. 1931.

[3]Von Mises, Richard, *Theory of Flight*, Dover Publications, Inc., New York, 1959.

[4]Lowry, John T., *Performance of Light Aircraft*, AIAA Education Series, AIAA, Reston, VA, 1999.

[5]Perkins, Courtland D., and Hage, Robert E., *Airplane Performance Stability and Control*, John Wiley and Sons, Inc., New York, 1967.

7
Jet Thrust Measurement In Flight

7.1 Introduction

The jet engine presents a separate set of problems for the flight test engineer when attempting to measure jet thrust or power. In many cases it may not be necessary to measure either one of these variables. Instead, the flight data may be referenced to some indication of jet thrust such as "referred rpm", $N/\sqrt{\theta}$, or fuel flow. The pilot does not need to know what thrust the engine is developing in order to operate the airplane as long as the performance information has been referenced to some related variable. However, we always have the problem in flight testing of knowing if our engine is developing the standard thrust for a given rpm or fuel flow. There are also times when we want to determine airplane drag and to do this we need to determine thrust. Before we examine the methods of in-flight thrust measurement, let us review the basic equations for the thrust of a jet engine.

7.2 Basic Theory[1]

A jet engine produces thrust by taking a mass of air in the intake, adding energy, and accelerating it out the tailpipe. The gross thrust developed by this process can be expressed by the equation:

$$F_G = Q_e V_{ex} + A_e(P_e - P_A) \tag{7.1}$$

where
F_G = gross thrust
Q_e = mass flow through tailpipe
V_{ex} = the air mass exit velocity
A_e = the exit area
P_e = the exit static pressure
P_A = ambient static pressure

A measurement of gross thrust does not consider forward motion of the airplane. As the aircraft's speed increases the air entering the intake is compressed prior to entering the aircraft compressor. This compression produces a force that opposes the motion of the airplane. This force is called ram drag and can be expressed as:

$$D_R = QV_T \tag{7.2}$$

where
D_R = ram drag
Q = mass flow rate of air in the inlet
V_T = true airspeed

The net thrust developed by the engine can then be determined by subtracting the ram drag from the gross thrust:

$$F_N = F_G - D_R \tag{7.3}$$

where
F_N = net thrust

If we assume that the mass flow of air in the inlet is the same as the mass flow at the exhaust then we can say:

$$F_N = Q(V_{ex} - V) \tag{7.4}$$

This is probably a reasonable assumption if we consider that the mass of fuel added to the flow is small in comparison to the total mass of air going through the engine.

In measuring the thrust of a jet engine, we must consider conditions at the tailpipe nozzle where the flow may be either subcritical (called unchoked) or sonic (choked). For subcritical or unchoked flow the exit static pressure is equal to the ambient static pressure or:

$$P_e/P_A = 1 \tag{7.5}$$

where
P_e = the static pressure at the nozzle exit
P_A = the ambient static pressure

and the exit temperatures are related by the equation:

$$T_{Tt}/T_e = (P_{Tt}/P_A)^{(\gamma-1)/\gamma} \tag{7.6}$$

where
T_{Tt} = the total temperature in the tailpipe
T_e = the static temperature at the exit
γ = the ratio of specific heats

The exit velocity for the unchoked case is given by the relation:

$$V_{ex} = \{(2\gamma R/\gamma - 1)T_{Tt}[1 - (P_A/P_{Tt})^{(\gamma-1)/\gamma}]\}^{1/2} \tag{7.7}$$

where
R = the gas constant for air

Using the previous assumption that the inlet mass flow rate is equal to the exit mass flow rate, we can express the mass flow rate by the equation:

$$Q = A_e V_{ex}(P_e/RT_e) \tag{7.8}$$

By substitution into Eq. (7.1) and simplifying, the equation for gross thrust of an engine operating in the unchoked condition becomes:

$$F_G/A_e = (2\gamma/\gamma - 1)P_A[(P_{Tt}/P_A)^{\gamma-1/\gamma} - 1] \tag{7.9}$$

If we introduce the pressure ratio into Eq. (7.9) we have:

$$F_G/\delta A_e = (2\gamma/\gamma - 1)P_{SL}[(P_{Tt}/P_A)^{(\gamma-1)/\gamma} - 1] \tag{7.10}$$

where

P_{SL} = the standard atmospheric pressure at sea level

In this equation the term $F_G/\delta A_e$ is only dependent on the ratio of the total tailpipe pressure to the ambient pressure.

A similar set of equations may be obtained for the case of choked flow. The choked flow case follows:

$$P_e/P_{Tt} = (2/\gamma - 1)^{\gamma/(\gamma-1)} \tag{7.11}$$

$$T_e/T_{Tt} = 2/\gamma + 1 \tag{7.12}$$

$$V_{ex} = (\gamma RT_e)^{1/2} \tag{7.13}$$

The mass flow equation is the same as for the unchoked case. Then, by substitution and simplification, the gross thrust equation becomes:

$$F_G/A_e = P_A[(\gamma + 1)(2/\gamma + 1)^{\gamma/(\gamma-1)} \cdot (P_{Tt}/P_A) - 1] \tag{7.14}$$

If we assume that the value for the specific heat ratio γ is approximately 1.33 for flow at elevated temperatures, then Eq. (7.14) can be further simplified to:

$$F_G/\delta A_e = P_{SL}[1.26(P_{Tt}/P_A) - 1] \tag{7.15}$$

For normal nozzles, the flow becomes choked at a pressure ratio of approximately 1.85. Eq. (7.10) would apply to pressure ratios below this number and Eq. (7.15) to those above it. The above equations are based on one-dimensional, isentropic flow.

Since we know that this is not the actual case in a jet engine exhaust, we must add a factor called the thrust coefficient C_f to Eqs. (7.10) and (7.15) to account for the error obtained by making these assumptions. The resulting equations are

$$F_G/\delta A_e = C_f(2\gamma/\gamma - 1)P_{SL}[(P_{Tt}/P_A)^{(\gamma-1)/\gamma} - 1] \tag{7.16}$$

for unchoked flow, and

$$F_G/\delta A_e = C_f P_{SL}[1.26(P_{Tt}/P_A) - 1] \tag{7.17}$$

for choked flow. For the choked flow condition the thrust coefficient is nearly constant varying only between 0.95 and 1.0. For the unchoked case the value of C_f gradually decreases as the pressure ratio decreases.

7.3 Methods of In-Flight Thrust Measurement

There are a number of methods available to the flight test engineer for in-flight thrust measurement on jet power aircraft. However, like power determination on a reciprocating engine-powered aircraft, they all have their weaknesses.

7.3.1 *Jet Flow Measurement*[1]

One of the more common methods is called the jet flow measurement method. This method works reasonably well on all types of jet engines and can be used as a check on other methods. The method is based on Eqs. (7.16) and (7.17), and gross thrust is determined by measuring the engine pressure ratio (EPR) and solving for gross thrust by use of the equations.

The thrust coefficient C_f is determined by measuring the thrust as a function of EPR during a ground static calibration and plotting C_f as a function of EPR as shown in Fig. 7.1. This ground static calibration should be conducted for each test installation and should be repeated if any engine or airframe component is changed during the test. Since during the ground static calibration it will not be possible to obtain as high a value for EPR as will be obtained in flight, the plot of C_f vs EPR must be extrapolated to the higher values of EPR. It is this extrapolation that is the most likely source of error in the method.

7.3.2 *Engine Manufacturer's Data*

Another method for in-flight thrust determination is the use of the engine manufacturer's thrust curves and engine calibration data. This is one of the least accurate methods since it does not account for installed thrust losses, and the engine thrust calibration is normally only conducted at sea level. This method would be used only when schedule time or budget restraints did not allow use of a more accurate method.

7.3.3 *Wind Tunnel Calibration*

If the facilities are available, a calibration of the engine in altitude-capable wind tunnel with the operational inlet and nozzles installed can provide accurate values of both gross and net thrust. This data may then be used for determining thrust in flight. However, care must be taken to insure that the engines used in the flight test and the wind tunnel calibration have the same thrust characteristics. It is preferred to use the same engine for the flight tests as was calibrated in the tunnel, since there may be considerable variation in thrust

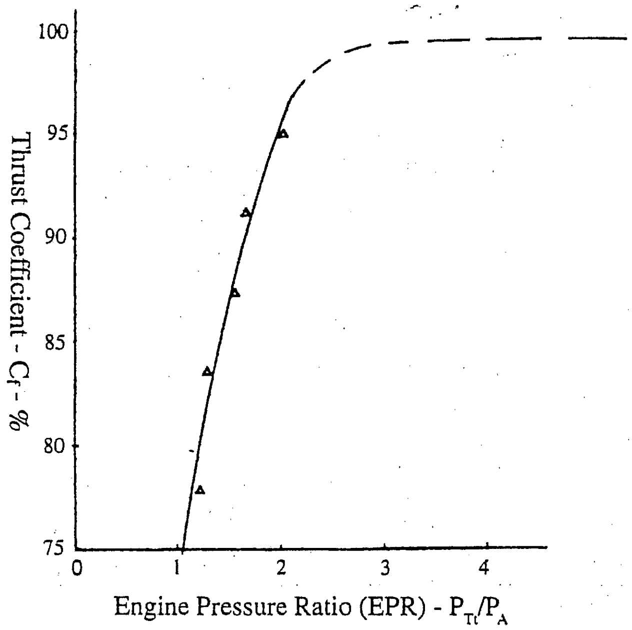


Fig. 7.1 Thrust coefficient vs EPR from static ground calibration.[1]

from engine to engine of the same design. This method costs a large sum of money and is normally only available to those testing military aircraft.

7.3.4 *Climb Performance Method*[2]

The jet flow method of thrust determination can be criticized for the fact that its accuracy is dependent upon the determination of the thrust coefficient C_f and that C_f values must be extrapolated to EPRs that can be obtained in flight. In addition, this method does not account for the thrust changes due to afterburner operation. There is also a question about the effects of altitude on C_f.

A method developed by the French called the climb performance method removes some of the inaccuracies of the jet flow method and is actually more of an extension of that method than a method that stands alone.

We can assume that the values of C_f obtained during the ground static calibration are accurate at low flight speeds and low altitudes. If we then determine the net thrust by this method in a low altitude climb at best rate-of-climb speed, we can assume it is accurate. If we compare the thrust at this altitude with the thrust obtained at a higher altitude, but with the same values of airspeed and $N/\sqrt{\theta}$, then we can determine if any change with altitude of C_f exists.

By using the same values of climb airspeed and $N/\sqrt{\theta}$ we can assume that the inlet drag is the same for both altitudes. If we also measure the rates of climb at both altitudes we can compare the net thrust at both altitudes by use of the thrust horsepower in excess equation.

$$(F_{NH} - D_H)/(F_{NL} - D_L) = (m_H/m_L)(RC_H/RC_L)(V_H/V_L)^{-1} \tag{7.18}$$

where
m = airplane mass
RC = rate of climb
V = true airspeed
D = airplane drag

subscripts
H = high altitude data
L = low altitude data

By mathematical manipulation we can rewrite Eq. (7.18) as:

$$\begin{aligned} F_{NH}/F_{NL} = {} & (m_H/m_L)(RC_H/RC_L)(V_H/V_L)^{-1} \\ & + \{D_H/D_L + (m_H/m_L)(RC_H/RC_L)(V_H/V_L)^{-1}\}D_L/F_{NL} \end{aligned} \tag{7.19}$$

In a climb the first term of Eq. (7.19) decreases with increasing altitude. The second term only changes slightly. At altitudes where the rate of climb ratio is greater than 0.5 it may be considered of second order and calculated from wind tunnel data.

In addition to providing data on any change in C_f with altitude, Eq. (7.19) may be used to determine the thrust increases due to afterburner operation.

References

[1]Herrington, R. M., Maj., Shoemacher, P. E., Bartlett, E. P., and Dunlap, E. W., "USAF Flight Test Engineering Handbook," AFFTCTRN 6273, Air Force Flight Test Center, Edwards AFB, CA, May 1961, rev. June 1964, Jan. 1966.

[2]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. I—Performance*, Pergamon Press, New York, 1959.

8
Level Flight Performance Theory

8.1 Introduction

Level flight performance may be described as steady state performance. It is a condition where all of the forces acting on the airplane are in balance or:

$$L = W - F \cos \alpha_F \tag{8.1}$$

$$F \cos \alpha_F = D \tag{8.2}$$

where
L = lift
W = airplane weight
F = thrust
α_F = the thrust angle with the horizontal

However, if we assume small angles we can say that thrust is equal to drag and lift is equal to weight.

In level flight performance we are primarily concerned with Eq. (8.2) or thrust F and drag D; however, we shall see that the lift L and weight W of Eq. (8.1) also play a part.

8.2 Thrust Required

Level flight performance is essentially a determination of airplane drag as a function of velocity. Since the thrust must equal the drag it could also be called the thrust required for a given velocity.

The drag of an airplane is composed of two components. They are 1) parasite drag; and 2) induced drag.

The parasite drag is composed of all forms of drag other than the drag due to creating lift which is the induced drag.

Parasite drag can be defined by the equation:

$$D_P = 1/2 C_{DP} \rho V_T^2 S \tag{8.3}$$

where
D_P = parasite drag
C_{DP} = parasite drag coefficient
S = reference area (normally taken as the wing planform area)

From this equation we can see that drag varies as a function of the true airspeed squared.

The induced drag is defined by the equation:

$$D_i = 2L^2/\rho V_T^2 S\pi ARe \tag{8.4}$$

where

D_i = induced drag
S = wing area
AR = aspect ratio
e = Oswald's efficiency factor

This equation shows that the induced drag varies as a function of $1/V_T^2$.

A plot of both parasite and induced drag as a function of airspeed is shown in Fig. 8.1.

If the profile and induced drag are added we arrive at the total drag of the airplane. This plot is also shown in Fig. 8.1. Since thrust must equal drag this curve may also be called the thrust required curve. There are several things that should be known about the thrust required curve for an airplane. First, it is limited on the low-speed end by the stalling speed. Second, the velocity for minimum drag occurs where the parasite and induced drag are equal. This point is the best lift to drag ratio $(L/D)_{\max}$ for the airplane. For a jet airplane this is the speed for both best endurance and best glide.

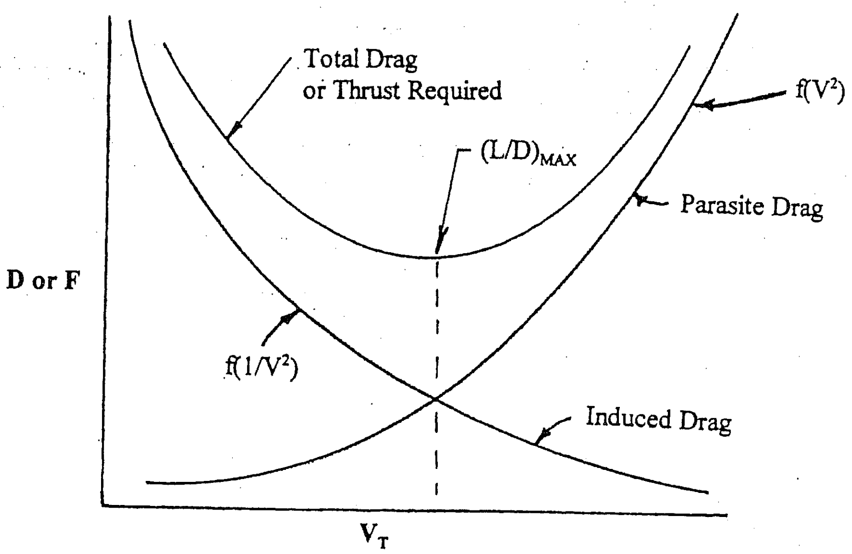


Fig. 8.1 Composition of thrust required curve.

8.3 Effects of Variables on Thrust Required

If one examines Eqs. (8.3) and (8.4) it may be seen that there are two variables other than those already discussed which can affect the thrust required. They are the lift and the air density. If we say that in steady state lift is equal to weight then in actuality the weight becomes the variable.

8.3.1 *Effects of Weight*

Since the lift, or weight, only appears in the equation for induced drag [Eq. (8.4)], we can say that the weight only affects the induced portion of the thrust required curve. This makes weight effects predominant at the low-speed end of the curve as is shown in Fig. 8.2. An increase in weight increases the thrust required. Fig. 8.2 also shows that the speed for $(L/D)_{max}$ increases as weight increases.

8.3.2 *Effects of Air Density*

In examining Eqs. (8.3) and (8.4) we can see that a change in density will affect the profile drag directly and the induced drag as a function of $1/\rho$. This has the effect of shifting the thrust required curve to the right as density decreases as is shown in Fig. 8.3. The value of minimum drag or thrust required remains the same, but the true airspeed at which it occurs increases. The effects of density on the thrust required curve can be negated if the thrust plotted against equivalent airspeed V_e. Then only one curve will result. Weight,

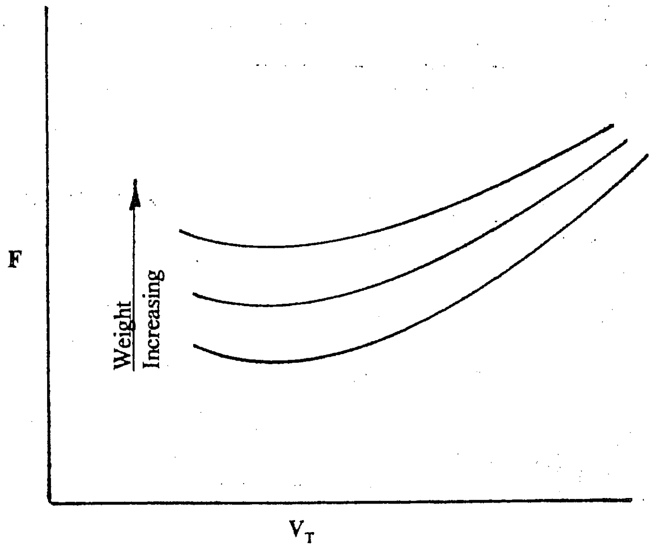


Fig. 8.2 Effects of weight on thrust required.

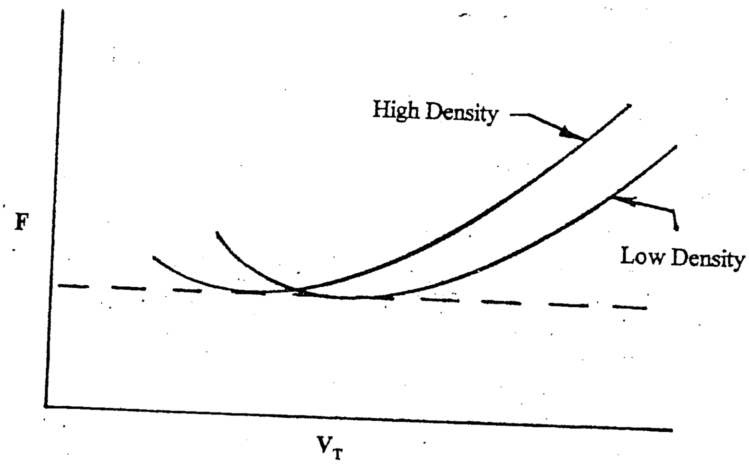


Fig. 8.3 Effects of air density on thrust required.

however, will still cause a shift in the curve even when plotted against V_e as is shown in Fig. 8.4.

8.4 Power Required

In propeller-driven airplanes, it is more convenient to use power required than thrust required. The thrust horsepower required can be stated by the equation:

$$THP_R = F_R V_T / 550 \tag{8.5}$$

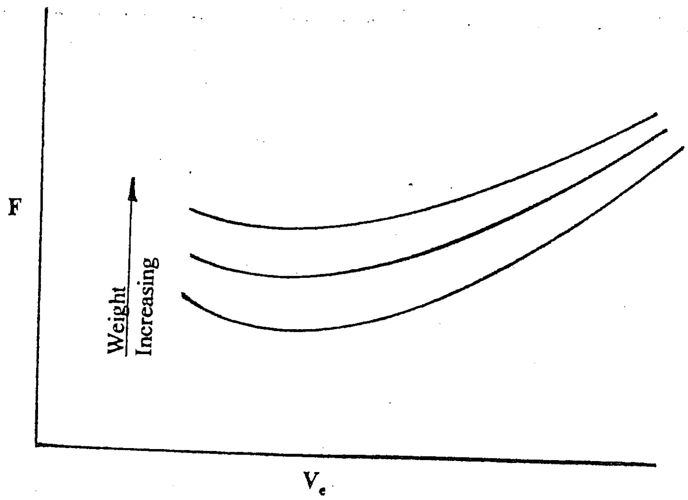


Fig. 8.4 Effects of weight on thrust required when plotted vs equivalent airspeed.

where
THP_R = thrust horsepower required
F_R = thrust required in pounds
V_T = true airspeed in feet per second

If we combine Eqs. (8.3) and (8.4) to obtain the thrust required and substitute that equation into Eq. (8.5) we have:

$$FHP_R = [1/2C_{DP}\rho V_T^3 S + 2L^2/\rho V_T S\pi ARe]/550 \tag{8.6}$$

From this equation we can see that the profile power required varies as the true airspeed cubed, and the induced power required varies as $1/V_T$. This is shown in Fig. 8.5. By adding the profile power required to the induced power required we have the total power required curve also shown in Fig. 8.5. On this curve the minimum drag or best L/D occurs at a higher speed than the speed for lowest FHP_R. The minimum FHP_R is the speed for best endurance on a propeller-driven airplane while the speed for $(L/D)_{max}$ is the best range speed.

8.5 Effects of Variables on the Power Required Curves

Although the same variables affect the power required curve as affect the thrust required curve, they affect them in a different manner.

8.5.1 Effects of Weight

Since weight affects the induced portion of the power required curve its greatest effect occurs at the low-speed end. The general effects of weight on the power required curve are shown in Fig. 8.6.

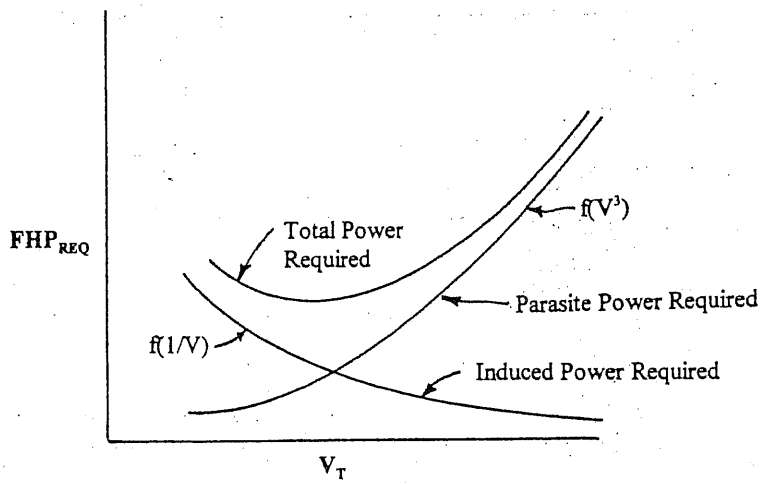


Fig. 8.5 Composition of power required curve.

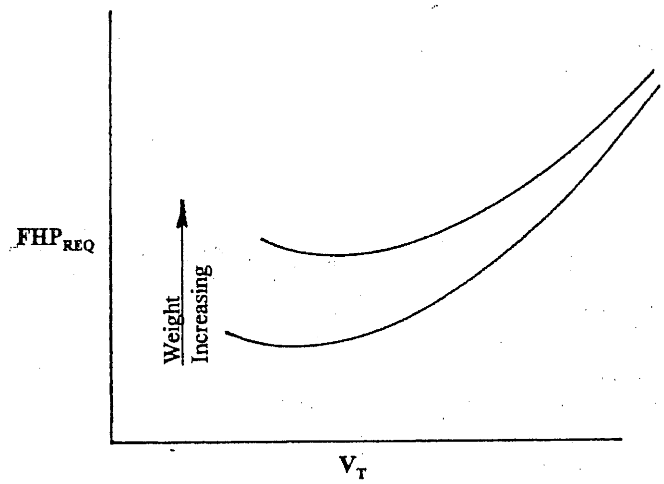


Fig. 8.6 Effects of weight on the power required curve.

In order for the airplane to maintain a given C_L, or angle of attack, when the weight changes one must vary the airspeed. This can be expressed by the relationship:

$$V_2/V_1 = \sqrt{W_2/W_1} \tag{8.7}$$

where
V_1 = the V of a specific C_L and weight, W_1
V_2 = the V for the same C_L but a different weight, W_2

This effect applies to the FHP_R curves and the airspeed will vary by the same relationship on these curves when the weight changes.

8.5.2 Effects of Density

Air density also affects the power required curves. Decreasing density tends to move the curves upward and to the right as shown in Fig. 8.7. The $(L/D)_{\max}$ moves up and to the right along a straight line as is shown. If we try to cancel the effects of density by plotting FHP_R vs V_e we find that we stop the movement of the curve to the right with decreasing density, but it still moves upward as is shown in Fig. 8.8. This is because if we substitute the relation $V_T = V_e\sqrt{\sigma}$ into Eq. (8.6) we find that we have:

$$FHP_R = [C_{DP}\rho_0 V_e^3 S/2\sqrt{\sigma} + 2L^2/\rho_0\sqrt{\sigma}V_e S\pi ARe]/550 \tag{8.8}$$

Although we have eliminated the density at altitude from this equation still remaining is the square root of the density ratio σ, so, we still have a density effect.

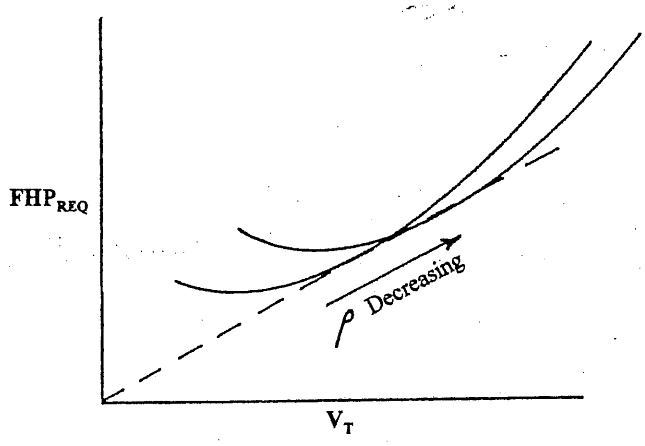


Fig. 8.7 Effects of density on power required.

8.6 Effects of High Mach Number

As an aircraft airspeed approaches Mach 1, the zero lift drag coefficient becomes a variable. (For low Mach numbers it is essentially constant.) This change is caused primarily by the introduction of wave drag.

The result of this change in drag is a rapid increase in the thrust required until Mach 1 is reached. After Mach 1 the drag and the thrust required begins to decrease. This phenomenon is described as the transonic drag rise or transonic thrust pinch.

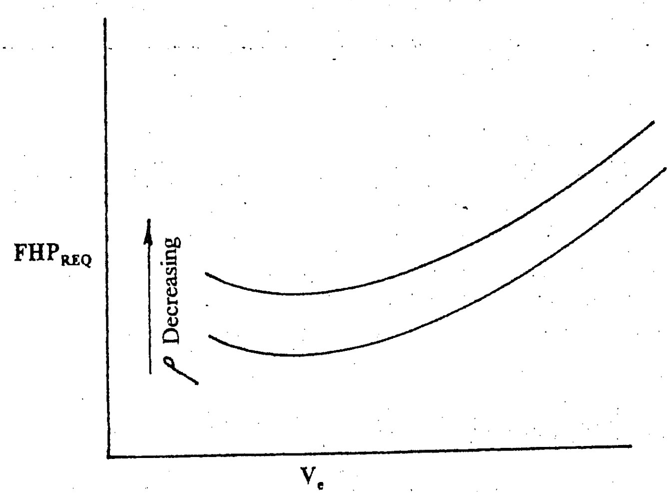


Fig. 8.8 Effects of air density on power required when plotted vs equivalent airspeed.

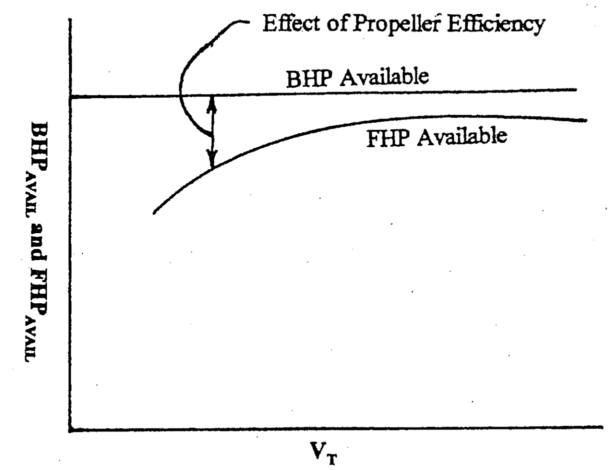


Fig. 8.9 Variation in thrust horsepower available due to propeller efficiency.

8.7 Thrust or Power Available

The thrust or thrust horsepower available is a function of the installed thrust or thrust horsepower of the engine.

Thrust or thrust horsepower will vary with air density; therefore, there will be a different value of thrust or power available for each set of altitude and temperature conditions. Since thrust or thrust horsepower available defines the maximum speed of the airplane, the maximum speed will vary as thrust or power available varies.

On propeller-driven airplanes the propeller efficiency is another variable that will affect the power available. Since propellers are designed to have their maximum efficiency in the normal cruising range of the airplane, the efficiencies will drop on either side of this range. This drop in efficiency occurs on the high-speed end due to tip losses caused by the propeller blades approaching their critical Mach number. On the low-speed end the drop in efficiency is caused by the blades operating at too high an angle of attack. Fig. 8.9 shows a plot of thrust horsepower available and how it varies with propeller efficiency.

References

[1]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Theory," FTC-TIH 64-2005, AFFTC, USAF Test Pilot School, Edwards AFB, CA, 1965.

[2]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH 64-2006, AFFTC, USAF Test Pilot School, Edwards AFB, CA, 1965.

[3]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. I—Performance*, Pergamon Press, New York, 1959.

[4]Hamlin, Benson, *Flight Testing*, The MacMillan Co., New York, 1946.

9

Level Flight Performance Flight Test and Data Reduction Methods for Propeller-Driven Aircraft

9.1 Introduction

There are a number of methods for level flight performance data reduction for propeller-driven airplanes. Most of these methods have problems with determining full throttle performance under standard conditions and are not readily useable for determining airplane drag coefficients. The method we will discuss here solves both of these problems. This method is called the PIW-VIW method.

9.2 Federal Aviation Administration Requirements

There are no FAA requirements for level flight performance. However, some of the acceptable methods for FAA climb performance require the use of parameters derived from the airplane drag polar which is obtained from level flight performance.

9.3 PIW-VIW Method

The PIW-VIW[1] method is derived by making the power required curve a single curve good for all air densities and airplane weights. To accomplish this we will make two assumptions. The first assumption is that at a given angle of attack both the lift coefficient C_L and the drag coefficient C_D are constant. When we assume that the angle of attack is the same for two different conditions of weight and altitude we can equate these two conditions through the following:

$$\frac{W_S}{W_T} = \frac{L_S}{L_T} = \frac{1/2\rho_{SL}\sigma_S V_S^2 S C_L}{1/2\rho_{SL}\sigma_T V_T^2 S C_L} \tag{9.1}$$

where
subscript S = standard conditions
subscript T = test conditions

If we cancel like terms through division and solve for the standard airspeed (V_S) which we will call VIW, and replace the true airspeed V_T and $\sqrt{\sigma}$ with equivalent airspeed V_e we have:

$$VIW = \frac{V_e}{\sqrt{W_T/W_S}} \tag{9.2}$$

The equivalent airspeed can be replaced by calibrated airspeed at low speeds and altitudes.

The power term, PIW, can be determined by assuming that the drag coefficient is constant at that angle of attack and using a similar mathematical approach to that used for VIW.

$$\frac{THP_S}{THP_T} = \frac{V_S/550(C_D 1/2\rho_{SL}\sigma_S V_S^2 S)}{V_T/550(C_D 1/2\rho_{SL}\sigma_T V_T^2 S)} \tag{9.3}$$

where
THP = the thrust horsepower and the subscripts have the same meaning as above

If we cancel like terms through division and substitute the VIW value for V_S we have:

$$THP_S = THP_T(W_S/W_T)^{3/2}\sigma_T^{1/2} \tag{9.4}$$

If we assume constant propeller efficiency, which is not a bad assumption for constant speed propellers on the front side of the power required curve, we can replace thrust horsepower with brake horsepower, BHP. Then by inverting the weight term we can rewrite the equation as:

$$PIW = \frac{\mathrm{BHP}_T\sqrt{\sigma_T}}{(W_T/W_S)^{3/2}} \tag{9.5}$$

Plotting PIW vs VIW provides one curve good for all air densities and airplane weights. It is the curve that the airplane would construct if flown at the standard weight (normally selected as the maximum takeoff gross weight) at sea level on a standard day.

9.4 Flight Test Method

The flight tests for level flight performance are simple and easy to perform. Testing should be conducted in smooth air as turbulence makes obtaining data difficult and will cause an apparent reduction in cruise airspeed for a given power setting. Clear, early mornings are preferred times to collect data since the air is usually smooth at all altitudes. As long as calibrations exist on the ship's instruments, they may be used for data collection and the data hand recorded. Nothing happens very fast during level flight performance tests so automatic data recording devices are not necessary.

9.4.1 *Method for Constant Speed Propeller Driven Airplanes*

For these aircraft, the test is initially conducted at maximum cruise rpm beginning at low altitude.

The procedure is to first stabilize at as low an airspeed as possible at the test altitude and record the ambient temperature. This provides an ambient temperature reading with little ram rise. Then the aircraft is accelerated at full throttle and maximum cruise rpm to maximum level flight speed. This may take some time (5 min or more). Once stabilized at maximum level flight speed, the following data is recorded:

1) airspeed
2) altitude
3) ambient temperature
4) rpm
5) manifold pressure (or torque for turbo-prop aircraft)
6) fuel quantity (for test weight determination)
7) fuel flow
8) carburetor air temperature (if available)

Once the data are collected, the manifold pressure is reduced an increment of 1 or 2 in. Hg, the airspeed allowed to stabilize, and the data collected once again. This procedure is repeated until the aircraft reaches the back side of the power required curve where the manifold pressure must be increased to maintain altitude and achieve a lower airspeed. It is worthwhile to obtain one or two data points on the back side of the power required curve. Data should be read in the sequence shown above and repeated in that sequence for each data point. The test should be repeated at medium and high altitudes to obtain a good data sample. After completing the high altitude test, full throttle manifold pressure should be obtained at each altitude during the descent so a manifold pressure vs pressure altitude curve can be obtained. If a secondary cruise rpm value is going to be given in the Pilot's Handbook, then the test should be repeated at this rpm.

9.4.2 *Method for Fixed-Pitch Propeller-Driven Airplanes*

The method for fixed-pitch propeller-driven airplanes is the same as for constant speed propeller airplanes except that rpm will change with each change in manifold pressure. Also on these airplanes, the low and medium altitude tests will be limited by maximum allowable rpm.

9.5 Reduction of Observed Data

The data reduction methods for propeller-driven airplanes can be divided into 1) those for constant speed propeller-driven airplanes; and 2) those for fixed-pitch propeller-driven airplanes.

9.5.1 Constant Speed Propeller

For airplanes with a constant speed propeller we use the PIW vs VIW method to reduce our data. In this method we must first determine our equivalent airspeed V_e (calibrated airspeed may be used for low speed airplanes), and our installed BHP at the test point. This is accomplished using the methods previously discussed in the chapters on airspeed calibration and determining engine power in-flight.

The second step in the reduction process is to determine the test density ratio σ. To accomplish this we use the equation:

$$\sigma = \delta/\theta \tag{9.6}$$

where
δ = pressure ratio (P_a/P_{SL})
θ = temperature ratio (T_a/T_{SL})

The pressure ratio can be obtained from a table lookup, entering the table with the calibrated pressure altitude Hp_C or by the equation given in Chapter 1.

The temperature ratio can be obtained by first determining the absolute temperature at the test altitude:

$$T_a = OAT_C + 273.16 \tag{9.7}$$

where
T_a = absolute temperature at the test altitude in degrees Kelvin
OAT_C = calibrated outside air temperature in degrees Celsius

and then dividing by the standard absolute temperature at sea level.

Once we have determined the test density ratio σ_T, the third step is to determine the test aircraft weight and the weight ratio W_T/W_S.

The next step is to determine values for PIW and VIW using Eq. 9.5 and 9.2 where all values are at the test data point.

We then plot the values of PIW and VIW for each data point as shown in Fig. 9.1.

9.5.2 PIW vs VIW Normalization

Since the values of PIW and VIW obtained above have data scatter and the proper fairing of a curve may be difficult, a normalization technique is used to make the curve into a straight line. It is then somewhat easier to fair a curve through the scattered data. We accomplish this mathematically by multiplying both PIW and VIW by VIW. This equation then becomes:

$$PIW \times VIW = f(VIW^4) \tag{9.8}$$

If we plot this equation we have a straight line as shown in Fig. 9.2. This plot does become nonlinear on both ends. On the high-speed end when our airplane

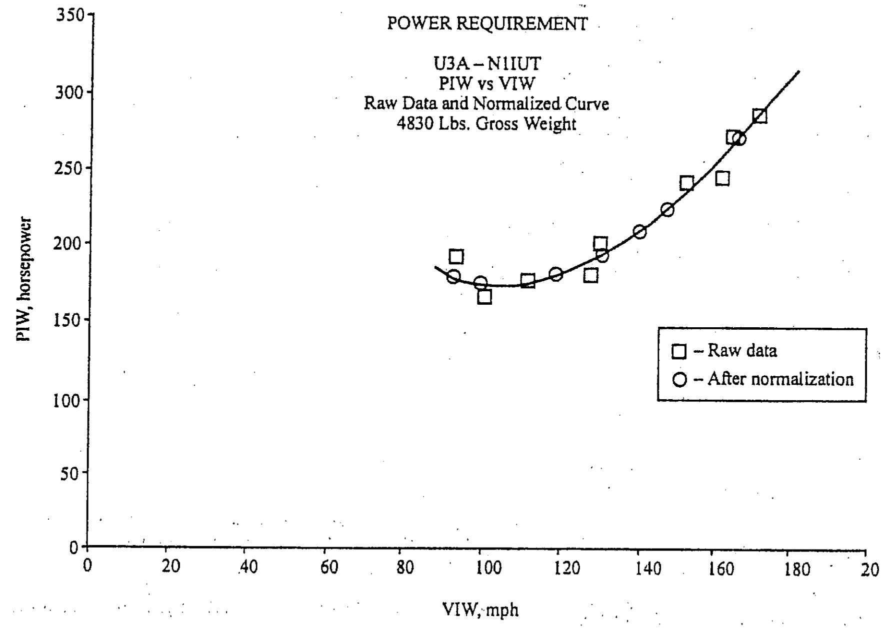


Fig. 9.1 PIW-VIW plot for a Cessna U3A. Reprinted with permission from Commander Aircraft.

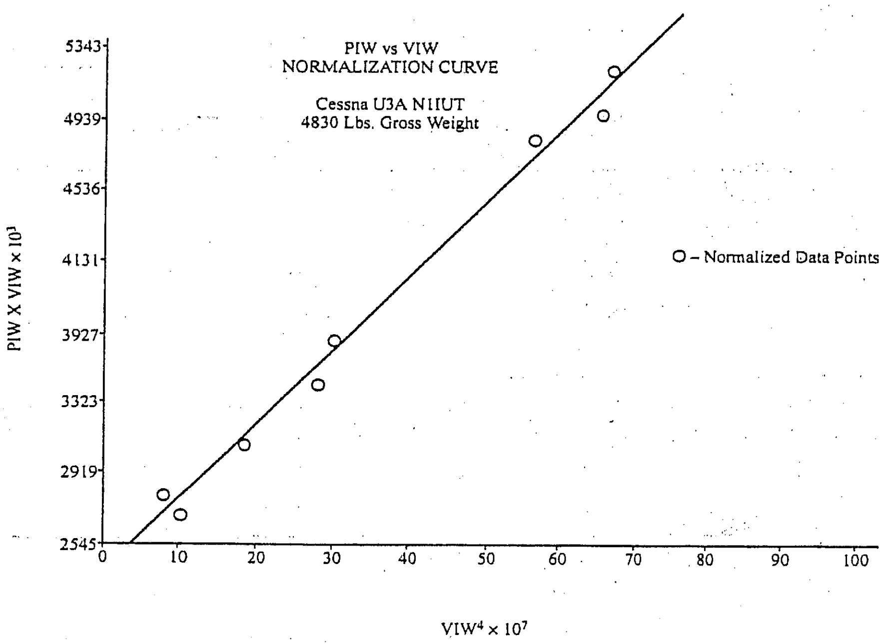


Fig. 9.2 PIW-VIW normalization from data of Fig. 9.1. Reprinted with permission from Commander Aircraft.

is fast enough to encounter the transonic drag rise and on the low-speed end where we start to get flow separation. At all other speeds it is a straight line, and useful in data fairing. Once we have obtained this straight line plot, we then read points from it, reconvert them to values of PIW and VIW, and use these values for our faired line in Fig. 9.1. We then have a smooth curve faired through the scattered data points.

9.5.3 *Fixed-Pitch Propellers*

For airplanes with fixed-pitch propellers the change in rpm is so great with change in power that we can no longer assume propeller efficiency to be constant. Also, since rpm is the only power parameter with which the pilot has to work, it is necessary to have some method to relate the power the engine is developing to the rpm. We accomplish this and account for the changes in propeller efficiency by correcting the rpm for density and weight in a manner similar to that used for the PIW and VIW terms and call it NIW.

$$NIW = \frac{rpm_I\sqrt{\sigma_T}}{\sqrt{W_T/W_S}} \tag{9.9}$$

We then plot these variables into three plots as shown in Figs. 9.3, 9.4, and 9.5.

Fig. 9.3, the plot of PIW vs VIW, is only used for developing a part of the maximum power performance since for all other powers we must be able to correlate rpm and power.

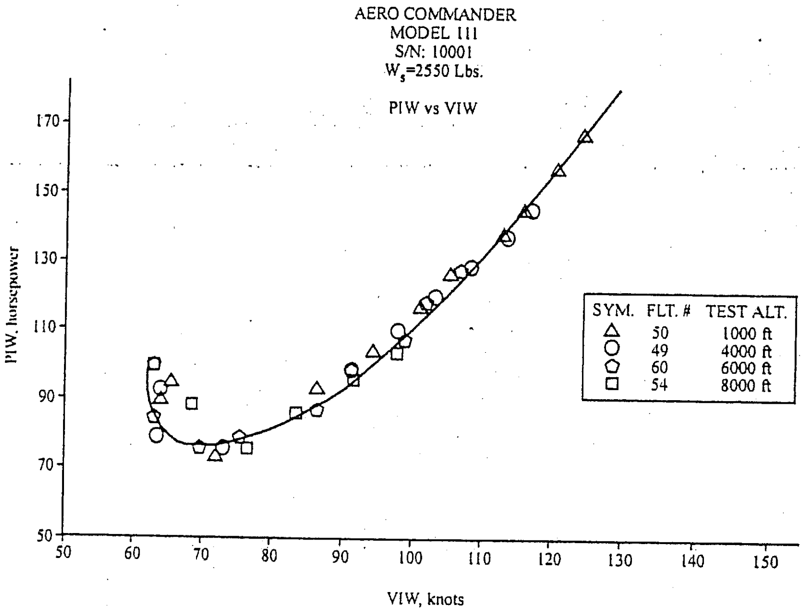


Fig. 9.3 PIW-VIW plot for determining maximum power performance. Reprinted with permission from Gulfstream Aerospace Corporation.

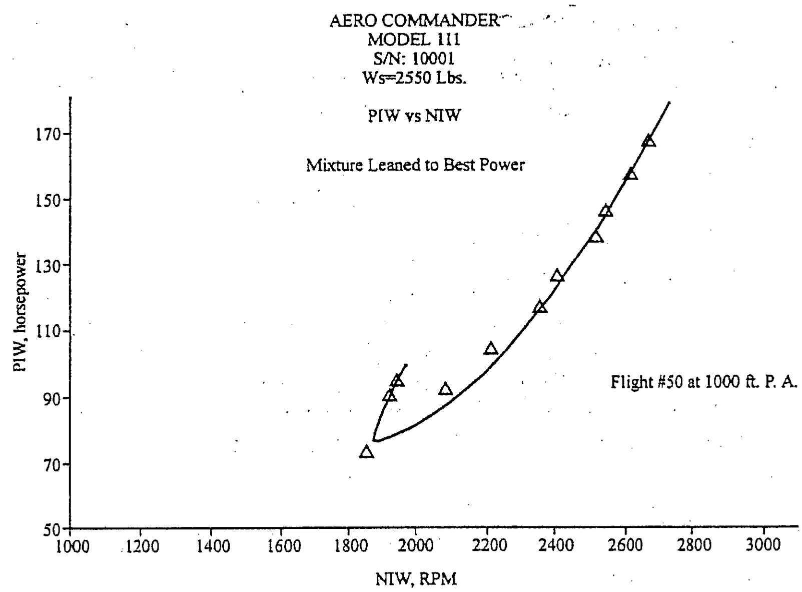


Fig. 9.4 PIW-NIW standardized power chart for a fixed pitch propeller airplane. Reprinted with permission from Gulfstream Aerospace Corporation.

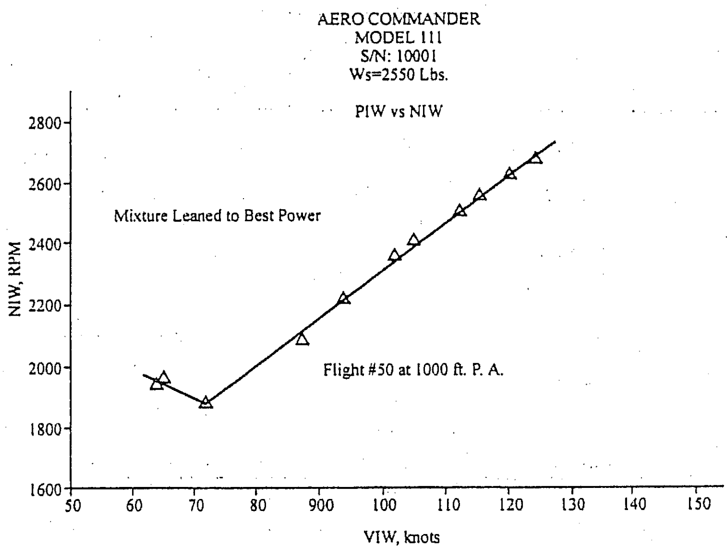


Fig. 9.5 NIW-VIW plot standardized rpm vs standardized airspeed. Reprinted with permission from Gulfstream Aerospace Corporation.

Fig. 9.4, the plot of PIW vs NIW, gives us the method of relating the engine power to the rpm.

Fig. 9.5, the plot of NIW vs VIW, allows us to determine the airspeed for a handbook performance plot. The method for developing this performance plot will be discussed later.

9.6 Expansion of Observed Data

The plots of PIW vs VIW or PIW vs NIW are not readily useable to the pilot. More useable to the pilot is a plot of true airspeed vs density altitude for a given power setting and a plot of percent power for a given rpm setting vs density altitude. In order to obtain such useful plots we must expand the PIW vs VIW, PIW vs NIW, and NIW vs VIW plots to non-standard conditions from the sea level standard conditions that they represent. Again, we divide the methods into those for constant speed propeller-driven airplanes and those for fixed-pitch propeller-driven airplanes.

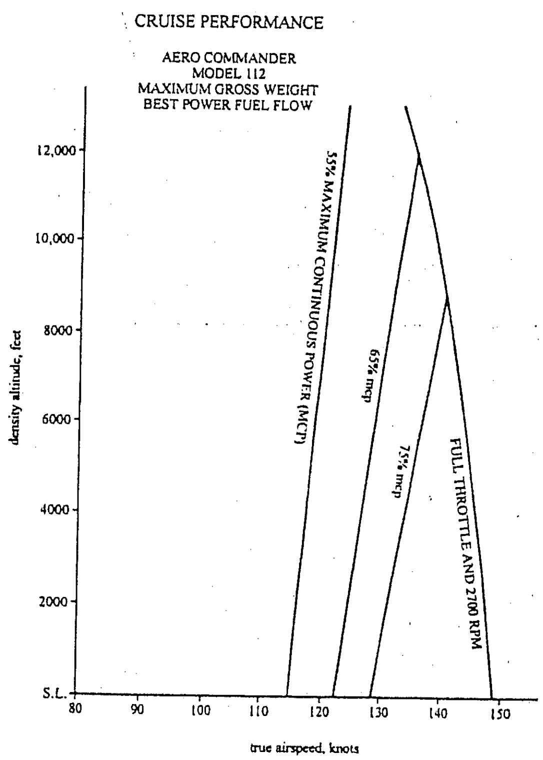


Fig. 9.6 Density altitude vs true airspeed Aero Commander 112. Reprinted with permission from Commander Aircraft.

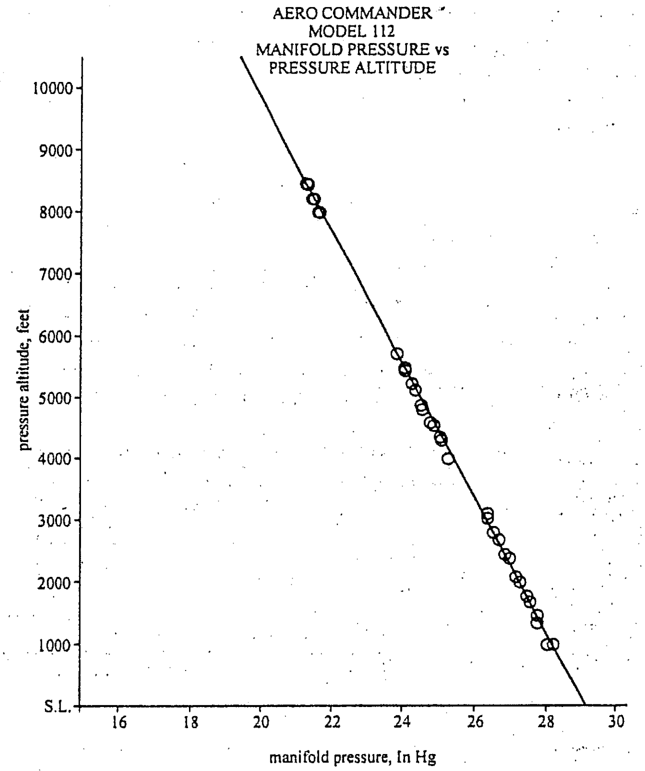


Fig. 9.7 Pressure altitude vs installed manifold pressure. Reprinted with permission from Gulfstream Aerospace Corporation.

9.6.1 Constant Speed Propeller Data Expansion

To develop the density altitude vs true airspeed plot (Fig. 9.6), we first develop the full throttle line at maximum rpm. To do this we must know how the full throttle manifold pressure varies with altitude at that rpm. We accomplish this by flying at full throttle and max rpm, in level flight, at several different altitudes, and record manifold pressure vs pressure altitude. We apply instrument corrections and plot this as shown in Fig. 9.7.

Using this information, we enter the standard engine power chart and determine the full throttle engine power at each altitude. This power information is then converted to values of PIW by determining the standard day square root of the density ratio σ for each altitude and multiplying that value times the horsepower determined from the power chart.

We then enter the PIW vs VIW plot at this value of PIW and determine VIW.

True airspeed is obtained by dividing VIW by the square root of the density ratio σ for the altitude in question.

Table 9.1 Cruise performance model XXX—Full throttle maximum gross weight

Standard Altitude—ft	Standard BHP	Installation BHP[a]	$\sqrt{\sigma}$	PIW[b]	VIW[c]	V_T-kn
Sea Level	200	198.1	1.000	198.1	148	148
2000	187.2	185.4	0.9710	180.0	140.2	144.4
4000	174.5	172.8	0.9424	162.84	133.3	141.5
6000	162.8	161.2	0.9143	147.38	127.0	138.9
8000	152.0	150.5	0.8866	133.43	122.2	137.8
10,000	140.7	139.2	0.8593	119.61	117.2	136.4
12,000	130.5	129.1	0.8326	107.48	111.3	133.6
14,000	121.0	119.7	0.8062	96.07	103.5	128.4

[a]Corrects standard BHP for known installation losses such as induction air heat rise.
[b]PIW = Inst. BHP × $\sqrt{\sigma}$.
[c]Read VIW from test PIW vs VIW plot using PIW calculated above.

A table of the just described approach is shown in Table 9.1. This same procedure may be used for the full throttle lines of rpm, other than maximum rpm, providing a full throttle manifold pressure vs altitude line is developed for these rpm.

To develop the lines of constant percent power the same approach is used. Except in this case, the horsepower developed by the engine does not change with altitude, so the values of PIW only change with the square root of the density ratio σ.

A table of data for a percent power line is shown in Table 9.2. The plot shown in Fig. 9.6 is actually a plot of density altitude vs true airspeed for a standard day. Since density altitude on a standard day is the same as pressure altitude, we use pressure altitude in developing the plot. However, we can enter this plot with a density altitude and get the correct true airspeed for that density altitude. To aid in doing this, a density altitude chart is sometimes

Table 9.2 Cruise performance model XXX—75% MCP maximum gross weight

Standard Altitude—ft	Standard BHP[a]	$\sqrt{\sigma}$	PIW[b]	VIW[c]	V_T-kn
Sea Level	150	1.0000	150.0	126.7	126.7
2000	150	0.9710	145.65	125.6	129.4
4000	150	0.9424	141.36	124.1	131.7
6000	150	0.9143	137.14	123.3	134.8
8000	150	0.8866	133.0	121.2	136.7

[a]The standard BHP for this case is 75% of the maximum rated power, which for this engine was 200 hp. No correction is made for installed losses since it is assumed that the pilot can increase the throttle to obtain 75% maximum continuous power.
[b]PIW is obtained in the same manner as in Table 9.1.
[c]VIW is read from the test PIW vs VIW plot.
Note: True airspeed is obtained by dividing VIW by square root of the standard density ratio.

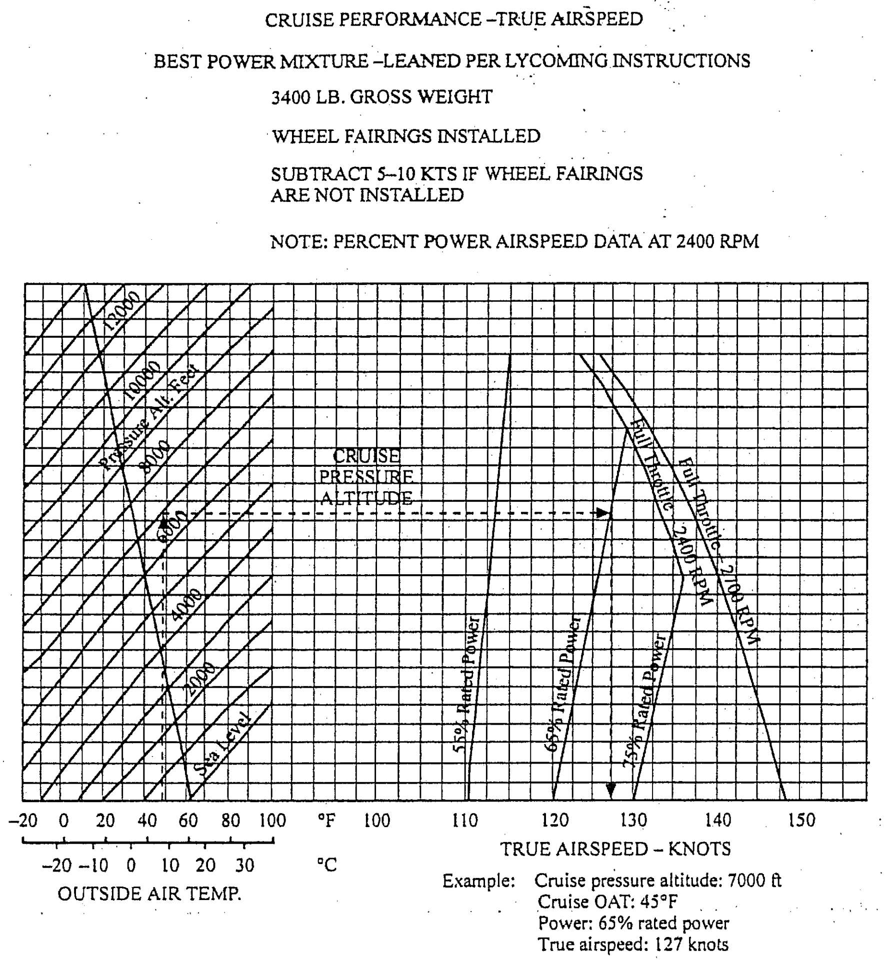


Fig. 9.8 Cruise performance plotted with density altitude conversion chart.

plotted to the left as shown in Fig. 9.8 so the pilot may enter with values of pressure altitude and temperature and determine his true airspeed.

9.6.2 Fixed-Pitch Propeller Data Expansion

To expand the fixed-pitch propeller data we use a slightly different approach than was used for the constant speed propeller data. The full throttle line becomes a maximum power line since the lower altitude part of it is limited by maximum rpm, while the higher altitude part is limited by full throttle. The lower part of this curve is obtained by entering the PIW vs NIW, and NIW vs VIW charts at NIW values for the maximum allowable rpm (i.e., max rpm $\times \sqrt{\sigma}$).

Fixed-pitch propellers are usually pitched to allow maximum rpm to the highest altitude where maximum cruise power can be obtained. We would then continue with the procedure just described until we reached this altitude.

Above this altitude we need to know how both the rpm and manifold pressure vary under standard conditions. This is difficult since on a fixed-pitch propeller installation both rpm and manifold pressure are interdependent and a change in one will affect the other. One method of accounting for this is to fly in level flight at full throttle at several altitudes above the altitude where maximum cruise power can be obtained and record values of rpm and manifold

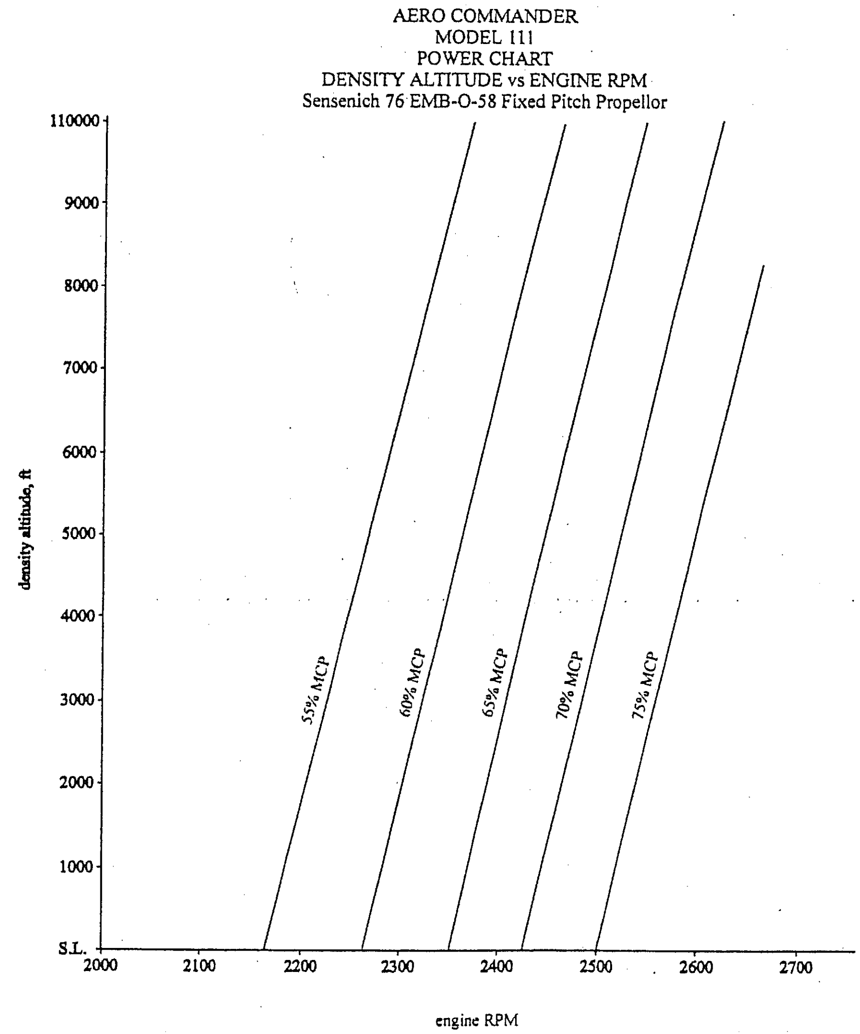


Fig. 9.9 Handbook power chart for fixed-pitch propeller-driven aircraft. Reprinted with permission from Gulfstream Aerospace Corporation.

pressure. We then plot these values, after correcting for instrument error, and determine the slopes of the lines. We may then use these slopes and the values for rpm and manifold pressure on a standard day at the maximum altitude where maximum cruise power can be obtained to determine the power characteristics at high altitude on full throttle.

The remaining portion of the maximum power line can be obtained by converting the values of power determined above to PIW and then entering the PIW vs VIW chart.

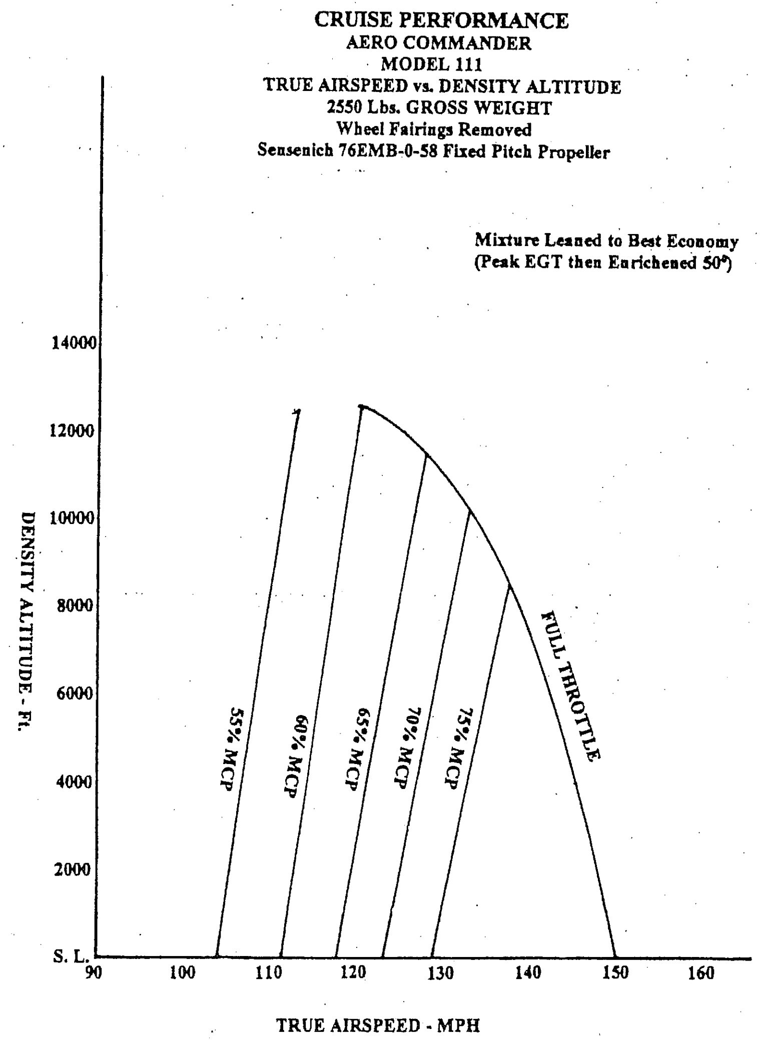


Fig. 9.10 Cruise performance chart for a fixed-pitch propeller-driven aircraft. Reprinted with permission from Gulfstream Aerospace Corporation.

To determine the percent power lines we first determine values of PIW for each altitude as was done in the constant speed propeller case. Using these values we then enter the PIW vs NIW curve and determine corresponding values of NIW. These NIW values may be converted to rpm and plotted vs altitude to obtain a cruise power chart as shown in Fig. 9.9. We also use the NIW values to enter the NIW vs VIW chart and determine values of VIW, from which true airspeed is determined.

In the above described manner we develop both a power chart in altitude vs rpm (Fig. 9.9) and a cruise speed chart (Fig. 9.10) which are interrelated.

In order that the pilot may correct for nonstandard conditions, a density altitude chart is added to the left of both of these charts as it was for constant speed propeller-driven airplanes.

Reference

[1]Hamlin, Benson, *Flight Testing*, The MacMillan Company, New York, 1946.

10
Level Flight Performance Jet Aircraft

10.1 Introduction

The level flight performance tests of jet-powered aircraft serve the same function as those for propeller-driven aircraft. The primary purpose for conducting these tests is to provide cruise data for the "Pilot's Operating Handbook" or flight manual. However, there are several other reasons that we may wish to conduct these tests; they are:

1) to assure design specifications are met
2) to determine the airplane drag polar
3) to determine engine characteristics in flight
4) to determine drag increments of small changes during development testing

The flight test techniques and theory of level flight performance tests of jet aircraft are similar in some respects to those for propeller-driven aircraft, but there are also some important differences. First let us examine the theory behind the level flight performance of a jet aircraft.

10.2 Theory[1,2]

If we start with the condition of steady state level flight where lift equals weight we can say:

$$L = W = \frac{\rho V_T^2 S C_L}{2} \tag{10.1}$$

If we also consider that $V_T = Ma$, and that $a = \sqrt{\gamma P_a/\rho}$, then we can write:

$$V_T^2 = M^2 \gamma \frac{P_a}{\rho} \tag{10.2}$$

where
M = Mach number
P_a = the ambient pressure
γ = specific heat ratio for air
a = the speed of sound

By substitution:

$$W = \frac{SC_L M^2 \gamma P_a}{2} \tag{10.3}$$

If we multiply Eq. (10.3) by P_{SL}/P_{SL} and substitute the pressure ratio δ for P_a/P_{SL}, then Eq. (10.3) becomes:

$$W = \frac{SC_L M^2 \gamma \delta P_{SL}}{2} \tag{10.4}$$

and by rearranging we have:

$$C_L = \frac{2(W/\delta)}{\gamma P_{SL} M^2 S} \tag{10.5}$$

For a given aircraft and set of conditions, γ, P_{SL}, and S are constants. Therefore:

$$C_L = f\left(\frac{W}{\delta}, M\right) \tag{10.6}$$

By the same approach an equation similar to Eq. (10.5) can be obtained for the drag coefficient:

$$C_D = \frac{2(D/\delta)}{\gamma P_{SL} M^2 S} \tag{10.7}$$

and likewise:

$$C_D = f\left(\frac{D}{\delta}, M\right) \tag{10.8}$$

Then if we have the engine thrust characteristics, the level flight data can be converted to values of C_D and C_L for the construction of an airplane drag polar. Once we have the airplane drag polar, we can calculate airplane performance characteristics for flight manual or other uses.

However, in many instances it is simpler if we have relationships that do not require the determination of C_L and C_D. If we rewrite Eq. (10.8) as:

$$D/\delta = f(C_D, M) \tag{10.9}$$

and say that:

$$C_D = C_{DP} + C_{Di} \tag{10.10}$$

we know that $C_{Di} = f(C_L)$, so also must:

$$C_D = f(C_L) \tag{10.11}$$

If we replace C_L in Eq. (10.11) with its function from Eq. (10.6), we have:

$$C_D = f\left(\frac{W}{\delta}, M\right) \tag{10.12}$$

or:

$$D/\delta = f(C_D, M) = f\left(\frac{W}{\delta}, M\right) \tag{10.13}$$

We now have created a functional relationship for drag which combines the aircraft and engine performance characteristics.

The variables that affect engine thrust are:

1) velocity (V)
2) temperature (T)
3) pressure (P)
4) engine speed (N)
5) engine size (D)

If we evaluate the variables using the Buckingham Π Theorem of dimensional analysis we can say:

$$\frac{F}{PD^2} = f\left(\frac{V}{\sqrt{T}}, \frac{ND}{\sqrt{T}}\right) \tag{10.14}$$

Since the engine size D is fixed we may eliminate it. We may also replace $V/\sqrt{T}$ with Mach number M and reference all pressures and temperatures to sea level by use of the temperature ratio θ and the pressure ratio δ. Eq. (10.14) then becomes:

$$\frac{F_N}{\delta} = f(M, N/\sqrt{\theta}) \tag{10.15}$$

A similar analysis will also show:

$$\frac{Q\sqrt{\theta}}{\delta} = f(M, N/\sqrt{\theta}) \tag{10.16}$$

where

Q = the mass flow of air through the engine

$$\frac{W_f}{\delta\sqrt{\theta}} = f(M, N/\sqrt{\theta}) \tag{10.17}$$

where
W_f = fuel flow

On engines which have the capability of reading engine pressure ratio (EPR), the value $N/\sqrt{\theta}$ may be replaced by EPR in Eqs. (10.15) through (10.17).

Since in level flight thrust is equal to drag, we can say:

$$f(M, N/\sqrt{\theta}) = f\left(\frac{W}{\delta}, M\right) \tag{10.18}$$

From this relation comes:

$$M = f\left(\frac{W}{\delta}, \frac{N}{\sqrt{\theta}}\right) \tag{10.19}$$

This relationship is quite important since it is the basis for the speed power level flight method for jet aircraft. If the pilot holds W/δ constant by varying his test altitude as weight changes, he may then obtain a plot of how Mach number varies with power setting (Fig. 10.1). Since all of the parameters are interrelated, a whole series of plots may be obtained using these basic relationships. Examples are shown in Figs. 10.2–10.6.

10.3 Flight Test Techniques[3,4]

As might be expected from the past discussion, the flight test technique for the speed power test of a jet aircraft is somewhat different from that for a propeller-driven aircraft. Since we intended to plot our data at values of constant W/δ, we must vary our test altitude as we burn off fuel in order to keep W/δ constant. This requires considerably more preflight planning than is required for level flight tests of propeller-driven airplanes. It also requires

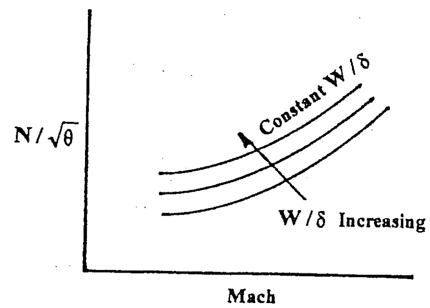


Fig. 10.1 Referred rpm vs Mach.

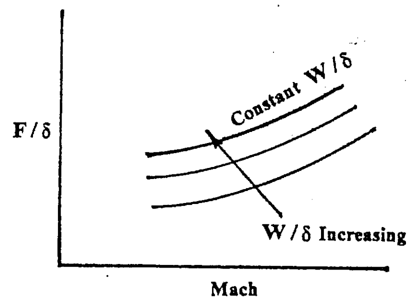


Fig. 10.2 Referred thrust vs Mach.

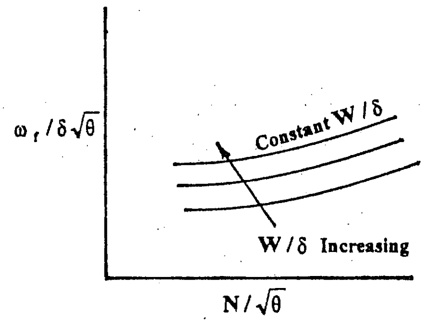


Fig. 10.3 Referred fuel flow vs referred rpm.

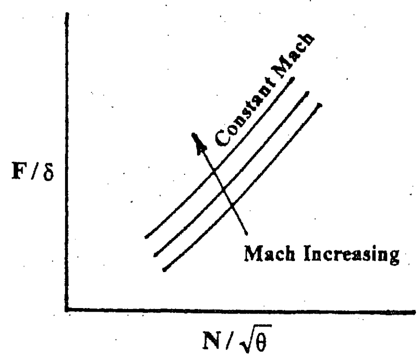


Fig. 10.4 Referred thrust vs referred rpm.

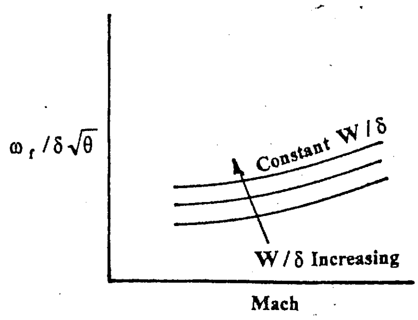


Fig. 10.5 Referred fuel flow vs Mach number.

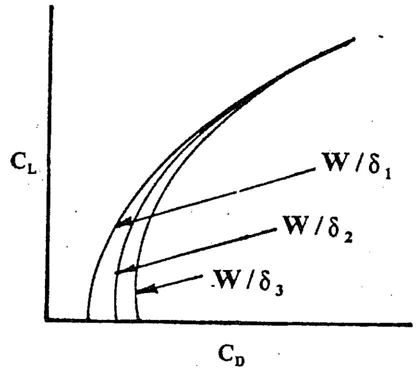


Fig. 10.6 Drag polar.

considerable skill on the part of the test pilot since a missed point cannot be repeated without an altitude change. The test is best performed by a pilot and flight test engineer working as a team. Even in single seat aircraft this can be accomplished via radio link.

In order to perform the test, the crew will need a plot of altitude vs aircraft weight for a constant W/δ (Fig. 10.7). They will also need a plot of the altimeter position error vs Mach number. The test team must determine the altitude at which the test data point is to be flown and allow sufficient time for the aircraft to arrive at that altitude and stabilize. This time averages about 8 min per data point.

In planning the test, estimate the airplane weight at the first data point, and then enter the altitude vs weight chart and find the aim altitude. This altitude must then be corrected for instrument and position error to give the pilot an aim indicated altitude. High-speed points should be obtained first, and then the speed reduced for each successive point. This seems to be the most efficient technique since some of the excess speed can be used in the climb. Also, most aircraft seem to stabilize faster when the data point is approached from the high-speed side.

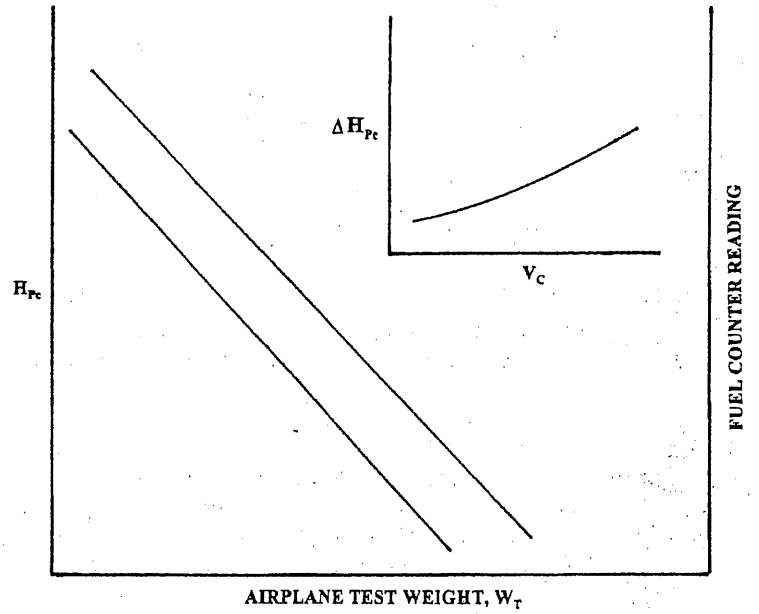


Fig. 10.7 Plot calibrated pressure altitude vs test weight for a constant W/δ. (Ref. 3)

Once established at the aim altitude, with the airspeed stabilized, allow 2–3 min to elapse after the last throttle movement before taking data. This will permit the engine to thermally stabilize. The rate of climb and aircraft acceleration should be zero. However, an acceptable tolerance is rate of climb less than 100 ft/min and airspeed 1.5 kn in 2 min. Even light turbulence will cause excursions greater than this and should be avoided.

10.4 Data Reduction

The data reduction sequence used for reducing the data obtained during the speed power tests will depend to a degree upon the type of jet engine in the aircraft and the way it is instrumented. For instance, the data from a straight turbojet-powered aircraft is handled differently than data from a turbofan engine. Also, if pressure ratio instrumentation is available the data is reduced in a different manner than if this instrumentation is not available.

All of these sequences are covered in the reference material, and due to the number and length we will not discuss them here.

References

[1]USAF Test Pilot School, FTC-TIH-70-1001, "USAF Performance Flight Testing Theory, Vol. II," Edwards AFB, CA, 1970, rev. 1973.

[2]Small, S. M., and Prueher, J. W., "Fixed-Wing Performance," USNTPS-FTM-No. 104, U.S. Naval Test Pilot School, Patuxent River, MD, 28 July 1972.

[3]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH-64-2006, AFFTC USAF Test Pilot School, Edwards AFB, CA, 1965.

[4]Martin, Kenneth, "Level Flight Performance Testing," UTSI Performance Flight Testing Short Course Notes, UTSI Tullahoma, TN, Oct. 1978.

11 Range and Endurance

11.1 Introduction

The airplane's ability to cover great distances by conversion of fuel energy into airspeed and its ability to use this same energy to stay aloft over a period of time are performance parameters of great importance. They take on even greater importance when one considers the dwindling world supplies of petroleum energy and the cost of acceptable substitutes.

The first of the above mentioned performance parameters we call range. The range parameter can be described in two different ways. First, the problem of flying a given distance with a minimum expenditure of fuel. Another method to describe this problem is the ability to obtain the maximum flying distance from a given fuel load. Either of these problems is approached the same way by the pilot, and it is the task of the test pilot and flight test engineer to provide the pilot with the information necessary to master them.

Second, the performance parameter of staying aloft for the maximum time on a minimum amount of fuel is called endurance. This parameter becomes important when the airplane must hold due to traffic delays and in certain other aircraft applications such as fish spotting or antisubmarine warfare missions that require long periods of time on station.

A convenient method for expressing the range, or endurance of an airplane is by the use of specific range and specific endurance.

Specific range can be defined in several ways as is shown below:

$$SR = \frac{\text{nautical miles traveled}}{\text{pounds of fuel consumed}} \tag{11.1}$$

or

$$SR = \frac{V_T}{\omega_F} \tag{11.2}$$

where

SR = specific range

V_T = true airspeed in knots

ω_F = fuel flow in pounds per hour

From Eq. (11.2) we can see that if we obtain values of fuel flow at the same time we are collecting other performance data, then we have the information for determining specific range.

Since range involves flying a distance while endurance involves flying a time, the specific endurance, SE, can be defined as:

$$SE = \frac{\text{hours flown}}{\text{pounds of fuel consumed}} \tag{11.3}$$

or

$$SE = \frac{1}{\omega_F} \tag{11.4}$$

If we recall our level flight performance theory, we will remember that the performance for propeller-driven airplanes is determined from a plot of THP vs V_e called the power required curve. Performance for the jet airplane, however, is determined from the thrust vs V_e curve. Since power is a function of both thrust and velocity, it is easy to see the relationships developed from the power required curve are going to be somewhat different than those developed from the thrust required curve. It is for this reason that we shall discuss range and endurance for propeller-driven airplanes and for jet airplanes separately.

11.2 Range—Propeller-Driven Airplanes

The propeller-driven airplane obtains its thrust by combining the propeller with one of three types of powerplants:

1) normally aspirated reciprocating
2) turbocharged or supercharged reciprocating
3) gas turbine

The fuel flow for each of these types is more directly a function of the horsepower delivered to the propeller than to the thrust produced. Also, the fuel flow characteristics of each of these types is somewhat different so we will address each type separately. First, let us review the range theory for propeller-driven airplanes.

11.2.1 Propeller-Driven Airplane Range Theory

The equations for range were first published by L. Breguet and are known as the Breguet range equations. They are based on the fact that an increment of range $\mathrm{d}s$ is equal to the velocity of the vehicle V_T times the increment of time traveled $\mathrm{d}t$

$$\mathrm{d}s = V_T \, \mathrm{d}t \tag{11.5}$$

and that for a propeller-driven aircraft the weight of fuel consumed in a given time is a function of the specific fuel consumption c and the power developed P.

$$\mathrm{d}W = cP \, \mathrm{d}t \tag{11.6}$$

If we solve Eq. (11.6) for dt and substitute it into Eq. (11.5) we have:

$$ds = \frac{-V_T\,dW}{cP} \tag{11.7}$$

or

$$R = \int ds = \int_{W_2}^{W_1} \frac{V_T\,dW}{cP} \tag{11.8}$$

where
R = range
W_1 = initial gross weight
W_2 = final gross weight

Since power for a propeller-driven airplane can be expressed as:

$$P = \frac{DV_T}{\eta_P} \tag{11.9}$$

where
D = airplane drag
η_P = propeller efficiency

and in level flight $L = W$. We can state Eq. (11.8) as:

$$R = \int_{W_2}^{W_1} \left(\frac{\eta_P}{c}\right)\left(\frac{L}{D}\right)\frac{dW}{W} \tag{11.10}$$

If we integrate Eq. (11.10) and express range in nautical miles and c in pounds per BHP-hours then we have the Breguet range equation:

$$R = 325.8\left(\frac{\eta_P}{c}\right)\left(\frac{C_L}{C_D}\right)\log_e\frac{W_1}{W_2} \tag{11.11}$$

If we want to express this equation in terms of specific range, we have:

$$SR = \frac{dR}{dW} = 325.8\left(\frac{\eta_P}{c}\right)\left(\frac{C_L}{C_D}\right)\left(\frac{1}{W}\right) \tag{11.12}$$

In evaluating Eq. (11.11) we can see that the specific range will be at a maximum when we have a maximum value for the combined term of $(\eta_P/c)(C_L/C_D)$. Keeping this in mind let us see how this is affected by each type of powerplant.

11.2.2 Normally Aspirated Reciprocating Powerplants

In evaluating reciprocating engine powered airplane range performance, it is useful to return to evaluating the effect of various parameters on the power required curve. Fig. 11.1 is a typical power required curve. The maximum range occurs at the point where a straight line from the origin is tangent to the curve. This is the point on the curve where the velocity achieved per unit power is the greatest. It is also the point where the maximum L/D occurs.

As altitude increases the power required curve (when plotted vs V_T) moves up and to the right as is shown in Fig. 11.2. So, as altitude increases, we get an increase in the true airspeed V_T and the power required P_{REQ} to maintain that true airspeed. If the true airspeed and power required for best range are known at one altitude they may be obtained for other altitudes by use of the following relations:

$$V_{T_{ALT}} = V_{T_{SL}}\sqrt{\frac{\sigma_{SL}}{\sigma_{ALT}}} \tag{11.13}$$

$$P_{REQ_{ALT}} = P_{REQ_{SL}}\sqrt{\frac{\sigma_{SL}}{\sigma_{ALT}}} \tag{11.14}$$

For a normally aspirated reciprocating engine at cruise power settings, the fuel flow is a function of engine power. This information and Eqs. (11.13) and (11.14) would indicate that the specific range for a normally aspirated reciprocating engine powered propeller-driven airplane would be unaffected by altitude. Except for the cases where fuel flow or propeller efficiency vary significantly with altitude, this is a true statement.

Now let us return to the power required curve and evaluate the effects of fuel burn-off, or weight change, on the specific range. Fig. 11.3 shows the

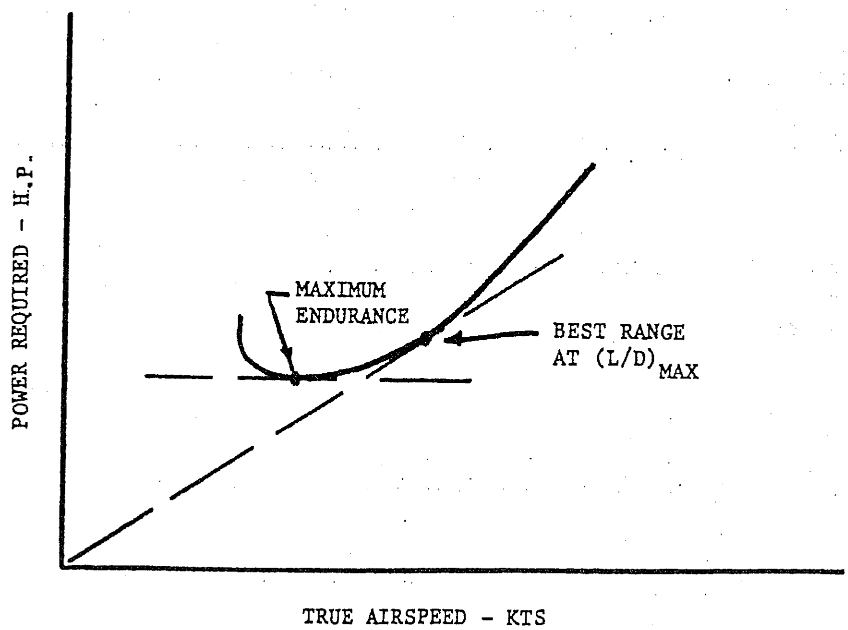


Fig. 11.1 Typical power required curve.

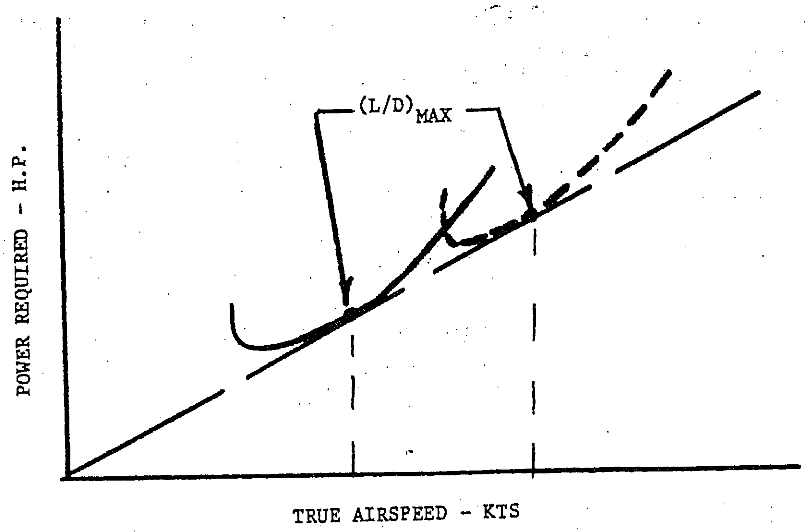


Fig. 11.2 Effects of altitude on power required and airspeed for best range.

effects of weight change on the power required curve, and the point of $(L/D)_{max}$. As we can see from this figure, the power required curve moves down and to the left as weight decreases. This says that the largest value of specific range occurs at the minimum flight weight and that the airspeed and power combination to achieve maximum specific range continually changes throughout the flight. The effects of weight on range related variables for the

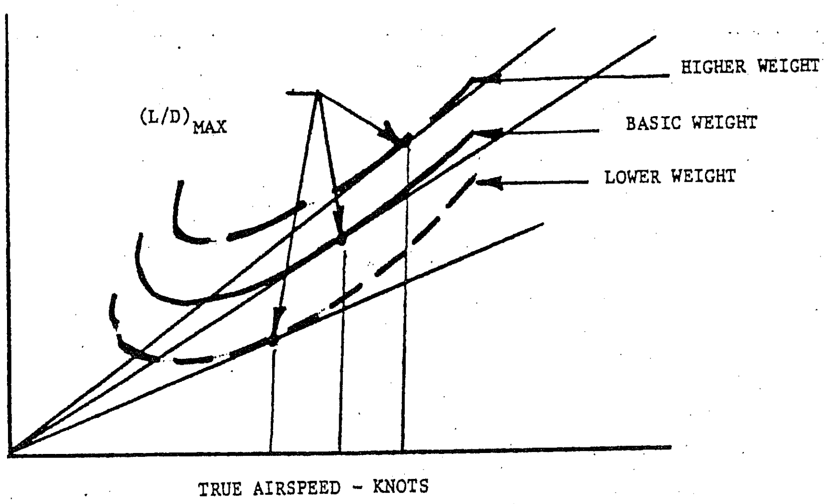


Fig. 11.3 Effects of weight on power required and airspeed for best range.

propeller-driven airplane may be determined by use of the following relations:

$$V_{e(W_2)} = V_{e(W_1)}\sqrt{\frac{W_2}{W_1}} \tag{11.15}$$

$$P_{REQ(W_2)} = P_{REQ(W_1)}\left(\frac{W_2}{W_1}\right)^{3/2} \tag{11.16}$$

$$SR_{(W_2)} = SR_{(W_1)}\left(\frac{W_2}{W_1}\right) \tag{11.17}$$

where
W_1 = aircraft gross weight at the initial condition
W_2 = aircraft gross weight at the condition in question

What we have just been discussing were the airframe effects, or the effects of various parameters on the L/D term of the Breguet range equation. Now lets turn our attention to the (η_P/c) term. The brake specific fuel consumption c for a normally aspirated reciprocating engine at cruise power settings is normally a constant. In other words the fuel flow will increase linearly with increased power. It does not vary significantly with altitude. This would say that the (η_P/c) term on a normally aspirated engine-powered, propeller-driven airplane is only affected by changes in propeller efficiency.

Propeller efficiency η_P is affected by a number of variables. Power, rpm, altitude, and aircraft velocity all have an effect on the propeller efficiency. The effects of these variables are all interrelated and it is difficult to determine which one will have the most powerful effect. This is especially true on a constant-speed propeller since the sensing mechanism adjusts the blade angle to maintain the rpm and not necessarily to operate at the most efficient angle. It is for this reason, that for range determination, it is preferred to obtain speed-power information at several values of rpm and altitude in order to determine the best combination.

11.2.3 Turbocharged Reciprocating Engine

Any range advantage from the turbocharged engine will come from its ability to operate a high altitude at cruise power settings and high true airspeeds. This advantage may be eliminated by the climb fuel requirement, and if the engine requires a higher brake specific fuel consumption in order to cool. Other aspects of the range of turbocharged engine powered airplanes are similar to the normally aspirated engine powered airplanes.

One additional advantage of the turbocharged engine powered airplane is that it may climb to altitudes where the prevailing winds are stronger and achieve some additional range advantage. The effects of wind on range will be discussed in detail later.

11.2.4 Turboprop Engines

The range of an airplane using a turboprop engine is affected by both the aircraft aerodynamics and the operating characteristics of the turbine engine.

From the aerodynamic standpoint the variables affecting range are the same as other propeller-driven airplanes.

The engine operating characteristics are quite different, however. For best fuel efficiency the turbine engine demands air at low temperature. This makes the turbine engine operate most efficiently at high altitude where the air temperature is low. In addition, the turbine engine is more efficient at higher power settings, and since the power required to achieve best range increases with altitude this, too, makes the turboprop engine more efficient at high altitude.

Since the propeller-driven airframe prefers low altitude for best range, while the turboprop engine prefers high altitude, the combination turboprop airplane will achieve its best range at some intermediate altitude. This makes flight testing for range determination more of a chore on turboprop airplanes than on other propeller-driven airplanes.

11.3 Range—Jet Aircraft

The range of a turbojet or fan-jet powered aircraft is affected by a larger number of variables than is the range of a propeller-driven aircraft. For instance, the specific fuel consumption of the jet engine is dependent upon the thrust being developed, the air density and temperature, and the speed of the aircraft. The operational envelope of a jet aircraft also affects its range. This is especially true of an aircraft that is capable of operating at supersonic speeds. For instance, aircraft designed for cruise at supersonic speeds may achieve best range while operating supersonic. The reverse would be true for those aircraft with supersonic capability that are designed for operation primarily at subsonic speeds.

Since so many variables do exist, let us divide them into those related to airframe and to powerplant as we did with propeller-driven airplanes.

First let us examine the aerodynamic theory relating to the range of a jet aircraft.

The fuel flow of a jet engine is primarily dependent upon the thrust produced rather than upon power as was the case for the propeller-driven airplane. This leads us to evaluate the range of the jet aircraft by use of the thrust required curve rather than by power required. Fig. 11.4 is a typical plot of thrust required for a subsonic jet aircraft. The best range point is located where a straight line from the origin is tangent to the curve. This point on the curve occurs where the ratio of the square root of the lift coefficient over the drag coefficient is at a maximum $(\sqrt{C_L}/C_D)_{\text{max}}$. At this point the induced drag is roughly 25% of the total drag. The jet airplane, therefore, is not nearly so dependent upon high aspect ratio wings for good range performance as is the propeller-driven airplane where best range occurs at $(L/D)_{\text{max}}$.

Let us now derive the range equations for a jet aircraft to determine why the difference between props and jets exist. The weight change of a jet aircraft due to fuel consumption can be expressed by the equation:

$$\mathrm{d}W = c_\tau T\,\mathrm{d}t \qquad (11.18)$$

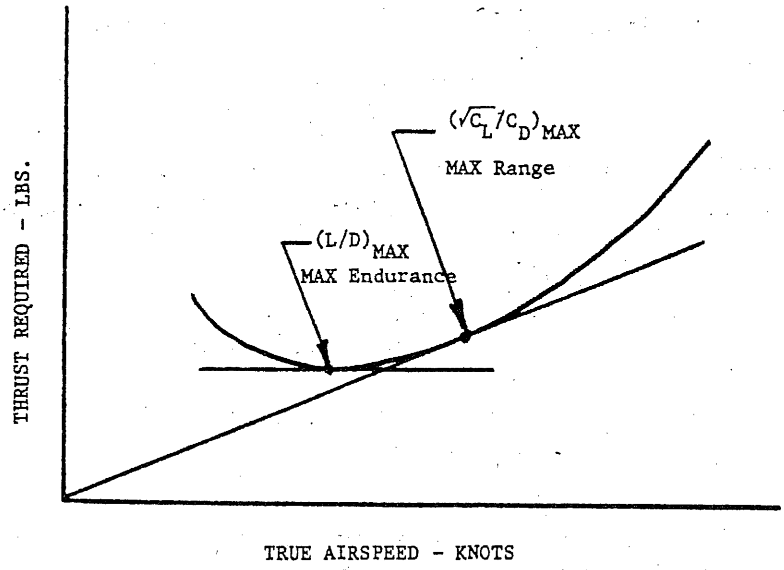


Fig. 11.4 Typical thrust required curve.

where
dW = weight change due to fuel consumption
c_τ = thrust specific fuel consumption in pounds of fuel/(lb thrust/s)
T = thrust in pounds
dt = time of flight in seconds

By solving for dt and substituting the result into the incremental range equation [Eq. (11.5)] we have:

$$ds = \frac{V_T \, dW}{c_\tau T} \tag{11.19}$$

and

$$R = \int_{W_2}^{W_1} \frac{V_T \, dW}{c_\tau T} \tag{11.20}$$

In level flight thrust is equal to drag and drag may be shown equal to:

$$T = D = W\left(\frac{C_D}{C_L}\right) \tag{11.21}$$

and true airspeed V_T is equal to:

$$V_T = \sqrt{\frac{2W}{\rho C_L S}} \tag{11.22}$$

By substitution of these identities into Eq. (11.20) and integrating we have:

$$R = \frac{2.828}{c_t\sqrt{\rho S}}\left(\frac{\sqrt{C_L}}{C_D}\right)\left(\sqrt{W_1} - \sqrt{W_2}\right) \tag{11.23}$$

Eq. (11.23) gives the range in ft. If we convert this to nautical miles and the thrust specific fuel consumption to pounds of fuel/(lb thrust h) we have:

$$R = \frac{1.674}{c_t\sqrt{\rho S}}\left(\frac{\sqrt{C_L}}{C_D}\right)\left(\sqrt{W_1} - \sqrt{W_2}\right) \tag{11.24}$$

Eqs. (11.23) and (11.24) show that the range for a jet aircraft is a function of $\sqrt{C_L}/C_D$ rather than C_L/C_D as was the case for propeller-driven airplanes. It also shows that jet range is very much a function of altitude since the air density ρ appears in the lower half of the equation.

If we neglect engine performance parameters the specific range SR of a jet airplane is affected by altitude according to the following relation:

$$SR_{ALT} = SR_{SL}\sqrt{\frac{\sigma_{SL}}{\sigma_{ALT}}} \tag{11.25}$$

These effects are graphically displayed when the thrust required is plotted against true airspeed V_T and shown for several altitudes. Figure 11.5 shows that true airspeed increases with altitude while thrust required remains constant. Therefore, for a jet airplane the aerodynamic parameters contribute to an improvement in range with an increase in altitude.

Now let us turn our attention to the engine effect on the range. Engine performance of the jet engine is improved by altitude in two ways. First, an increase in altitude causes a decrease in inlet air temperature. A decrease in inlet air temperature causes a reduction in the thrust specific fuel consumption. This is true up to the tropopause where air temperature becomes constant. The second benefit of increased altitude on the performance of a jet engine is caused by the increase in engine rpm required to generate the thrust required. Jet engines operate more efficiently at higher rpm and increased altitude provides this desired effect.

From the preceding discussion we can see that the range of a jet aircraft is very dependent upon altitude. In fact, the optimum altitude for the beginning of cruise flight is the altitude where maximum continuous thrust will maintain the best range airspeed. If that altitude is maintained, the best range airspeed will decrease. This is due to the aircraft weight change with fuel burn-off.

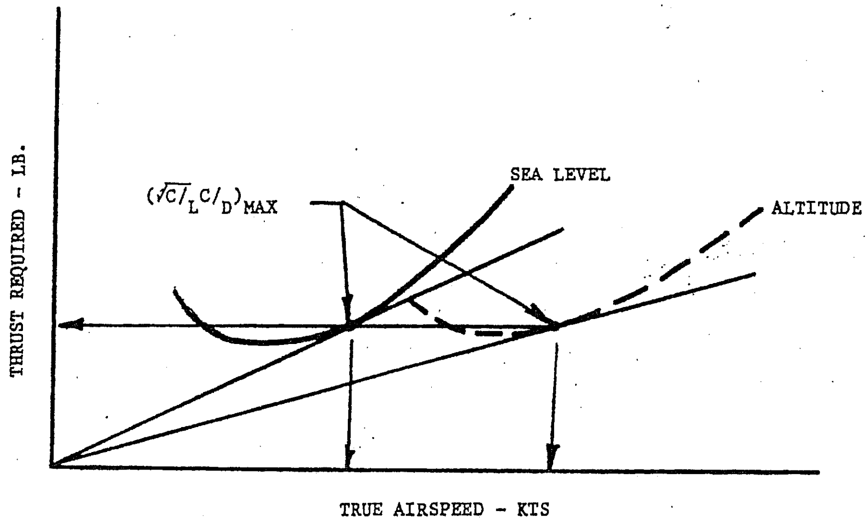


Fig. 11.5 Effects of altitude on thrust required and airspeed for best range.

Although it may be necessary to maintain a constant altitude cruise for air traffic control reasons, this is not the most efficient method for cruise flight in a jet aircraft.

If we return to the dimensional analysis method of evaluating jet aircraft performance that we used earlier for other level flight performance parameters we find that specific range may be defined as:

$$SR = \frac{\omega_f}{\delta\sqrt{\theta}M} = f(C_L, M) \tag{11.26}$$

This equation may also be stated as:

$$SR = \frac{\omega_f}{\delta\sqrt{\theta}M} = f\left(\frac{W}{\delta}, M\right) \tag{11.27}$$

It is this last equation that gives us the key to the most efficient method of cruise operation. Since the specific range is a function of W/δ and Mach, we could hold the initial optimum condition of specific range if we held these values constant. This is accomplished by holding Mach number constant and allowing the aircraft to climb as fuel is burned so W/δ remains constant. This is the concept of the cruise climb and is the most efficient method of cruise for a jet aircraft.

If we compare the specific ranges of the constant altitude method of cruise with the cruise climb we find the specific ranges vary according to the following relations:

$$SR_2 = SR_1\sqrt{\frac{W_1}{W_2}} \tag{11.28}$$

for a constant altitude cruise, and

$$SR_2 = SR_1\left(\frac{W_1}{W_2}\right) \tag{11.29}$$

for a cruise climb.

These relations show the cruise climb to be a much more efficient method of operating a jet aircraft. They also help to point out the considerable difference in range performance between jet and propeller-driven airplanes.

11.4 Effects of Wind on Range

The range theory we have been discussing was for the case of zero wind. Very seldom in real-life flight situations do we experience a zero wind condition. This leads us then to consider the effects of wind on aircraft range. It is easy to understand that a tailwind will increase the aircraft's range while a headwind will reduce it. The effects of headwinds or tailwinds on range may be determined by the following equation:

$$R_{WIND} = R_{NOWIND} \pm (V_{WIND})t \tag{11.30}$$

where
t = time of flight

It can be seen from Eq. (11.30) that the time of flight plays an important role in the effect of wind on airplane range. For a headwind condition we would want to reduce the flight time, and for a tailwind we would want to increase it. The method available to the pilot to accomplish this task is to increase or decrease airspeed.

The amount that the airspeed should be increased or decreased to maintain optimum range conditions may be determined by reference to Figs. 11.6 and 11.7. These figures show typical power required and thrust required curves for propeller-driven and jet aircraft. From our previous discussion we know that best range, no wind, is achieved at the airspeed where a straight line drawn from the origin is tangent to the curve. If we plot the wind speed along the abscissa as is shown in Figs. 11.6 and 11.7, and draw a straight line from that point tangent to the curve, then the best range airspeed for that wind condition is the airspeed where the line is tangent to the curve.

For the propeller-driven airplane determining at what altitude to fly to achieve optimum range depends on the altitude where most favorable winds exist. However, there is a trade-off in the benefit derived from the wind vs the fuel required to climb to that altitude.

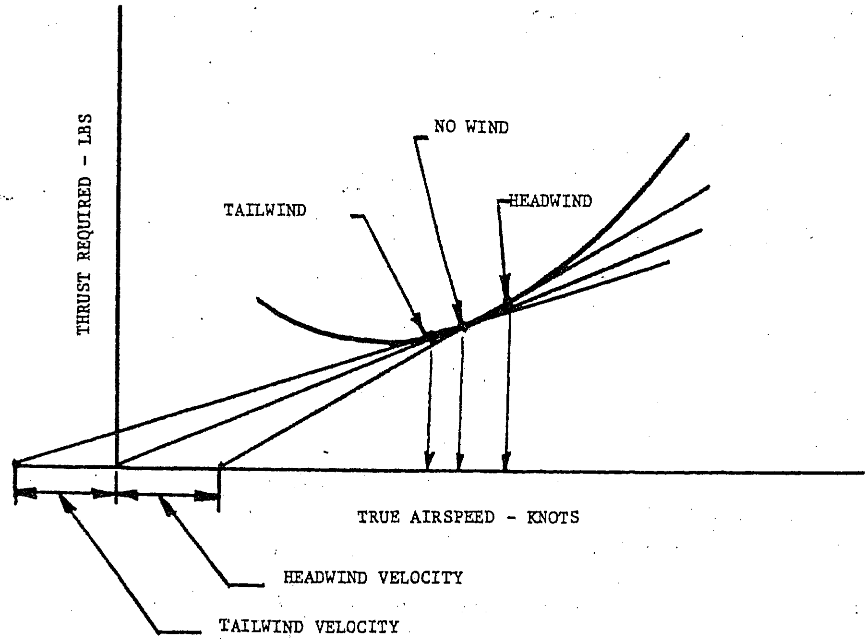


Fig. 11.6 Effects of wind on best range airspeed for a jet-propelled airplane.

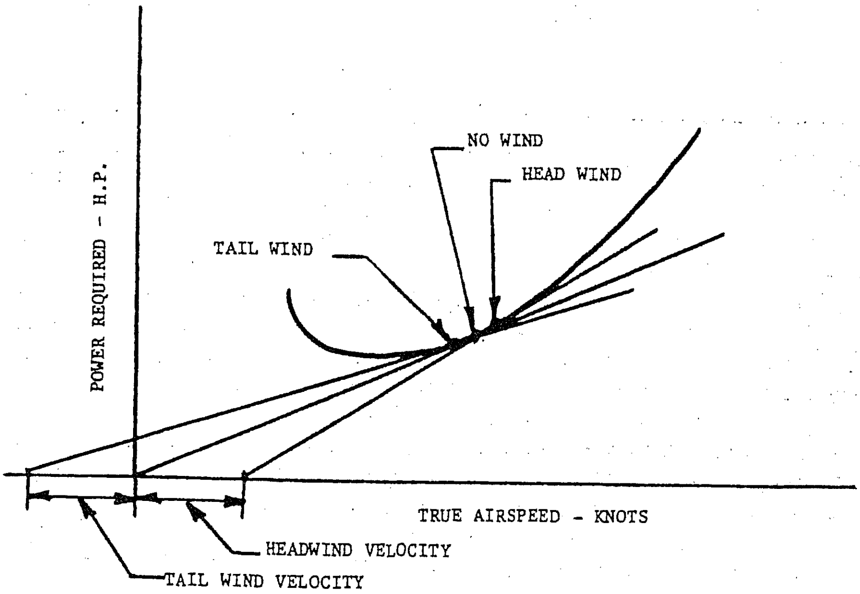


Fig. 11.7 Wind effects on best range airspeed for a propeller-driven airplane.

The jet aircraft presents a much more complicated problem when we try to determine optimum range altitude when under the influence of wind. Since the range of a jet aircraft increases with increasing altitude, it may be worthwhile to select altitudes with unfavorable winds. This of course will depend upon the trade-off between range gained with increasing altitude and range lost due to unfavorable winds.

11.5 Endurance—Propeller-Driven Aircraft

Now let us turn our attention from range to that of endurance or staying aloft for the maximum amount of time. As was discussed earlier the maximum endurance for an airplane occurs at the condition of level flight where fuel flow is lowest. For the propeller-driven airplane this point occurs at the minimum power required. This is shown in Fig. 11.1 for a typical propeller-driven airplane.

Breguet also determined the equations for endurance of a propeller-driven airplane. By use of Eq. (11.6) we may solve for time and derive the Breguet range equation for a propeller-driven airplane. The derivation is similar to that used for the range equation with the following results:

$$E = 778\left(\frac{\eta_P}{c}\right)\left(\frac{C_L^{3/2}}{C_D}\right)\sqrt{\rho S}\left(\frac{1}{\sqrt{W_2}} - \frac{1}{\sqrt{W_1}}\right) \tag{11.31}$$

and

$$SE = 550\left(\frac{\eta_P}{c}\right)\left(\frac{C_L^{3/2}}{C_D}\right)\left(\rho\frac{S}{2}\right)\left(\frac{1}{W^{3/2}}\right) \tag{11.32}$$

where
E and SE are in hours

From Eqs. (11.31) and (11.32) we can see that the endurance and specific endurance equations both contain a density term, and as the density decreases the endurance also decreases. These equations hold for the reciprocating engine-powered propeller-driven airplane that achieves its best endurance at sea level.

In the case of the turboprop airplane the density term in Eqs. (11.31) and (11.32) may be canceled by engine effects. The turbine engine prefers both low inlet air temperature and high power settings for best efficiency. The trade-off between these factors and the aerodynamic factor expressed in Eqs. (11.31) and (11.32) may cause the turboprop airplane to prefer some medium altitude for best endurance.

The engines preference for high power is the reason that multiengine turboprop patrol aircraft often operate with one or more engines shutdown.

11.6 Endurance—Jet Aircraft

The fuel flow of a jet aircraft is a function of thrust, therefore the lowest fuel flow, and maximum endurance, will occur at the level flight point for minimum thrust. From our previous discussions of jet aircraft we know that this point occurs at $(L/D)_{\max}$ (see Fig. 11.4). We also know from these discussions that the thrust required does not vary with altitude and that from the aerodynamic standpoint endurance is not a function of altitude. This may be determined mathematically by deriving the endurance equations for a jet aircraft. If we solve Eq. (11.18) for time we have:

$$\mathrm{d}t = -\frac{\mathrm{d}W}{c_\tau T} \tag{11.18}$$

To obtain endurance E we integrate:

$$E = \int_{W_2}^{W_1} \frac{\mathrm{d}W}{c_\tau T} \tag{11.33}$$

If we then substitute Eq. (11.21) for thrust we have:

$$E = \int_{W_2}^{W_1} \frac{\mathrm{d}W}{W}\left(\frac{1}{c_\tau}\right)\left(\frac{C_L}{C_D}\right) \tag{11.34}$$

which, when integrated, becomes:

$$E = \frac{1}{c_\tau}\left(\frac{C_L}{C_D}\right)\log_e \frac{W_1}{W_2} \tag{11.35}$$

and for specific endurance SE

$$SE = \frac{1}{c_\tau}\left(\frac{C_L}{C_D}\right)\frac{1}{W} \tag{11.36}$$

We can see from Eqs. (11.35) and (11.36) that the jet aircraft is insensitive to altitude aerodynamically.

As we discussed in the range section, the same is not true for the jet engine. It is the most efficient at or near the tropopause.

Considering both of these factors, the jet airplane will also achieve its greatest specific endurance at or near the tropopause.

References

[1]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. I—Performance*, Pergamon Press, NY, 1959.

[2]USAF Test Pilot School, FTC-TIH-70-1001, "USAF Performance Flight Testing Theory, Vol. II," Edwards AFB, CA, 1970, rev. 1973.

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

[4]Dommasch, D. O., Sherby, S. S., and Connolly, T. F. *Airplane Aerodynamics*, Pitman Publishing Co., New York, fourth edition 1967.

12
Climb Performance Theory

12.1 Introduction

Climb performance is a basic requirement in performance testing. Both the FAA Regulations and Military Contract Specifications require a certain value of climb performance. Other performance items such as takeoff also depend on the airplane's ability to climb.

There are two different techniques used to determine climb performance: 1) steady climb and 2) level acceleration.

The steady climb method is used generally for low-speed aircraft and all FAA climb performance, while the level acceleration method is used for high performance military aircraft. Both of these techniques come from the same equation and background theory.

12.2 Climb Theory

The theory of climb performance can be derived using two separate approaches: 1) vector approach and 2) energy approach.

12.2.1 Vector Approach

If we examine an aircraft in a climb (Fig. 12.1) and assume that: 1) the angle of attack is small; 2) the thrust line acts along direction of flight; and 3) the airplane is both climbing and accelerating in the direction of flight. We can say that:

$$L = W\cos\gamma \tag{12.1}$$

$$F = D + W\sin\gamma \tag{12.2}$$

where
γ = climb angle

Using Newton's second law we find that:

$$F - D - W\sin\gamma = Ma = \frac{W}{g}\frac{\mathrm{d}V}{\mathrm{d}t} \tag{12.3}$$

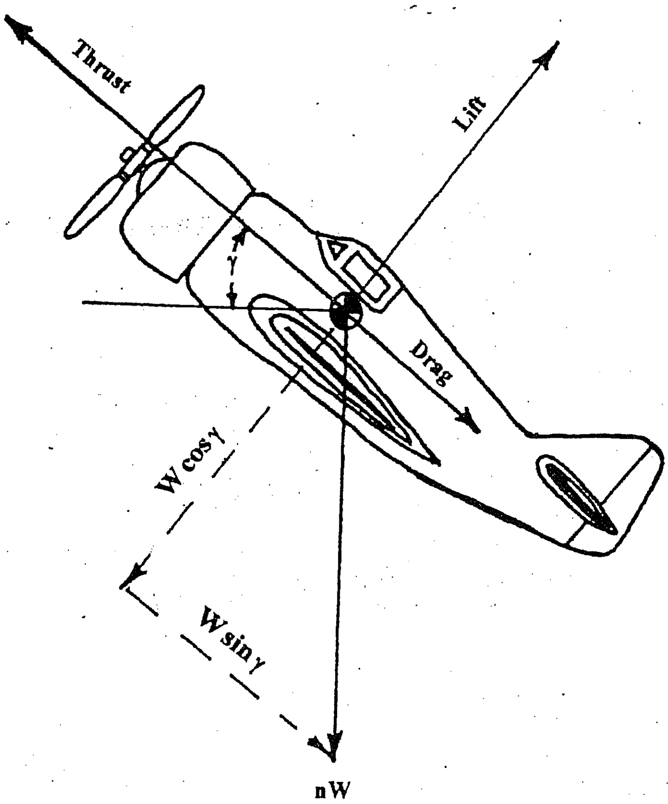


Fig. 12.1 Forces acting on an aircraft during climb.

If we transpose $W \sin\gamma$, multiply through by true airspeed, V, and divide through by weight, we have:

$$\frac{V(F-D)}{W} = V \sin\gamma + \frac{V}{g}\frac{\mathrm{d}V}{\mathrm{d}t} \qquad (12.4)$$

If we draw a velocity diagram of Fig. 12.1 (see Fig. 12.2), we find that:

$$R.O.C. = \frac{\mathrm{d}H}{\mathrm{d}t} = V_T \sin\gamma \qquad (12.5)$$

where
$R.O.C.$ or $\mathrm{d}H/\mathrm{d}t$ = rate of climb

Also, if we multiply Eq. 12.4 through by weight we find that the left-hand side may be expressed as the total thrust horsepower minus the drag thrust horsepower, or thrust horsepower required. This quantity is the excess horsepower available over that which is necessary to overcome drag. It is called thrust horsepower in excess and is the horsepower that the pilot has available to maneuver or climb.

$$FHP_{inexcess} = W\left(\frac{dH}{dt}\right) + \left(\frac{W}{g}\right)\left(\frac{dV}{dt}\right)V_T \tag{12.6}$$

In this form we have thrust horsepower in excess defined by rate of climb and acceleration. If we are in a steady climb at a constant true airspeed the acceleration term becomes zero. It is from this remaining equation that the steady climb technique comes.

$$FHP_{inexcess} = W\left(\frac{dH}{dt}\right) \tag{12.7}$$

A similar thing happens if we hold the altitude constant and let the airplane accelerate. Then the climb term becomes zero and we have:

$$FHP_{inexcess} = \left(\frac{W}{g}\right)\left(\frac{dV}{dt}\right)V_T \tag{12.8}$$

It is from this equation that the level acceleration technique comes.

In airplanes that climb at a constant calibrated airspeed the acceleration term of Eq. (12.4) cannot be assumed to be zero. In such instances we must account for the change in true airspeed with a change in altitude dV/dh. If we say that:

$$\frac{dV}{dt} = \left(\frac{dV}{dH}\right)\frac{dH}{dt} \tag{12.9}$$

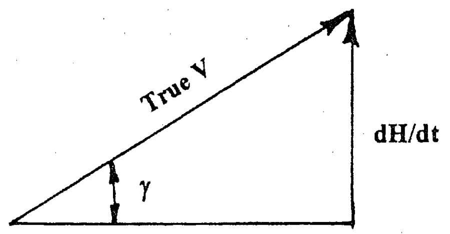


Fig. 12.2 Climb vector diagram.

and make that substitution into Eq. (12.4) we have:

$$\frac{V(F-D)}{W}=\frac{\mathrm{d}H}{\mathrm{d}t}=\frac{V}{g}\left(\frac{\mathrm{d}V}{\mathrm{d}H}\right)\frac{\mathrm{d}H}{\mathrm{d}t} \tag{12.10}$$

Factoring and solving for rate of climb $(\mathrm{d}H/\mathrm{d}t)$ we have:

$$R.O.C.=\frac{\mathrm{d}H}{\mathrm{d}t}=\frac{\left(\frac{F-D}{W}\right)V}{1+\frac{V}{g}\left(\frac{\mathrm{d}V}{\mathrm{d}H}\right)} \tag{12.11}$$

The term $V/g(\mathrm{d}v/\mathrm{d}H)$ is dimensionless and is known as the acceleration factor. It will be discussed further in a later section.

If we examine a plot of thrust horsepower available and thrust horsepower required (Fig. 12.3) we see that these two curves essentially define the total performance of the airplane. The thrust horsepower available can be defined in two ways. If we know the available BHP and the propeller efficiency η_P then we can define the thrust horsepower available.

$$FHP_{avail}=(BHP_{avail})\eta_P \tag{12.12}$$

Also, if we know the FHP required and the FHP in excess, then we may

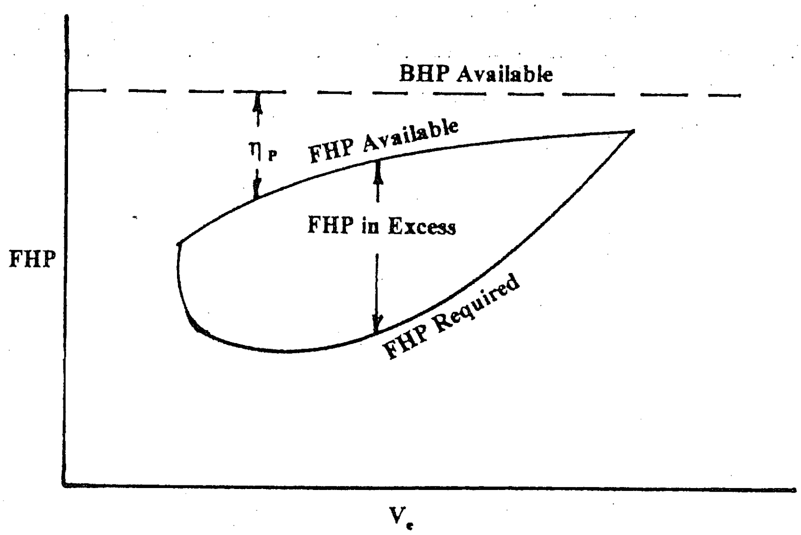


Fig. 12.3 Airplane power curves.

obtain *FHP* available.

$$FHP_{avail} = FHP_{req} + FHP_{inexcess} \tag{12.13}$$

However, thrust horsepower required or thrust horsepower available are difficult values to determine by flight test because they contain propeller efficiency η_P. Thrust horsepower in excess may be determined by use of climb or acceleration methods. If we plot thrust horsepower in excess vs equivalent airspeed we find that it takes the shape shown in Fig. 12.4. The point where *FHP* in excess becomes zero on the left-hand side of the curve, is the power-on stalling speed and on the right-hand side of the curve, the maximum level flight speed. If we substitute the rate of climb for the *FHP* in excess by use of Eq. (12.7), then we may plot Fig. 12.4 as rate of climb vs equivalent airspeed (Fig. 12.5) and see that the maximum rate of climb occurs at the speed where the maximum *FHP* in excess occurs. Both of these figures also show that thrust horsepower in excess or rate of climb decreases with increasing altitude.

By taking the tangent to the top of either curve we may generate a plot of the best rate of climb speed vs attitude (Fig. 12.6). Also, by drawing a line from the origin tangent to each curve we can obtain a plot of best angle-of-climb speed vs altitude as is also shown in Fig. 12.6. By plotting the maximum rates of climb (for each altitude) vs altitude we may determine the maximum rate of climb at sea level, the service ceiling, and the absolute ceiling of the aircraft (Fig. 12.7).

In actual climb testing Fig. 12.5 is only used to determine the best rate and best angle-of-climb speeds. This is because, to determine these curves, climbs are done at several airspeeds and a curve faired through the resulting data.

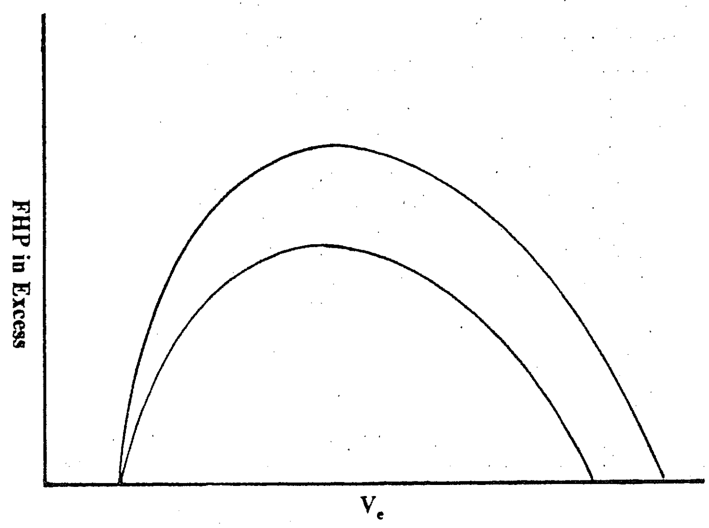


Fig. 12.4 Thrust horsepower in excess vs equivalent airspeed.

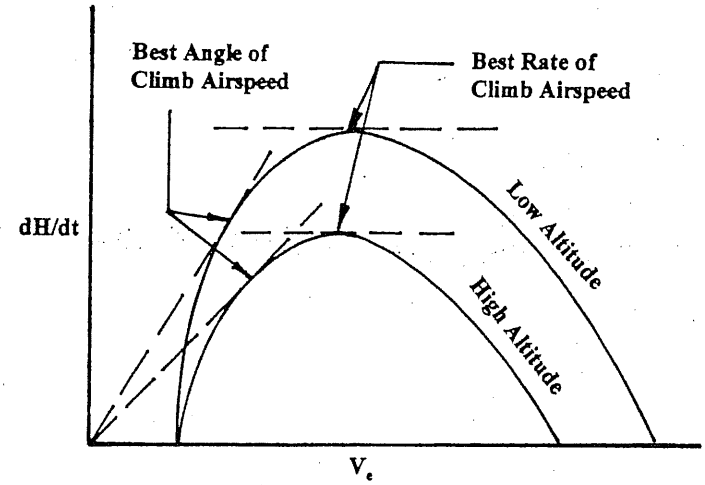


Fig. 12.5 Rate of climb vs equivalent airspeed.

Since one of these speeds may or may not coincide with the best rate-of-climb speed, it may be difficult to determine the true value for the maximum rate of climb. Therefore, in order to define Fig. 12.7 a series of climbs are conducted at the best rate-of-climb speed at several altitudes, the data reduced, and Fig. 12.7 determined directly.

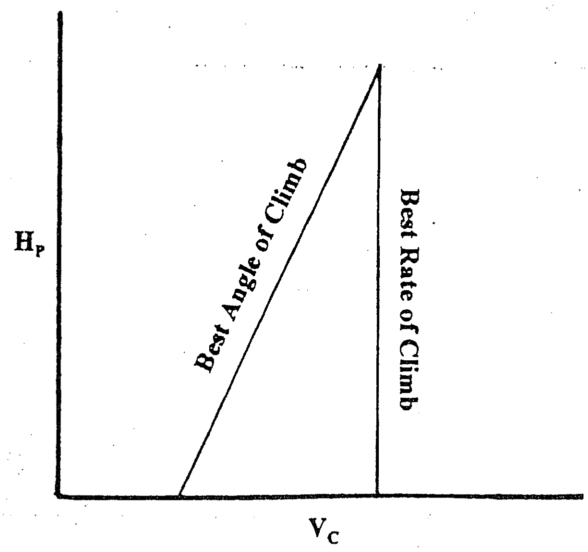


Fig. 12.6 Best rate and best angle-of-climb speeds vs altitude.

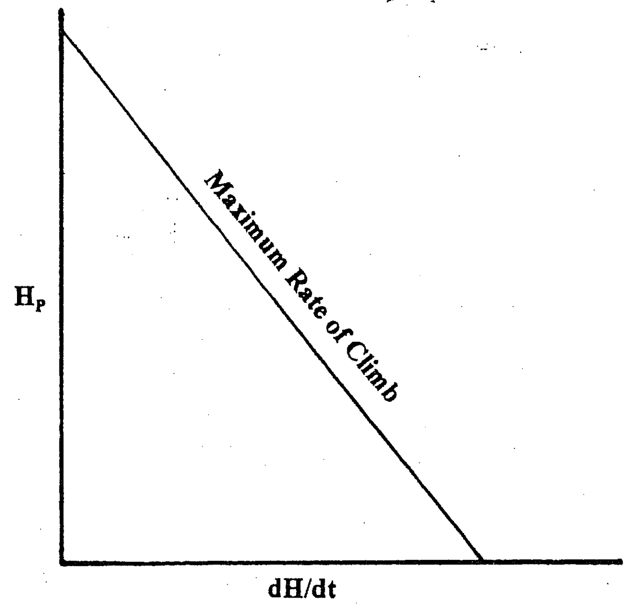


Fig. 12.7 Maximum rate of climb vs altitude.

12.2.2 Energy Approach

Another method of determining the rate of climb is the energy method. If we say that the total energy of a vehicle is the sum of the potential and kinetic energy it possesses:

$$E_{total} = mgH + 1/2mV^2 \tag{12.14}$$

where
E_{total} = total energy
m = vehicle mass

then the specific energy h_e in feet can be obtained by dividing Eq. (12.14) through by vehicle weight.

$$h_e = H + \frac{V^2}{2g} \tag{12.15}$$

If we differentiate Eq. (12.15) with respect to time we have:

$$\frac{dh_e}{dt} = \frac{dH}{dt} + \left(\frac{V}{g}\right)\left(\frac{dV}{dt}\right) \tag{12.16}$$

We can see then from this equation that by multiplying through by weight we are back to the form of Eq. (12.6) for *FHP* in excess.

References

[1]Hendrix, George D., et al., "Performance Flight Testing Theory," FTC-TIH 64-2005, USAF Test Pilot School, Edwards AFB, CA, 1964.

[2]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH-64-2006, AFFTC, USAF Test Pilot School, Edwards AFB, CA, 1964.

[3]Hamlin, Benson, *Flight Testing*, The MacMillan Company, New York, 1946.

13
Climb Performance Methods, Data Reduction, and Expansion

13.1 Introduction

The method of climb data reduction and expansion to be used depends upon the test method used. As one would suspect, the methods that have been in use the longest have the largest number of reduction methods in use. There are currently several methods available for reducing steady climb data that are both accepted and valid.

13.2 Federal Aviation Administration Regulations

The regulations for climb in CAR 3 and FAR Part 23 are divided into those for airplanes under 6000 lbs. takeoff gross weight and those in excess of 6000 lb takeoff gross weight.

For single engine airplanes, there are requirements for climb in a clean configuration, in a takeoff configuration, and in a rejected landing configuration.

For multi-engine airplanes, the requirements are for climb in a clean configuration, in an engine out configuration, and in a rejected landing configuration.

13.2.1 Civil Air Regulation 3 (Ref. 1)

CAR 3.85 is for airplanes with takeoff gross weights in excess of 6000 lb It contains three climb requirements. CAR 3.85(a) is for the normal climb condition. It requires a steady rate of climb at sea level of at least 300 ft/min and a steady climb angle of at least 1:12 for landplanes or 1:15 for seaplanes. This test is to be performed with maximum continuous power in the clean configuration with the cowl flaps set in the position used for the cooling test. CAR 3.85(b) is the requirement for climb with an inoperative engine for multi-engine airplanes. For multi-engine airplanes with a stalling speed at maximum gross weight of greater than 70 mph, the steady rate of climb shall be at least $0.02V_{SO}^2$ in ft/min at an altitude of 5000 ft with the critical engine inoperative. The remaining engine is at maximum continuous and the propeller is feathered on the inoperative engine. The landing gear is retracted and the wing flaps are in the most favorable position. Cowl flaps are set in the same position as for the cooling tests. CAR 3.85(c) covers the balked landing climb condition. It requires that the steady angle of climb at sea level be at least 1:30 with takeoff power on all engines, the landing gear and flaps in the landing position, and the aircraft weight at maximum landing weight. If rapid retraction of the wing flaps is safely possible without loss of altitude or exceptional piloting skill, the flaps may be retracted for this climb test.

CAR 3.85a covers airplanes with takeoff gross weights of less than 6000 lbs. CAR 3.85a(a) is the takeoff climb condition. It requires that the aircraft must have a steady rate of climb at the maximum takeoff weight of 10 V_{S1} or 300 ft/min, whichever is greater, with takeoff power, landing gear extended, wing flaps in the takeoff position, and the cowl flaps in the position used for cooling tests. This is a difficult requirement for small retractable gear airplanes with horsepowers in the 200 horsepower range. On airplanes in that category this requirement usually determined the takeoff gross weight of the airplane. CAR 3.85a(b) is essentially the same as the requirements for larger airplanes covered in 3.85(b). CAR 3.85a(c) the requirement for the balked landing condition is the same as CAR3.85(c) except that the climb requirement changes to 5 V_{SO} or 200 ft/min, whichever is greater.

13.2.2 Federal Aviation Regulation Part 23 (Ref. 2)

In the early FAR Part 23, climb is separated into three specific regulations. FAR 23.65 covers all engines operating climb, FAR 23.67 covers one engine inoperative climb, and FAR 23.77 covers balked landing climb.

FAR 23.65(a) is the corresponding regulation to CAR 3.85(a) and reads the same. FAR23.65(b) is the takeoff climb requirement of CAR 3.85a(a). It also is the same except that the requirement has been changed to 11.5 V_{S1} ft/min where V_{S1} is in knots rather than miles per hour.

FAR 23.67 the one engine inoperative climb requirement has (a) and (b) parts which cover airplanes with takeoff gross weights above (a) and below (b) 6000 lb takeoff gross weight. These are essentially the same as CAR 3.85(b) and CAR 3.85a(b) except that the requirements are based upon airspeeds in knots rather than miles per hour. However, for airplanes that have a stalling speed less than 61 kn, FAR 23.67(c) requires that the rate of climb be measured at 5000 ft with the critical engine inoperative. A positive rate of climb is not required, but the value even if it is a descent must be measured at that altitude.

FAR 23.77 the balked landing climb requirement has an (a) and (b) part for airplanes above and below 6000 lb takeoff gross weight. These are essentially the same as the CAR requirements except that the requirements have been converted to knots rather than miles per hour.

Later versions of FAR Part 23 starting with Amendments 23-34 have rewritten the climb requirements and although some are the same their letter designations may have changed. This has led to some confusion in supplemental type certificate programs on older airplanes. For instance, in both FAR 23.65 and 23.77 the requirements which apply to airplanes with takeoff gross weights above and below 6000 lb have been reversed. Where in the early regulation FAR 23.65(a) was for airplanes over 6000 lb takeoff gross weight, in today's regulation it is 23.65(b). In addition, new regulations were added, including those for commuter category airplanes, and the climb requirements have been changed from feet per minute to climb gradients. These requirements are too numerous to discuss in this text and the regulation should be consulted before starting any new airplane program.

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

Advisory Circular 23-8A discusses acceptable methods for complying with the climb regulations. Under section 23.65 of the AC the sawtooth climb technique is discussed. In this section the AC discusses that if the climb speed is increased to meet cooling requirements, then the same speed must be used for the climb testing. The AC also allows the use of continuous climbs where the objective is to determine if the new climb performance is better than the original in cases such as the installation of a more powerful engine. For data extrapolation, the AC recommends that climb data not be extrapolated more than 3000 ft.

Section 23.67 the single engine climb section provides a summary of the climb requirements for the various categories of airplanes. Acceptable methods for determining commuter category climb gradients are also covered in section 23.67

Section 23.77 discusses the acceptable methodology for balked landing climb. The sawtooth climb method discussed later in this chapter is also an acceptable method for determining balked landing climbs. The AC also defines the rapid retraction of the wing flaps as a flap system that will retract in less than 2 s. It also states that if the flaps will not retract in 2 s, then the rate of climb with the flaps in a position they have reached at the end of 2 s would be considered in an equivalent level of safety finding. Items for commuter category airplanes balked landing climb tests are also covered under this section.

13.3 Test Methods

13.3.1 Steady Climbs

The steady climb method, also called sawtooth climbs, requires considerable flight test time to complete. First, a series of climbs are performed at various airspeeds and several altitudes. These climbs are from 3 to 5 min duration at each airspeed, and two climbs are conducted at each airspeed and altitude in opposite directions in order to cancel wind gradient acceleration effects on the rate of climb. Also, the climbs are conducted crosswind in order to help minimize wind effects. Plots of pressure altitude vs time are then made for each airspeed and altitude as shown in Fig. 13.1. An average curve is drawn between the data from opposite direction climbs, and the slope is taken at the test altitude in question to determine the rate of climb dH/dt, at the altitude, for that airspeed. This process is repeated for each airspeed and altitude and the resultant rates of climb vs airspeed are plotted as shown in Fig. 13.2. Since this data is only used to determine best rate-of-climb speed V_y and best angle-of-climb speed V_x, and these values are not significantly affected by nonstandard conditions other than weight, it is customary to ignore corrections to this data and plot the values directly. By drawing tangents to the top of each curve a plot of best rate-of-climb speed V_y vs altitude H_p may be determined as

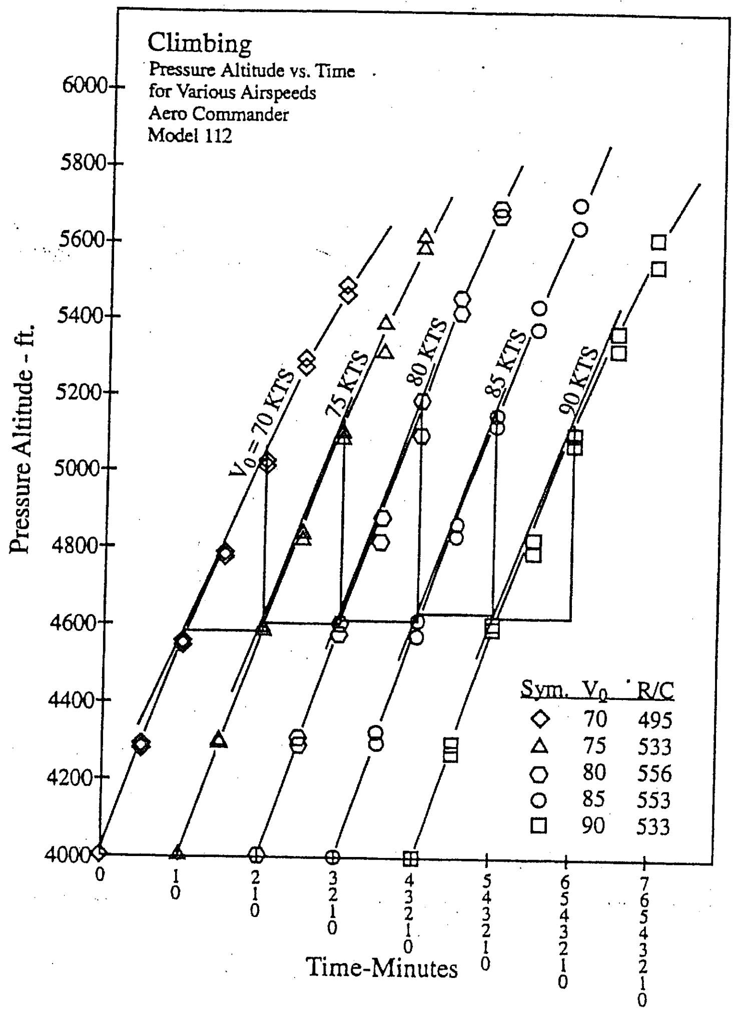


Fig. 13.1 Pressure altitude vs time for various climb airspeeds. Reprinted with permission from Commander Aircraft.

shown in Fig. 13.3. By drawing lines from the origin tangent to the curves as shown in Fig. 13.2, we may also obtain a plot of best angle-of-climb speed V_x vs altitude H_p also shown in Fig. 13.3.

Once we have the best rate-of-climb speed V_y, we then conduct climbs at several altitudes using only this speed. These climbs are sometimes referred to as check climbs. Again, we conduct the climbs in opposite directions (crosswind) in order to minimize wind acceleration effects, make plots of altitude vs time, and take slopes in order to determine dH/dt for that altitude (Fig. 13.4). Once we have this information we are ready to start one of the reduction methods.

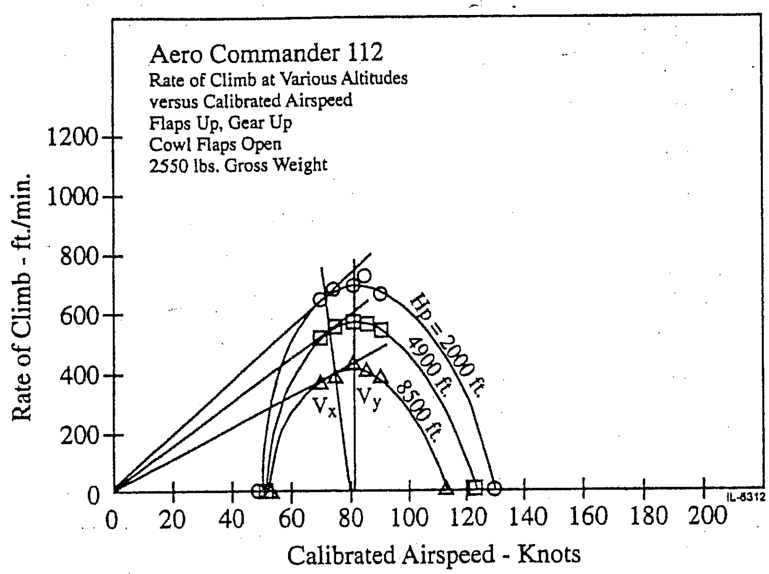


Fig. 13.2 Rate of climb vs calibrated airspeed for three altitudes. Reprinted with permission from Commander Aircraft.

13.3.2 Single Heading Climbs with INS

Since the advent of the Inertial Navigation System (INS) some testing has been conducted using sawtooth climbs on only one heading.[5] Resultant data indicate that (if certain onboard test stability criteria are observed along with the use of the INS for kinetic energy corrections) the single heading method adequately defines climb performance without compromising accuracy.

13.4 Reduction Methods for Steady Climbs

There are many accepted methods for steady climb data reduction; some are:

1) PIW vs CIW method
2) density altitude method[4]
3) equivalent attitude method[4]
4) dimensionless method[6]

The PIW vs CIW method and equivalent altitude method are simple and straightforward. The density altitude method is more complicated. All of these three methods are essentially methods for propeller-driven airplanes. PIW vs CIW and equivalent altitude are used with both constant speed-propellers and fixed-pitch propellers, while density altitude is used with constant speed-propellers.

The reduction scheme for PIW vs CIW is shown in Table 13.1. Once we have values for PIW and CIW for several altitudes we plot them as shown in

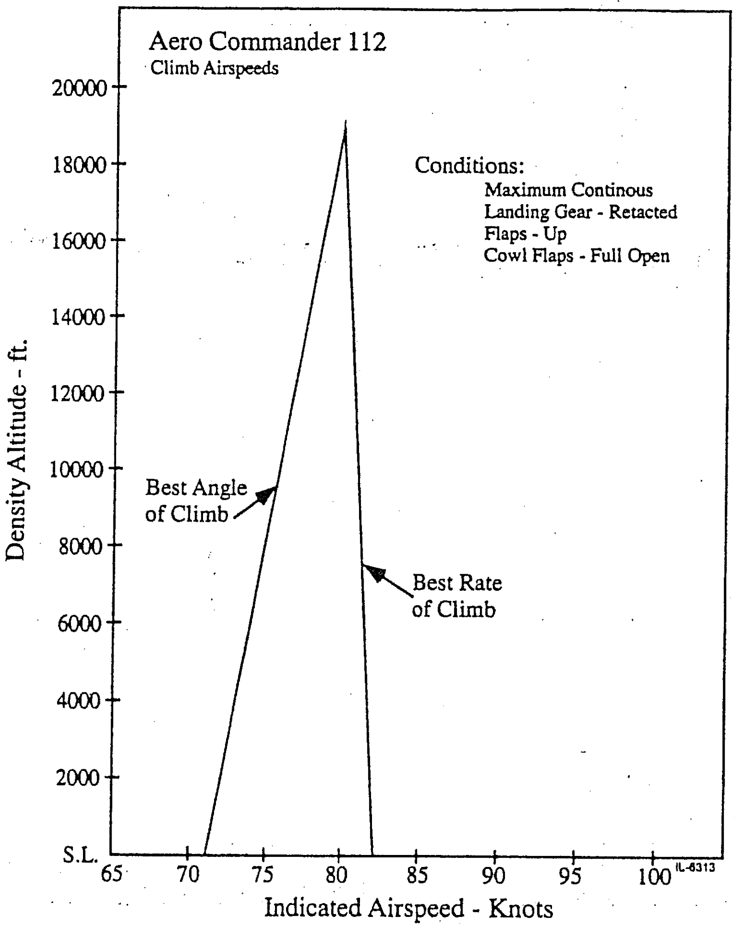


Fig. 13.3 Best rate-of-climb and best angle-of-climb airspeed vs altitude. Reprinted with permission from Commander Aircraft.

Fig. 13.5. We may then obtain the maximum rate of climb at sea level and are ready to expand the data to nonstandard conditions.

The reduction scheme for the density altitude method is shown in Table 13.2. Since this method uses density altitude as a parameter, its final results plot in density altitude vs rate of climb.

The data reduction approach for the equivalent altitude method is shown in Table 13.3. This method also plots out directly into equivalent altitude vs rate of climb.

The dimensionless method of climb plotting may be used for both propeller-driven and jet aircraft, however it is used more frequently with jet aircraft.

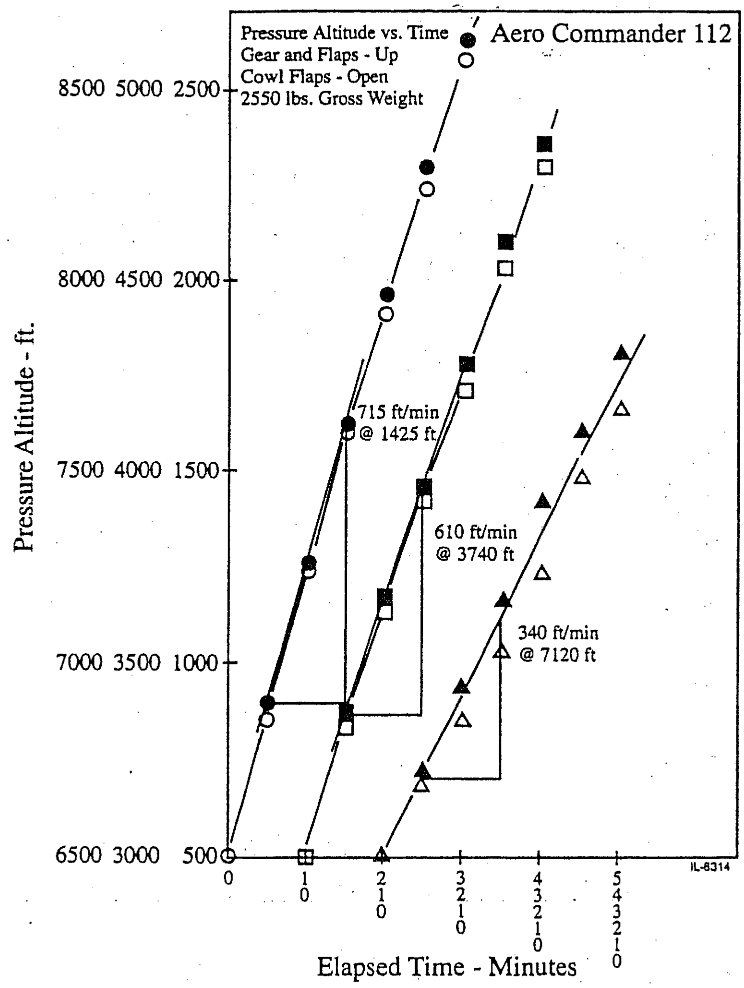


Fig. 13.4 Check climbs at V_y and three altitudes. Reprinted with permission from Commander Aircraft.

Through dimensional analysis it can be shown that:[6]

$$\frac{FHP_R}{\delta\sqrt{\theta}} = f(M, W/\delta) \tag{13.1}$$

and

$$\frac{FHP_A}{\delta\sqrt{\theta}} = \left(\frac{F_{na}}{\delta}\right) KM = f(N/\sqrt{\theta}, M) \tag{13.2}$$

Table 13.1 PIW-CIW data reduction and expansion

Step number	Quantity	Reference	Units	Climb #1	Climb #2	Climb #3
1	V_O	flight data	kn			
2	V_I	instrument calibration	kn			
3	V_C	position calibration	kn			
4	H_{P_O}	flight data	ft			
5	H_{PI}	instrument calibration	ft			
6	$T_O(OAT)$	flight data	°C			
7	T_I	instrument calibration	°C			
8	T_a	273.16 + #7	K			
9	CAT_O	flight data	°C			
10	CAT_I	instrument calibration	°C			
11	rpm_O	flight data	rpm			
12	rpm_I	instrument calibration	rpm			
13	$M.P._O$	flight data	in.Hg			
14	$M.P._I$	instrument calibration	in.Hg			
15	dry/wet temperature	flight data	°_ / °_			
16	$\Delta M.P.$	psychrometer chart	in.Hg			
17	Effective $M.P.$	#14 − #16	in.Hg			
18	T_S at H_{PI}	altitude table	K			
19	T_{CAT}	273.16 + #10	K			
20	T_S/T_{CAT}	#18/#19	n/d			
21	$(T_S/T_{CAT})^{1/2}$	$(\#20)^{1/2}$	n/d			
22	$BHP_{ALT.}$	engine chart	hp			
23	BHP_{TC}	#22 × #21	hp			
24	δ at H_{PI}	altitude table	n/d			
25	θ	#8/288.16	n/d			
26	σ	#24/#25	n/d			
27	$(\sigma)^{1/2}$	$(\#26)^{1/2}$	n/d			
28	W_T	flight data	lb			
29	W_S	arbitrary	lb			
30	W_T/W_S	#28/#29	n/d			
31	$(W_T/W_S)^{1/2}$	$(\#30)^{1/2}$	n/d			
32	$(W_T/W_S)^{3/2}$	$(\#30)^{3/2}$	n/d			
33	PIW	(#23 × #27)/#32	hp			
34	$(dH/dt)_O$	flight data	ft/min			
35	$(dH/dt)_{TC}$	(#34 × #8)/#18	ft/min			
36	CIW	(#35 × #27)/#31	ft/min			
37	plot PIW vs CIW	#33 vs #36	——	——	——	——

(cont.)

Table 13.1 PIW-CIW data reduction and expansion (continued)

Step number	Quantity	Reference	Units	Climb #1	Climb #2	Climb #3
R/C	EXPANSION	CALCULATIONS				
1	Altitude	arbitrary	ft	3000	6000	9000
2	Standard $(\sigma)^{1/2}$	table at #1	n/d			
3	$M.P.$ at #1	MP vs H_{PI} plot	in.Hg			
4	W	arbitrary	lb			
5	W_S	arbitrary	lb			
6	W/W_S	#4/#5	n/d			
7	$(W/W_S)^{1/2}$	$(\#6)^{1/2}$	n/d			
8	$(W/W_S)^{3/2}$	$(\#6)^{3/2}$	n/d			
9	BHP_S	power chart	hp			
10	PIW	(#9 × #2)/#8	hp			
11	CIW	from plot in #37	ft/min			
12	rate of climb	(#11 × #7)/#2	ft/min			
13	Plot altitude vs R/C	#1 vs #12	——	——	——	——

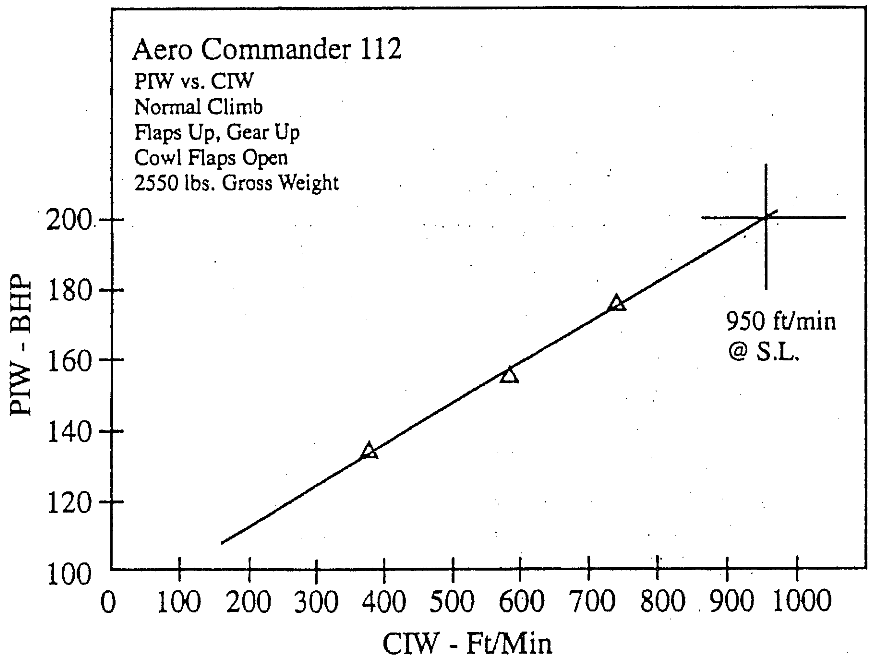


Fig. 13.5 Plot of PIW vs CIW showing sea level rate of climb. Reprinted with permission from Commander Aircraft.

Table 13.2 Climb data reduction by the density altitude method

Step number	Quantity	Reference	Units	Climb #1	Climb #2	Climb #3
1	V_O	flight data	kn			
2	V_I	Instrument calibration	kn			
3	V_C	position correction	kn			
4	H_{PO}	flight data	ft			
5	H_{PI}	instrument calibration	ft			
6	$T_O(OAT)$	flight data	°C			
7	T_I	instrument calibration	°C			
8	T_a	273.16 + #7	K			
9	T_S at H_{PI}	altitude table	K			
10	δ at H_{PI}	altitude table	n/d			
11	θ	#8/288.16	n/d			
12	σ	#10/#11	n/d			
13	$(\sigma)^{1/2}$	$(\#12)^{1/2}$	n/d			
14	V_T	#3/#13	kn			
15	H_D	$145539(1-(\#13)^{0.4699})$	ft			
16	$(dH/dt)_O$	flight data	ft/min			
17	T_a/T_S	#8/#9	n/d			
18	$(dH/dt)_T$	#16 × #17	ft/min			
19	W_T	flight data	lb			
20	W_S	arbitrary	lb			
21	R/C_{WC}	#18(1 − #19/#20)	ft/min			
22	b	airplane geometry	ft			
23	e	Oswald's efficiency factor	n/d	0.8	0.8	0.8
24	$q\pi eb^2$	$(\#3)^2\pi(\#23)(\#22)^2/295$	lb			
25	ΔD_i	$((\#20)^2 - (\#19)^2)/\#24$	lb			
26	R/C_D	101.27(#25)(#14)/#20	ft/min			
27	rpm_O	flight data	rpm			
28	rpm_I	instrument calibration	rpm			
29	$M.P._O$	flight data	in.Hg			
30	$M.P._I$	instrument calibration	in.Hg			
31	$C.A.T._O$	flight data	°C			
32	$C.A.T._I$	instrument calibration	°C			
33	$C.A.T._a$	#32 + 273.16	K			
34	$(T_s/C.A.T._a)^{1/2}$	$(\#9/\#33)^{1/2}$	n/d			
35	BHP at H_{PI}	engine power chart	BHP			
36	BHP_T	#35 × #34	BHP			
37	T_S at H_D	altitude table	K			
38	inlet heat rise	#33 − #8	K			
39	BHP at H_D	engine power chart	BHP			
40	$(T_S/C.A.T._a)_D^{1/2}$	$(\#37/(\#37 + \#38))^{1/2}$	n/d			
41	BHP_{TD}	#39 × #40	BHP			
42	η_P	propeller charts[a]	%			

(cont.)

Table 13.2 Climb data reduction by the density altitude method (continued)

Step number	Quantity	Reference	Units	Climb #1	Climb #2	Climb #3
43	ΔTHP	#42 × (#41 − #36)	hp			
44	R/C_{THP}	#43 × 33,000/#20	ft/min			
45	$(dH/dt)_S$	#18 − #21 − #26 + #44	ft/min			
46	H_D vs $(dH/dt)_S$	plot #15 vs #45	—	—	—	—

[a]If a propeller chart is not available this item may be estimated. At climb airspeeds values may be between 0.65 and 0.75.

Table 13.3 Climb data reduction by the equivalent altitude climb method

Step number	Quantity	Reference	Units	Climb #1	Climb #2	Climb #3
1	V_O	flight data	kn			
2	V_I	instrument calibration	kn			
3	V_C	position correction	kn			
4	H_{PO}	flight data	ft			
5	H_{PI}	instrument calibration	ft			
6	$T_O(OAT)$	flight data	°C			
7	T_I	instrument calibration	°C			
8	T_a	#7 + 273.16	K			
9	δ at H_{PI}	altitude table	n/d			
10	θ	#8/288.16	n/d			
11	σ	#9/#10	n/d			
12	$(\sigma)^{1/2}$	$(\#11)^{1/2}$	n/d			
13	H_D	$145539(1 - (\#12)^{0.4699})$	ft			
14	H_e	#5 − 0.36(#5 − #13)	ft			
15	T_S at H_{PI}	altitude table	K			
16	T_a/T_S	#8/#15	n/d			
17	$(dh/dt)_O$	flight data	ft/min			
18	R/C_{TC}	#16 × #17	ft/min			
19	W_T	flight data	lb			
20	W_S	arbitrary	lb			
21	W_T/W_S	#19/#20	n/d			
22	R/C_{WC}	#18 × #21	ft/min			
23	$(W_T/W_S)^2$	$(\#21)^2$	n/d			
24	$1 - (W_T/W_S)^2$	1 − #23	n/d			
25	b	aircraft geometry	ft			
26	e[a]	Oswald's efficiency factor	n/d	0.8	0.8	
27	K	$(9504.8 \times \#20)/\#26 \times (\#25)^2$				
28	$K/V_C(\sigma)^{1/2}$	#27/#3×#12				
29	ΔD_i	#24 × #28	ft/min			
30	$(dH/dt)_S$	#22 − #29	ft/min			
31	H_e vs $(dH/dt)_S$	plot #14 vs #30	—	—	—	

[a]If actual Oswald's efficiency factor use in place of 0.8, otherwise use 0.8.

where
FHP_R = thrust horsepower required
FHP_A = thrust horsepower available
F_{na} = net thrust available
K = constant to correct M to $V/\sqrt{T_a}$

From these equations it may also be shown that:
Propeller driven

$$\frac{dH/dt}{\sqrt{\theta}} = f\left(\frac{FHP_a}{\delta\sqrt{\theta}}, M, W/\delta\right) \tag{13.3}$$

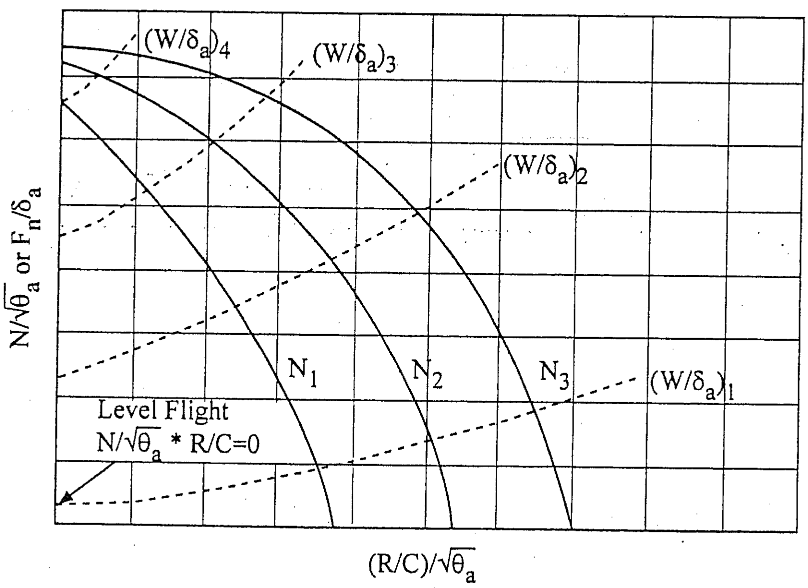
or

Turbojet

$$\frac{dH/dt}{\sqrt{\theta}} = f\left(\frac{F_{na}}{\delta}, M, W/\delta\right) \tag{13.4}$$

Figs. 13.6, 13.7, and 13.8 (Ref. 6) show typical dimensionless climb data plots.

Table 13.4 gives a method of climb data reduction for jet aircraft.



Typical Dimensionless Climb Data at Constant
Mach Number and Various W/δ_a Parameters

Fig. 13.6 Typical dimensionless climb data at constant Mach number.[6]

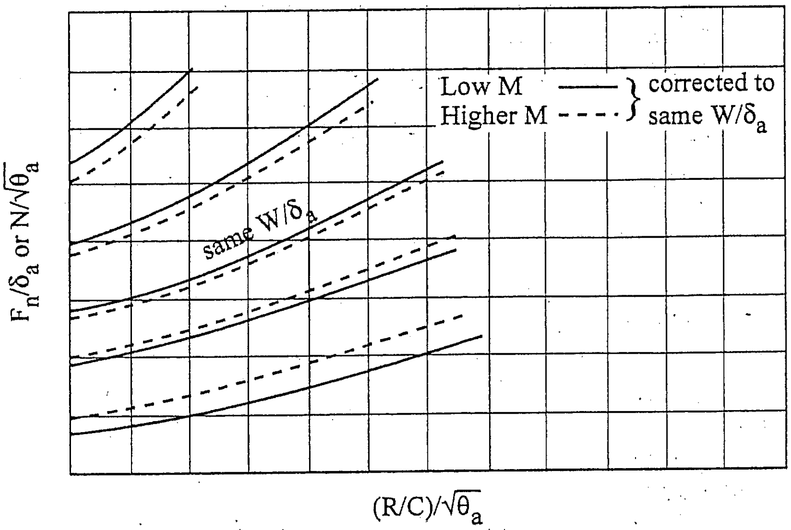


Fig. 13.7 Typical dimensionless climb data plot for two Mach numbers.[6]

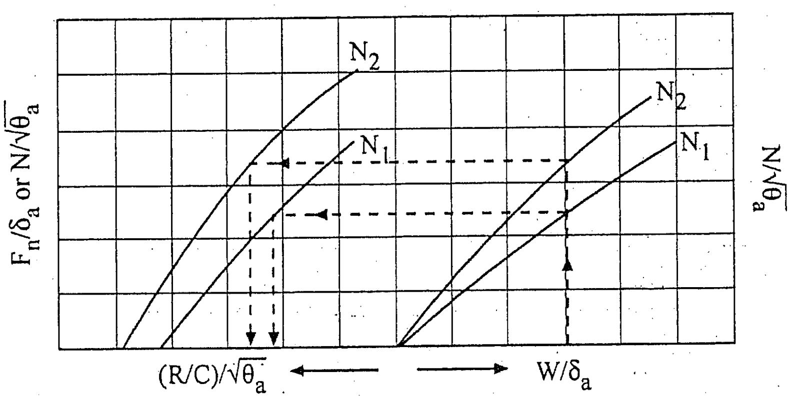


Fig. 13.8 Dimensionless climb data corrected to constant W/δ_a (Ref. 6).

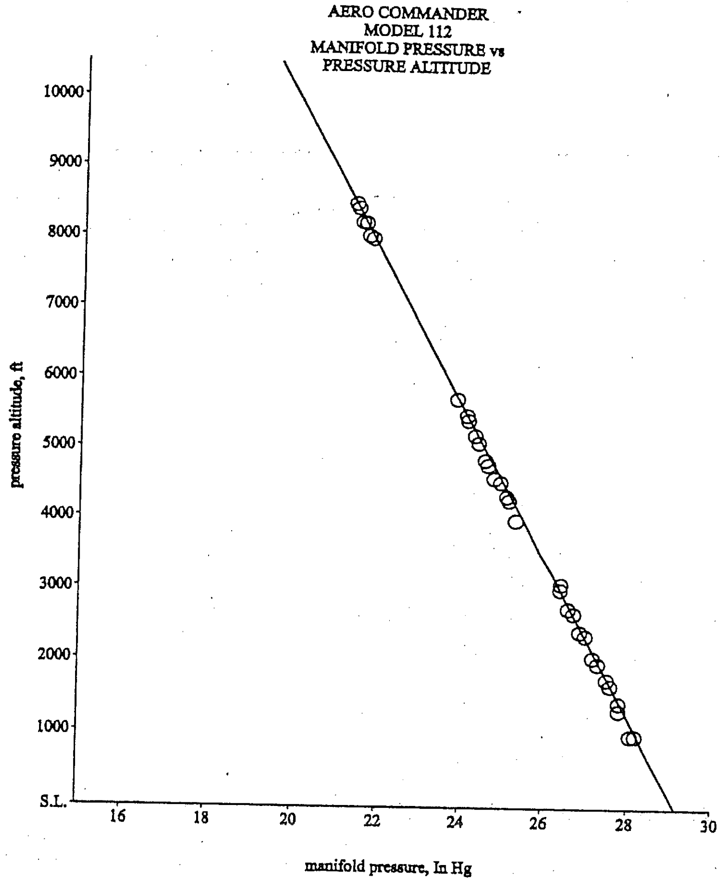


Fig. 13.9 Plot of installed manifold pressure vs pressure altitude. Reprinted with permission from Commander Aircraft.

13.5 Expansion Methods

Of the steady climb methods only the PIW vs CIW method requires expanding to nonstandard conditions. The expansion method is shown in the second part of Table 13.1. In order to complete this expansion one must have a plot of pressure altitude vs engine manifold pressure for the engine as installed in the airplane. Such a plot is shown in Fig. 13.9. Once the expansion is complete using the expansion part of Table 13.1, the climb data may be plotted in handbook format as is shown in Fig. 13.10.

The other steady climb methods plot directly into a plot that may be used for either standard or nonstandard conditions.

Again, as in the level flight performance, these plots may be presented opposite a density altitude chart so that rates of climb for nonstandard temperature may be obtained.

Table 13.4 Jet aircraft climb data reduction[3]

Step number	Quantity	Reference	Units	Climb #1	Climb #2
1	H_{PO}	flight data	ft		
1	H_{PI}	instrument correction	ft		
3	H_{PC}	position correction	ft		
4	V_O	flight data	kn		
5	V_I	instrument correction	kn		
6	V_C	position correction	kn		
7	M	chart 8.5[3] & #6 and #3	Mach		
8	V_{TS}	chart 8.5[3] & #6 and #3	kn		
9	T_O	flight data	°C		
10	T_I	instrument calibration	°C		
11	T_{at}	chart 8.2[3] & #10 and #7	K		
12	$\theta^{1/2}$	$(\#11/288.16)^{1/2}$	n/d		
13	$\theta_S^{1/2}$ at H_{PC}	altitude tables	n/d		
14	$(T_{at}/T_{aS})^{1/2}$	#12/#13	n/d		
15	W_T	flight data	lb		
16	W_S	arbitrary	lb		
17	ΔW	#16 − #15	lb		
18	$\Delta W/Engine$	#17/number of engines	lb		
19	$W_T/Engine$	#15/number of engines	lb		
20	$\mathrm{d}H/\mathrm{d}t$	flight data	ft/min		
21	$(\mathrm{d}H/\mathrm{d}t)_{TC}$	#20 × #14	ft/min		
22	wind gradient	chart 5.41[3] when required	n/d		
23	$(\mathrm{d}H/\mathrm{d}t)_W$	#21 × #22 when required	ft/min		
24	acceleration factor	chart 5.51[3] for constant V_C climb	n/d		
24-1	acceleration factor[a]	chart 5.51 & 5.52 for non-constant V_C climb	n/d		
25	N_O	engine rpm - flight data	rpm		
26	N_I	instrument correction	rpm		
27	N_T	average of all engines	rpm		
28	$N_T/\theta^{1/2}$	#27/#12	rpm		
29	$N_T/\theta_S^{1/2}$	#27/#13	rpm		
30	$\Delta F_N/\delta_a$[a]	chart like 5.21[3] & #28, #29 & #7			
31	$(\Delta R/C_1)/(\Delta F_N/\delta_a)$	chart 5.22,[3] #3, #7 & #19			
32	$(\Delta R/C_1)_a$	(#31 × #30)/#24	ft/min		
33	R/C_1	#21 or #23 + #32	ft/min		
34	$\Delta R/C_2$	(−#17 × #33)/#15	ft/min		
35	$(\Delta R/C_3)\Delta W$	chart 5.31,[3] #3, #7 and wing span			
36	$(\Delta R/C_3)_a$	(#35 × #17)/#24	ft/min		
37	R/C_S	#33 + #34 + #36	ft/min		
38	H_{PC} vs R/C_S	plot #3 vs #37	———	———	———

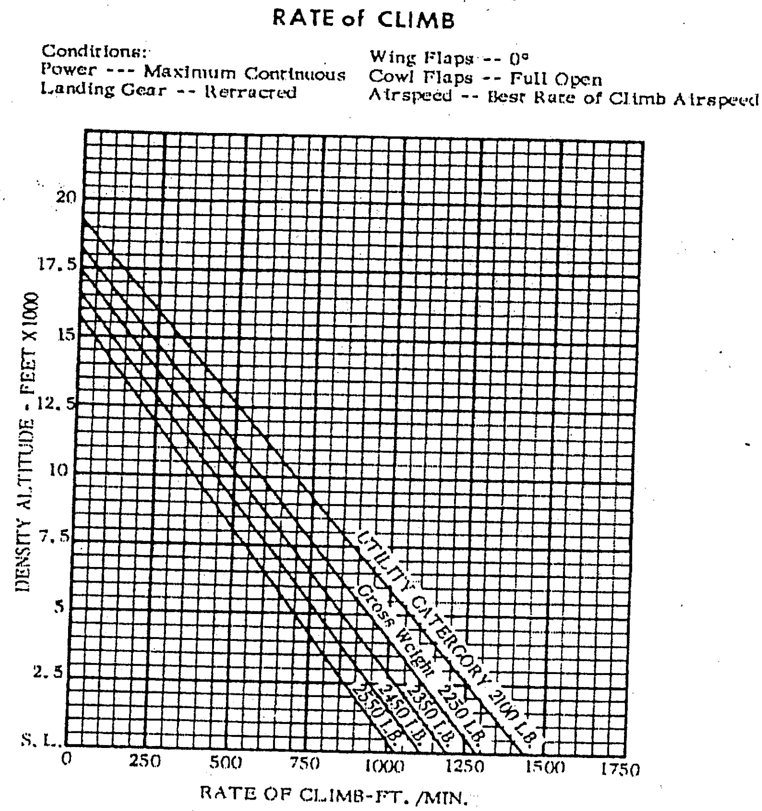


Fig. 13.10 "Pilot's Operating Handbook" climb performance. Reprinted with permission from Commander Aircraft.

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility and Acrobatic Categories," Federal Aviation Agency, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airplane Airworthiness; Normal, Utility and Acrobatic Airplanes," Federal Aviation Administration, Washington, D.C., 1965.

[3]Federal Aviation Administration Advisory Circular AC No: 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, Washington, D.C., 1989.

[4]Federal Aviation Administration Order 8110.7, "Engineering Flight Test Guide for Small Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 20 June 1972.

[5]De Pasquale, A. J., and Baillie, I. J., "Single Heading Climbs—an Alternate Technique for Determining Performance," A.IAA-86-9757, paper presented at the AIAA/AHS/CASI/DGLR/IES/ISA/ITEA/SETP/SFTE 3rd Flight Testing Conference, Las Vegas, Nevada, 1986.

14
Energy Approach to Performance Flight Testing

14.1 Introduction

With the advent of high performance and jet aircraft, conventional "steady state" approaches to determining airplane performance began to develop flaws. For instance, the change in true airspeed with increase in altitude (dV/dh) in climb performance had generally been ignored. However, with these aircraft it became a significant factor.

The method developed (sometimes called the Rutowski energy methods for Edward S. Rutowski, a Douglas Aircraft Company engineer who is credited with developing them),[1] is based on the total energy possessed by the flight vehicle. As we examine this approach, it is interesting to note that it can be used to determine the performance of any object from airplanes to barn doors.

It should be noted that the method has not been accepted by the FAA for determining compliance with the FARs.

14.2 Theory

The Rutowski method is based on the fact that the total energy of an aircraft in flight is a sum of its potential energy, as reflected by its altitude and its kinetic energy, as shown by its airspeed or Mach number.

$$E = PE + KE \tag{14.1}$$

$$E = Wh + \frac{WV^2}{2g} \tag{14.2}$$

where

E = the total aircraft energy
W = the aircraft gross weight
h = the altitude
V = the true airspeed
g = the acceleration due to gravity

As in other flight test methods it is preferable to have terms that will allow comparison of airplanes that have different gross weights, or to be able to plot data for a given airplane that has the weight normalized to some value such as TOGW. The way this is accomplished with Eq. (14.2) is to divide through by the weight and to redefine the term on the left-hand side of the equation as the specific energy E_S.

$$E_S = h + \frac{V^2}{2g} \tag{14.3}$$

where
$E_S = E/W$

This new specific energy E_S term has units that are in feet or meters and for this reason is sometimes called energy height. It may be thought of as the altitude a flight vehicle would reach if all of its energy were converted into potential energy. Likewise, it may be said to be the maximum speed the aircraft could achieve if all of its potential energy were converted to kinetic energy. This makes it possible to plot lines of constant specific energy on an altitude-velocity diagram such as is shown in Fig. 14.1. Any point on this diagram is defined by its potential (altitude) energy and its kinetic (velocity) energy. If the total energy of the vehicle remains constant, movement on the diagram can only be along the lines of constant specific energy. Points A to B or B to A.

What we have described so far has only been a method to define the energy state of an aircraft. It does not tell us what its performance capabilities are. To define the performance capabilities we must know the ability of a given aircraft to change its energy level in a given time. This is accomplished by differentiating Eq. (14.3) with respect to time.

$$\frac{dE_S}{dt} = \frac{dh}{dt} + \frac{V}{g}\left(\frac{dV}{dt}\right) \tag{14.4}$$

This equation should appear familiar since the right-hand side is the same as the thrust-horsepower-in-excess equation with the weight divided out. As a result the left-hand side of the equation is called the specific excess power P_S

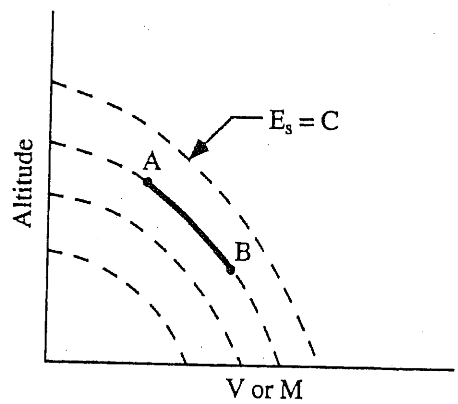


Fig. 14.1 Plot of specific energy.

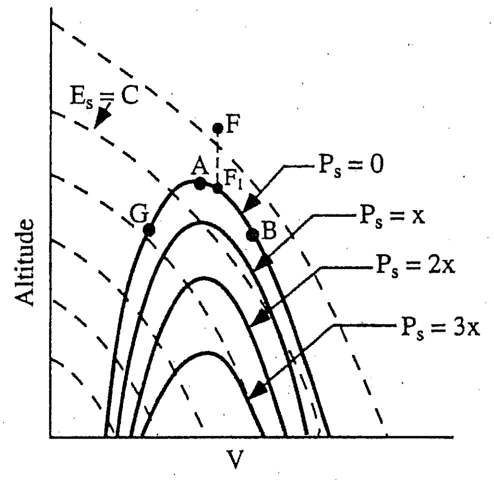


Fig. 14.2 Plot of specific excess power for a subsonic aircraft.

and may also be defined as:

$$\frac{dE_S}{dt} = P_S = \frac{(F - D)V}{W} = \frac{FHP_{xcess}}{W} \tag{14.5}$$

With the specific excess power for a given airplane known, we may then plot lines of constant P_S upon our specific energy plot and define the performance capabilities of the airplane. Figs. 14.2 and 14.3 are two such plots: Fig. 14.2 for a subsonic airplane, Fig. 14.3 for a supersonic airplane. The line of $P_S = 0$ might be defined as the performance envelope of the airplane. At any point on the $P_S = 0$ curve the airplane is in steady level flight at full throttle at the altitude and airspeed shown. The aircraft cannot operate outside this envelope in steady level flight. It may dive or zoom climb-out of the envelope to higher airspeeds or altitudes as is illustrated by points E and F, but it cannot maintain these energy levels. It will fall back to the $P_S = 0$ line as is indicated by points E' and F'. An aircraft located at point A can neither climb nor accelerate, and if it wants to go to point B must dive to achieve a higher level of P_S so it can climb or accelerate to point B. An aircraft at point G is on the back side of the more familiar thrust required curve.

14.3 Application to Climb Performance

As we can see from Eq. (14.4), known values of P_S may be used to obtain either the rate of climb or the acceleration capabilities of the aircraft. The values of rate of climb and acceleration obtained by this method are also more accurate than those obtained by conventional methods since the change in

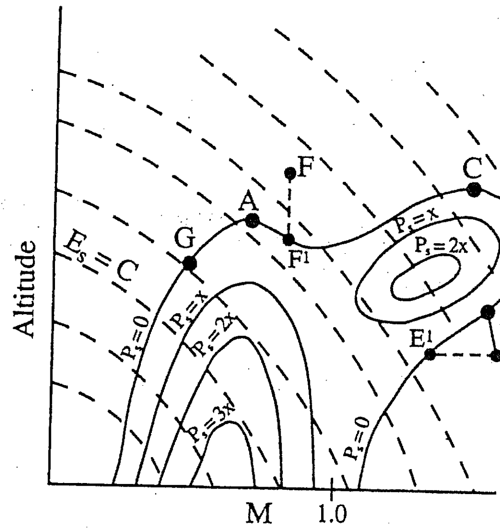


Fig. 14.3 **Plot of specific excess power for a supersonic aircraft.**

aircraft weight during the climb or acceleration is included.

$$\frac{dE_S}{dt} = \frac{d(E/W)}{dt} = P_S \tag{14.6}$$

The problem we face in flight test is to obtain the values of P_S for a given airplane. As was mentioned earlier there may be some difficulty in obtaining values of P_S by measuring rate of climb dh/dt. For instance in modern jet fighters with thrust-to-weight ratios in excess of one, it is not possible to achieve a steady climb at reasonable altitudes. Even in aircraft that do not have this thrust-to-weight it may not be possible to achieve a steady climb without some acceleration due to the change in true airspeed with an increase in altitude dV/dh. Although it is possible to make this correction, a simpler flight test method, which can be used by any aircraft, is to perform a level acceleration. This technique can be used for measuring both P_S and climb performance.

In the level acceleration technique the rate of climb dh/dt term of Eq. (14.4) is held to zero or near zero values, and the aircraft's ability to accelerate in level flight is measured. This information may then be converted mathematically to values of P_S, or rate of climb, at the altitude tested.

The flight test method and data reduction techniques for the level acceleration test will be covered later. First, let us turn our attention to some of the performance information that may be gleaned from knowing the P_S envelope of our aircraft.

14.4 Other Applications of Energy Methods

The energy approach may be applied to a large number of performance problems. This is accomplished by relating the fuel energy used by the aircraft to the aircraft energy state by use of an energy balance equation. Rutowski[1] writes such an equation as follows:

$$dE = \eta_0 H_C \, dW_f - E_S \, dW_f - DV \, dt \tag{14.7}$$

where

η_0 = the thermal and mechanical efficiency of the powerplant
H_C = the heat content of the fuel
W_f = the fuel weight
D = the airplane drag
V = the true airspeed

The first term on the right-hand side of Eq. (14.7) represents the conversion of fuel energy to power the aircraft. The second term is the aircraft's change in specific energy E_S due to its weight change in burning fuel. The third term is the energy dissipated by the aircraft due to aerodynamic drag. The total equation is an application of the principle conservation of energy over a time interval dt (Ref. 1).

If we return to Eq. (14.2) and factor out the weight we find it may be written as:

$$E = E_S W \tag{14.8}$$

If we differentiate Eq. (14.8), we have:

$$dE = W \, dE_S + E_S \, dW \tag{14.9}$$

The dW in the second term of this equation is the change in aircraft weight. This weight change is equal in magnitude to the fuel consumed, but opposite in sign.

$$dW = -dW_f \tag{14.10}$$

If we substitute this relation into Eq. (14.9) and set Eq. (14.9) and (14.7) equal we may write:

$$W \, dE_S = \eta_0 H_C \, dW_f - DV \, dt \tag{14.11}$$

If we now divide this equation through by $W \, dt$ we have:

$$\frac{dE_S}{dt} = \frac{\eta_0 H_C}{W}\left(\frac{dW_f}{dt}\right) - \frac{DV}{W} \tag{14.12}$$

Recalling Eqs. (14.4) and (14.5) you will recognize the left-hand side of

Eq. (14.12) as being P_S, and we now have created an equation for P_S that relates both engine and airframe performance characteristics. This basic relationship may be used to solve a number of climb-related problems.[1,2]

In addition, the problem of aircraft range may be addressed by use of Eq. (14.11) and the following relation:

$$dR = V\,dt \tag{14.13}$$

where

dR = the increment of range the aircraft will travel in time dt

By substituting Eq. (14.13) into Eq. (14.11) and solving for dR we have:[1]

$$dR = \frac{\eta_0 H_C}{D}\,(dW_f) = \frac{W}{D}\,dE_S \tag{14.14}$$

To apply Eqs. (14.11) and (14.14) requires the maximizing or minimizing of a given variable. For instance, if we wish to determine the minimum time to reach a specific energy level then we want to maximize the rate of energy increase. This can be done by maximizing the integral:[3]

$$E_S = \int_{t_1}^{t_2} P_S\,dt \tag{14.15}$$

This integral is maximized when:

$$\left[\frac{\partial P_S}{\partial V}\right]_{E_S=K} = 0 \tag{14.16}$$

or

$$\left[\frac{\partial P_S}{\partial H}\right]_{E_S=K} = 0 \tag{14.17}$$

In order to solve these expressions, we must use the calculus of variations. However, a much simpler approach is to solve them graphically. In the altitude velocity plane, Eqs. (14.16) and (14.17) are satisfied at the points where the contours of constant E_S are tangent with the contours of constant P_S (Ref. 3). If these points are connected we have the path for maximum energy increase, or the minimum time to reach a given energy level (see Figs. 14.4 and 14.5).

Similarly, the minimum fuel-to-climb path may be obtained by plotting contours of constant excess specific power divided by fuel flow. P_S/W_f and connecting the locus of points where these values are tangent with E_S (see Fig. 14.6).

The maximum range glide path can be obtained by plotting contours of constant drag and connecting the locus of points where these contours are tangent with the contours of constant E_S (see Fig. 14.7).

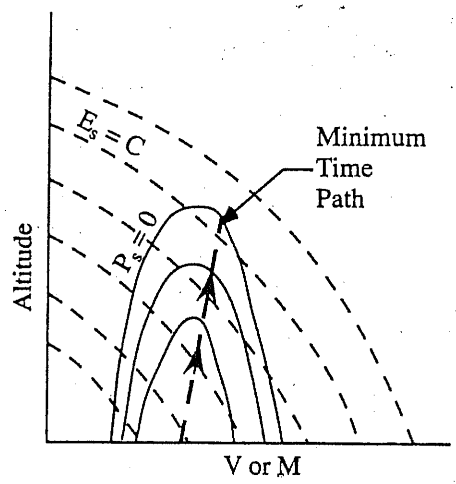


Fig. 14.4 Minimum time-to-climb path for a subsonic airplane.

From these examples it may be seen that energy methods may be used to solve a number of airplane performance problems. However, in order to do this, we must know the P_S and drag characteristics of the airplane in question. For this reason we must return to the discussion of the level acceleration flight test method.

14.5 Level Acceleration Flight Test Method

Although intended for use with jet-propelled aircraft, the level acceleration method can also be used for propeller-driven aircraft. There are two approaches

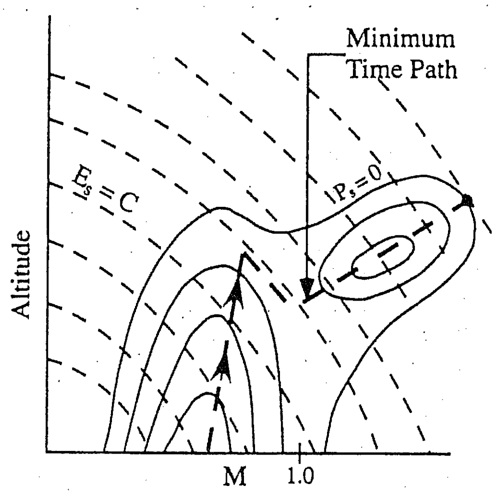


Fig. 14.5 Minimum time-to-climb path for supersonic airplane.

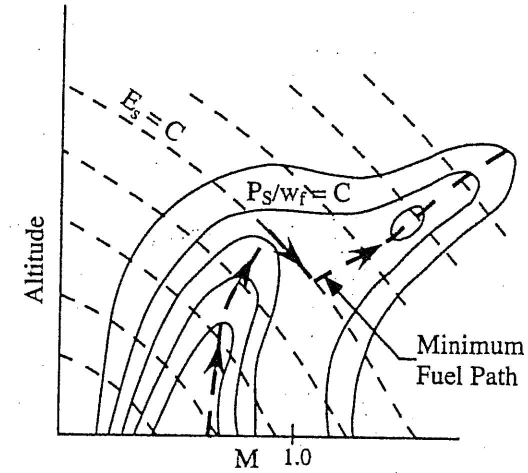


Fig. 14.6 Minimum fuel-to-climb path for a supersonic airplane.

to conducting the test. One method is to slow the aircraft to just above the stall, stabilize at the test altitude, start the instrumentation, and then add the desired test power setting. The aircraft is then held at the test altitude by rotating the nose downward as the aircraft increases airspeed. Data are recorded until the aircraft stabilizes at its maximum level flight airspeed. Airspeed and altitude vs time should be recorded during the run along with power or thrust settings, ambient temperature, and aircraft weight. Altitude should be held as

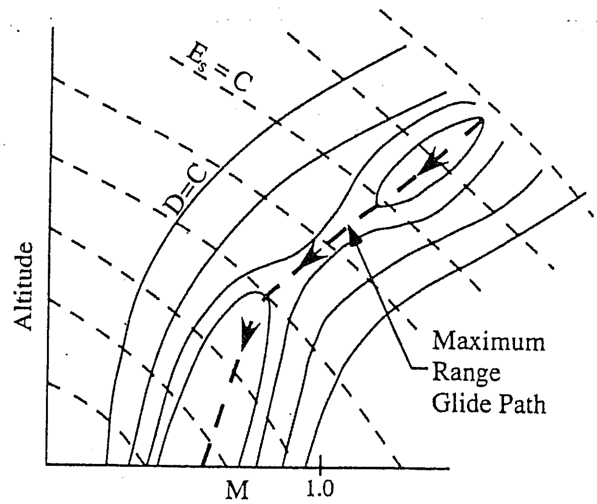


Fig. 14.7 Maximum range glide path for a supersonic airplane.

close as possible to the test altitude during the acceleration and any deviations and their magnitude from the test altitude should be recorded.

A second approach to performing the acceleration is to slow the aircraft to just above stall speed at the test power setting by entering a level turn and increasing the load factor to slow the airplane. Once at the target airspeed, the wings are rolled level and the aircraft allowed to accelerate to its maximum airspeed for that altitude and power setting. This method may work best for jet aircraft where the engine may require a certain time to stabilize at the selected thrust setting.

Data are reduced by determining dV/dt at even increments of airspeed and if excursions exist dH/dt should also be obtained for these airspeeds. Excess specific energy is then determined for each selected airspeed using Eq (14.4). Thrust horsepower in excess may then be determined using Eq (14.5). Plots of rate of climb vs airspeed such as is shown in Fig. 13.2 may be determined by performing the test at several altitudes.

References

[1]Rutowski, E. S., "Energy Approach to the General Aircraft Performance Problem," Douglas Aircraft Co., Report No. SM-14875, Santa Monica, CA, July 1953.

[2]Bryson, A. E. Jr., Desai, M. N., and Hoffman, W. C., "The Energy-State Approximation in Performance Optimization of Supersonic Aircraft," AIAA Paper No. 68-877, AIAA Guidance, Control, and Flight Dynamics Conference, Pasadena, CA, Aug. 1968.

[3]Rutowski, E. S., "Energy Approach to the General Aircraft Performance Problem," *Journal of the Aeronautical Sciences*, March 1954, pp. 187–195.

15
Turning Performance

15.1 Introduction

The measurement of turning performance is of primary importance in the evaluation of combat aircraft, since the turn radius R and turn rate w are very important in air combat. For other aircraft turning performance is not so important and in many cases is ignored. This is true of most civilian aircraft and there are no current FAA regulations for turning performance.

For military fighter and attack type aircraft, a certain level of turning performance is specified in the contract. It is then up to the flight test organization to determine if this level of performance has been met.

15.2 Forces on an Aircraft During a Level Turn

Fig. 15.1 shows the forces acting on an aircraft during a level turn. The centrifugal force F_c of the turn can be expressed by the equation:

$$F_C = ma = \frac{W}{g}\frac{V_T^2}{R} \tag{15.1}$$

where
m = mass
a = acceleration
W = weight
V_T = true airspeed
g = acceleration due to gravity
R = radius of turn

The load factor n can be derived as follows:

$$(nW)^2 = (F_C)^2 + W^2 = \left(\frac{WV_T^2}{gR}\right)^2 + W^2 \tag{15.2}$$

$$n^2 = \frac{V_T^4}{g^2R^2} + 1 \tag{15.3}$$

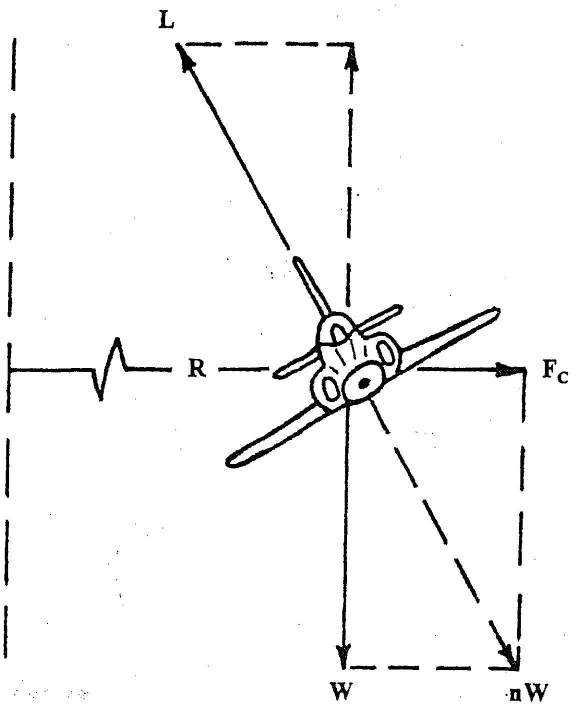


Fig. 15.1 Forces acting on an aircraft in a level turn.

The radius of turn R can be obtained by solving the above equation for R.

$$R = \frac{V_T^2}{g\sqrt{n^2 - 1}} \tag{15.4}$$

The rate of turn is obtained from:

$$\omega = \frac{V_T}{R} = \frac{g\sqrt{n^2 - 1}}{V_T} \tag{15.5}$$

where
ω is in rad/s

From flight tests we are able to obtain accurate values of V_T and ω. We can then enter Eq. (15.5) and solve for the load factor n. With this information we can enter Eq. (15.4) and solve for the turn radius R. We may then obtain plots of load factor vs airspeed and radius of turn vs airspeed for various altitudes as is shown in Figs. 15.2 and 15.3. We may then use the values obtained in level turns at constant airspeed to compare with contract specifications or other aircraft.

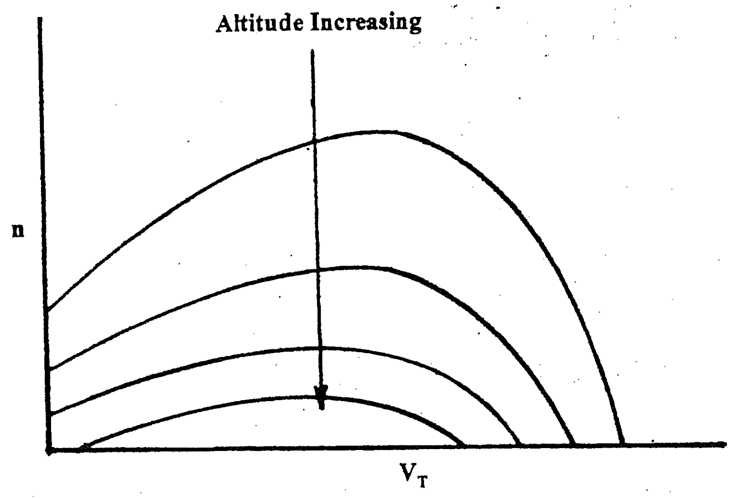


Fig. 15.2 Turning load factor vs true airspeed.

15.3 Turning Performance Limitations

There are four primary limitations on the turning performance of an airplane. They are:

1) thrust available
2) drag characteristics
3) $C_{L\max}$
4) structural strength

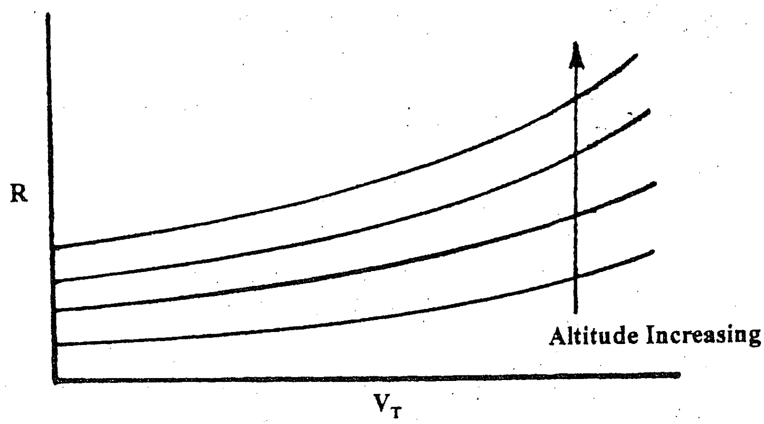


Fig. 15.3 Turn radius vs true airspeed.

15.3.1 Thrust and Drag Limitations

In the combat situation thrust and drag are the limiting factors on continuous turns without altitude loss. Since the drag must increase in the turn due to the increased load factor, the excess thrust available will decrease for any given flight speed. Eq. (15.4) shows that the greater the load factor the smaller the level turn radius for a given true airspeed. However, the greater the load factor the greater the drag and the smaller the excess thrust, so any of the factors that affect thrust or drag affect turning performance. Corrections must therefore be applied to turning performance data for:

1) temperature effects on thrust
2) pressure altitude effects on thrust
3) weight effects on induced drag, which are multiplied by increased load factor
4) effects on the test load factor resulting from nonstandard temperature, altitude, and weight

The effects of these variables are shown in Fig. 15.4.

15.3.2 $C_{L\max}$ Limitations

The $C_{L\max}$ limitation on turning performance occurs during that portion of the flight envelope where the load factor that the aircraft can pull is limited by aerodynamic stall or other angle of attack limitation. This limitation covers the low-speed portion of the flight envelope as is illustrated by the airspeed, V, vs load factor, n, diagram shown in Fig. 15.5.

All aircraft are capable of being flown to this limitation, and flight test to determine it are called lift boundary investigations.

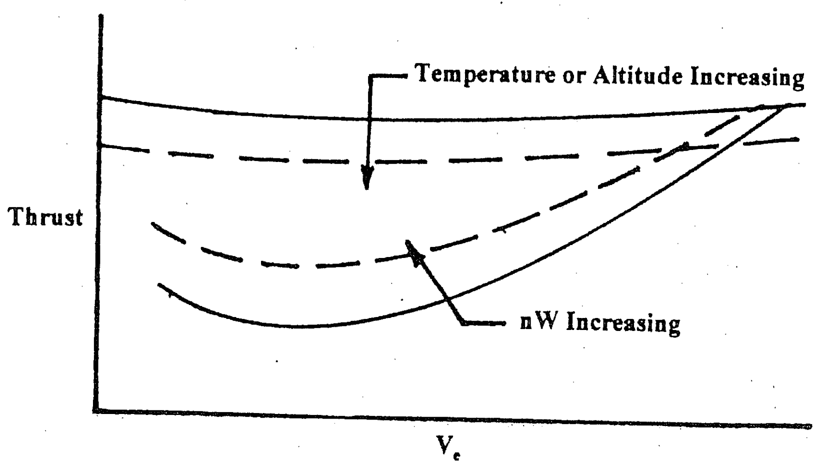


Fig. 15.4 Factors affecting turning performance.

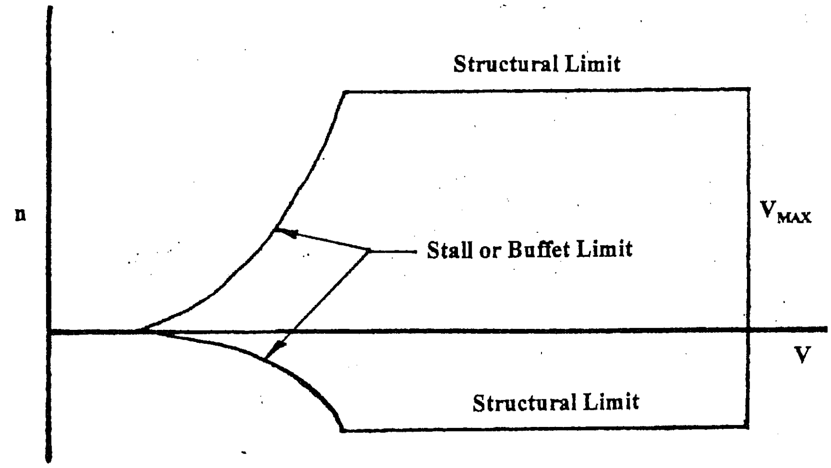


Fig. 15.5 Airspeed vs load factor (V-n) diagram.

15.3.3 Structural Limitations

Structural limitations on turning performance have not been a significant factor on older combat aircraft, since most were not capable of achieving their limit load factor in a level turn. They are becoming more of a factor on today's aircraft that have high power-to-weight ratios and are capable of sustaining the limit load factor in level flight through a larger portion of the flight envelope.

As we can see from Eq. (15.4), once we have reached limit load factor the turn radius will increase as true airspeed increases. The reverse is true of turn rate as can be seen from Eq. (15.5).

15.4 Flight Test Method

15.4.1 Preflight Procedure

A pilot's data card should be prepared for the recording of the following data in each stabilized turn:

observed airspeed (V_o)
observed pressure altitude (H_{P_o})
normal acceleration (n_Z)
fuel used or remaining (W_F)
outside air temperature (T_o)

In addition, the card should contain a table of how n_Z varies with bank angle, ϕ. It is also helpful if a blank plot of n_Z vs V_o is available so the pilot can make a rough "how goes it" plot during the flight.

15.4.2 Data Collection

First obtain a 1 g trim point at full throttle and test altitude. Be sure to allow the airspeed to stabilize. Record all data. The difference in the recorded n_Z and 1 g should be considered a zero error, or tare, and included in data corrections.

While maintaining a constant altitude, establish a 30 deg bank and allow the airspeed to stabilize. Trim the aircraft and allow it to remain in a condition of stable equilibrium for 5 s prior to recording data.

Obtain data points at increasing bank angles until the maximum n_Z in level flight is achieved. The bank angle increments should become smaller as n_Z increases. At bank angles in excess of 60 deg it may not be possible to hold a steady bank angle. At that point the pilot should switch to holding a constant airspeed.

Some propeller-driven airplanes have different turning performance in right and left turns, so data should be collected in both directions of turn.

15.4.3 Data Reduction[4]

Once data have been collected at several different altitudes they may be reduced using the following procedure:

1) Apply instrument and position corrections. Be sure to apply any tare correction to n_Z data.
2) Determine the aircraft test gross weight, W_T.
3) Correct n_Z to standard weight.

$$n_{z_S} = n_{z_T}\left(\frac{W_T}{W_S}\right) \tag{15.6}$$

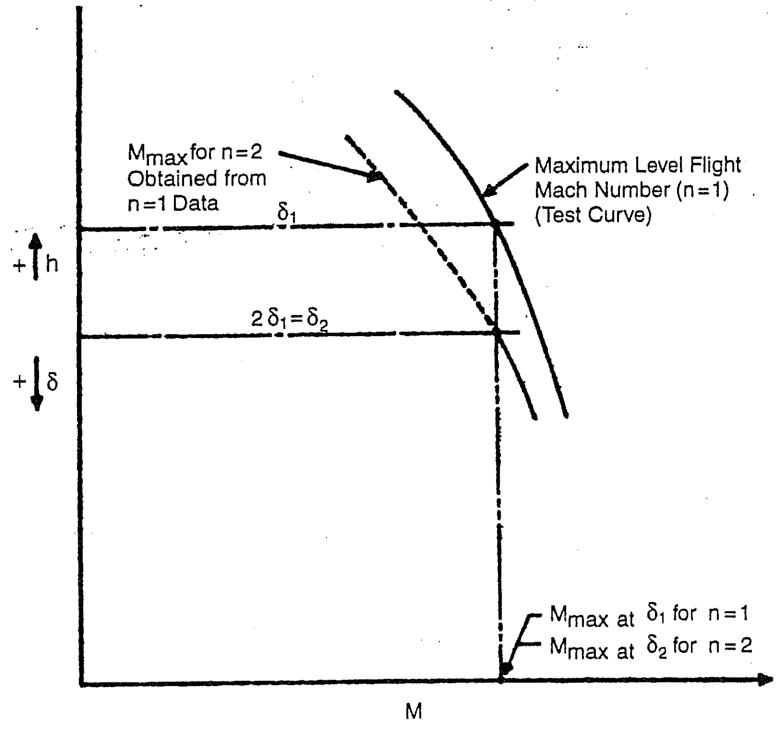
4) Construct a plot of n_{ZS} vs M or V_T.
5) Calculate turn radius, R, from Eq. (15.4).
6) Calculate turn rate, ω, from Eq. (15.5).
7) Plot n_{Z_S} vs M or V_T.
8) Plot R vs M or V_T.
9) Plot ω vs M or V_T.

15.4.4 Normalization

In many cases it may not be possible to obtain complete data for turning performance due to the amount of data required. However, Perkins and Dommasch[3] show that:

$$\frac{n_Z W}{\delta} = f(M, F_N/\delta) \tag{15.7}$$

From this functional relation we can see that for constant values of W, M, and F_N/δ the normal load factor n_Z is a direct function of the atmospheric pressure ratio δ. Through this relationship the turning performance at one altitude may



For Constant M and $N/\sqrt{\theta}$; $\frac{n_1}{n_2} = \frac{\delta_1}{\delta_2}$.

Fig. 15.6 Plot of altitude vs Mach showing load factor and pressure ratio relation.[3] [The original version of this material was first published by the Advisory Group for Aerospace Research and Development, North Atlantic Treaty Organization (AGARD/NATO) in *AGARD Flight Test Manual, Volume I—Performance.*]

be related to the turning performance at any other altitude, or:

$$\frac{n_{Z1}}{n_{Z2}} = \frac{\delta_1}{\delta_2} \tag{15.8}$$

A plot demonstrating this ratio is shown in Fig. 15.6.

References

[1]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH 64-2006, AFFTC, USAF Test Pilot School, Edwards AFB, CA, 1964.

[2]Hendrix, G. D., Godwin, O. D., Frazier, F. D., and Durnin, R. E., "Performance Flight Testing Theory," FTC-TIH 64-2005, USAF Test Pilot School, Edwards AFB, CA, 1964.

[3]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. I—Performance*, Pergamon Press, New York, 1959.

[4]Small, S. M., Prueher, J. W., Williamson, R. C., and Gemmill, A. M., "Fixed-Wing Performance Theory and Flight Test Techniques," United States Naval Test Pilot School, Flight Test Manual No. 104, Naval Air Test Center, Patuxent River, MD, July 1977.

16
Methods for Drag Determination in Flight

16.1 Introduction

An accurate determination of drag is one of the most difficult tasks undertaken in flight testing. It is, however, a very important item since if the drag polar of an aircraft is accurately determined and the thrust, or thrust horsepower, available is known then all of the performance characteristics of the subject airplane may be calculated.

There are a number of methods for determining drag through flight testing. The objective of this discussion is to examine several of these methods. We will discuss the theory of the methods along with the strengths and weaknesses of each.

The four methods we will investigate are:

1) speed power method
2) prop-feathered sinks method
3) incremental drag method
4) incremental power method

16.2 Speed Power Method[1–3,5]

This method is based on the fact that for level, unaccelerated flight thrust is equal to drag. Therefore, if we can determine the thrust, or thrust horsepower, at a given level flight speed, we may also determine the drag at that given speed. Since we have already discussed the theory behind the speed power or PIW-VIW method, we will not repeat it here.

The equation on which the speed power polar is based is:

$$\frac{BHP_{\mathrm{REQ}}\sqrt{\sigma}}{(W_T/W_W)^{3/2}} = f\left(\frac{V_T\sqrt{\sigma}}{\sqrt{W_T/W_S}}\right) \tag{16.1}$$

The left side of this equation is called PIW and the right side VIW. A typical plot of this curve is shown in Fig. 16.1. For a given configuration and propeller the PIW-VIW plot will define speed power performance at any altitude and weight. From the PIW-VIW plot we may obtain the drag polar for the airplane

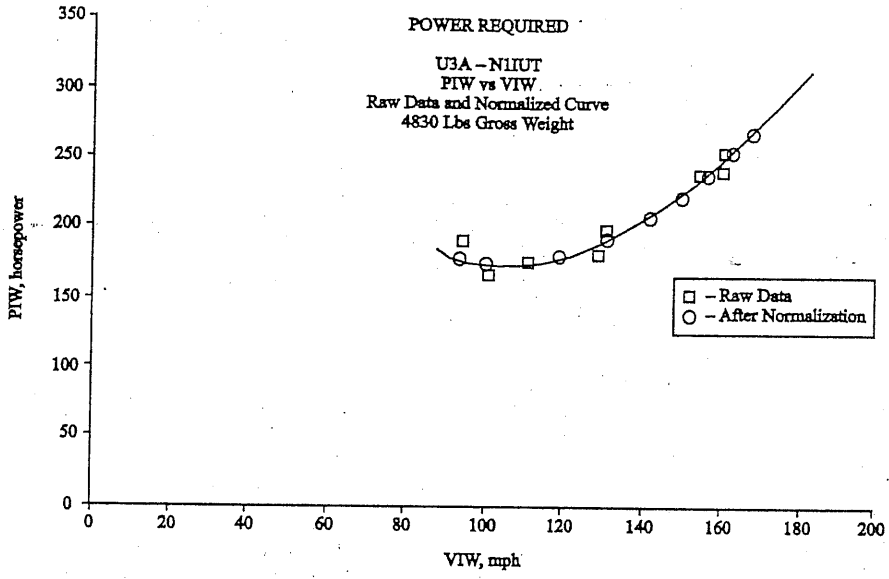


Fig. 16.1 PIW vs VIW plot for Cessna U3A.

by using the following equations for drag and lift coefficients:

$$C_D = \frac{2(550)(PIW)\eta_P}{\rho_0(VIW)^3 S} \tag{16.2}$$

$$C_L = \frac{2W_S}{\rho_0(VIW)^2 S} \tag{16.3}$$

where

$\rho_0 = .00237$ slugs/ft^3
C_D = drag coefficient
η_P = propeller efficiency
S = wing area
C_L = lift coefficient

The propeller efficiency η_P may be assumed to be a constant value such as 0.83 or determined for each point from propeller efficiency charts. The latter method being the most accurate.

Plots of C_D vs C_L and C_D vs C_L^2 developed by the above methodology from the data in Fig. 16.1 are shown in Figs. 16.2 and 16.3.

The strength of this method is that it is relatively simple to perform, and the data is easy to reduce. It also does not require sophisticated instrumentation to obtain reasonable data.

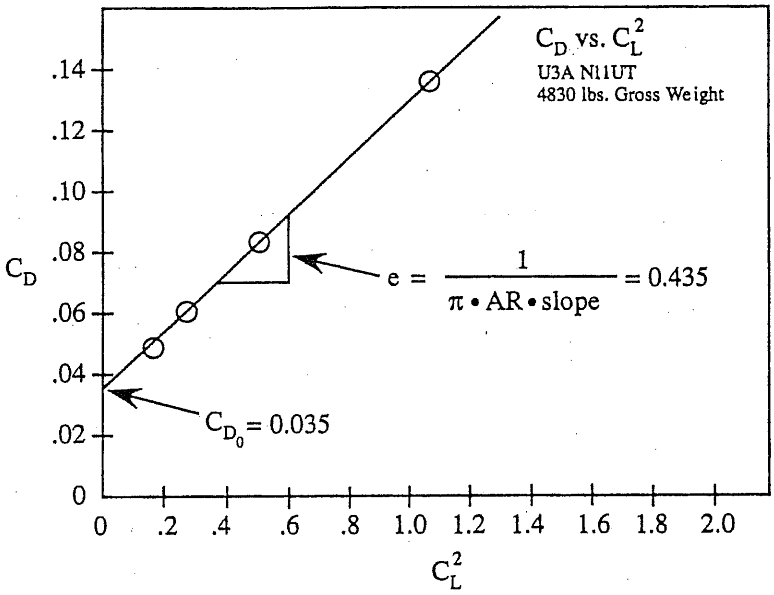


Fig. 16.2 Plot of C_D vs C_L^2 for Cessna U3A.

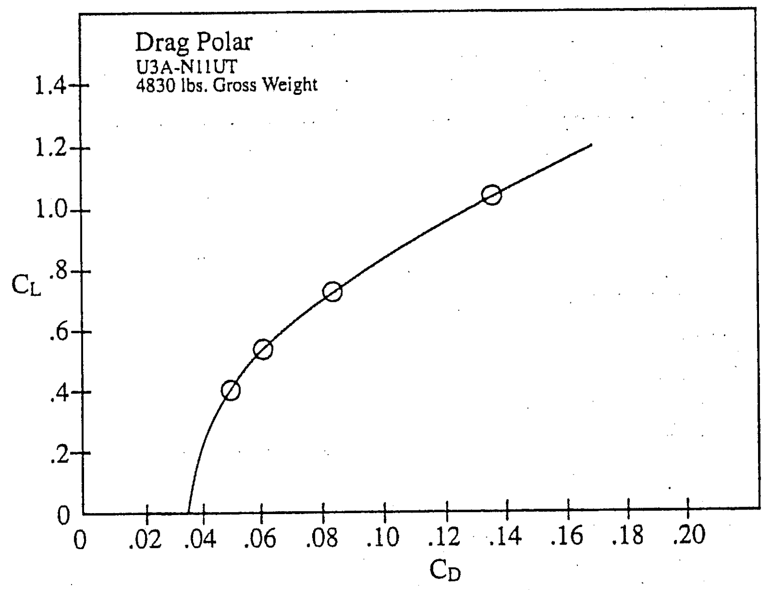


Fig. 16.3 Plot of C_L vs C_D for Cessna U3A.

Its problems fall in the area of determining the installed power and in assuming that propeller efficiency remains constant, which it does not. Installed power is difficult to determine without the use of torque meters or other devices and most of these have some limits on their accuracy. Propeller efficiency is also difficult to accurately determine and recent full-scale wind tunnel tests of the Advanced Technology Light Twin (ATLIT) have shown that the old methods, such as Gray Charts, have yet to be much improved on.

For a jet aircraft, speed power data is obtained using the constant W/δ method. As was discussed in that section, drag may be extracted directly from the flight test data.

The strengths of the method for jet aircraft are the same as for propeller-driven aircraft. Its problem is an accurate determination of installed thrust.

16.3 Prop-Feathered Sinks or Glide Polars

16.3.1 *Federal Aviation Administration Requirements*[6]

Prior to 1996 neither CAR 3 nor FAR Part 23 contained requirements for glide performance. However, in 1996 the FAA added FAR 23.71 entitled "Glide: Single-Engine Airplanes." This regulation required that the maximum horizontal distance traveled in still air per 1000 ft of altitude lost be determined in nautical miles. The speed to achieve this distance is to also be determined with the engine inoperative and the landing gear and flaps in the most favorable position. The propeller is to be in the minimum drag position.

16.3.2 *Theory*

If we examine the forces acting on an airplane in a steady-state glide (Fig. 16.4) we can see that the only forces remaining to vectorially balance the weight are the lift and the drag. In examining Fig. 16.4 we can see that, in this case:

$$L = W_T \cos \gamma \tag{16.4}$$

$$D = -W_T \sin \gamma \tag{16.5}$$

Then from the flight test we need to know the glide angle γ, and the test weight W_T.

In order to determine the glide angle γ we must determine the rate-of-descent $\mathrm{d}H_P/\mathrm{d}t$ and the true airspeed V_T. Glides are usually made through the same altitude increment at several different airspeeds in order that a plot of rate of descent vs airspeed can be constructed (Fig. 16.5). It is from this plot that the drag polars may be generated by use of Eqs. (16.4) and (16.5) and the coefficients obtained by the basic lift and drag equations.

In order to obtain the rate of descent vs airspeed plot the data must first be converted to a tapeline rate of descent. This is accomplished by first correcting

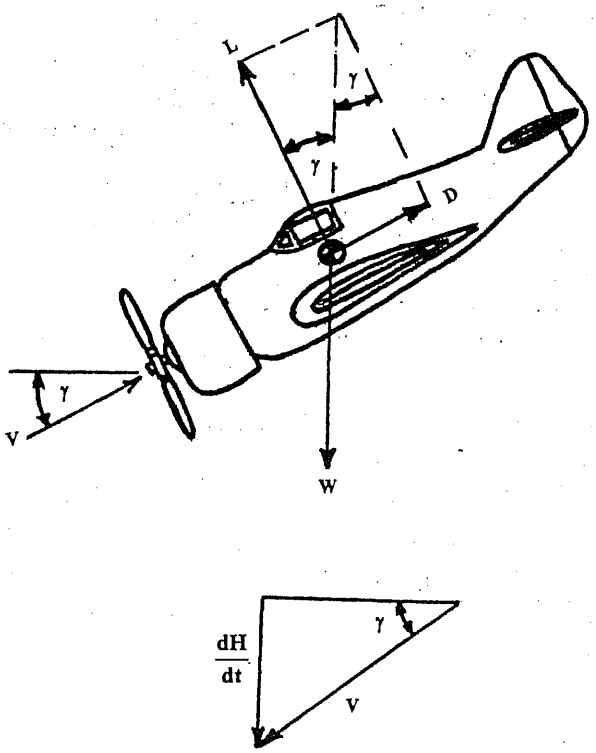


Fig. 16.4 Forces and their components during a glide.

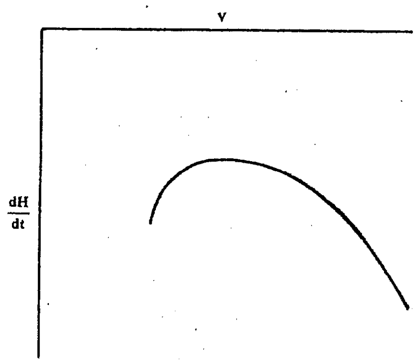


Fig. 16.5 Plot of rate of descent vs airspeed.

the rate of descent data for nonstandard temperature.

$$\left(\frac{\mathrm{d}H}{\mathrm{d}t}\right)_{TC} = \left(\frac{\mathrm{d}H}{\mathrm{d}t}\right)_{OBS}\left(\frac{T_O}{T_S}\right) \tag{16.6}$$

where
T_O = observed temperature at the test altitude
T_S = standard temperature at the test altitude

The airspeed side of the plot may be handled in the same manner as VTW.

The strengths of this method are that it is simple and straightforward, and it does not require the determination of power or propeller efficiency. It also works well for jet aircraft where the motoring characteristics of the engine may be determined.

The problems associated with the method are that for propeller-driven aircraft we cannot account for the drag due to slipstream, the drag due to the feathered propeller, or the differences in cooling drag between a cold and a hot engine. In addition, the method presents some hazard since, in most high-wing, loaded aircraft the descent is quite rapid and the time allowed for restarting the engines may be small. Therefore, it is best attempted in an area where dead stick landings can be made without difficulty.

16.4 Incremental Drag Method[4]

The incremental drag method is based on the relation that:

$$FHP = DV \tag{16.7}$$

where
FHP = thrust horsepower
D = drag
V = true airspeed

and that if a known increment of drag ΔD is added, then to maintain straight and level flight at the same airspeed an increment of thrust horsepower must also be added or:

$$FHP + \Delta FHP = (D + \Delta D)V \tag{16.8}$$

If we consider that $FHP = (BHP)\eta_P$ and that change in propeller efficiency (η_P) due to the change in drag is small it can be shown that:

$$D = \Delta D\left(\frac{BHP}{\Delta BHP}\right) \tag{16.9}$$

Since the terms on the right side of the equation can be determined we may also determine drag.

Some of the problems associated with the method are: 1) obtaining the known drag increment and 2) determining power, which is essentially the same problem as in other methods.

To the author's knowledge this method is not currently being used by flight test organizations, however, Mississippi State University conducted some research into the method that, along with the derivation of Eq. (16.9), is reported in SAE Paper 770470 (Ref. 4).

16.5 Incremental Power Method

This method is best performed on multiengine airplanes such that exactly one half of the power can be reduced. The test is performed by establishing level flight with both engines operating at a constant airspeed. This condition may be described by our equation:

$$\eta_P BHP_2 = DV \tag{16.10}$$

where

BHP_2 = the total horsepower developed by both engines

We then feather one engine while maintaining the same airspeed and record the rate of descent. The second condition may be described by the equation:

$$BHP_1 = DV + W_T(\sin\gamma) \tag{16.11}$$

$$BHP_1 = \frac{1}{2} BHP_2 \tag{16.12}$$

Then by substitution we can show that:

$$D = -2W_T(\sin\gamma) \tag{16.13}$$

As you can see this method is similar to the prop-feathered sink method and the data reduction is the same.

The method's strength is that it is much safer than the prop-feathered sink.

It does, however, have all of the problems associated with the prop-feathered sink, while gaining one new one due to the rudder deflections required to trim the dead engine. This may not be too significant at high airspeed since the deflection would be small, but as speed approaches minimum control speed it becomes much more significant.

References

[1]Herrington, R. M., Maj., Shoemacher, P. E., Bartlett, E. P., and Dunlap, E.W., "USAF Flight Test Engineering Handbook," AFFTCTRN 8273, Air Force Flight Test Center, Edwards AFB, CA, 1966.

[2]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH 64-2006, USAF Test Pilot School, Edwards AFB, CA, 1965.

[3]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test

Theory," FTC-TIH 64-2005, USAF Test Pilot School, Edwards AFB, CA, 1965.

[4]Bridges, P. D., Cross, E. J., Jr., and Boatwright, D. W., "Flight Test Evaluation of a Method to Determine the Level Flight Performance of a Propeller Driven Aircraft," SAE Report 770470, Society of Automotive Engineers, April 1977.

[5]Hamlin, Benson, *Flight Testing*, The MacMillan Company, New York, 1946.

[6]Federal Aviation Regulations Part 23, "Airworthiness Standards: Normal, Utility, Acrobatic, and Commuter Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974, rev. 1998.

17
Airspeed vs Flight Path Angle Performance Method for Powered-Lift Aircraft

17.1 Introduction

For powered-lift aircraft both lift and drag are functions of the blowing coefficient C_J in addition to the angle of attack. This fact presents certain problems when the conventional speed power performance techniques are used.

The conventional techniques, such as the constant W/δ techniques, assume that the lift coefficient C_L is not a function of thrust. Since this assumption will not hold for a powered-lift aircraft, constant W/δ data will no longer normalize into a single curve good for all altitudes and temperatures. The ability to normalize data into a single curve good for all temperatures and altitudes is one of the main reasons for using the W/δ method. Therefore, some other technique must be applied to powered-lift aircraft.

17.2 V-γ Method

The technique most commonly applied to powered-lift aircraft is the airspeed V vs flight path angle γ method. This method is based upon the assumption that, for a given thrust, flight path angle is a function of airspeed. The V-γ relationship defines a level of excess thrust for each combination of V and γ. As a result, this excess thrust may be converted into common aircraft performance parameters, such as rate of climb, acceleration in level flight, maximum level flight speed, and so on. Since the rate of climb, or acceleration in level flight, is a function of excess thrust, excess thrust may be determined in flight test by using either of these two items as a flight test technique.

The steady climb technique generates a plot of altitude vs time that, when corrected for instrument error, is plotted as shown in Fig. 17.1. To obtain rate of climb the slope of the H_P vs time curve must be taken at a given altitude. This is done using a digital computer that averaged each pair of points and calculated the slope of the resulting line. This slope is then corrected for temperature to obtain the rate of climb dH/dt.

Once the rate of climb dH/dt is obtained the flight path angle γ is obtained using the following relationship:

$$\sin\gamma = \frac{dH/dt}{V_T} \tag{17.1}$$

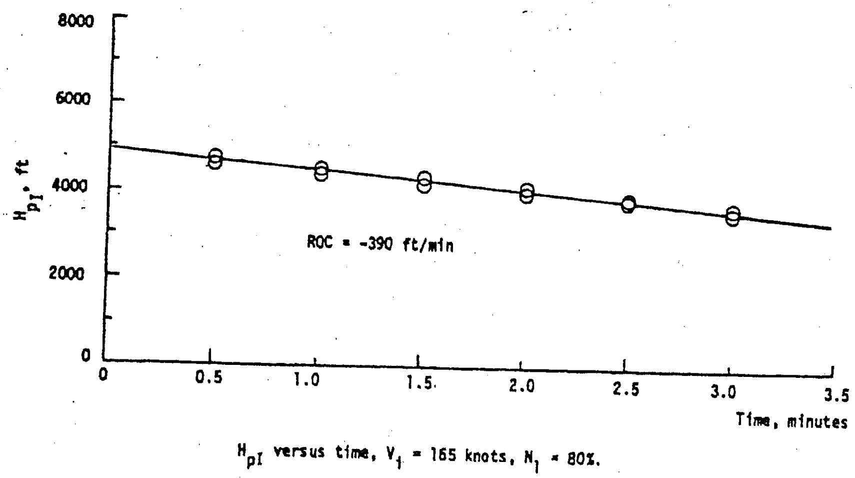


Fig. 17.1 Pressure altitude vs time at a constant airspeed and thrust.

For a steady climb the excess thrust (F_{EX}) can be found from the equation:

$$F_{EX} = F_G - D = W(\sin\gamma) \tag{17.2}$$

where
$D =$ the total airplane drag including ram drag

In a steady climb the lift is related to the weight through the following relation:

$$L = W(\cos\gamma) \tag{17.3}$$

If we solve for the weight in Eq. (17.3) and then substitute for the weight in Eq. (17.2) we have, written in coefficient form:

$$C_{FEX} = C_L(\tan\gamma) \tag{17.4}$$

Using this equation we may obtain plots of excess thrust coefficient C_{FEX} vs lift coefficient C_L. However, these plots would only be good for the altitude, temperature, and aircraft gross weight conditions where the test data were obtained. What is needed is a method to correct the data to a single plot at sea level standard conditions. Then, by using a reverse technique, the data could be extrapolated to any condition of altitude, temperature, or weight. Such a method was developed by Parks[7] during the YC-14 program.

Parks's method is based upon the assumption that, for a given configuration, the value of $\tan\gamma$ for steady state conditions will not vary if lift coefficient C_L and blowing coefficient C_J are held constant. Parks arrived at this conclusion

Table 17.1 V-γ method sample data reduction[8]
$V_O = 165$ kn; $N_1 = 80\%$; clean configuration

Step number	Quantity	Reference	Units	Value
1	V_O	flight data	kn	165
2	V_I	instrument calibration	kn	165
3	V_C	position correction	kn	167.5
4	H_{PO}	flight data	ft	4325
5	H_{PI}	instrument calibration	ft	4305
6	H_{PC}	position correction	ft	4370
7	δ at H_{PC}	altitude tables	n/d	0.8519
8	T_O	flight data	°C	22.6
9	T_a	#8 + 273.16	K	295.76
10	T_S at H_{PC}	altitude tables	K	279.50
11	θ	#9/288.16	n/d	1.02641
12	$\sqrt{\theta}$	$\sqrt{\#11}$	n/d	1.01312
13	σ	#7/#11	n/d	0.8300
14	$\sqrt{\sigma}$	$\sqrt{\#13}$	n/d	0.9110
15	V_T	#3/#14	kn	183.9
16	$(\mathrm{d}H/\mathrm{d}t)_O$	flight data	ft/min	−390
17	$(\mathrm{d}H/\mathrm{d}t)_{TC}$	#16 × (#9/#10)	ft/min	−412
18	$\sin\gamma$	#17/(#15 × 101.34)	n/d	−0.02212
19	γ	arcsin(#18)	°	−1.27
20	N_1 at H_{PC}	flight data	% rpm	80
21	$N_1/\sqrt{\theta}$	#20/#12	% rpm	79.0
22	F_G/δ	#15, #21 and plot	lb	1150
23	F_G	#22 × #7	lb	980
24	W_T	flight data	lb	3477.3
25	W_S	arbitrary	lb	3600
26	W_T/W_S	#24/#25	n/d	0.96592
27	$(W_T/W_S)^{1/2}$	$\sqrt{\#26}$	n/d	0.98281
28	F_{GS}	#23/#26	lb	1015
29	V_{EW}	#3/#27	kn	170.4
30[a]	q	$0.5 \times \rho_0 \times (\#3 \times 1.689)^2$	lb/ft²	95.12
31[b]	C_J	#23/(#30 × S)	n/d	0.0976
32	α_O	flight data	°	8.0
33	α_I	instrument calibration	°	7.0

[a]$\rho_0 = 0.0023769$ slugs/ft³ — [b]S = 105.6 ft²

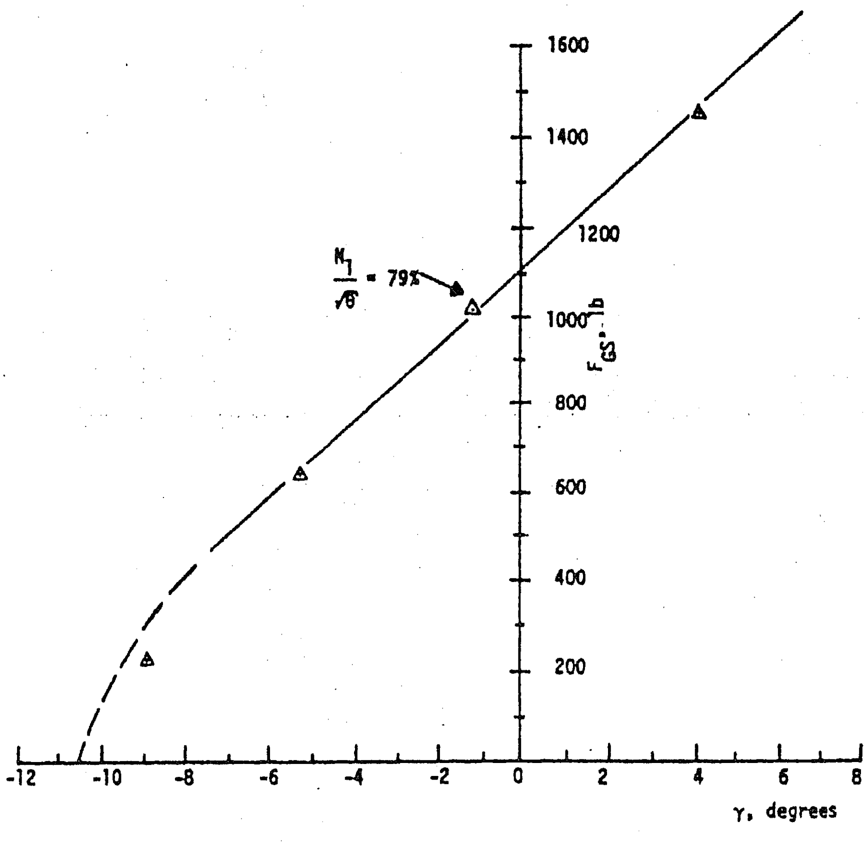
by using the relation for excess thrust coefficient C_{FEX} developed by Williams.[9]

$$C_{FEX} = C_J - \frac{KC_L^2}{\pi A + 2C_J} - C_{D0} \tag{17.5}$$

Note: The two dimensional blowing coefficient $C\mu$ of Williams's equation has been replaced by the three-dimensional blowing coefficient C_J, and C_{D0} contains all nonlift dependent drag terms.

If Eq. (17.5) is divided through by C_L it is then possible to see that, for values of constant C_L and C_J, C_{FEX} must also be constant since other variables are fixed. If we then return to Eq. (17.4) we can see that the flight path angle γ must also be constant. If γ is constant, then the lift coefficients from two different flight conditions in the same configuration may be set equal. This relation will then allow us to correct data to different conditions such as sea level standard conditions. If the variables are referenced to a climb or descent at standard weight, the only correction required is that for weight.

For instance the airspeed correction may be derived by equating the lift coefficients for the test and standard conditions and solving for the corrected



F_{GS} versus γ, sample plot, V_i = 165 knot, V_{EW} = 171.5 knots, clean configuration.

Fig. 17.2 Standardized gross thrust vs flight path angle for a constant airspeed.

equivalent airspeed V_{EW} as shown below.

$$\frac{W_S(\cos\gamma)}{1/2\rho_0 V_{EW}^2 S} = \frac{W_T(\cos\gamma)}{1/2\rho_0 V_E^2 S} \tag{17.6}$$

$$V_{EW} = V_E\left(\frac{W_S}{W_T}\right)^{1/2} \tag{17.7}$$

When C_L and C_J are constant, as was assumed by Parks, then the ratio of C_J/C_L must also be constant for different flight conditions. This equality allows us to correct the observed thrust F_G to standard conditions F_{GS} through the following relations:

$$\frac{F_{GS}}{W_S(\cos\gamma)} = \frac{F_G}{W_T(\cos\gamma)} \tag{17.8}$$

Solving for F_{GS} we have:

$$F_{GS} = F_G\left(\frac{W_S}{W_T}\right) \tag{17.9}$$

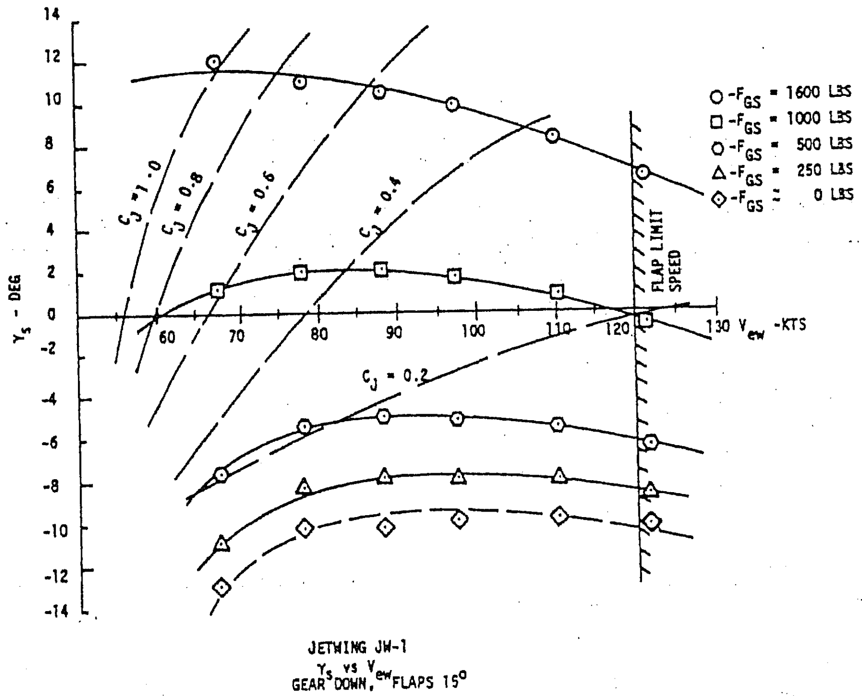


Fig. 17.3 Performance map for Jetwing JW-1, gear down, flaps 15 deg, 3600 lb gross weight.

Since the flight path angles γ were equated their correction is not required, and a plot may be constructed of γ vs V_{EW} for various values of F_{GS}.

Data can be collected by making climbs or descents at a constant airspeed and power setting. Several power settings should be used at each constant airspeed, and plots of H_P vs t, such as is shown in Fig. 17.1, obtained for each airspeed. Once the slope of H_P vs t has been obtained from a plot, the data are reduced to sea level standard conditions using the reduction sequence shown in Table 17.1.

Once corrected the data are plotted on individual plots for each airspeed as shown in Fig. 17.2. With plots such as this for each airspeed and configuration tested, the combined V-γ maps, such as is shown in Fig. 17.3, are constructed.

References

[1]"Jetwing from Colorado," *Air International*, Vol. 14, No. 2, pp. 69–71, Fine Scroll Limited, London, England, Feb. 1978.

[2]Maskell, E. C., and Spence, D. A., "A Theory of the Jet Flap in Three Dimensions," Royal Aircraft Establishment, Farnborough, Hants, England, Dec. 1958.

[3]Department of the Air Force and Naval Air Systems Command, "Flying Qualities of Piloted Airplanes," MIL-F-8785B(ASG), Aug. 1969.

[4]Langdon, S. D., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," United States Naval Test Pilot School Flight Test Manual No. 103, Naval Air Test Center, Patuxent River, MD, Aug. 1969.

[5]Perkins, C. D., and Hage, R. E., *Airplane Performance Stability and Control*, John Wiley and Sons, Inc., New York, 1963.

[6]Cochrane, J. A., and Riddle, D. W., "Joint Navy/NASA QSRA Flight Research Program Preliminary Summary," paper presented at the American Institute of Aeronautics and Astronautics—Aircraft Systems and Technology Meeting, 4 Aug. 1980.

[7]Parks, E. K., "A Note on Performance Flight Testing of STOL Airplanes and the Application of Inertial Navigation Systems for Performance Testing," office memo, Performance and Flying Qualities Branch, Air Force Flight Test Center, Edwards AFB, CA, Sept. 1975.

[8]Trimbach, J. M., "The V-γ Performance Method as Applied to the Ball-Bartoe Jetwing STOL Aircraft," masters thesis, University of Tennessee, Knoxville, June 1981.

[9]Williams, J. S., Butler, F. J., and Wood, M. N., "The Aerodynamics of Jet Flaps," Aeronautical Research Council Reports and Memoranda No. 3304, Her Majesty's Stationery Office, London, England 1963.

[10]Herrington, R. M., Maj., Shoemacher, P. E., Bartlett, E. P., and Dunlap, E. W., "USAF Flight Test Engineering Handbook," AFFTCTRN 6273, Air Force Flight Test Center, Edwards AFB, CA, May 1961, rev. June 1964, Jan. 1966.

[11]Kimberlin, R. D., "Performance Flight Test Evaluation of the Ball-Bartoe JW-1 Jetwing STOL Research Aircraft," paper presented at the Society of Flight Test Engineers Eleventh Annual Symposium Flight Testing in the Eighties, Atlanta, GA, Aug. 1980.

[12]Rooke, K. A., "An Evaluation of Nozzle Pressure Ratio as a Means of Determining In-Flight Gross Thrust of the Ball-Bartoe Jetwing with the Pratt and Whitney JT15D-1 Turbofan Engine Installed," masters thesis, University of Tennessee, Knoxville, July 1980.

[13]Small, S. M., Prueher, J. W., Williamson, R. C., and Gemmill, A. M., "Fixed-Wing Performance Theory and Flight Test Techniques," United States Naval Test Pilot School, Flight Test Manual No. 104, Naval Air Test Center, Patuxent River, MD, July 1977.

[14]McPherson, R. L., "YC-14 Flight Test Results," paper presented at the American Institute of Aeronautics and Astronautics—Aircraft Systems and Technology Meeting, Seattle, WA, Aug. 1977.

[15]Cheney, H. K., "YC-15 STOL Performance Flight Test Methods," paper presented at the Society of Flight Test Engineers Eighth Annual Symposium—Flight Test Technology, Baltimore, MD, Aug. 1977.

[16]Martin, J. L., "The Quiet Short-Haul Research Aircraft (QSRA)," *SETP Technical Review*, Vol. 14, No. 4, Sept. 1979, pp. 77–93.

18
Takeoff and Landing Theory and Methods

18.1 Introduction

Takeoff and landing distances are some of the most difficult and costly flight test data to obtain. They are difficult to obtain due to the large number of variables involved with some variables, such as pilot technique, being essentially uncontrollable. They are costly due to the size of the test team required, the amount of specialized test equipment required, and the amount of data reduction involved.

Also, takeoff and landing data may only be considered to be ballpark answers due to the large factor that pilot technique plays. This is especially true where less skilled pilots are involved and may be the reason why the FAA in CAR 3 and early FAR Part 23 did not have a regulatory requirement to collect takeoff and landing data for airplanes of less than 6000 lb gross weight.

For airplanes of more than 6000 lb gross weight, regulations do exist and are divided into those for light aircraft (aircraft of less than 12,500 lb gross weight) and transport aircraft (aircraft in excess of 12,500 lb gross weight). These categories are covered by FAR Part 23 for light aircraft and FAR Part 25 for transport aircraft. There are several differences between these regulations but the main difference is the obstacle height that must be cleared. For FAR Part 23 (light aircraft) it is 50 ft. For FAR Part 25 (transport aircraft) it is 35 ft for takeoff and 50 ft for landing. This seems to be reversed since the transport regulation is supposed to be a more difficult regulation with which to comply. However, if one considers the quality of airports from which these aircraft operate, one may begin to understand some of the reasoning behind it.

18.2 Federal Aviation Administration Regulations

18.2.1 Civil Aeronautics Regulation 3.84 Takeoff

For airplanes with a takeoff gross weight in excess of 6000 lb, the distance to takeoff and climb over a 50-ft obstacle shall be determined under the most unfavorable condition of weight and c.g. This is normally taken to be most forward c.g. at maximum TOGW. The engines must be operating within approved limitations and the cowl flaps should be set for the normal takeoff setting. Upon reaching the 50-ft obstacle, the airplane must have accelerated to an airspeed not less than $1.3V_{S1}$ unless a lower speed of V_X plus 5 can be shown to be safe under all conditions including turbulence and a complete engine failure. The takeoff distances along with configurations, speeds, and so forth, shall be included in the airplane flight manual.

The policy guidance of CAR 3.84 says that in addition to measuring the complete takeoff to 50 ft, a piece method may be used where the acceleration distance to an airspeed of $1.3V_{S1}$ is measured by camera and then a climb segment in the takeoff configuration at an airspeed of $1.3V_{S1}$ may be added to the ground distance segment.

18.2.2 CAR 3.84a Takeoff Requirements; Airplanes of 6000 lb or Less

This regulation covers airplanes of 6000 lb or less takeoff gross weight. The same wording applied to airplanes under 6000 lb TOGW in FAR Part 23 up until about the time of Amendment 34. For these airplanes there is no requirement to measure the takeoff distance, but these airplanes must demonstrate the ability to lift the nosewheel on the takeoff roll in the takeoff configuration (if nosewheel airplanes) at $0.85V_{S1}$. If they are tail wheel airplanes they must be able to lift the tailwheel at a speed of not more than $0.8V_{S1}$.

18.2.3 CAR 3.86 Landing

The landing distance to come to a complete stop from a point where the aircraft is at a height of 50 ft above the landing surface must be determined for airplanes over 6000 lb gross weight. Prior to reaching the 50-ft height, a steady gliding approach shall have been maintained at a calibrated airspeed of at least $1.3V_{S0}$. The landing shall be made in a manner that the vertical acceleration does not cause a bounce, nose over, ground loop, or porpoise, and it shall be made in such a manner that it does not require an exceptional degree of skill on the part of the pilot or exceptionally favorable conditions. The distance so obtained shall be entered in the airplane flight manual along with the airplane configuration in which they were obtained.

The policy guidance included with CAR 3.86 says that the FAA will not approve the use of reverse thrust for determining landing distances. It also says that the landing distance should be determined photographically.

18.2.4 CAR 3.87 Landing Requirements; Airplanes of 6000 lb or Less

For airplanes of 6000 lb or less it shall be demonstrated that the airplane can be safely landed and brought to a stop without requiring an exceptional degree of piloting skill, and without excessive vertical acceleration, tendency to bounce, nose over, ground loop, or porpoise.

18.2.5 FAR Part 23 Takeoff

Cover the takeoff in the modern regulation. FAR Parts 23.55, 23.57, 23.59, and 23.61 apply to commuter airliners and will not be discussed here. FAR Part 23.51 discusses the takeoff speeds for all categories of airplanes including commuter category airplanes. For normal, utility, and acrobatic categories a rotation speed, V_R, is defined as the speed at which the pilot makes a control input with the intention of lifting the airplane out of contact with the runway. For multiengine airplanes it must not be less than the greater of $1.05V_{MC}$ or $1.10V_{S1}$. For single engine airplanes it must not be less than V_{S1}.

The speed at 50 ft above the takeoff surface must be the higher of $1.10V_{MC}$ or $1.20V_{S1}$ for multiengine airplanes. For single-engine airplanes, the higher of a speed shown to be safe under all reasonably expected conditions including complete engine failure or $1.20V_{S1}$.

FAR 23.53 states that the takeoff distance must be determined using the speeds obtained under FAR 23.51 and the distance to takeoff and climb to 50 ft must be determined for each weight, altitude, and temperature within the operating limits established for takeoff with takeoff power on each engine, wing flaps in the takeoff position, and the landing gear extended.

18.2.6 FAR Part 23.73 Reference Landing Approach Speed

This regulation provides the criteria for determining the landing approach speed. For all airplanes except commuter category airplanes the landing approach speed, V_{REF}, shall not be less than the greater of V_{MC} or $1.3V_{S0}$.

18.2.7 FAR Part 23.75 Landing Distance

The landing distance to come to a complete stop must be determined from a point 50 ft above the landing surface. This must be accomplished using a steady approach speed not less than V_{REF} and an approach gradient that is not steeper than 5.2% (3 deg) down to the 50-ft point. The applicant may demonstrate that a gradient steeper than 5.2% is safe, but the demonstrated gradient must be established as an operating limitation and the pilot must have an instrument to display the gradient. A constant configuration must be maintained throughout the maneuver, and the landing must be made without excessive vertical acceleration or the tendency to bounce, nose over, ground loop, or porpoise. It must be possible to transition to a balked landing at the maximum landing weight or landing weight, altitude, and temperature as appropriate. The brakes must be used so as to not cause excessive wear on the brakes or the tires. Retardation devices, other than brakes, may be used if they are shown to provide consistent results and are safe and reliable. If a landing device depends upon the operation of an engine and the failure of that engine results in the increase of the landing distance, then the landing distance must be determined with that engine inoperative.

18.2.8 Advisory Circular 23-8A

This advisory circular discusses takeoff and landing in some detail and provides some clarification of the regulation. For instance, the term complete engine failure is taken to mean only one engine on a multiengine airplane. The circular also permits the use of the piece method for determining takeoff distances as did the CAR policy. It also clarifies the aircraft loading for the tests (forward c.g.) and cautions that the tire speed limits for Technical Standard Order tires should be observed for the high altitude, hot day cases. For landing the advisory circular clarifies the term steady gliding approach to mean an approach where power is used to control sink rate. The terms reliable and safe are discussed regarding landing as is the use of reverse thrust. Considerable detail is provided in the advisory circular and it should be consulted prior to conducting takeoff and landing tests.

18.2.9 Advisory Circular 23-15

Since it was the intent of the FAA, at industry insistence, to remove some of the more burdensome methods of AC 23-8A for small airplanes, this advisory circular allows the use of simple methods for measuring takeoff and landing distances such as the sighting bar method discussed later. If the project is a small airplane, AC 23-15 should be consulted prior to starting the test.

18.3 Theory

In considering either takeoffs or landings, it is customary to divide them into two segments: 1) ground distance segment; and 2) air distance segment.

18.3.1 Takeoff

First, let us consider the ground distance segment of the takeoff. The takeoff ground roll is affected by at least seven factors. They are:

1) wind
2) runway slope
3) aircraft weight
4) air density
5) air temperature
6) pilot technique
7) runway surface condition

Fig. 18.1 shows the forces acting on an aircraft during the takeoff roll. The equation for the aircraft acceleration during this roll may be stated as:

$$F - [D + \mu(W - L)] = \frac{W}{g}a \tag{18.1}$$

where
μ = the coefficient of friction for the tires on the runway

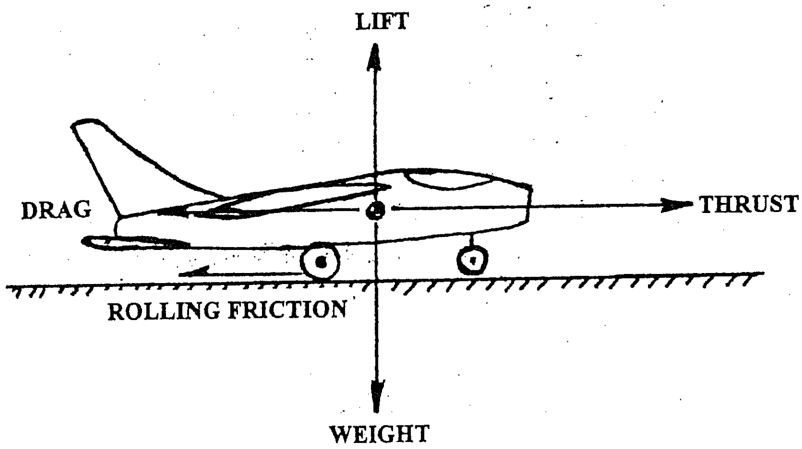


Fig. 18.1 **Forces acting on the airplane during takeoff roll.**

The term inside the brackets is the total resistance to the takeoff roll.

$$
\begin{aligned}
R &= D + \mu(W - L) \\
&= \left(C_{DP} + \frac{C_L^2}{\pi ARe}\right)qS + \mu(W - C_L qS)
\end{aligned}
\tag{18.2}
$$

where
R = the total resistance during the takeoff roll

By differentiating Eq. (18.2) with respect to C_L or determining the change in resistance with change in C_L we have:

$$\frac{\mathrm{d}R}{\mathrm{d}C_L} = \frac{2C_L}{\pi ARe}qS - \mu qS \tag{18.3}$$

If this change is zero we have the optimum C_L for achieving the minimum resistance R.

$$C_{Lopt} = \frac{\pi ARe\mu}{2} \tag{18.4}$$

Determining C_{Lopt} may be somewhat difficult since values of Oswald's efficiency factor e may not be available for operation in ground effect. In addition, rotation to the angle of attack necessary for C_{Lopt} may not be possible due to control power limitations. If this problem does not exist the possibility of overrotation does. Since induced drag is a function of the square of the lift coefficient, overrotation could destroy any gains obtained by operating near C_{Lopt}. As a result, the procedure of rotating the aircraft to the angle of attack for C_{Lopt} during the takeoff roll is seldom used.

If we return again to Eq. (18.1) we can see that all of the variables are a function of aircraft velocity and are continuously changing during the takeoff roll. This would make an exact equation for the ground roll difficult to obtain. So, in order to simplify this somewhat, we take a mean value of the net thrust and resistance. The excess thrust can be expressed by the equation:

$$F_{xcess} = F_{net} - R \tag{18.5}$$

The work done during the takeoff roll is equal to the gain in kinetic energy. This can be expressed as:

$$(F_{net} - R)S_{gTO} = \left(\frac{W}{2g}\right)V_{TO}^2 \tag{18.6}$$

where
S_{gTO} = the takeoff ground roll distance
V_{TO} = the takeoff velocity

Solving for the ground roll we have:

$$S_{gTO} = \frac{W}{g(F_{net} - R_{mean})}\left(\frac{V_{TO}^2}{2}\right) \tag{18.7}$$

This equation shows that the ground roll distance is very dependent on the takeoff airspeed. Therefore, the use of flaps or high-lift devices is a very effective means of reducing the takeoff roll.

The shortest takeoff roll may not mean the shortest total distance over an obstacle since we also must consider the air distance. In examining this phase of the takeoff we should consider the following factors:

1) wind
2) wind shear
3) aircraft weight
4) air density
5) pilot technique
6) air temperature
7) ground effect

During the air distance phase of the takeoff, the aircraft both climbs and accelerates to the 50-ft obstacle. Again, it is much simpler if we take mean values for thrust and drag and examine the problem from the "work done" standpoint. In this case:

Work done = potential energy gain + kinetic energy gain or

$$(F_{net} - D)_{mean} S_{aTO} = 50W + \frac{W(V_{50}^2 - V_{TO}^2)}{2g} \tag{18.8}$$

where
S_{aTO} = the air distance for takeoff
V_{50} = the aircraft velocity at the 50-ft obstacle

If we solve Eq. (18.8) for the air distance we have:

$$S_{aTO} = \frac{W\left[50 + \left(\frac{V_{50}^2 - V_{TO}^2}{2g}\right)\right]}{(F_{net} - D)_{mean}} \tag{18.9}$$

By summing Eqs. (18.7) and (18.9) we arrive at the total distance over a 50-ft obstacle.

$$S_{total} = S_{gTO} + S_{aTO} \tag{18.10}$$

18.3.2 Landing

The landing problem is evaluated in a manner similar to the takeoff problem. It is based on the same equation and suffers from many of the same shortcomings in test methods.

The landing case is broken into two segments as was the takeoff case. Both segments of the landing are derived from Eq. (18.1).

In deriving the equations for the landing ground roll and landing air distance, we use the same methodology as was used for takeoff. We must, however, keep in mind that the landing is a reverse case to the takeoff.

Using this approach the landing ground roll can be stated as:

$$S_{gL} = -\left(\frac{W}{2g}\right)\frac{V_{TD}^2}{(F - R)_{mean}} \tag{18.11}$$

where
S_{gL} = the landing ground roll
V_{TD} = the touchdown speed

While the air distance is given by:

$$S_{aL} = -\frac{W}{(F - R)_{mean}}\left(\frac{V_{50}^2 - V_{TD}^2}{2g} + 50\right) \tag{18.12}$$

where
S_{aL} = the landing air distance
V_{50} = the aircraft speed at the 50-ft obstacle

18.4 Test Methods

There are a number of test methods used for takeoff and landing tests. These methods vary from relatively simple and inexpensive methods to very complex and expensive methods.

18.4.1 Sighting Bar Method

One of the simplest and least expensive methods is the sighting bar method. This method consists of one or two sighting bars located at a known distance from the runway. By the use of these devices, a stopwatch, runway observers, and hand-recorded flight data, takeoff or landing data may be obtained. A typical sighting bar takeoff and landing course is shown in Fig. 18.2. Variations of this basic method include a single sighting bar that obtains both horizontal and vertical distance information. The sighting bar method is very dependent upon the observer operating the sight bar and upon the runway observers. This dependence reduces the accuracy of the method. However, the loss in accuracy is compensated for by the simplicity of the method. Since the raw data is immediately available without film reading or other preliminary reduction, it is possible to conduct more data runs and obtain a larger statistical sample.

18.4.2 Strip Camera Method

The strip camera method uses a camera that takes a picture at fixed time intervals while tracking the test aircraft and records the strip picture on a photographic plate. In this manner, one takeoff or landing run is recorded on a single photographic plate as a series of strip pictures. By knowing the location of the camera site with respect to the runway, the takeoff distance and height information along with a time reference can be obtained from the photographic plate. This method is more expensive than the sighting bar method due to the cost of processing the film and obtaining the data from it. Runway observers are still required to help pin down the liftoff or touchdown point and collect wind and temperature information. This method does provide a permanent record of the takeoff or landing and is accurate.

18.4.3 Movie Theodolite

The movie theodolite method is similar to the strip camera method except that a movie camera is substituted for the strip camera. Fig. 18.3 shows a movie theodolite takeoff and landing course. As can be seen from this figure, the movie film can be used to record other data such as time and azimuth in addition to filming the airplane. This method offers accuracy equal to or slightly better than the strip camera method. Accuracy may also be improved if there is correlation between the movie theodolite and the data recording devices onboard the aircraft. This method is more sophisticated and costly than the previously discussed methods, and the data film is more difficult and time-consuming to analyze.

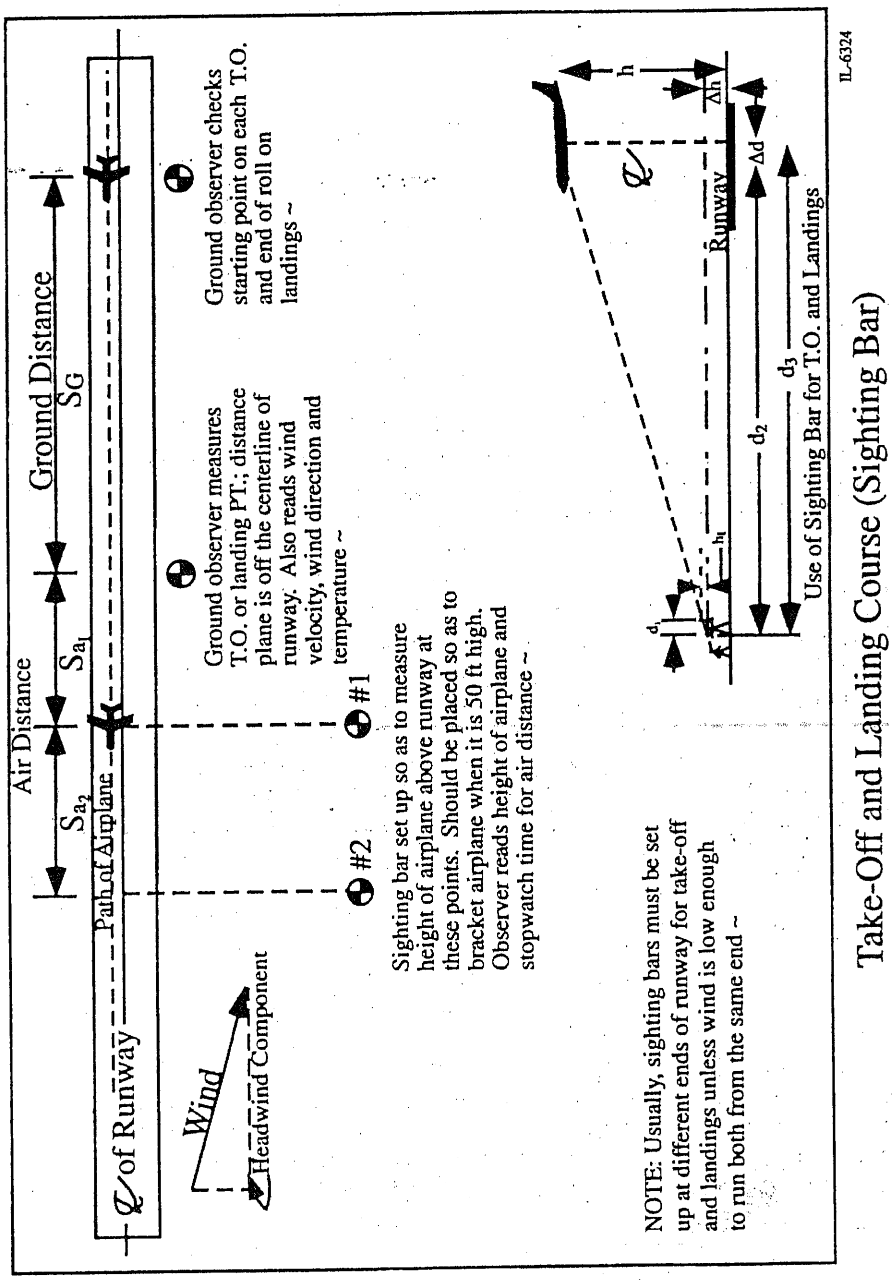
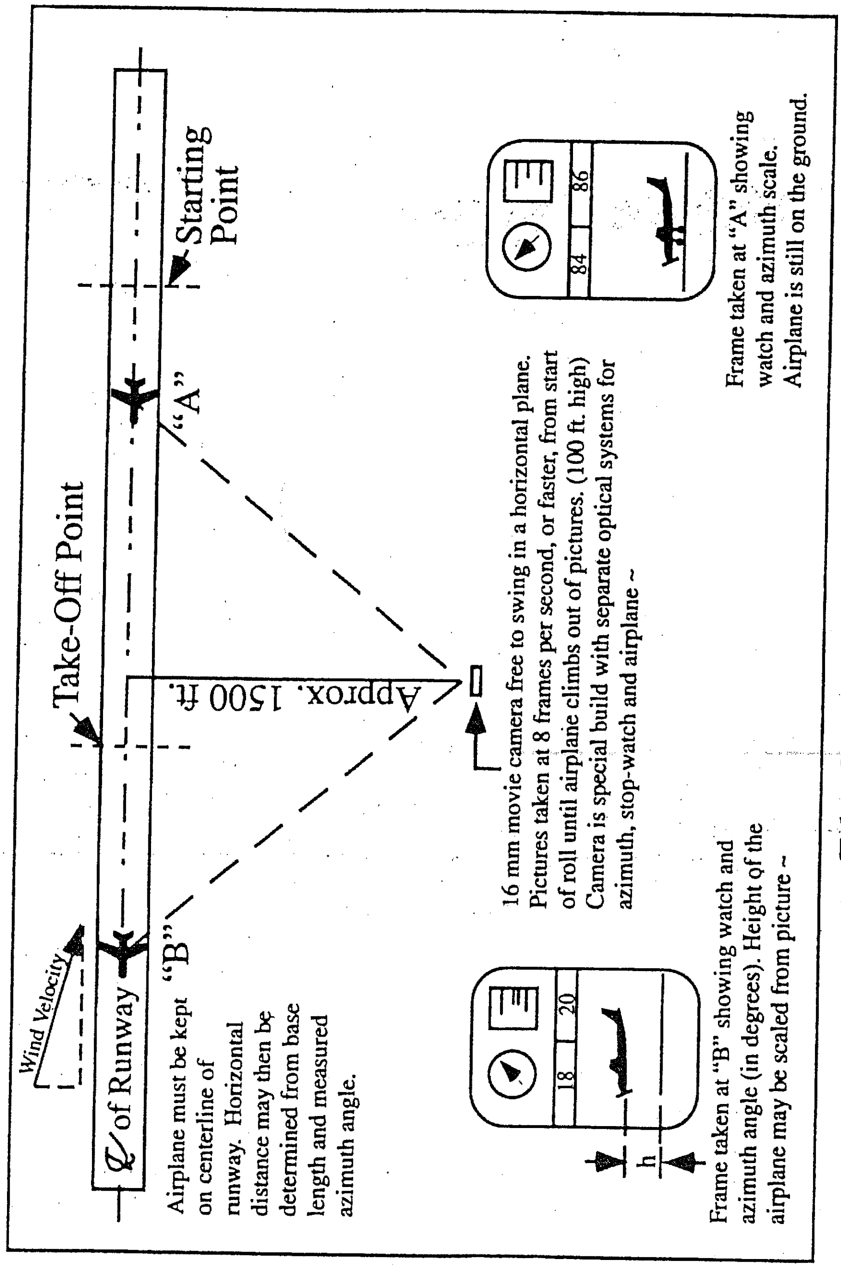


Fig. 18.2 Sighting bar takeoff and landing course.[4]



Take-Off and Landing Course

Fig. 18.3 Movie theodolite takeoff and landing course.[4]

18.4.4 Onboard Theodolite

A more recent method than those previously described is the on-board theodolite. In this method a camera is mounted on the airplane to obtain three-axis position information. Runway lights and other objects on or along the runway of known size are used with photogrammetric techniques and perspective geometry to obtain airplane position and altitude. A time readout is also displayed, usually on the edge of the film. This system also provides for easy coordination with other onboard data systems since they are all contained within the test airplane. An onboard system is expensive and requires special equipment for film analysis. It offers the advantage of not requiring a large test crew.

18.4.5 Del Norte Trisponder

A new system for determining takeoff and landing distances has recently been developed by Del Norte Technology, Inc., called the Trisponder 202. This unit measures horizontal distance and, when combined with a radio altimeter for height information, provides all the distance and height information necessary for determining takeoff and landing distances.

The Trisponder consists of a distance measuring unit (DMU), a master transponder, a remote transponder, and associated antennae, cables, and power sources. The master transponder and DMU can be located on the aircraft while the remote transponder is located on the ground or vice versa. If space and loading limitations allow, it is preferable to locate the DMU and master transponder onboard the aircraft since this will simplify the problem of time correlating the distance and height data.

The beauty of this system is that it does not require any surveyed test site. This allows the system to be loaded aboard the test aircraft and flown to another site should weather or traffic prevent conducting the test at the primary test site.

The drawback to the system is its initial cost and its electronic complexity.

18.5 Test Procedures

Since there are a large number of uncontrollable variables in takeoff and landing testing, every effort should be made to control those variables that can be controlled. First, let us look at items that apply for both takeoff and landing.

The atmospheric variables (wind, outside air temperature, and pressure altitude) should be recorded at the time of the test run for each run. The wind velocity and direction at both the 50-ft obstacle height and 6 ft above the runway should be recorded for each run. Tests should not be conducted if the wind velocity exceeds 10 kn, since wind corrections become unreliable above that wind speed.

18.5.1 Takeoff

Now let us turn our attention to the takeoff procedure. In order to reduce data scatter the aircraft should be stopped at the starting point, the power increased to takeoff power, allowed to stabilize, and then the brakes released. The pilot technique for ground roll, rotation, and climb should be, as nearly as possible, the same for each series of takeoffs. Experimentation to determine the optimum technique should be conducted prior to taking data and not during the actual tests.

Flight data should be collected during the entire takeoff and climb to the obstacle on a photo panel or other suitable recording device. A time reference and some method of correlating flight data with distance and height data such as event lights is also useful. If the events of brake release and liftoff are shown on both flight and height-distance data along with time, correlation between these two sets of data is greatly simplified and accuracy improved. If it is not possible to continuously collect flight data during the takeoff run then flight data should be recorded 1) just prior to brake release; 2) at liftoff; and 3) at the approximate 50-ft obstacle height.

18.5.2 Landing

The procedures for landing tests are somewhat like the takeoff only in a reverse direction. In the landing, test power information is not so important as in the takeoff; however, it should be monitored closely to insure that residual power does not remain after touchdown. Braking should be applied to the maximum without skidding the tires. The brakes should be given enough time to cool between runs so that they do not drag on the next takeoff or fade during the next landing. As in the takeoff, the piloting and braking technique should be as consistent as possible.

At the end of the landing run the airplane should be brought to a complete stop and held there several seconds so the end of the run may be easily identified on the film or other data trace. Event lights, or marks, are also handy in this identification.

18.6 Data Reduction

The final data plot for each takeoff or landing run from any of the test methods should look similar to those obtained from the Trisponder and shown in Figs. 18.4 and 18.5. Once we have this data and flight data we can start the data reduction to standard conditions.

18.6.1 Takeoff

The first step in reducing the takeoff data is to make the correction for wind. The wind correction must be made to both the observed ground distance S_{gO} and the observed air distance S_{aO} as shown in the following equations:

$$S_{gT} = S_{gO}\left(1 + \frac{V_W}{V_{TOw}}\right)^{1.85} \tag{18.13}$$

$$S_{aT} = S_{aO} + V_W t \tag{18.14}$$

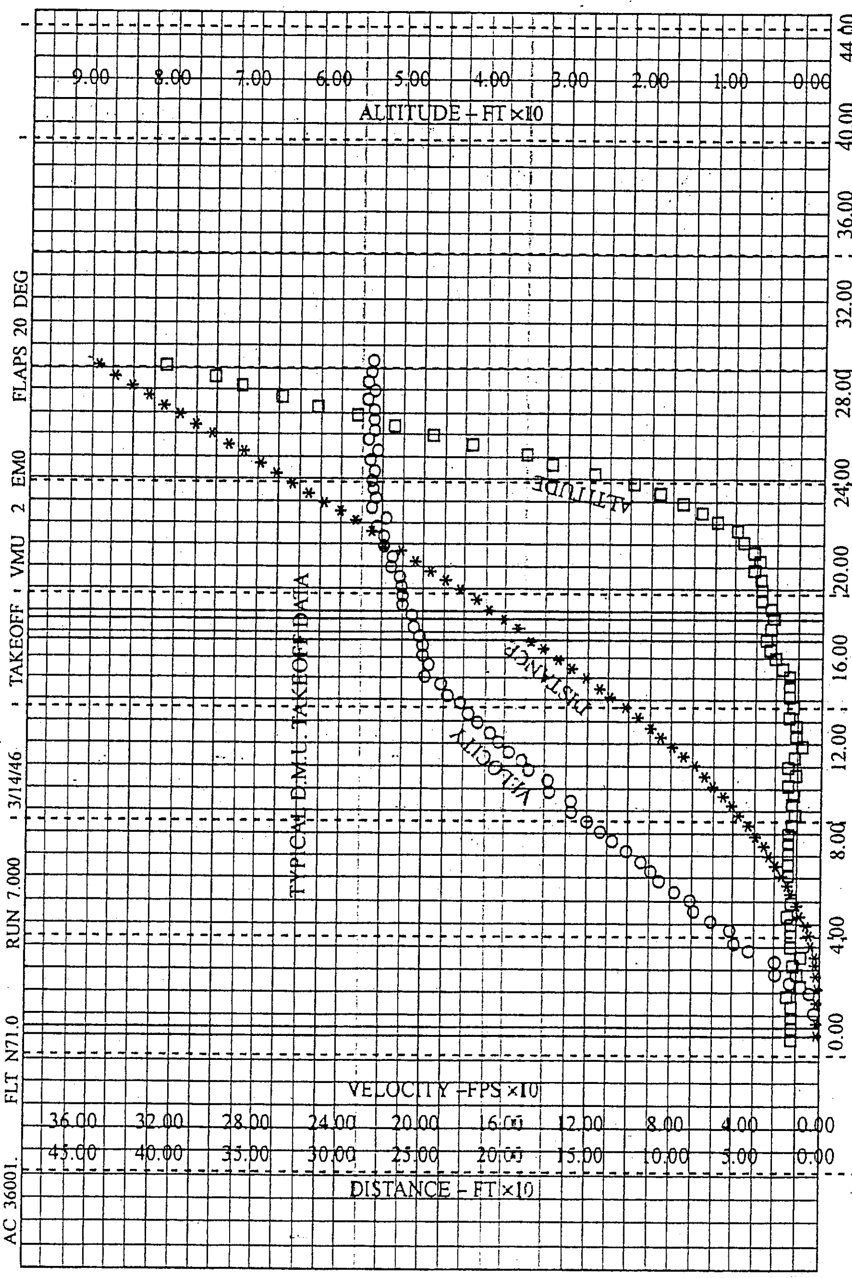


Fig. 18.4 Typical Trisponder takeoff data.[5] (Reprinted with permission from SAE 770477 ©1977 SAE International.)

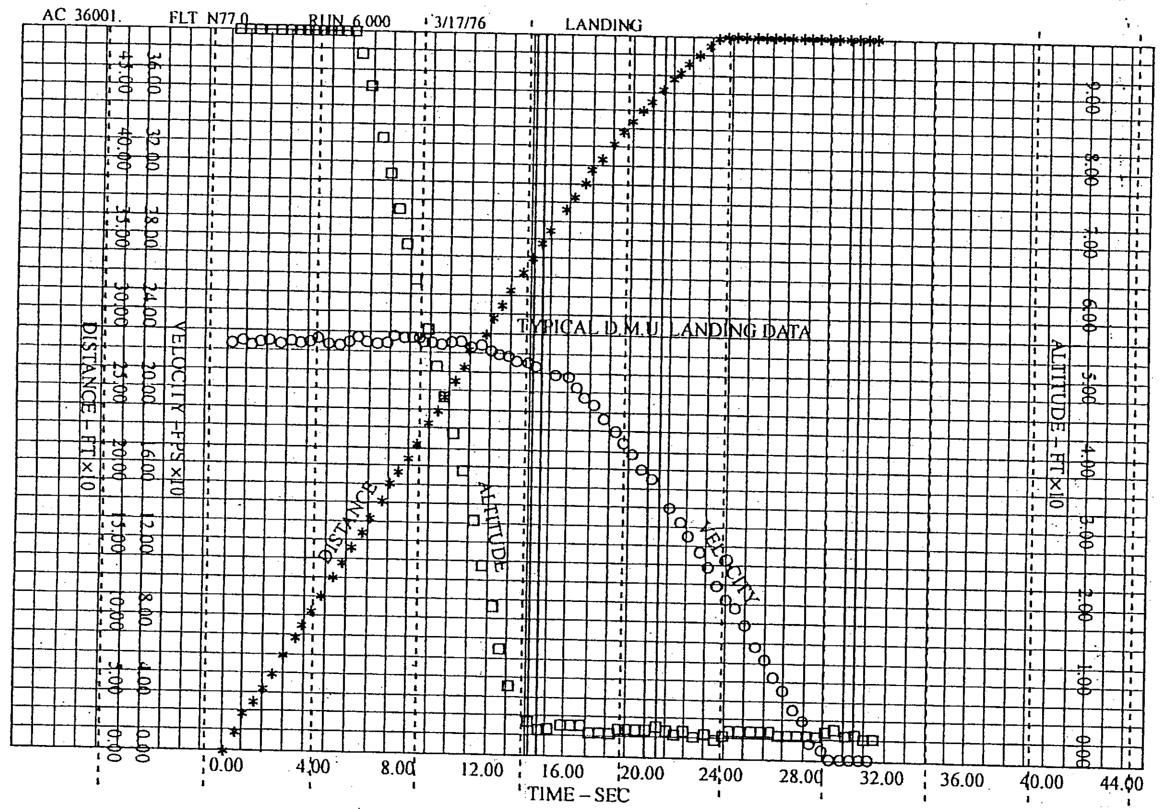


Fig. 18.5 Typical Trisponder landing data.[5] (Reprinted with permission from SAE 770477 ©1977 SAE International.)

where

S_{gT} = the test ground distance or wind corrected ground distance
S_{aT} = the test air distance or wind corrected air distance
V_W = the component of the wind velocity along the runway
V_{TOw} = the true ground speed at takeoff
t = the time from liftoff to the 50-ft obstacle

Once the wind corrected distances have been obtained it is necessary to correct the distances to sea level standard conditions. To do this by exact methods requires the knowledge of thrust, thrust minus resistance, and thrust minus drag. Since this information is not usually available for the takeoff configuration, empirical forms have been developed through experience that provide a reasonable correction. The empirical equations for correcting takeoff distances to standard conditions for jet and constant speed propeller-driven airplanes are given below. Empirical equations for other types of aircraft (fixed-pitch propeller and so on) are given in the "Flight Test Engineers Handbook."[3]

18.6.1.1 Jet aircraft.

$$S_{gS} = S_{gT}\left(\frac{W_S}{W_T}\right)^{2.3}\left(\frac{\sigma_T}{\sigma_S}\right)\left(\frac{F_T}{F_S}\right)^{1.3} \tag{18.15}$$

$$S_{aS} = S_{aT}\left(\frac{W_S}{W_T}\right)^{2.3}\left(\frac{\sigma_T}{\sigma_S}\right)^{0.7}\left(\frac{F_T}{F_S}\right)^{1.6} \tag{18.16}$$

where

S_{gS} = sea level standard ground distance
S_{aS} = sea level standard air distance
F_T = test thrust
F_S = standard thrust

18.6.1.2 Constant speed propeller-driven aircraft.

$$S_{gS} = S_{gT}\left(\frac{W_S}{W_T}\right)^{2.6}\left(\frac{\sigma_T}{\sigma_S}\right)^{1.9}\left(\frac{N_T}{N_S}\right)^{0.7}\left(\frac{BHP_T}{BHP_S}\right)^{0.5} \tag{18.17}$$

$$S_{aS} = S_{aT}\left(\frac{W_S}{W_T}\right)^{2.6}\left(\frac{\sigma_T}{\sigma_S}\right)^{1.9}\left(\frac{N_T}{N_S}\right)^{0.8}\left(\frac{BHP_T}{BHP_S}\right)^{0.6} \tag{18.18}$$

where

N_S = maximum takeoff rpm
N_T = actual rpm during the test run

Once the standard ground and air distances are obtained by one of the above methods, the standard distance over a 50-ft obstacle S_{50} is obtained by adding the air and ground distances together.

$$S_{50} = S_{gS} + S_{aS} \tag{18.19}$$

This process is done for each test run and an average taken of all the runs to determine the final distance.

After the average distances have been determined the data may be expanded to nonstandard conditions, such as those shown in Fig. 18.6, by reverse use of the above equations.

18.6.2 Landing

The correction of landing distances is much simpler than is the correction of takeoff distances. The reason for this is that the power is either at idle or held constant at some low value from the obstacle height to touchdown. This removes the power corrections from the empirical equations and greatly simplifies them.

The landing data, like the takeoff data, is first corrected for wind. The equations for wind correction of landing distances are quite similar to the takeoff wind corrections and are shown below.

$$S_{gT} = S_{gO}\left(\frac{V_{TD} + V_W}{V_{TD}}\right)^{1.85} \tag{18.20}$$

$$S_{aT} = S_{aO} + V_W t \tag{18.21}$$

where

V_{TD} = the touchdown airspeed

To correct these distances to standard conditions the following empirical equations may be used:

$$S_{gS} = S_{gT}\left(\frac{W_S}{W_T}\right)^2\left(\frac{\sigma_T}{\sigma_S}\right) \tag{18.22}$$

It has been found that the weight correction to the landing ground distance is not accurate and that weight does not affect landing ground distance in any predictable form. For this reason landing tests are usually conducted as close to maximum gross weight as possible and the weight correction of Eq. (18.22) ignored.

$$S_{gS} = S_{gT}\left(\frac{\sigma_T}{\sigma_S}\right) \tag{18.23}$$

$$S_{aS} = S_{aT} \tag{18.24}$$

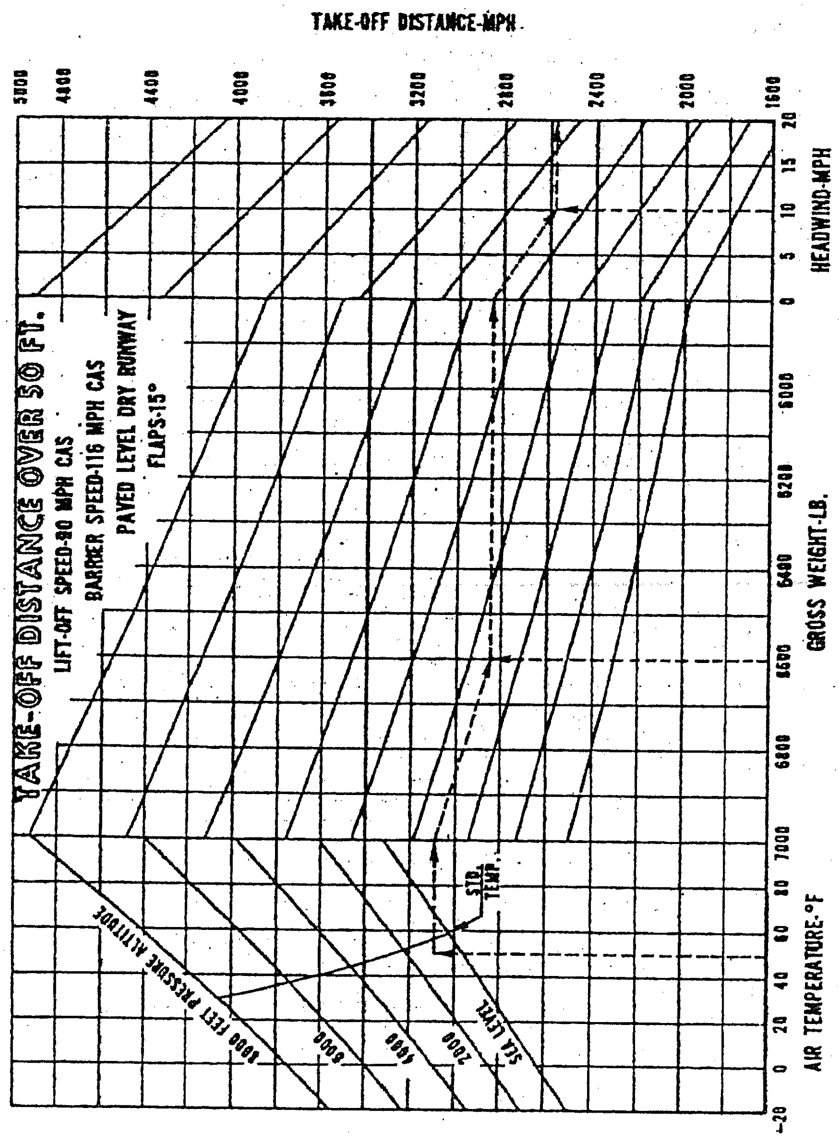


Fig. 18.6 Expanded handbook takeoff data.[6]

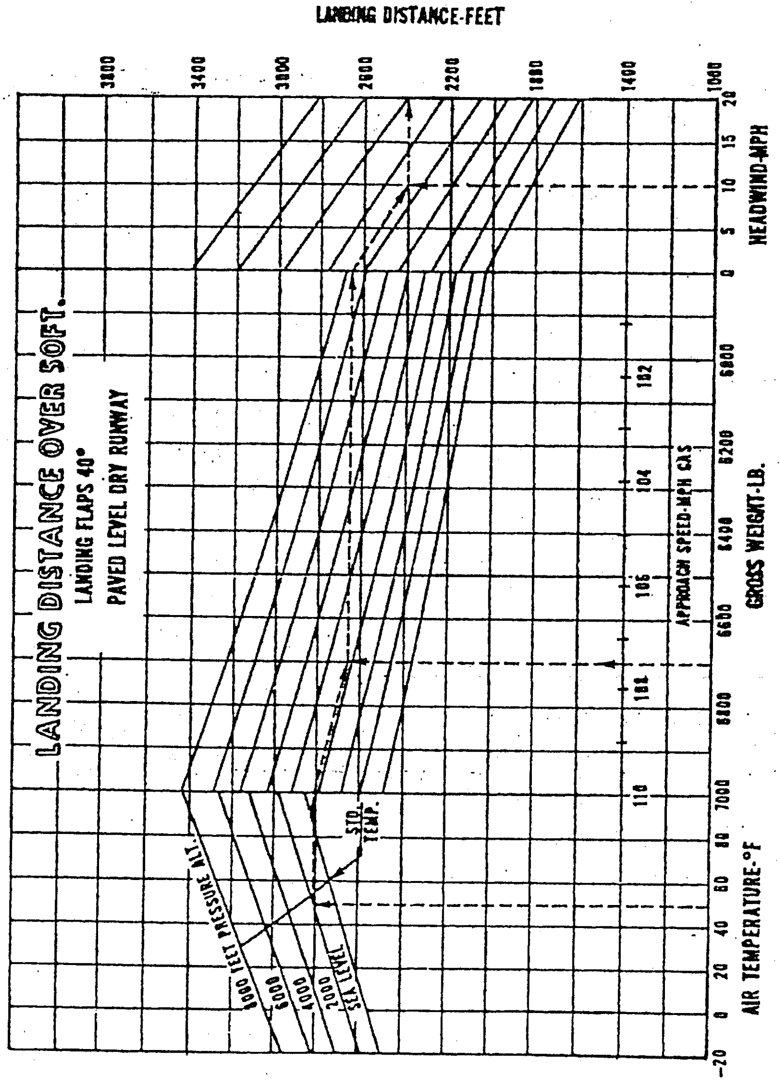


Fig. 18.7 Expanded handbook landing data.[6]

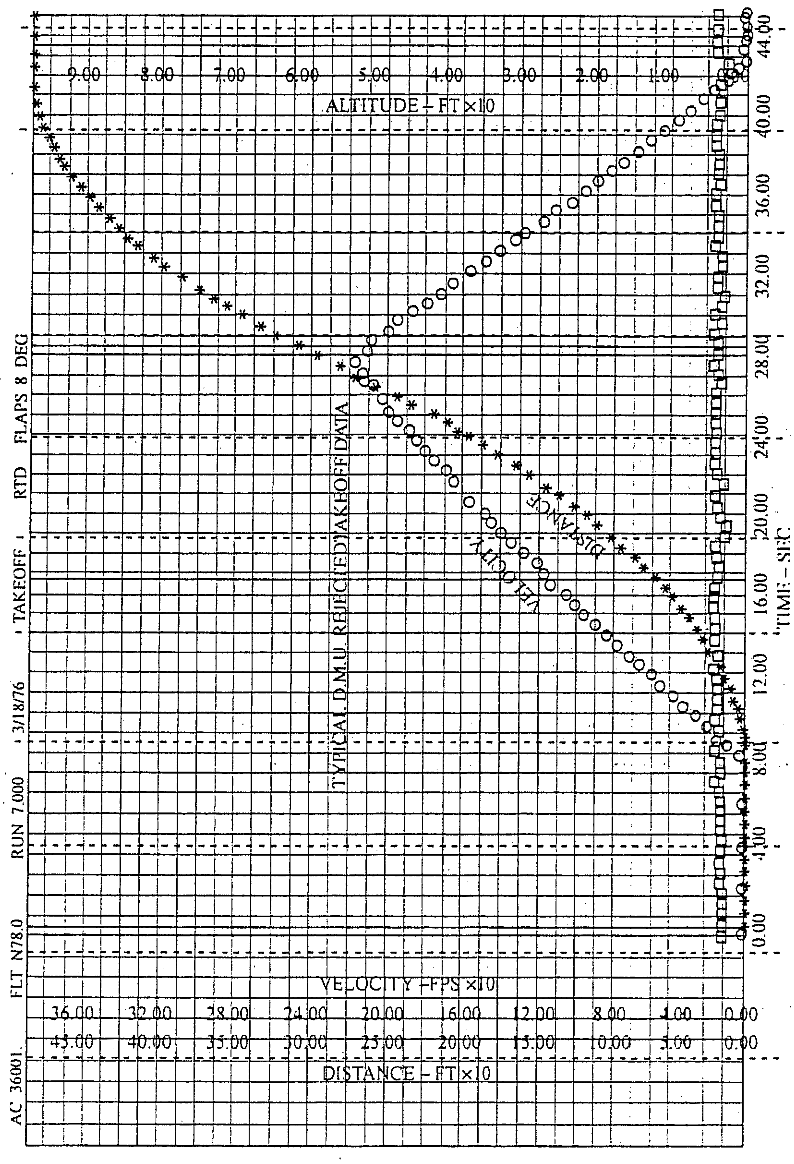


Fig. 18.8 Typical Trisponder rejected takeoff data.[5] (Reprinted with permission from SAE 770477 ©1977 SAE International.)

NAVAJO CHIEFTAIN

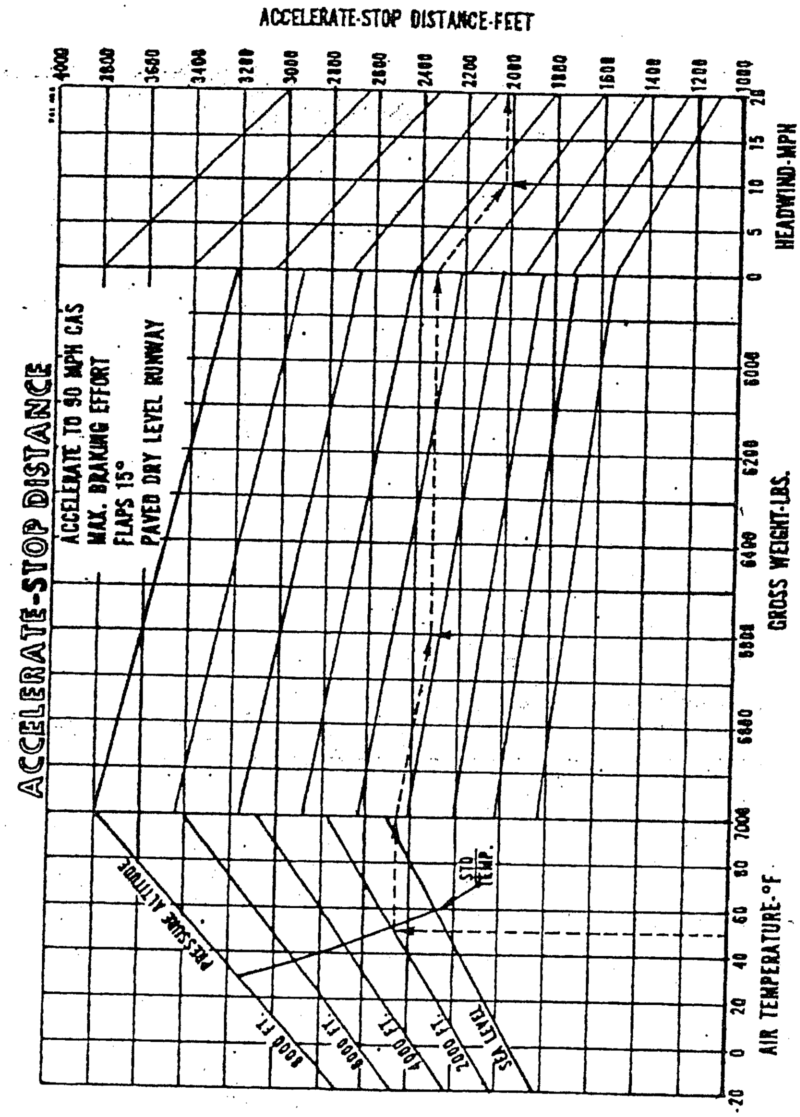


Fig. 18.9 Expanded handbook rejected takeoff data.[6]

The total standard landing distance can then be expressed as:

$$S_{50} = S_{gT}\left(\frac{\sigma_T}{\sigma_S}\right) + (S_{aO} + V_W t) \tag{18.25}$$

Expansion to nonstandard conditions, such as is shown in Fig. 18.7, may be accomplished by a reverse use of the correction equations.

18.7 Rejected Takeoff Distances

The distance required to accelerate to takeoff speed and then abort the takeoff and stop is called a rejected takeoff. It is also sometimes called balance field length. It applies primarily to multiengine airplanes where the takeoff is rejected because of an engine failure at takeoff speed.

The test is conducted by accelerating the aircraft to liftoff speed as in the takeoff test, failing an engine, waiting a specified number of seconds to simulate failure recognition time, then closing the throttle on the good engine and applying brakes as in the landing ground run. Data is recorded, and time, distance, and velocity plots similar to the one shown in Fig. 18.8 are generated. Since this procedure is essentially a takeoff ground run tied to a landing ground run with an intermediate segment in the middle, the data reduction to standard conditions can be accomplished by using the takeoff and landing ground distance reduction methods.

Data may be expanded to nonstandard conditions as shown in Fig. 18.9 by reverse use of these equations.

References

[1]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Theory," FTC-TIH 64-2005, USAF Test Pilot School, Edwards AFB, CA, 1965.

[2]Godwin, O. D., Frazier, F. D., and Durnin, R. E., "USAF Performance Flight Test Techniques," FTC-TIH 64-2006, USAF Test Pilot School, Edwards AFB, CA, 1965.

[3]Herrington, R. M., Maj., Shoemacher, P. E., Bartlett, E. P., and Dunlap, E. W., "USAF Flight Test Engineering Handbook," AFFTCTRN 6273, Air Force Flight Test Center, Edwards AFB, CA, May 1961, rev. June 1964, Jan. 1966.

[4]Bilke, Paul F., "Performance Flight Test Methods in Use by the Flight Section," Army Air Forces Technical Report No. 5069, Army Air Forces Material Command, Dayton, OH, 1944.

[5]Schick, F. D., "An Inexpensive Electronic Method for Measuring Takeoff and Landing Distances," SAE Report 770477, Society of Automotive Engineers, April 1977.

[6]Piper Aircraft Corp., *Navajo Chieftain Pilot's Operating Handbook*, Lock Haven, PA, 1972.

PART 2
Stability and Control Flight Testing

19
Introduction to Stability and Control Flight Testing

19.1 Introduction

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

19.2 Regulations

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

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

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

19.3 Reference Axes Systems

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

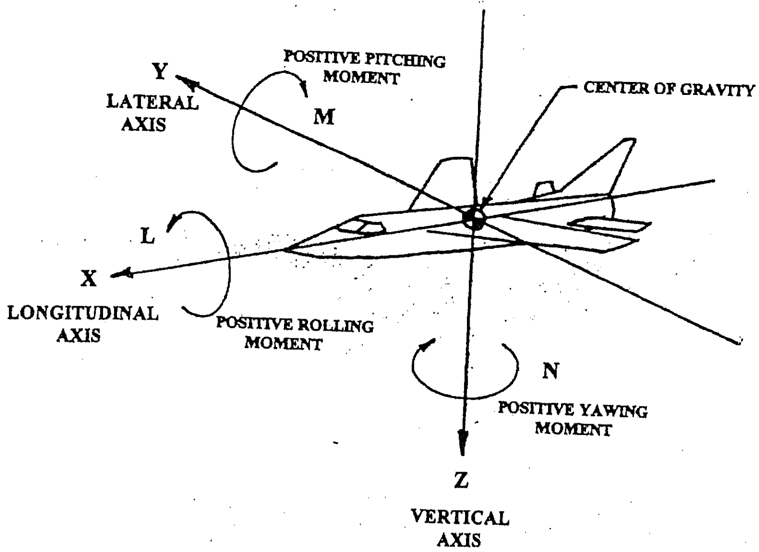


Fig. 19.1 Body axis reference system.[1]

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

19.4 Definitions of Stability and Controllability

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

19.4.1 Static Stability[1,2]

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

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

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

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

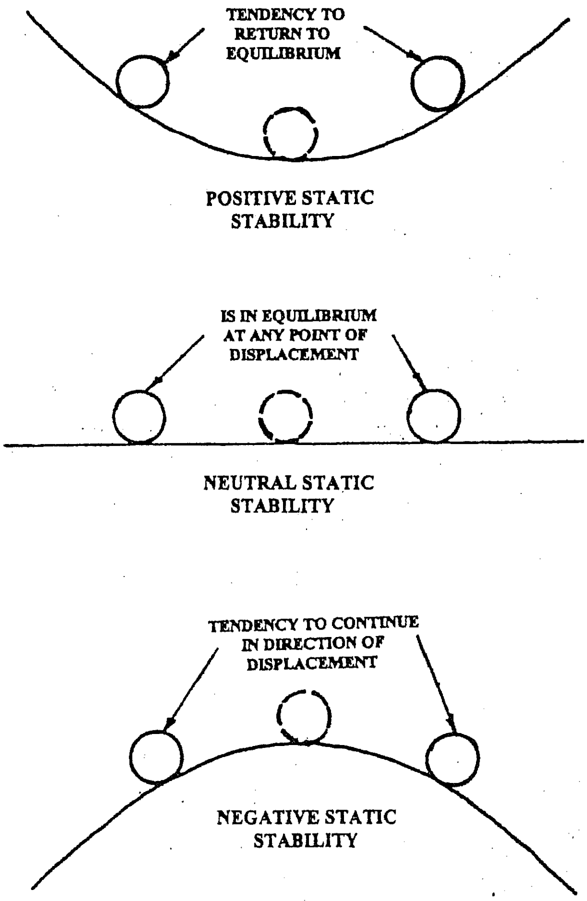


Fig. 19.2 Static stability.[1]

19.4.2 Dynamic Stability

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

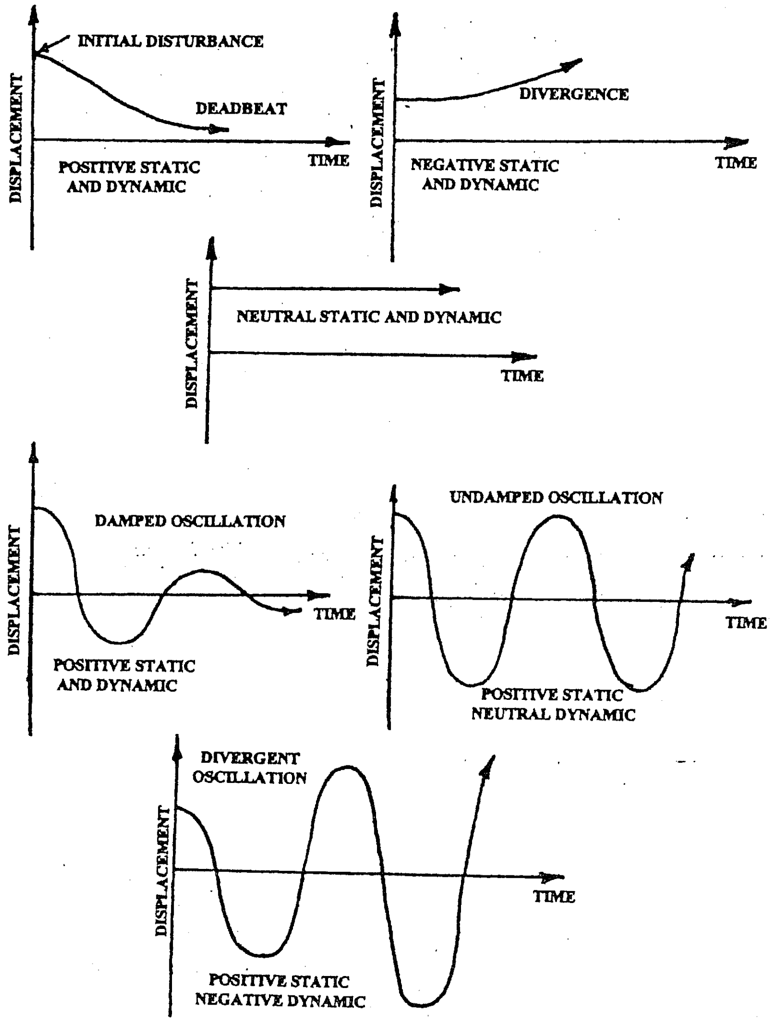


Fig. 19.3 Dynamic stability.[1]

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

19.4.3 Trim[1]

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

19.4.4 Controllability[1]

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

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

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

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

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

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

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

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

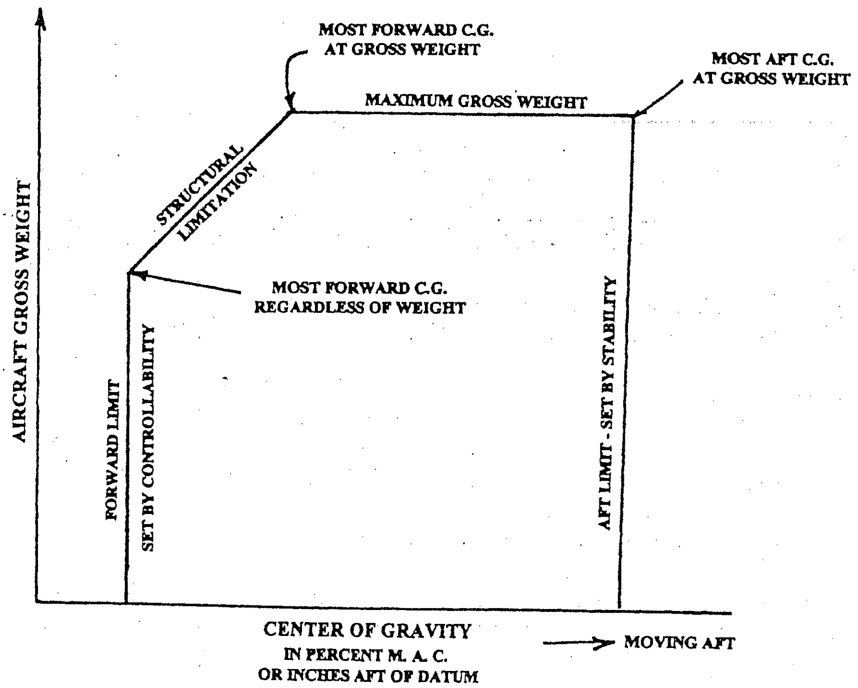


Fig. 19.4 Aircraft c.g. envelope.

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

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

19.6 Control System Characteristics

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

19.6.1 Control System Friction

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

19.6.2 Control System Breakout Force

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

19.6.3 Control System Gearing

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

References

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

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

20
Static Longitudinal Stability Theory

20.1 Introduction

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

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

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

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

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

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

20.2 Stick-Fixed Static Longitudinal Stability

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

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

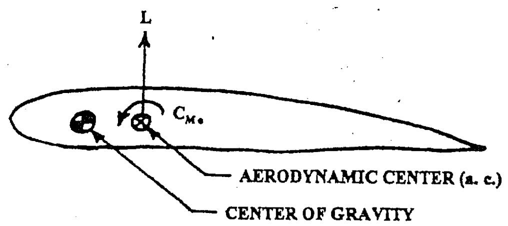
The fuselage contribution is also generally unstable.[3,4] This is due to its shape and the upwash and downwash of the wing acting on it, causing nose up pitching moments with increasing C_L. It is the fuselage ahead of the wing that is the largest problem, and long noses should be avoided where possible.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

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

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

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

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



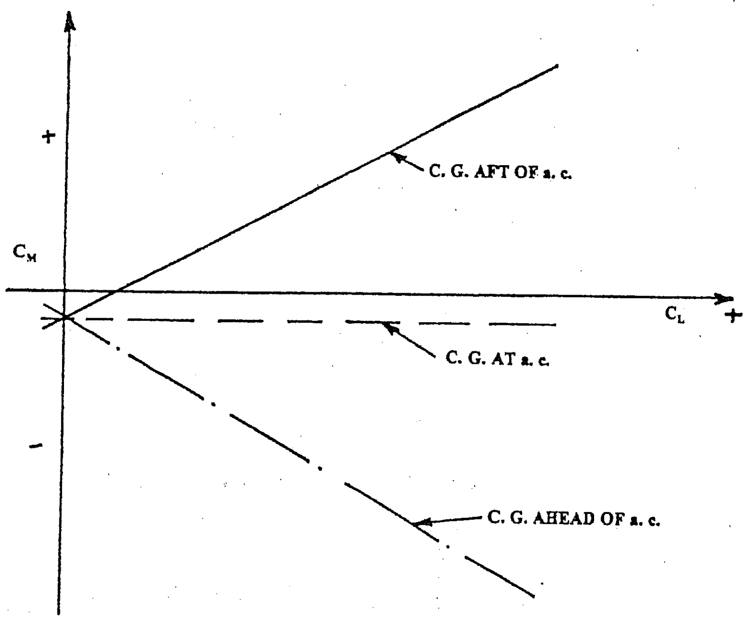


Fig. 20.1 Wing contribution to stability.[3]

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

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

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

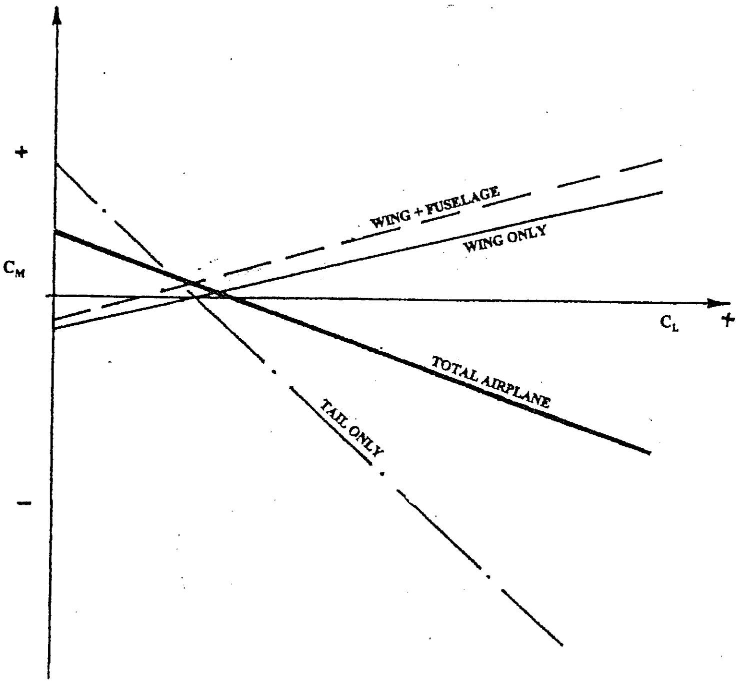


Fig. 20.2 Contribution of airplane components to stability.[3]

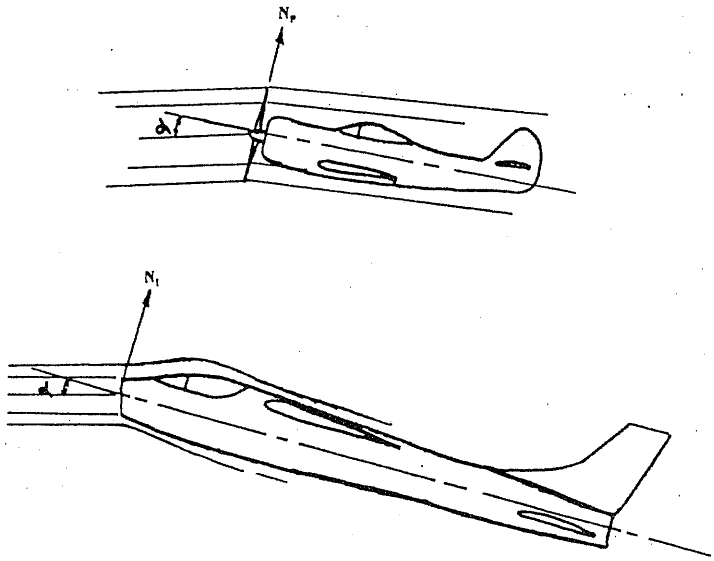


Fig. 20.3 Normal force created by propeller or jet intake.[3]

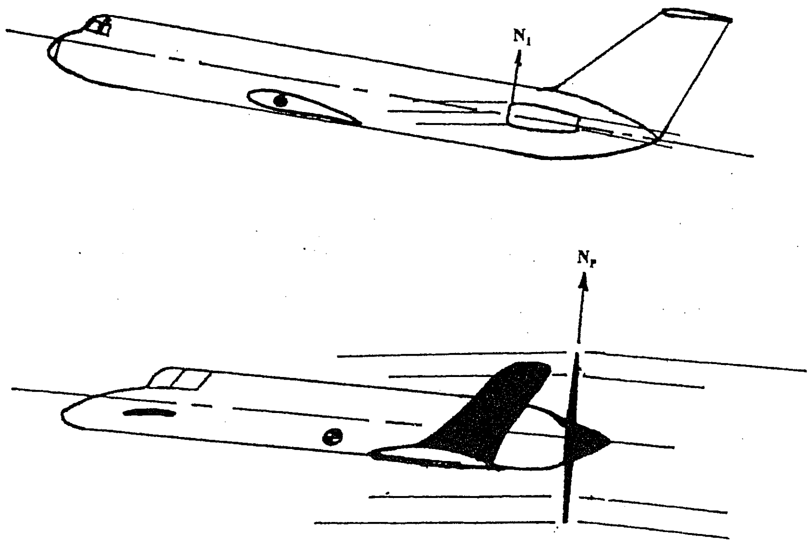


Fig. 20.4 Effects of aft-mounted jet engines or propellers.

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

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

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

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

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

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

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

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

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

where

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

a_t = lift curve slope of the horizontal tail

a_w = lift curve slope of the wing

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

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

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

where

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

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

20.3 Longitudinal Control

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

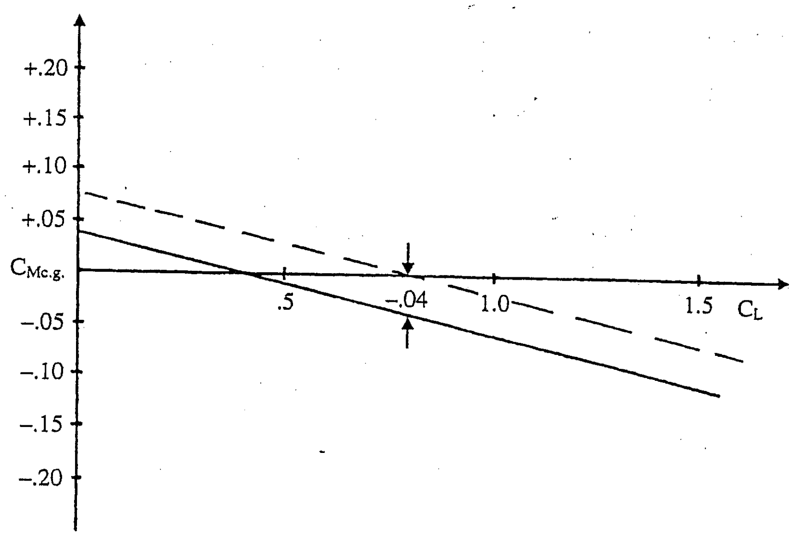


Fig. 20.5 Longitudinal control mechanics.[1]

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

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

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

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

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

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

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

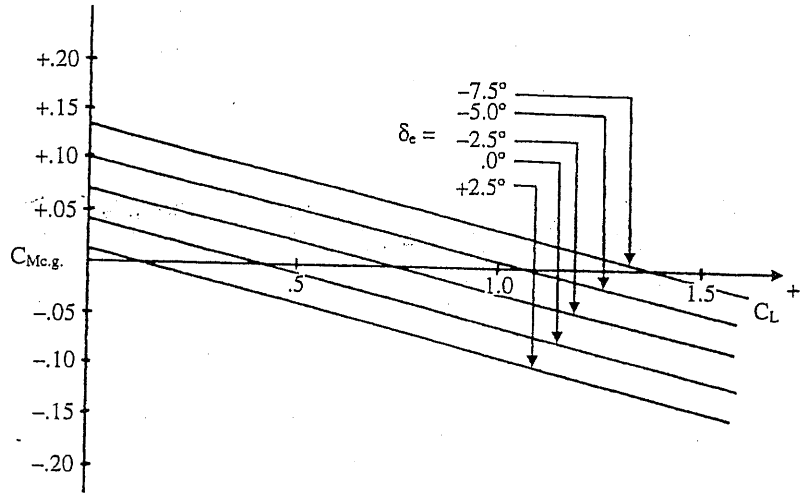


Fig. 20.6 Variation of $C_{m_{c.g.}}$ vs C_L for various elevator angles.[1]

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

20.4 Elevator Position Stability

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

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

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

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

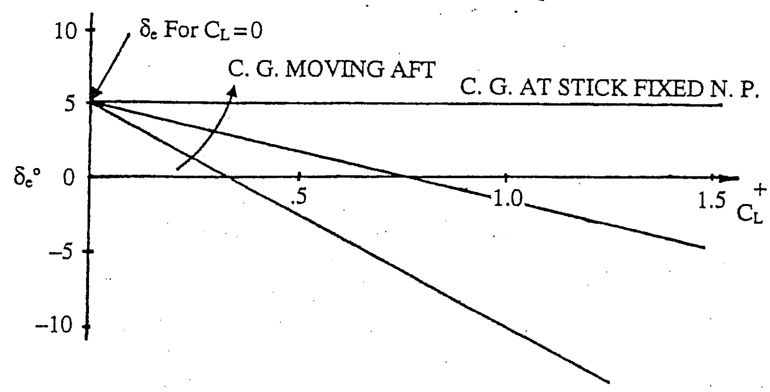


Fig. 20.7 Elevator position vs C_L for various c.g. positions.[1]

where

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

For a stabilizer, elevator configuration:[4]

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

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

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

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

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

20.5 Stick-Free Longitudinal Stability

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

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

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

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

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

where

C_{he} = the total elevator hinge moment coefficient

C_{ho} = the elevator hinge moment due to camber, zero for a symmetrical section

$C_{h_{\delta_t}}$ = the elevator hinge moment due to trim tab deflection

δ_t = the trim tab deflection

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

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

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

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

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

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

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

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

20.6 Control Force Stability

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

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

$$\frac{dF_s}{dV_e} = 2K\frac{W}{S_W}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{dC_m}{dC_L}\right)_{free}\frac{V_e}{V_{e_{trim}}^2} \tag{20.13}$$

where

dF_s/dV_e = the longitudinal control force variation with equivalent airspeed about the trim airspeed

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

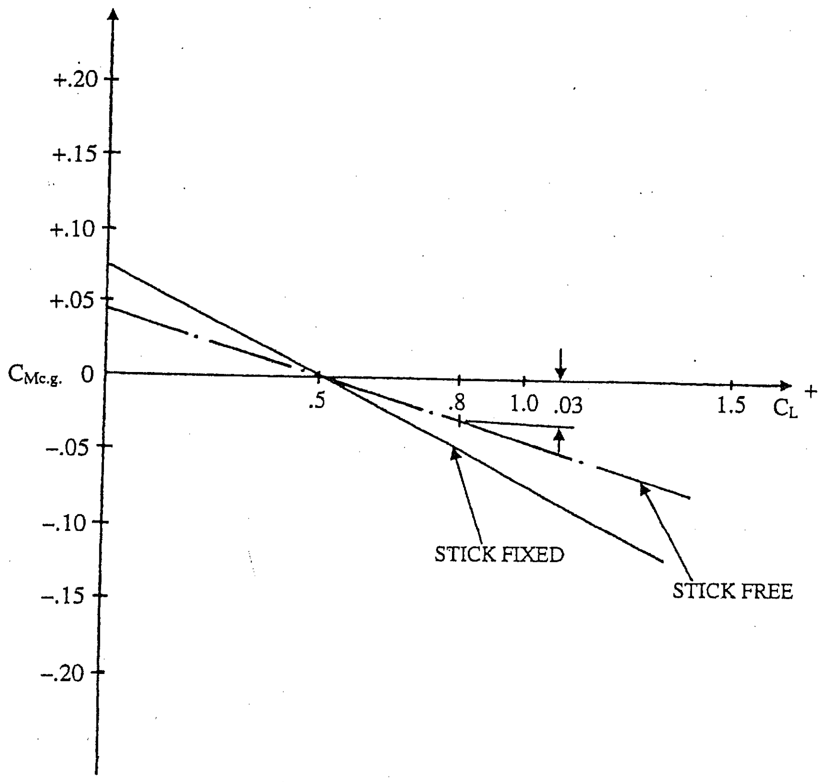


Fig. 20.8 $C_{m_{c.g.}}$ **vs** C_L **stick fixed and stick free.**[1]

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

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

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

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

References

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

[2]Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

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

[4]Staff, USAF TPS, "Stability and Control Flight Test Theory, Vol. 1," AFFTC-TIH-77-1, Feb. 1977.

21
Static Longitudinal Stability Flight Test Methods

21.1 Introduction

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

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

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

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

21.2 Federal Aviation Administration Regulations

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

21.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

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

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

21.2.2 Federal Air Regulations Part 23 (Ref. 2)

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

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

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

21.3 Stick-Fixed Neutral Point Determination

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

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

$$\frac{d\delta_e}{dC_L} = \frac{(dC_m/dC_L)_{\text{fixed}}}{C_{m_{\delta_e}}} \tag{21.1}$$

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

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

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

21.4 Stick-Free Neutral Point Determination

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

$$\frac{\mathrm{d}F_s}{\mathrm{d}V_e} = 2K\frac{W}{S_W}\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}}\left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{free}\frac{V_e}{V^2_{e_{trim}}} \tag{21.2}$$

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

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

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

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

where

$A = -KS_eC_e$

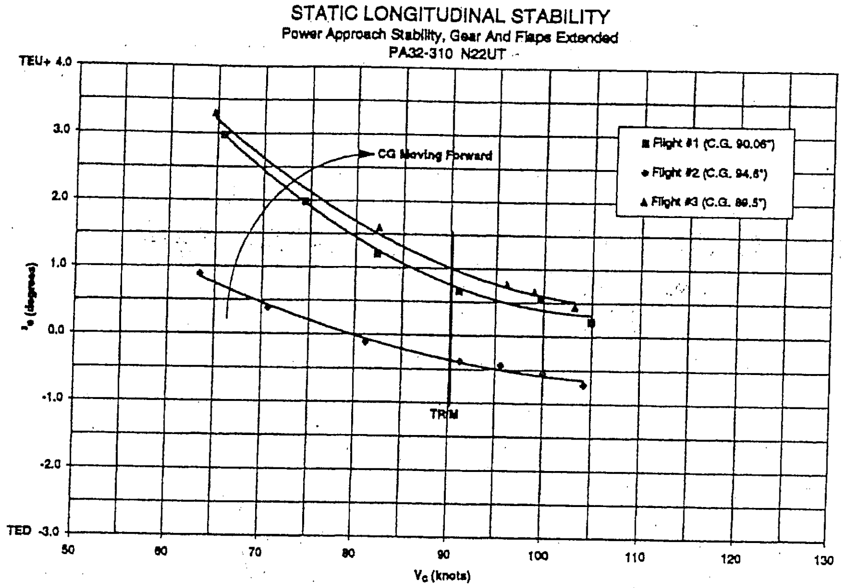
S_e = elevator area

C_e = elevator MAC

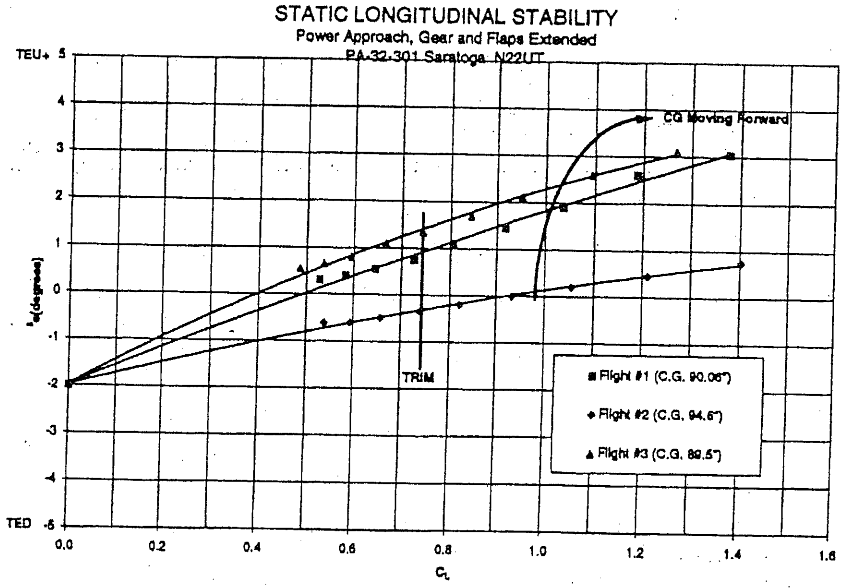
K = control system gearing constant

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

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

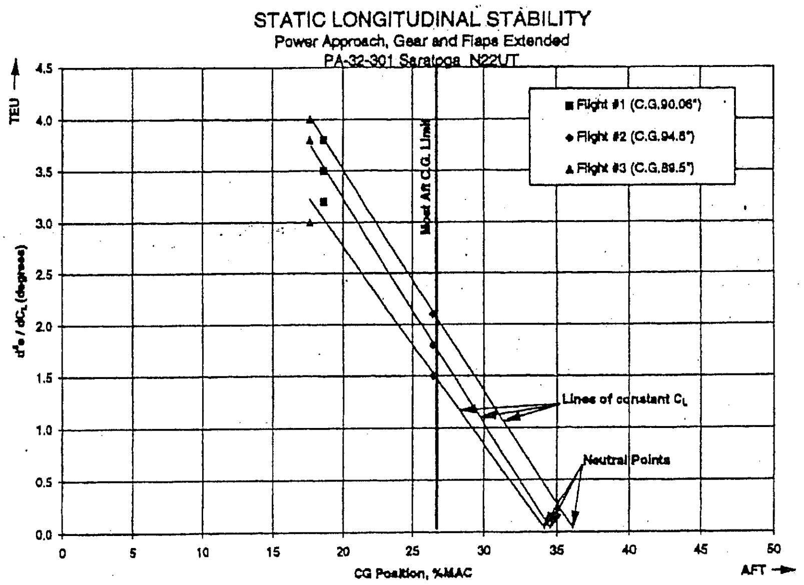


Step 1. Plot data of elevator position (δ_e) vs calibrated airspeed (V_C) for each flight at different c.g. positions and fair a smooth curve through the data. Mark the trim airspeed on the plot.

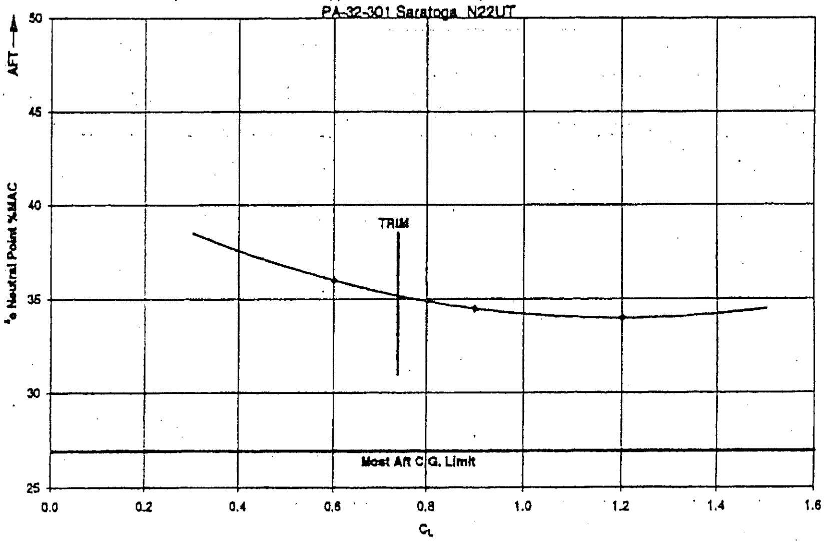


Step 2. From the smooth curves of Step 1 plot elevator position δ_e vs lift coefficient C_L. Select airspeed at which the flight test data were obtained to calculate C_L but obtain the δ_e from the faired lines. Note that these curves all ray from the δ_e for $C_L = 0$.

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

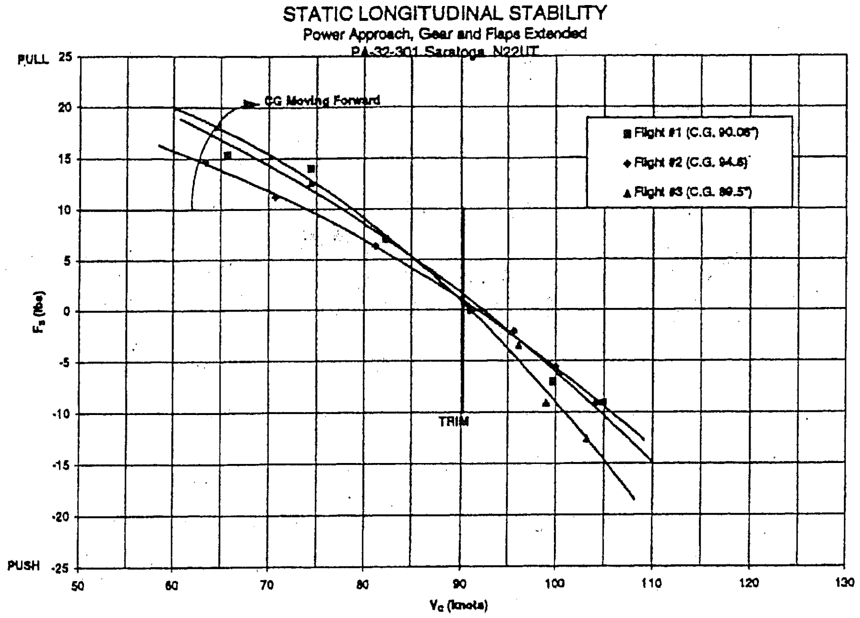


Step 3. Take slopes $d\delta_e/dC_L$ at even increments of C_L from each of the curves and plot c.g. position. Fair curves through the points for each respective C_L and extrapolate to zero. This is the c.g. position of the neutral point for that lift coefficient.

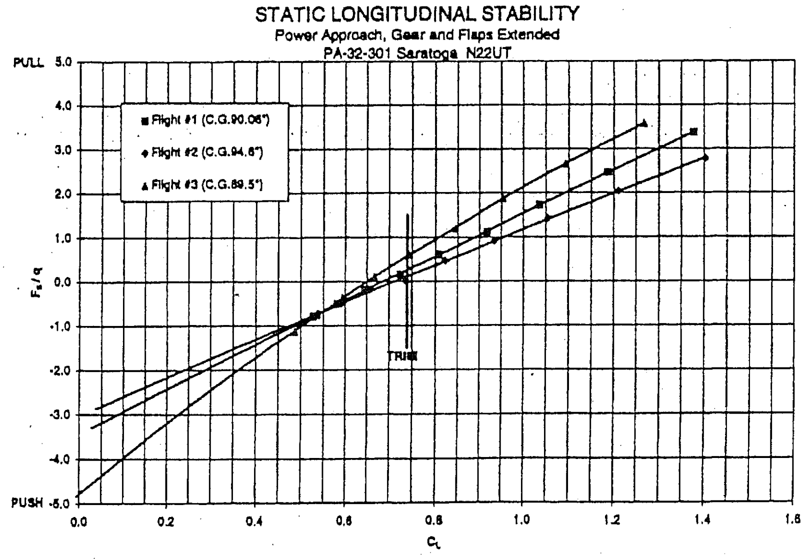


Step 4. Plot the locus of neutral points for each C_L vs C_L and compare with the desired most aft c.g. position. Mark trim C_L neutral point since this is most important neutral point.

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

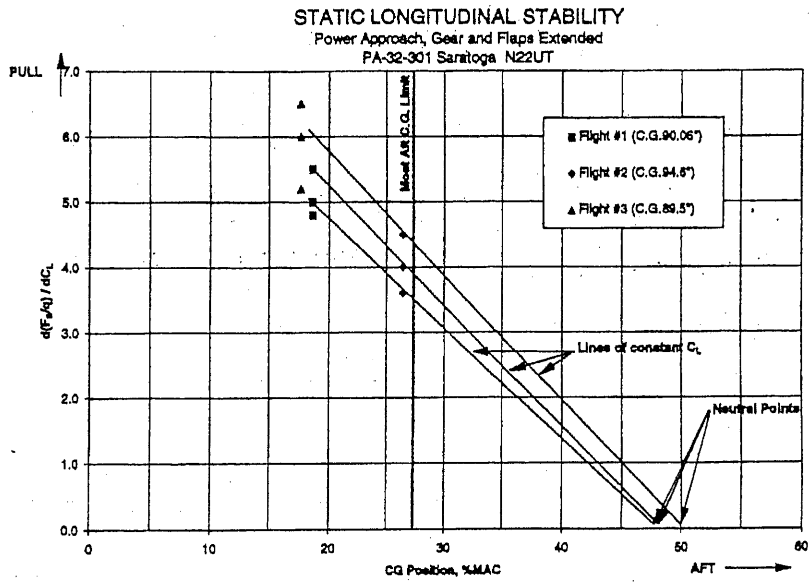


Step 1. Plot elevator control force F_S vs calibrated airspeed V_C for each c.g. position tested and fair a smooth curve through the data points. Mark the trim airspeed on the plot. For FAA testing this is all that is required by the regulations.

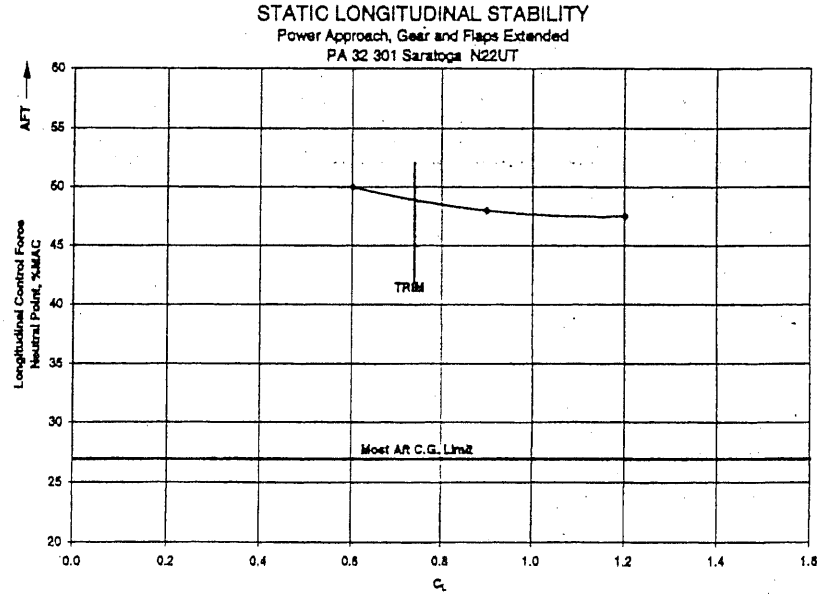


Step 2. Using even increments of airspeed obtain the F_S from the previous plot using the faired lines and not the data points and plot F_s/q vs C_L for each c.g. position tested. These lines should cross at or near the trim C_L.

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



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



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

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

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

21.5 Flight Test Method for Determination of Neutral Points

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

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

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

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

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

21.6 Other Static Longitudinal Stability Tests

21.6.1 Speed Stability[7]

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

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

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

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

21.6.2 Flight Path Stability[5,7]

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

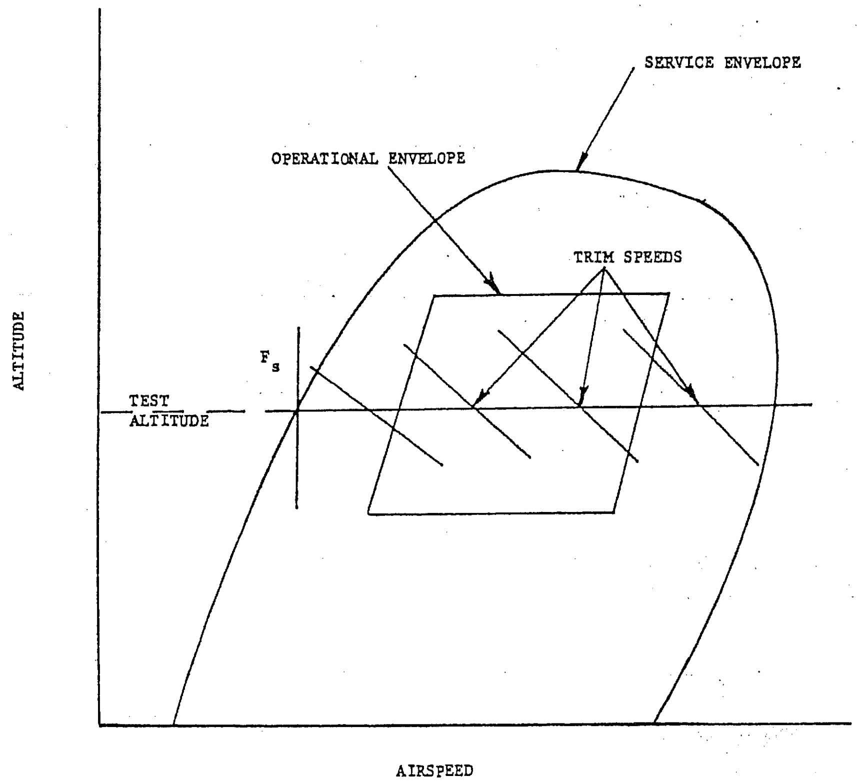


Fig. 21.3 Speed stability measurement.[7]

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

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

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

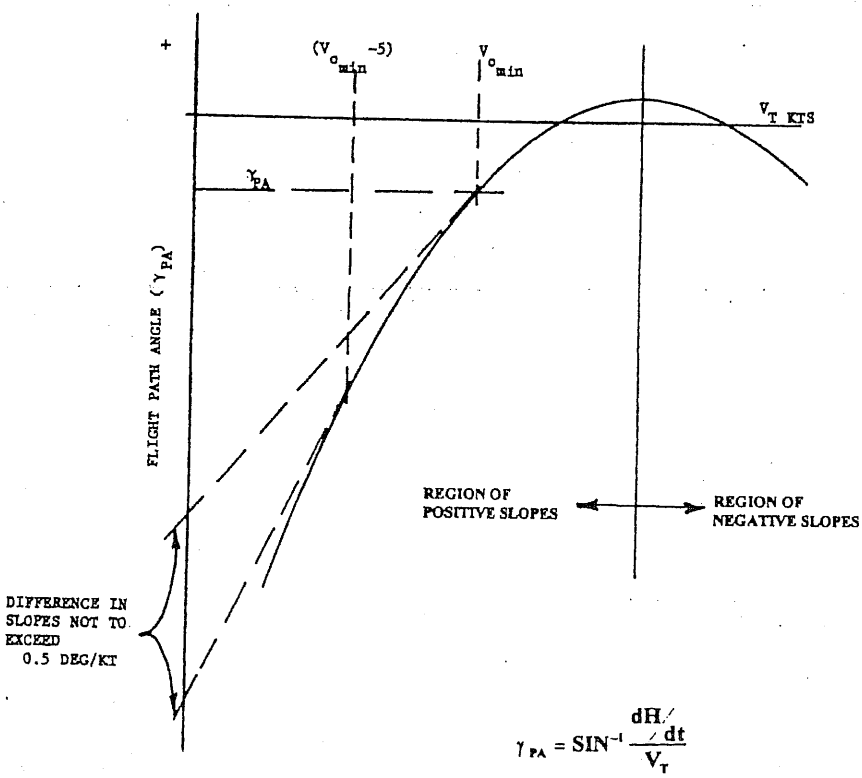


Fig. 21.4 Flight path stability measurement.[5]

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

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

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[3]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

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

[5]Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM No. 103, Jan. 1975, rev. 1 Aug. 1977.

[6]Staff, USAFTPS, "Stability and Control Flight Test Theory, Vol. 1," AFFTC-TIH-77-1, Feb. 1977.

[7]Staff, USAFTPS, "Stability and Control Flight Test Techniques, Vol. II," AFFTC-TIH-77-1, Feb. 1977.

[8]Allison, Rodney, "Static Longitudinal Stability Flight Test Report," University of Tennessee Space Institute, WO4, Tullahoma, TN, Feb. 1997.

22 Dynamic Longitudinal Stability Theory

22.1 Introduction

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

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

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

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

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

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

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

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

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

22.2 Theory

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

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

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

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

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

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

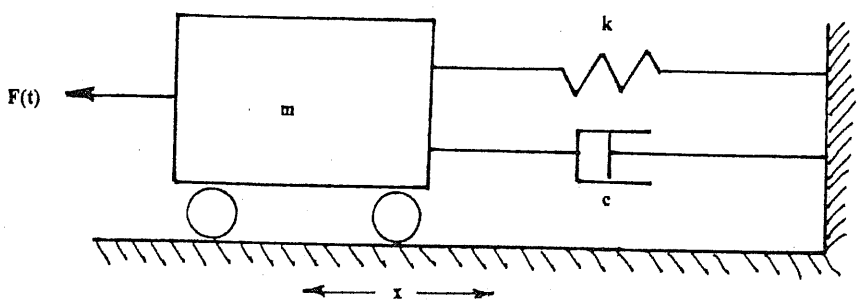


Fig. 22.1 Spring-mass-damper system.

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

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

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

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

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

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

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

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

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

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

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

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

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

22.3 Long Period or Phugoid

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

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

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

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

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

where

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

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

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

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

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

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

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

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

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

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

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

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

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

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

1) Natural frequency ω_p

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

2) Period P_p

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

where
u_o is in ft/s

3) Damping ratio ζ_p

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

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

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

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

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

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

22.4 Short Period

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

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

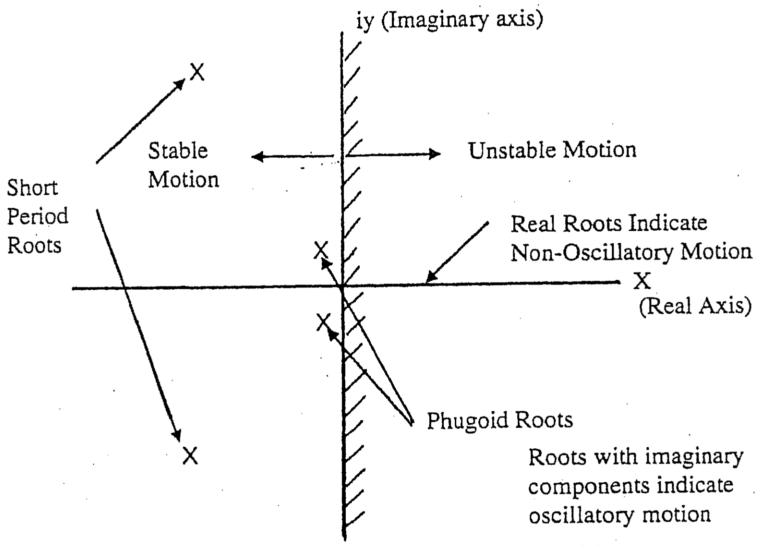


Fig. 22.2 Root locus plot for dynamic longitudinal stability.[1]

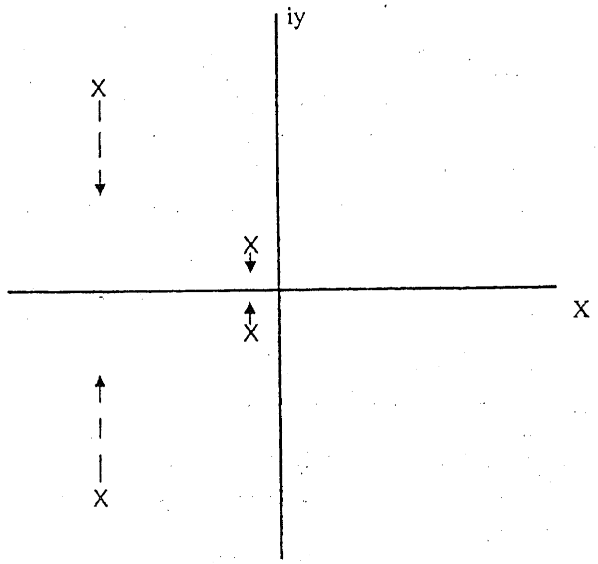


Fig. 22.3 Effects of aft movement of the c.g.[1]

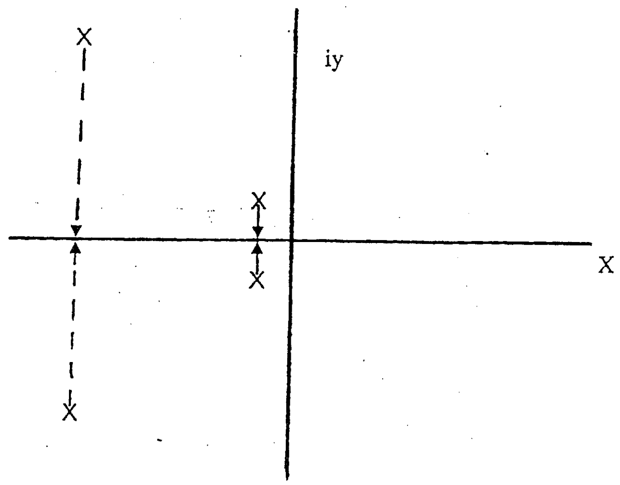


Fig. 22.4 Longitudinal dynamic stability roots when the c.g. is just ahead of the neutral point.[1]

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

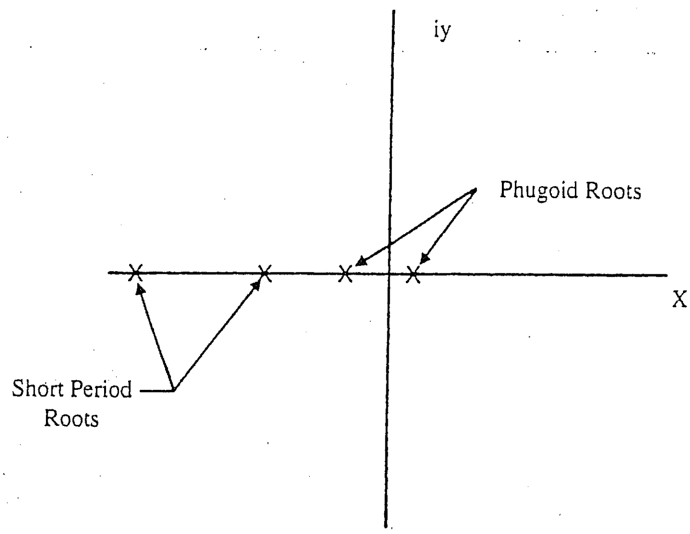


Fig. 22.5 Effects upon dynamic longitudinal stability roots when the c.g. is moved aft of neutral point.[1]

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

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

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

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

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

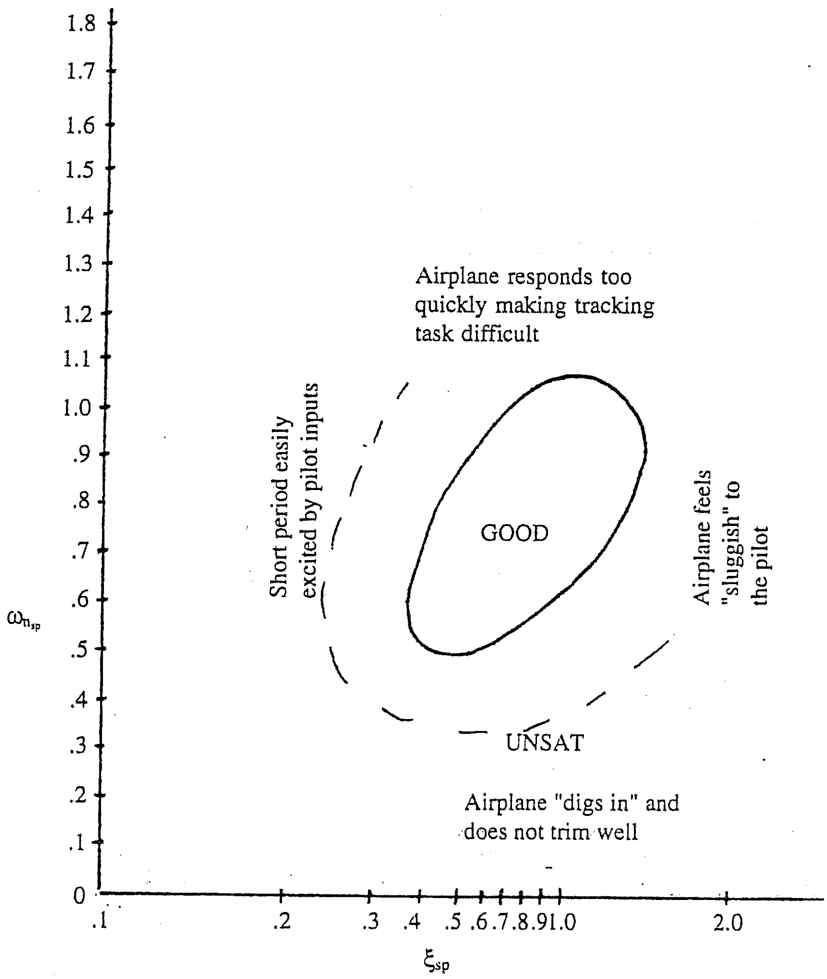


Fig. 22.6 Short period thumbprint.[1]

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

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

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

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

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

$$\omega_{n_{SP}} = \sqrt{\frac{\gamma/2P_a M_2}{I_{yy}} S\bar{c}C_{L_\alpha}\left(\frac{X_{cg}}{\bar{c}} - N_m\right)} \tag{22.12}$$

where

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

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

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

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

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

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

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

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

where
W = aircraft weight

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

22.5 Elevator Short Period

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

References

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

[2]Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103. 1 Jan. 1975, rev. 1 Aug. 1977.

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

[4]Staff, USAFTPS, "Stability and Control Flight Test Theory, Vol. 1," AFFTC-TIH-77-1, Feb. 1977.

23 Dynamic Longitudinal Stability Flight Test Methods and Data Reduction

23.1 Introduction

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

23.2 Federal Aviation Administration Regulations

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

23.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

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

23.2.2 Federal Aviation Regulations Part 23 (Ref. 2)

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

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

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

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

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

23.3 Flight Test Methods for Evaluating the Phugoid

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

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

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

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

23.4 Phugoid Data Reduction

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

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

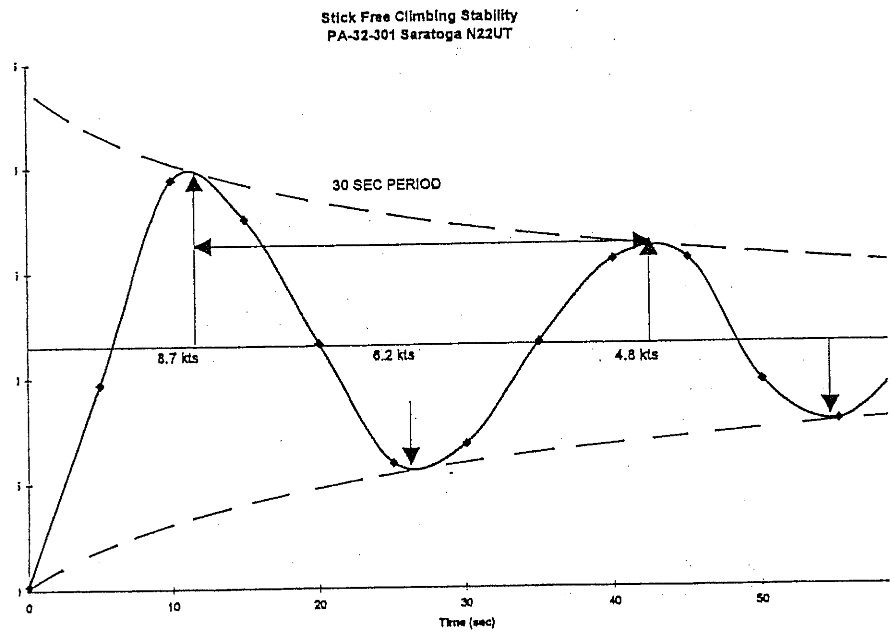


Fig. 23.1 Plot of climb phugoid airspeed vs time.

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

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

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

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

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

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

23.5 Short Period Flight Test Methods

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

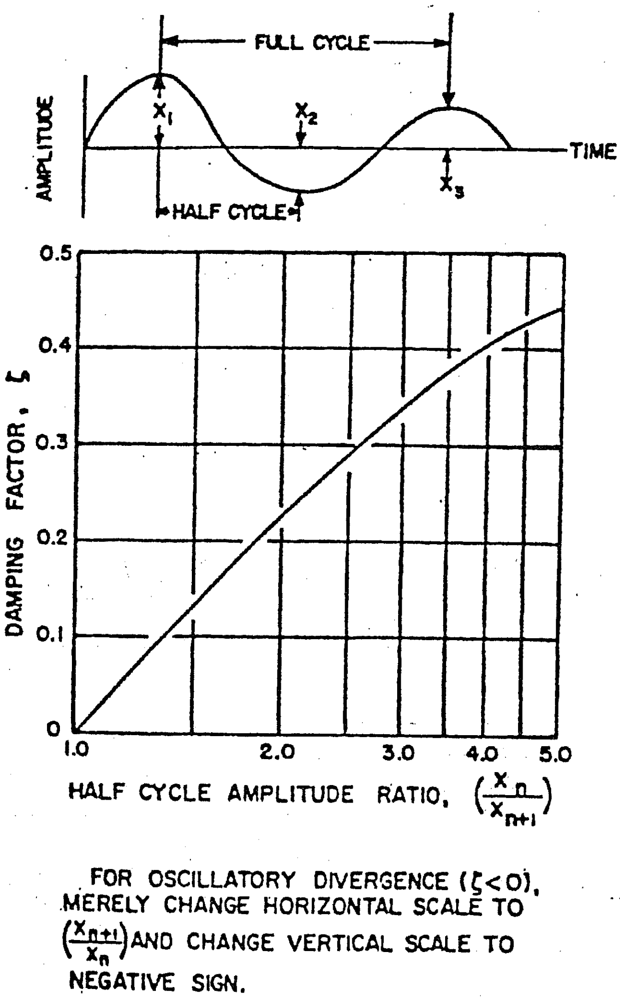


Fig. 23.2 Determination of damping ratio for lightly damped system.[4]

23.5.1 Airplane Short Period[3-5,7]

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

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

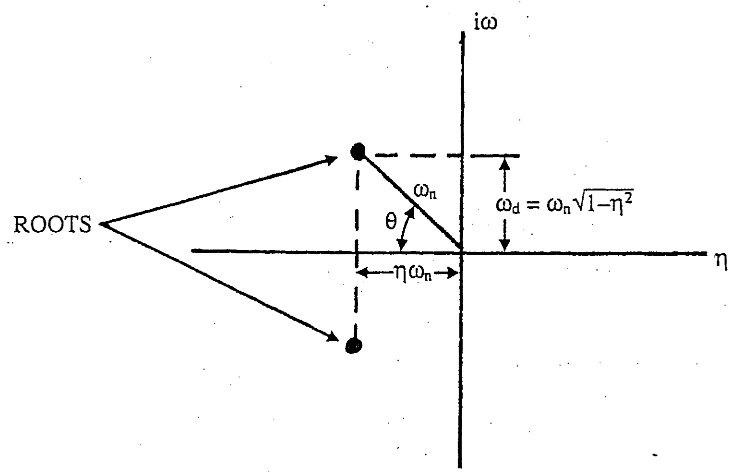


Fig. 23.3 Root locus plot.

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

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

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

23.5.2 Elevator Short Period

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

23.6 Short Period Data Reduction

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

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

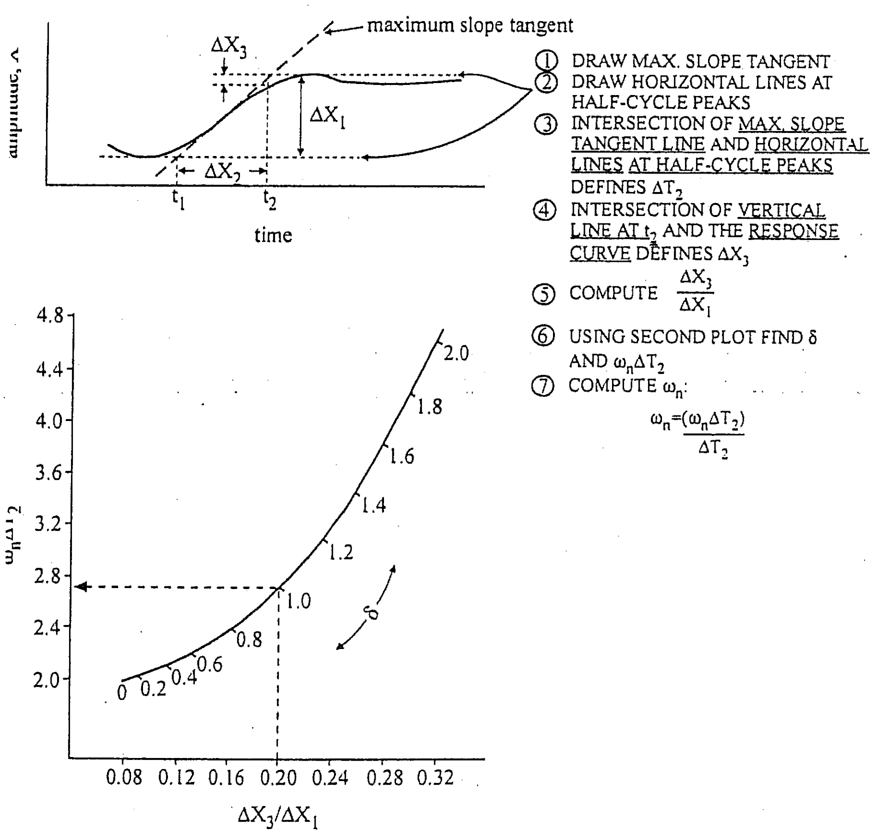


Fig. 23.4 Determination of second order response characteristics for a heavily damped system.[4]

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

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[3]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

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

[5]Langdon, S. D., and Cross, W. V., "Fixed Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

[6]Staff, USAFTPS, "Stability and Control Flight Test Theory, Vol. 1," AFFTC-TIH-77-1, Feb. 1977.

[7]Staff, USAFTPS, "Stability and Control Flight Test Techniques, Vol. II," AFFTC-TIH-77-1, Feb. 1977.

24
Longitudinal Maneuvering Stability Theory

24.1 Introduction

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

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

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

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

Steady pull-up

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

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

Steady level turn

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

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

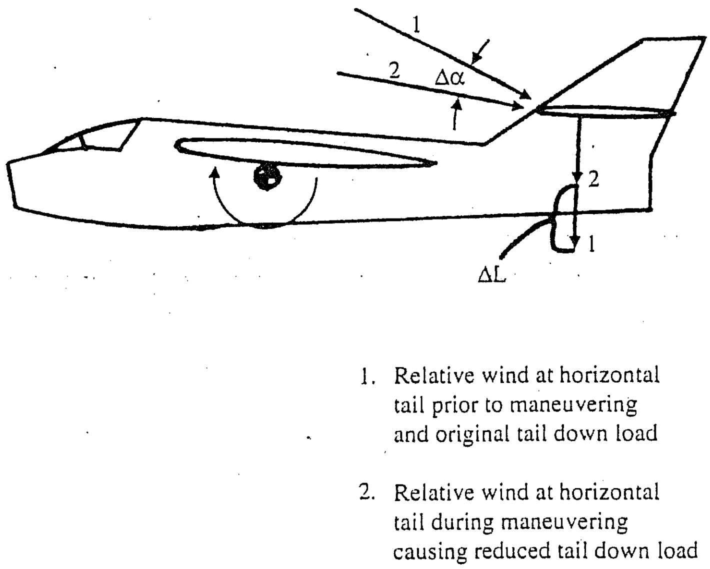


Fig. 24.1 Change in relative wind at the horizontal tail due to maneuvering.[3]

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

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

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

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

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

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

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

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

24.2 Elevator Position Maneuvering Stability

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

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

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

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

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

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

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

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

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

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

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

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

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

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

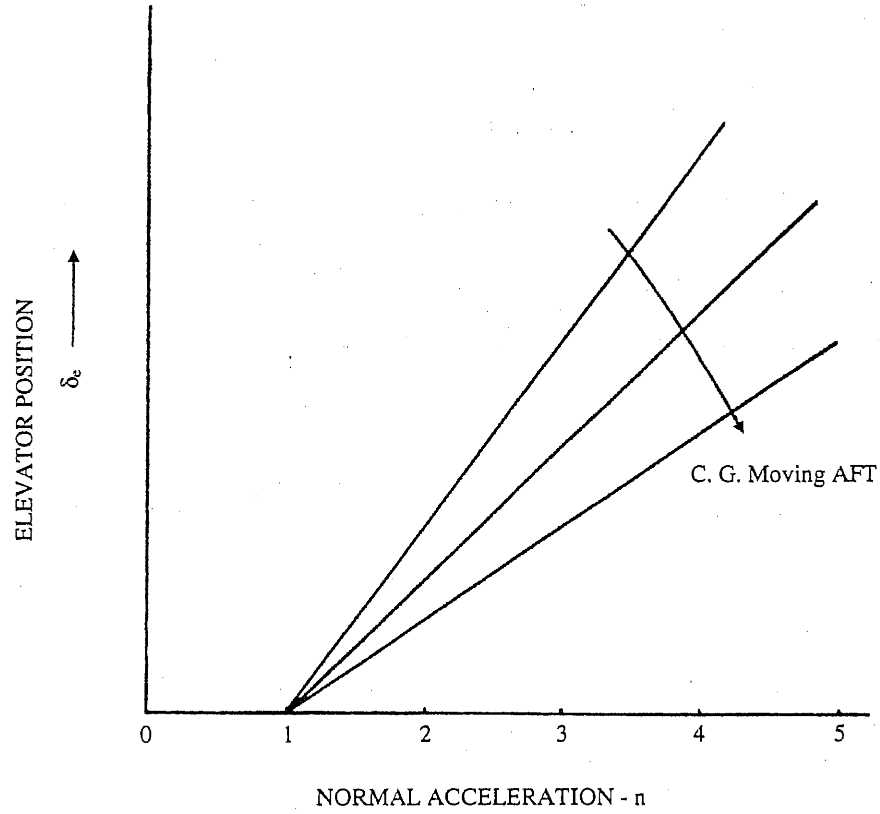


Fig. 24.2 Effects of c.g. movement on $d\delta_e/dn$ (Ref. 1).

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

24.3 Stick-Force Maneuvering Stability

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

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

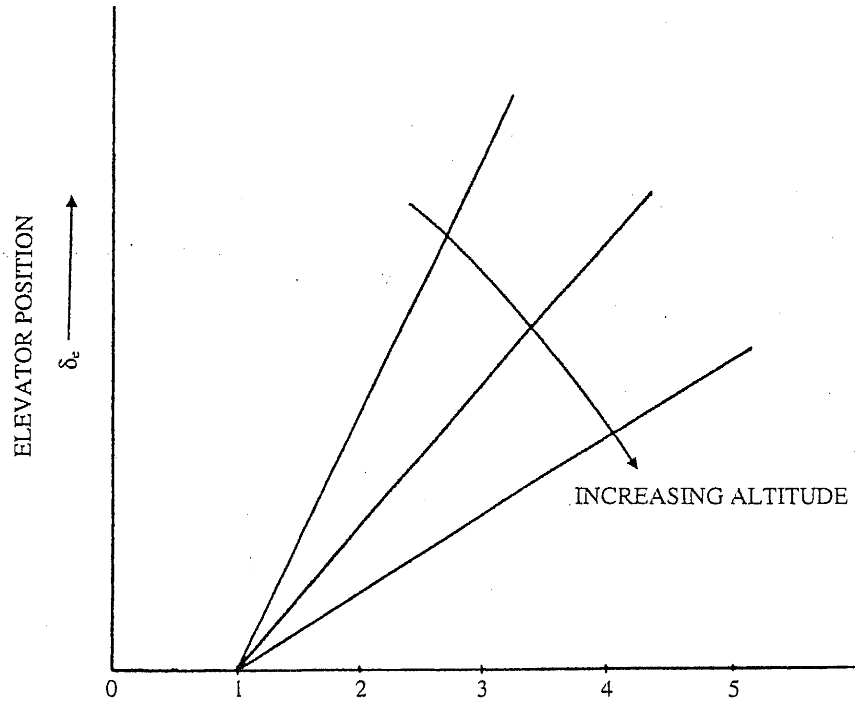


Fig. 24.3 Effects of altitude on $d\delta_e/dn$ at a constant equivalent airspeed.[1]

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

24.3.1 Stick-Force Maneuvering Stability—Reversible Control System

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

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

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

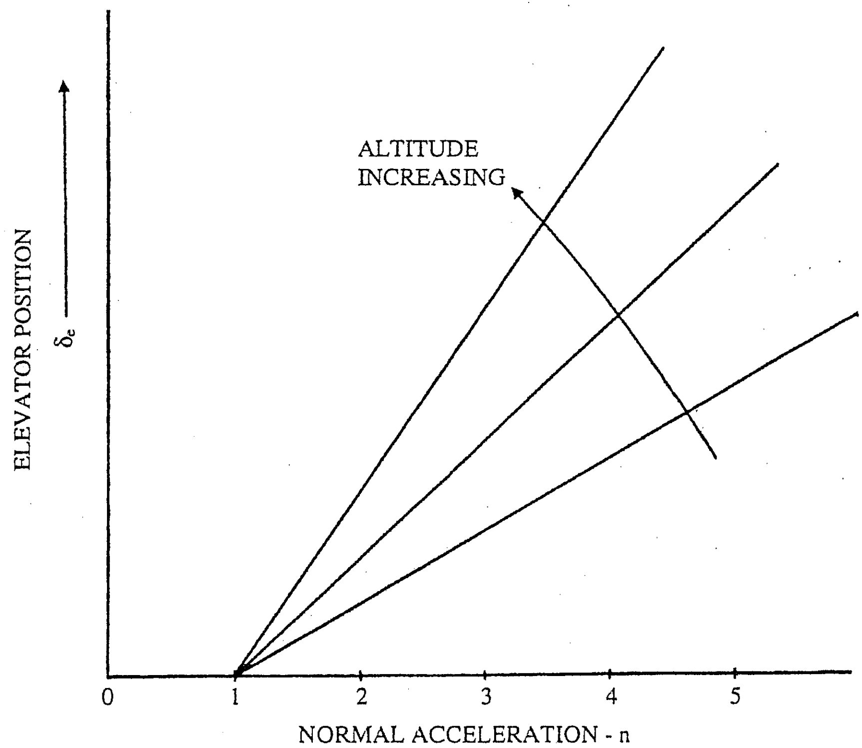


Fig. 24.4 Effects of altitude on $d\delta_e dn$ at a constant Mach number.[1]

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

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

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

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

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

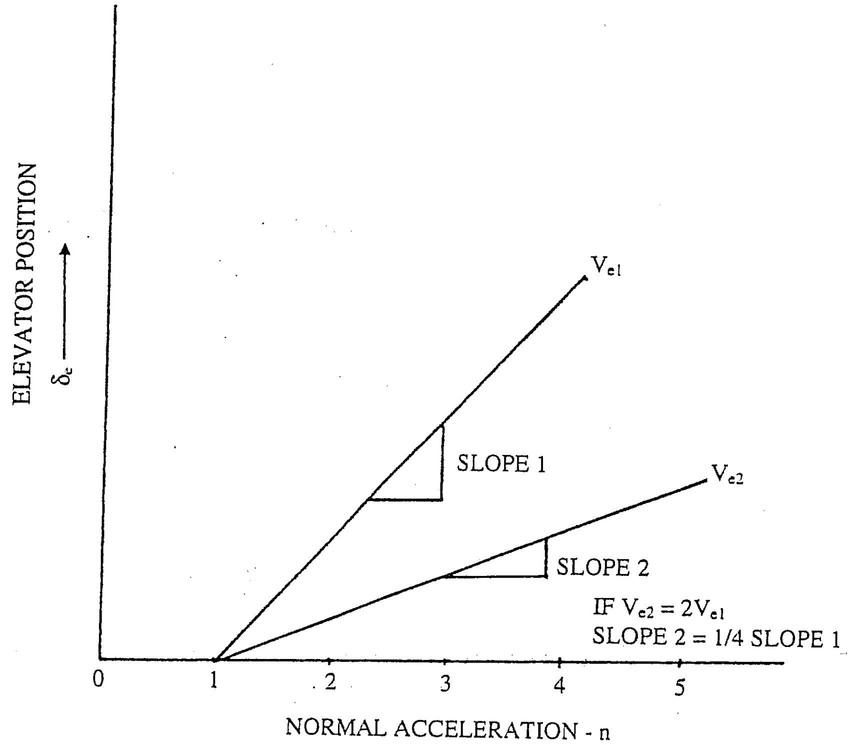


Fig. 24.5 Effects of varying equivalent airspeed on $d\delta_e/dn$ (Ref. 1).

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

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

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

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

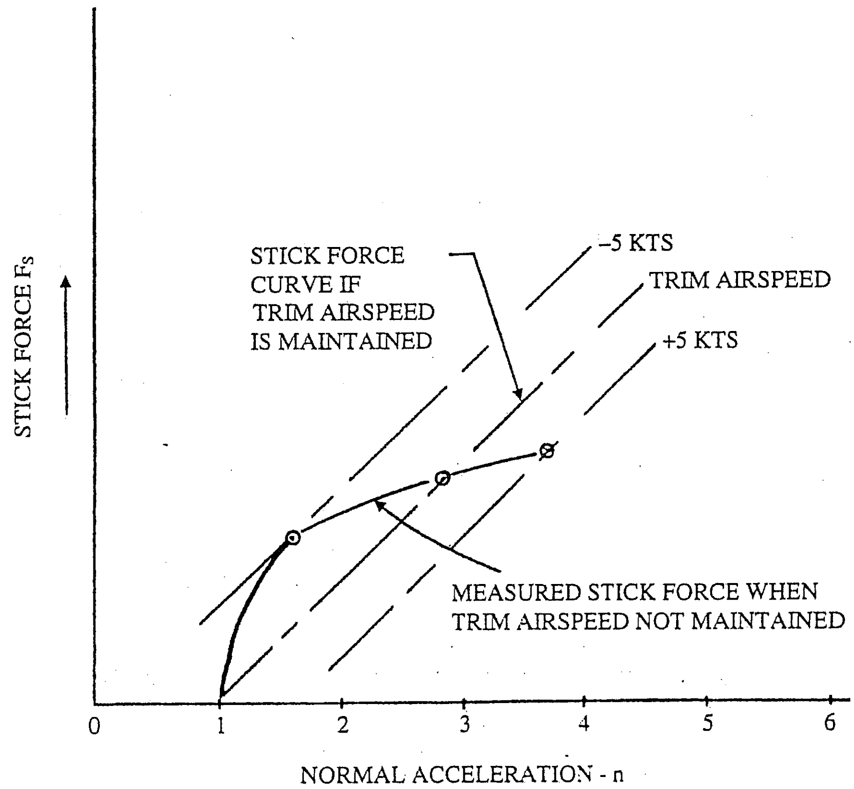


Fig. 24.6 Effects of not maintaining trim airspeed while collecting F_s vs n data.[1]

24.3.2 Stick-Force Maneuvering Stability—Irreversible Control Systems

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

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

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

where

K_1 = a constant that describes the characteristics of the system

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

$$\left(\frac{\mathrm{d}F_s}{\mathrm{d}n}\right)_{ST} = -\frac{K_1}{C_{m_{\delta_e}}} \frac{W/S}{1/2\rho_0 V_{e_{trim}}^2} \left\{ \left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{fixed} + \frac{C_{m_{\dot\theta}} \rho g \bar{c}}{4(W/S)}(1 - 1/n^2) \right\} \tag{24.14}$$

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

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

$$F_s = K_2 q \Delta\delta_e \tag{24.15}$$

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

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

$$\left(\frac{\mathrm{d}F_s}{\mathrm{d}n}\right)_{ST} = -\frac{K_2(W/S)}{C_{m_{\delta_e}}} \left\{ \left(\frac{\mathrm{d}C_m}{\mathrm{d}C_L}\right)_{fixed} + \frac{C_{m_{\dot\theta}} \rho g \bar{c}}{4(W/S)}(1 - 1/n^2) \right\} \tag{24.16}$$

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

24.4 Compressibility Effects

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

References

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

[2]Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

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

25 Maneuvering Stability Methods and Data Reduction

25.1 Introduction

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

25.2 Federal Aviation Administration Regulations

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

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

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

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

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

25.3 Evaluation by Pilot Opinion

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

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

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

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

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

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

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

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

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

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

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

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

25.4 Flight Test Methods for Quantitative Evaluation

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

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

25.4.1 Steady Pull-Ups[3,4]

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

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

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

25.4.2 Steady Pushovers[3]

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

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

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

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

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

25.4.5 Constant g Method[4]

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

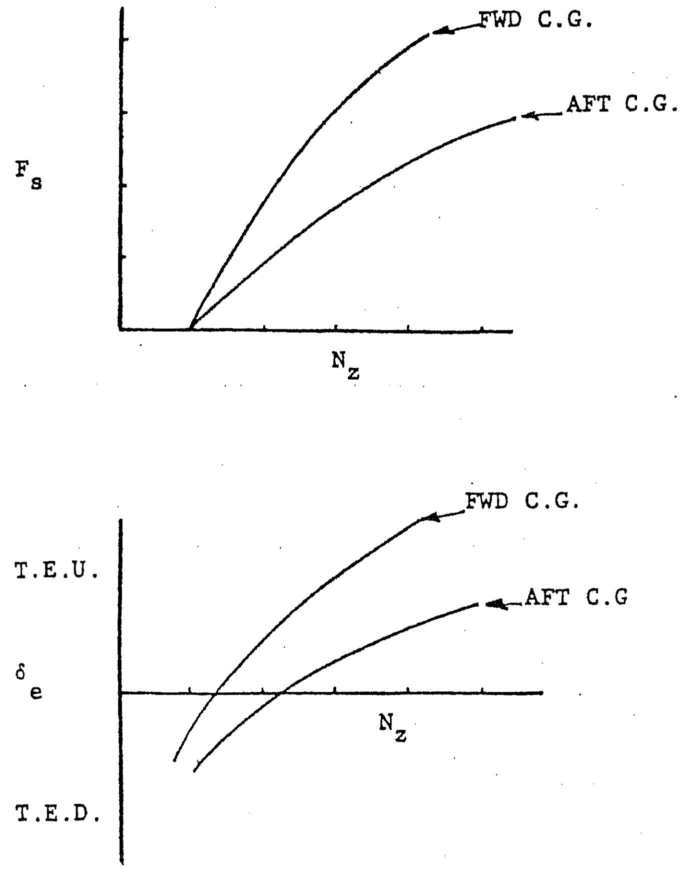


Fig. 25.1 Maneuvering stability data plots.[4]

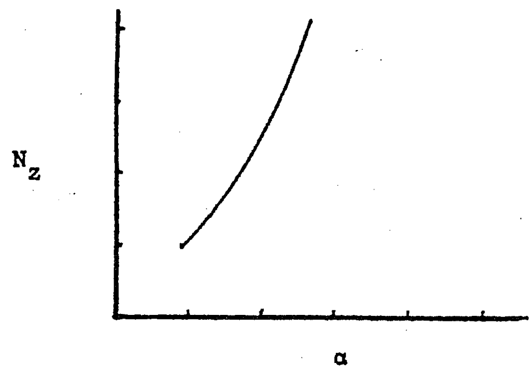


Fig. 25.2 Load factor vs angle of attack.[4]

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

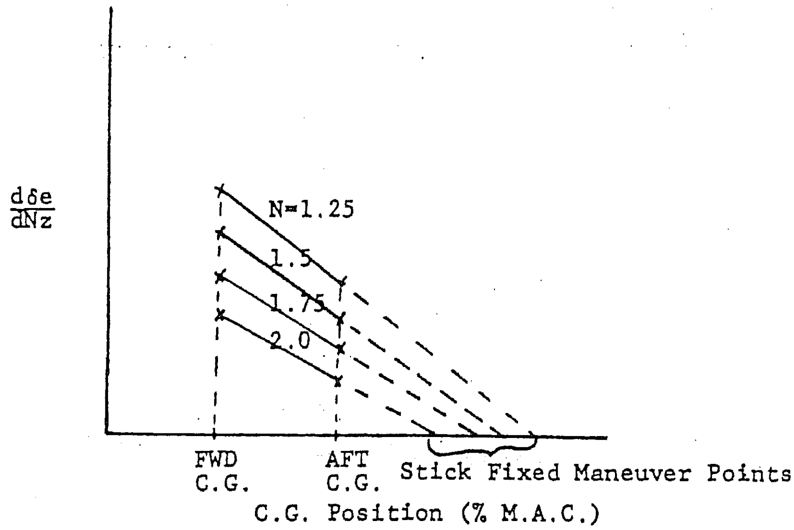


Fig. 25.3 Maneuver point extrapolation.[4]

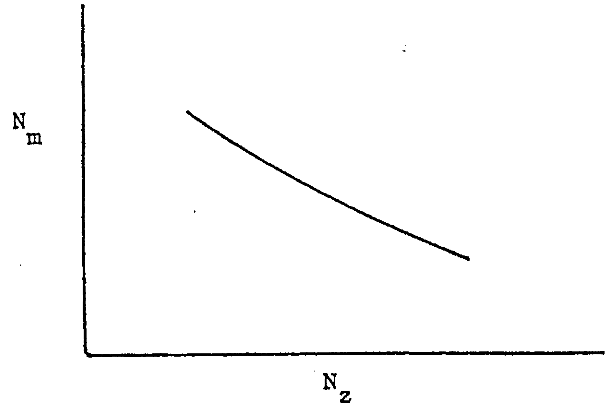


Fig. 25.4 Stick-fixed maneuver points vs load factor.[4]

25.5 Data Reduction Techniques

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

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

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

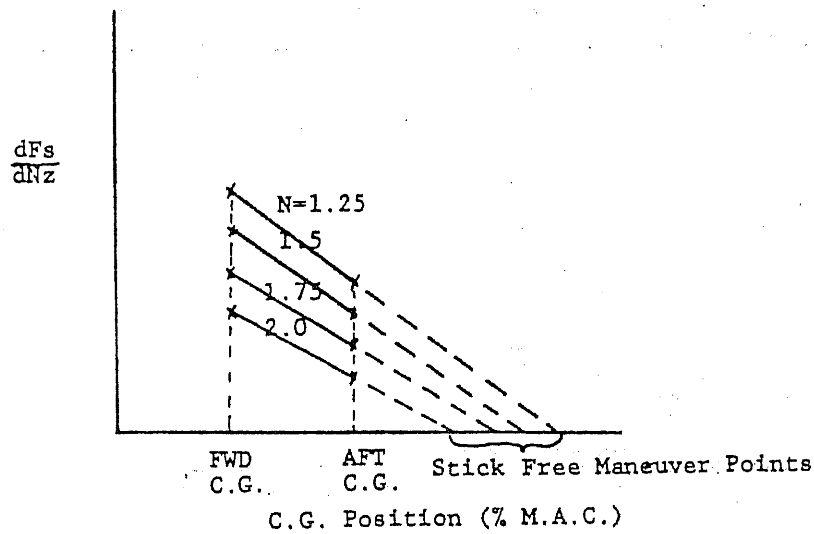


Fig. 25.5 Stick-free maneuver point extrapolation.[4]

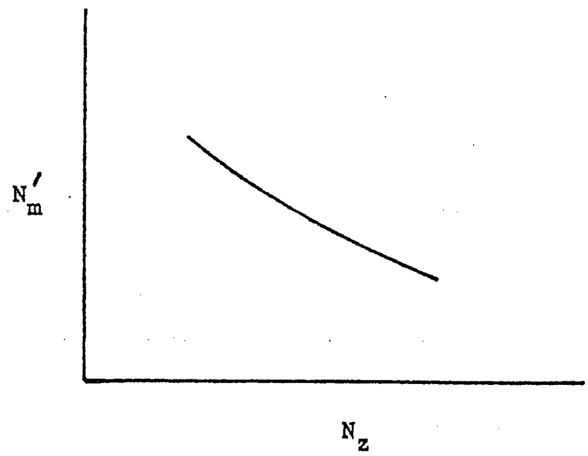


Fig. 25.6 Stick-free maneuver point vs load factor.[6]

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

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

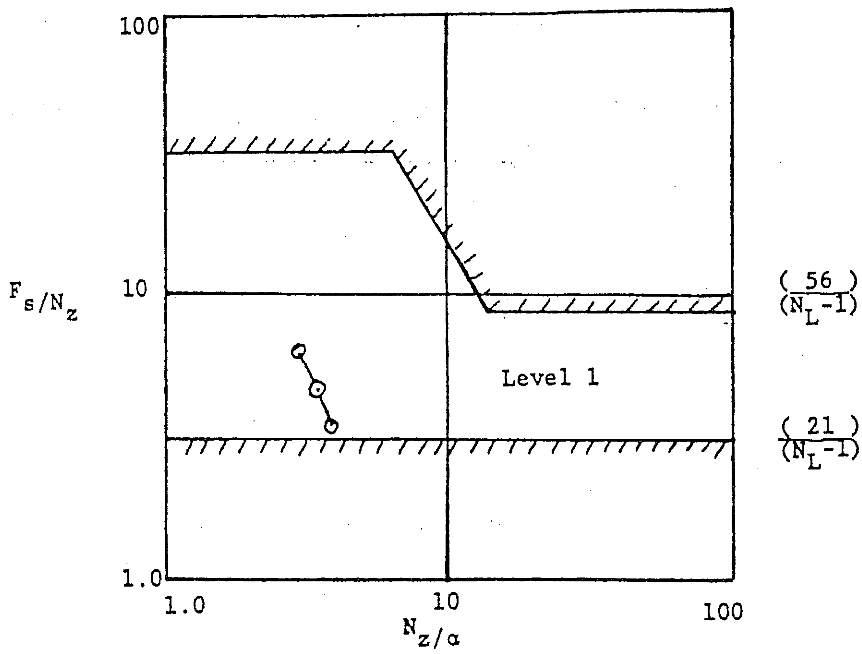


Fig. 25.7 Stick-force per g vs N_Z/α (Ref. 5).

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

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

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

References

[1]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[2]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1989.

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

[4]Staff, USAFTPS, "Stability and Control Flight Test Techniques, Vol. II," AFFTC-TIH-77-1, Feb. 1977.

[5]MIL-F-8785B(ASG), "Flying Qualities of Piloted Airplanes," 7 August 1969 (including Interim Amendment 1 [USAF], dated 31 March 1971).

26
Longitudinal Control and Trim Theory and Flight Test Methods

26.1 Introduction

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

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

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

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

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

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

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

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

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

26.2 Federal Aviation Administration Regulations

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

26.2.1 *Civil Aeronautics Regulation 3 (Ref. 3)*

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

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

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

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

26.2.2 Federal Aviation Regulation Part 23 (Ref. 4)

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

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

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

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

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

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

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

26.3 Longitudinal Control

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

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

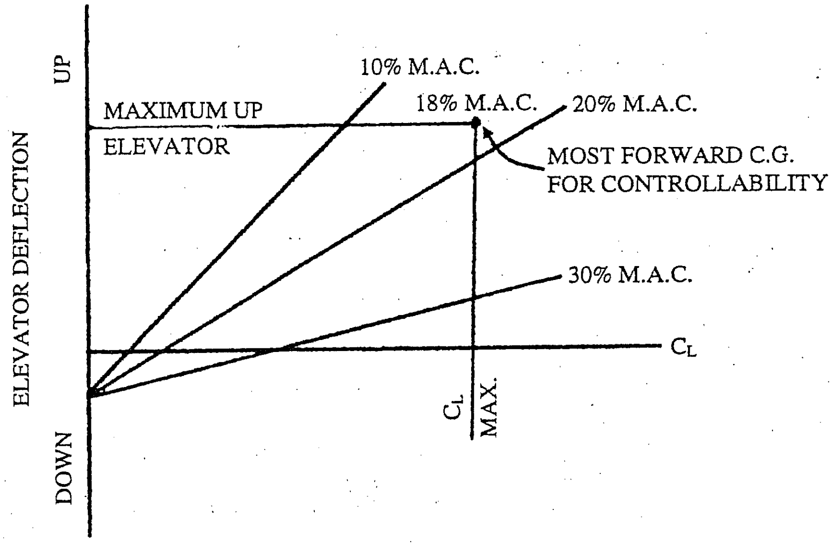


Fig. 26.1 Most forward c.g. determination.[2]

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

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

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

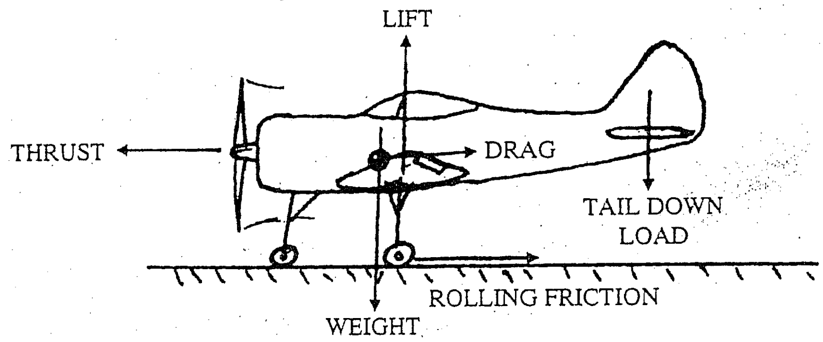


Fig. 26.2 Forces acting on aircraft during takeoff.[2]

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

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

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

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

26.4 Longitudinal Trim

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

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

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

SECTION V, FLIGHT CHARACTERISTICS

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

23.141 3. **Longitudinal Control**
23.145

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

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

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

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

SECTION V, FLIGHT CHARACTERISTICS

B. Most Rearward C.G.

23.145 5. Controllability - Longitudinal:

CONFIGURATION (GEAR EXTENDED EACH CASE)	INFORMATION		DOES IT REQUIRE A CHANGE IN THE TRIM CONTROL OR THE EXERTION OF MORE CONTROL FORCE THAN CAN BE READILY APPLIED WITH ONE HAND FOR A SHORT PERIOD TO.......
	OBSERVED DATA	TRIM SPEED	
A. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) FLAPS RETRACTED ( )	H_p_____FT. OAT_____°F RPM_____	_____V_{S1} _____ -IAS _____ -CAS	EXTEND THE FLAPS AS RAPIDLY AS POSSIBLE TO LANDING POSITION _____° WHILE MAINTAINING APPROX. 40 PERCENT ABOVE THE INSTANTANEOUS STALL SPEED? NO YES F_E_____LBS.
B. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS_____°	H_p_____FT. OAT_____°F RPM_____	_____V_{S0} _____ -IAS _____ -CAS	RETRACT THE FLAPS AS RAPIDLY AS POSSIBLE, WHILE MAINTAINING APPROX. 40 PERCENT ABOVE THE INSTANTANEOUS STALL SPEED? NO YES F_E_____LBS.
C. CLIMB M.C. POWER ( ) GEAR EXTENDED ( ) LANDING FLAPS_____°	H_p_____FT. OAT_____°F RPM_____	_____V_{S0} _____ -IAS _____ -CAS	RETRACT THE FLAPS AS RAPIDLY AS POSSIBLE, WHILE MAINTAINING APPROX. 40 PERCENT ABOVE THE INSTANTANEOUS STALL SPEED? NO YES F_E_____LBS.
D. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) FLAPS RETRACTED ( )	H_p_____FT. OAT_____°F RPM_____	_____V_{S1} _____ -IAS _____ -CAS	APPLY T.O. POWER QUICKLY WHILE MAINTAINING THE SAME SPEED? NO YES F_E_____LBS.
E. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS_____°	H_p_____FT. OAT_____°F RPM_____	_____V_{S0} _____ -IAS _____ -CAS	APPLY T.O. POWER QUICKLY WHILE MAINTAINING THE SAME SPEED? NO YES F_E_____LBS.
F. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS_____°	H_p_____FT. OAT_____°F RPM_____	_____V_{S0} _____ -IAS _____ - CAS	OBTAIN AND MAINTAIN SPEED AT $1.1V_{S1}$? _____ -IAS _____ CAS NO YES F_E_____LBS.
G. GLIDE THROTTLES CLOSED ( ) GEAR EXTENDED ( ) LANDING FLAPS_____°	H_p_____FT. OAT_____°F RPM_____	_____V_{S0} _____ -IAS _____ -CAS	OBTAIN AND MAINTAIN SPEED AT THE LOWER OF $1.7V_{S1}$ OR V_F NO YES F_E_____LBS.

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

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

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

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

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

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

References

[1]Langdon, S. D., "Fixed Wing Stability and Control—Theory and Flight Test Techniques," USNTPS-FTM-103, Patuxent River, MD, 1 Aug. 1969.

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

[3]Civil Aeronautics Manual 3, "Airplane Airworthiness: Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[4]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[5]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

[6]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual Vol. II—Stability and Control*, Pergamon Press, New York, 1959.

[7]Federal Aviation Administration Form 8110-18, "Flight Test Report Guide, FAR 23 Aircraft," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1975.

[8]Federal Aviation Regulation Part 25, "Airworthiness Standards: Transport Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1975.

[9]Bunker, R. S., Maj., Hendrix, G. D., Maj., et al., "Stability and Control," AFFTC-TIH-64-2004, Air Force Flight Test Center, Edwards AFB, CA, rev. Nov. 1964.

27
Methods for Improving Longitudinal Stability and Control

27.1 Introduction

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

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

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

27.2 Control System Gadgets

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

27.2.1 Stick Centering Springs[1–4]

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

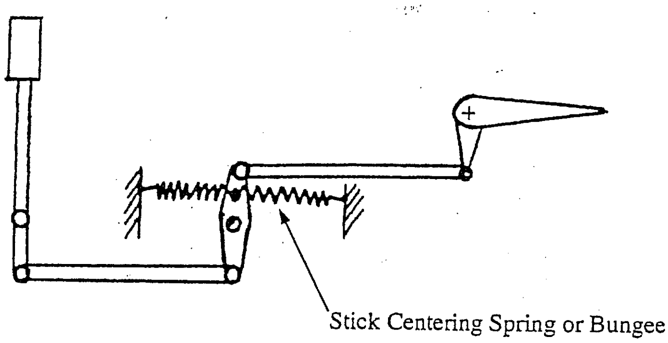


Fig. 27.1 Stick centering spring.[3]

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

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

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

27.2.2 Elevator Down-Spring[1–4]

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

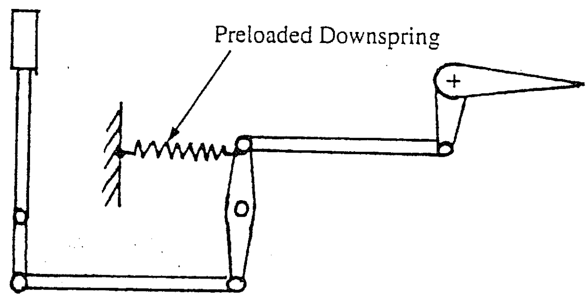


Fig. 27.2 Elevator down-spring.[3]

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

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

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

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

27.2.3 Bobweight[1–4]

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

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

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

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

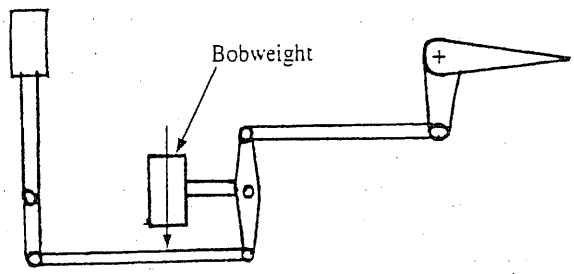


Fig. 27.3 Elevator bobweight.[3]

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

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

27.2.4 Spring-Weight Combinations

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

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

27.3 Elevator Tabs

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

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

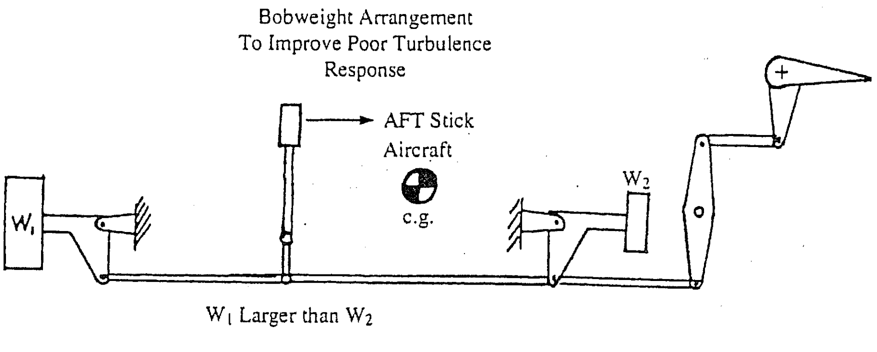


Fig. 27.4 Bobweight arrangement to improve turbulence response.[3]

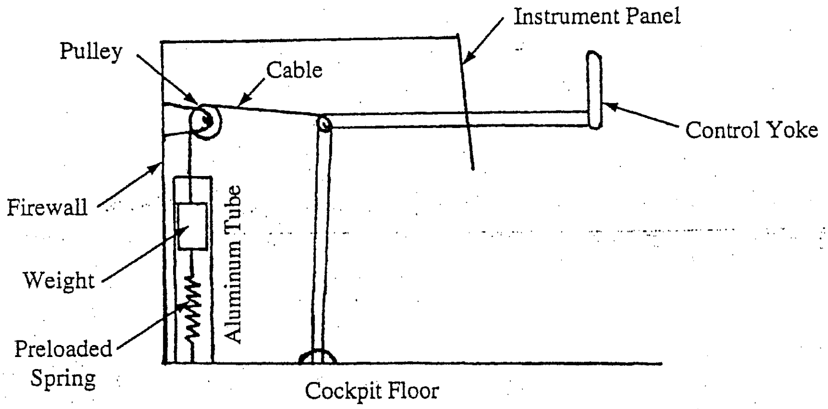


Fig. 27.5 Down-spring/bobweight combination.

27.3.1 Balance Tab[1–4]

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

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

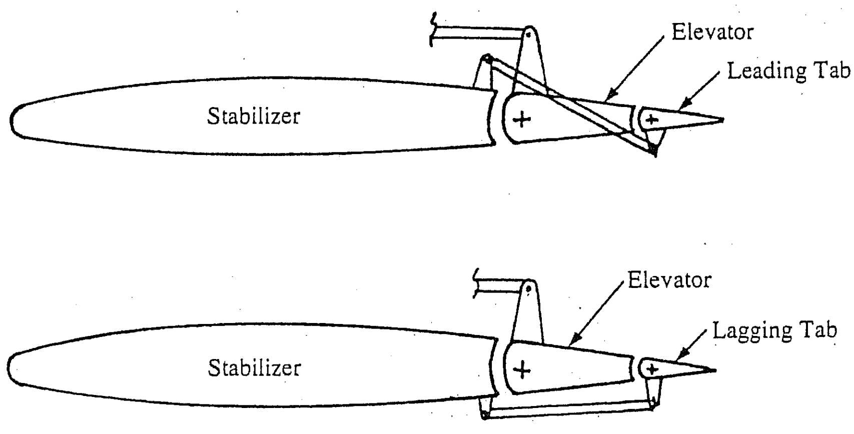


Fig. 27.6 Leading and lagging balance tabs.[3]

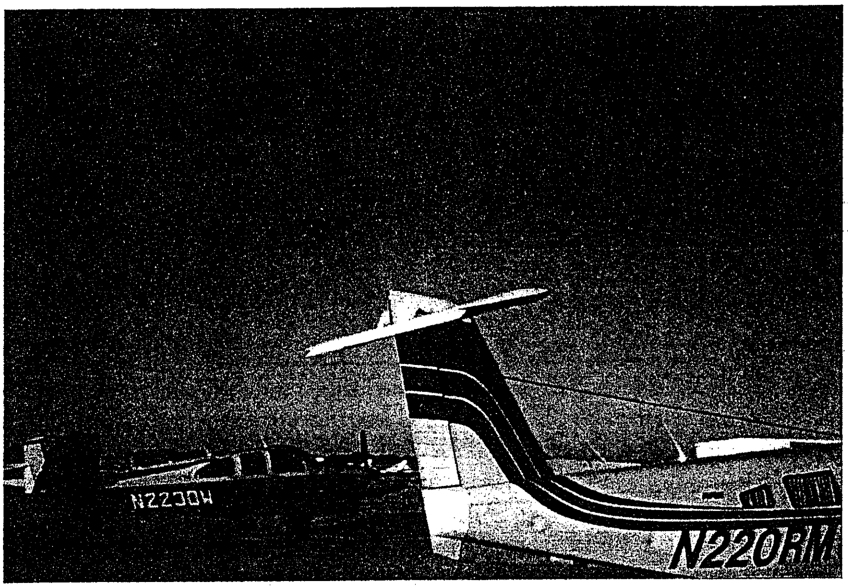


Fig. 27.7 Piper T-tail Lance with leading balance tab.

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

27.3.2 Servo Tab[1–4]

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

27.3.3 Spring Tab[1–4]

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

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

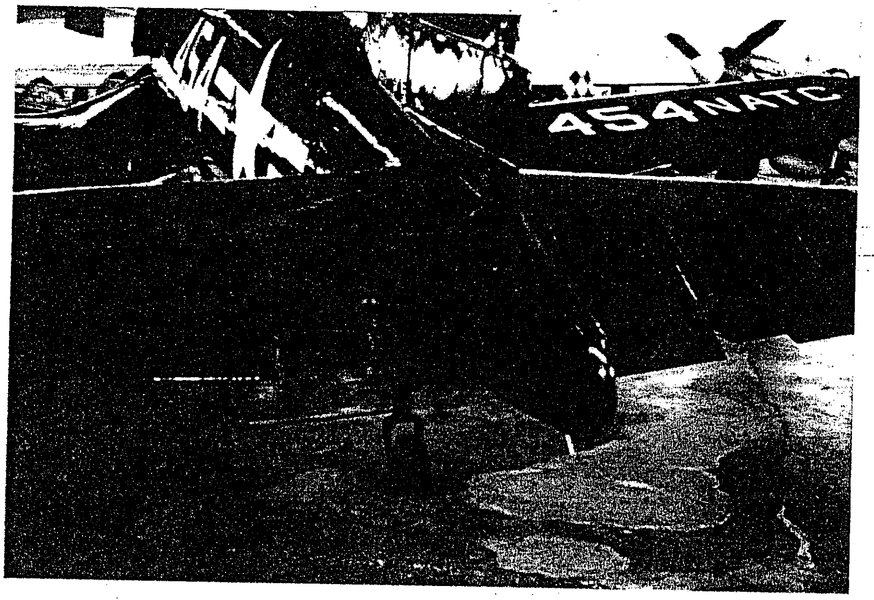


Fig. 27.8 F4U Corsair horizontal tail with leading balance tab.

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

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

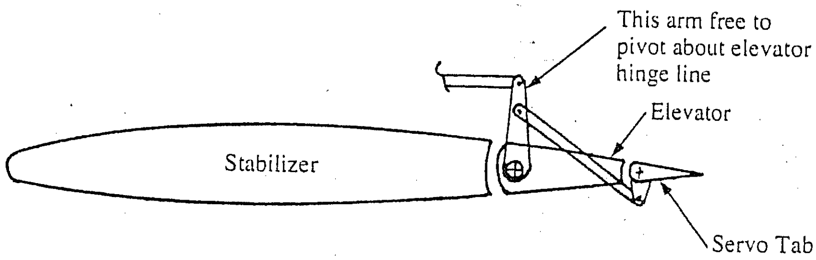


Fig. 27.9 The servo tab.[3]

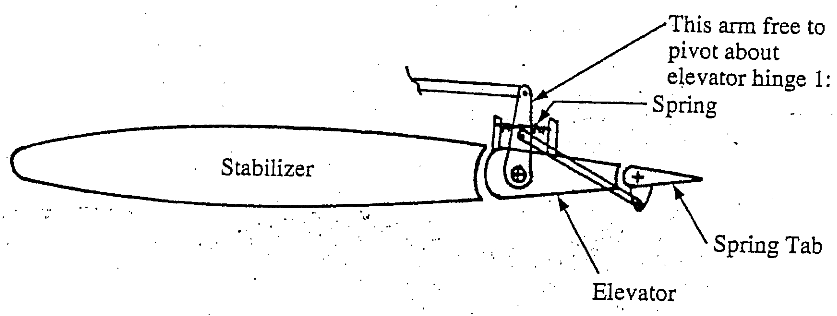


Fig. 27.10 The spring tab.[3]

27.4 Aerodynamic Balance

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

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

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

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

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

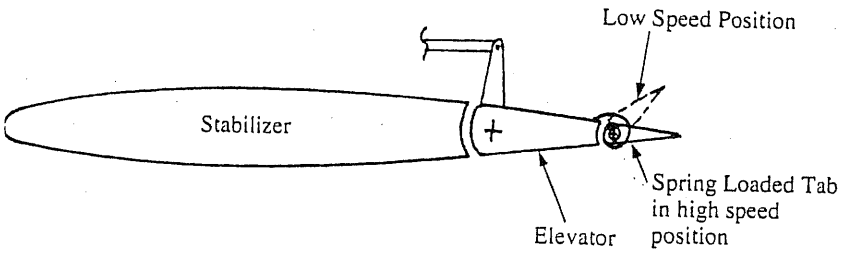


Fig. 27.11 Spring-loaded tab.[3]

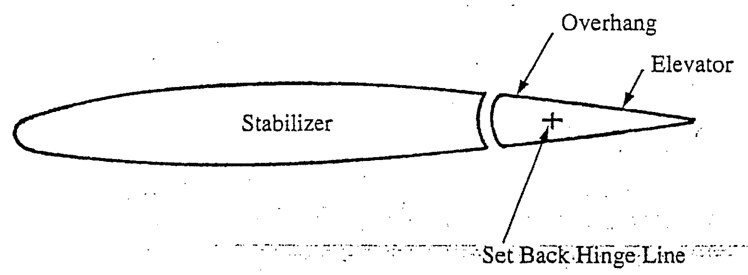


Fig. 27.12 Overhang or set-back hinge line balance.[3]

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

27.4.2 Horn Balance[2,3]

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

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

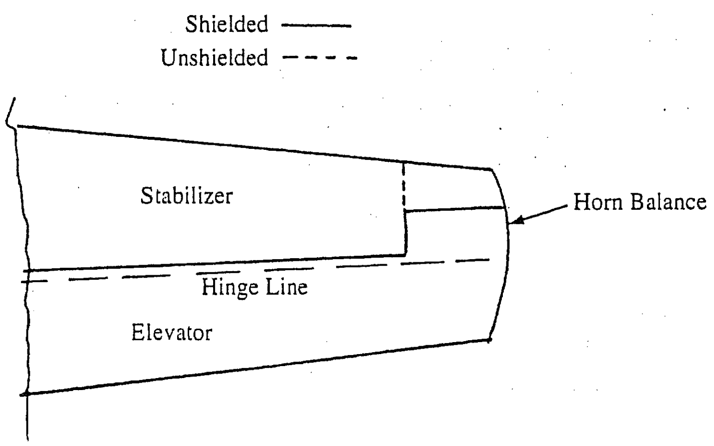


Fig. 27.13 Shielded and unshielded horn balance.[3]

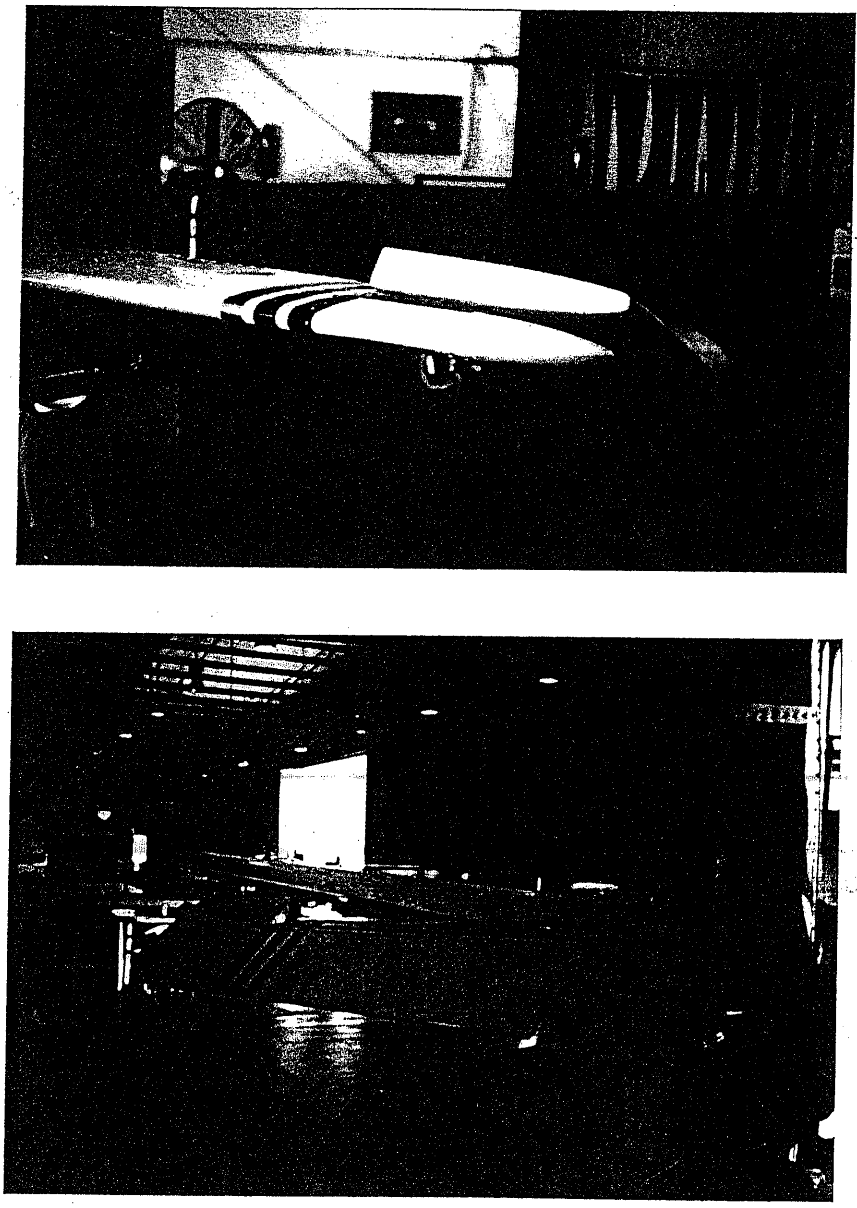


Fig. 27.14 Unshielded horn balance on the Micco 145A.

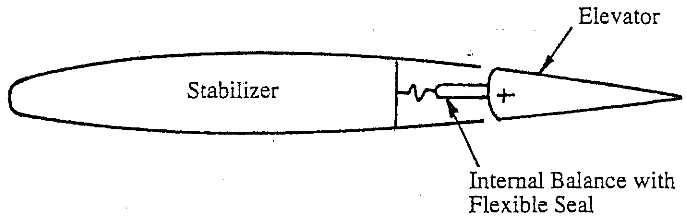


Fig. 27.15 Internal balance with a flexible seal.[3]

27.4.3 Internal Balance with Flexible Seal[2,3]

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

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

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

27.4.5 Beveled Trailing Edge[2,3]

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

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

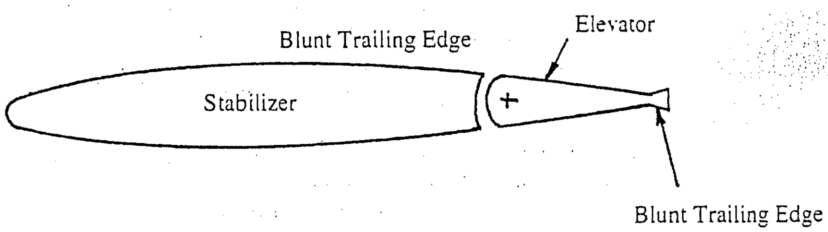


Fig. 27.16 Blunt elevator trailing edge.

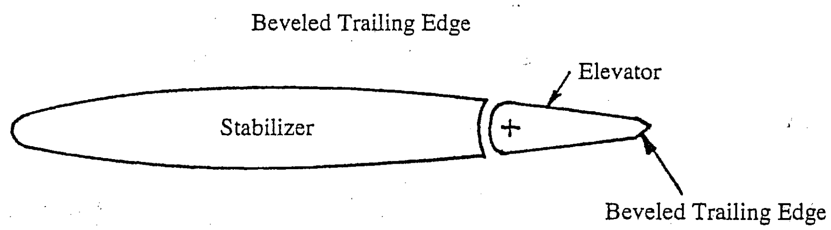


Fig. 27.17 Beveled elevator trailing edge.[3]

References

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

[2]Staff, USAFTPS, "Stability and Control Flight Test Theory, Vol. 1," AFFTC-TIH-77-1, Feb. 1977.

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

[4]Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

28 Lateral-Directional Stability Theory and Flight Test Methods

28.1 Introduction

A study of lateral-directional stability is a study of the reaction of the airplane when its flight path deviates from the plane of symmetry. The angle that the plane of symmetry makes with the relative wind is called the sideslip angle β. The angle that the plane of symmetry makes with some fixed reference is called the yaw angle ψ. The yaw angle is positive when the nose is displaced to the right of the reference, while sideslip is positive when the relative wind is coming from the right side of the airplane. The sideslip can be compared to the angle of attack, while the yaw angle is similar to the pitch angle. Also, like angle of attack and pitch, the two will only always agree in magnitude in the wind tunnel. It should be noted that it is possible to have yaw without sideslip, and to have yaw rates and moments without sideslip. It might also be noted that sideslip acts in a different plane than does the angle of attack and its effects are quite different. In lateral-directional stability and control a sideslip will not only generate a yawing moment, but it will also generate rolling moments and side-force. This is considerably different from the longitudinal case where a change in angle of attack only generates a pitching moment. Also, where angle of attack has great usefulness to the pilot, sideslip has little use except for crosswind landing situations.[1–3]

28.2 Federal Aviation Administration Regulations

Both CAR 3 and FAR Part 23 contain requirements for lateral-directional stability.

28.2.1 Civil Aeronautics Regulation 3 (Ref. 4)

CAR 3.118(a)(1) requires that three control airplanes exhibit static directional stability for all landing gear and flap positions and symmetric power conditions for all airspeeds from $1.2V_{S1}$ up to the maximum airspeed. This directional stability shall be demonstrated by recovery from a skid (sideslip) with the rudder free.

CAR 3.118(a)(2) requires that for three control airplanes the static lateral stability as demonstrated by the ability to raise the low wing during a sideslip shall be positive at the maximum permissible airspeed and not be negative at a speed of $1.2V_{S1}$. This, too, shall be demonstrated for all landing gear and flap positions and symmetrical power conditions.

This regulation also requires in CAR 3.118(a)(3) that in straight steady sideslips the rudder and aileron control displacements and forces shall increase steadily as sideslip angle is increased. However, these increases do not have to be in constant proportion. The control forces, displacements, and sideslip angles should be appropriate for the type airplane. In addition, up to full rudder, or a rudder force of 150 lb, the rudder pedal forces shall not reverse and increased rudder deflection shall produce increased angles of sideslip. The bank angles during these sideslips should be sufficient to indicate departure from straight flight.

CAR 3.118(b) covers two controlled airplanes (airplanes where the rudder and aileron are interconnected through a single control, usually the lateral control). For these airplanes 3.118(b)(1) covers static directional stability. It requires that these airplanes shall be rolled from one 45 deg bank to the opposite 45 deg bank without exhibiting dangerous skidding characteristics.

Static lateral stability for two controlled airplanes as stated in 3.118(b)(2) is demonstrated by showing that the airplane will not assume a dangerous attitude or airspeed if the controls are abandoned for a period of two minutes. This demonstration is conducted at an airspeed of $0.9V_H$ (or at V_C if lower) at an aft c.g. loading.

28.2.2 Federal Aviation Regulations Part 23 (Ref. 5)

FAR 23.177 covers static directional and lateral stability. Early versions of the regulation are separated into three-control and two-control airplanes as was CAR 3. However, in current versions of FAR Part 23 the requirements for two control airplanes have been dropped, since no one has designed a two-control airplane for many years.

FAR 23.177(a)(1) in the early Part 23 and 23.177(a) in current versions read nearly the same as the directional stability requirements of CAR 3. Added, however, is a statement requiring that the rudder deflections and forces be proportional to the sideslip angle and another statement requiring that the rudder pedal forces shall not reverse at airspeeds between $1.2V_{S1}$ and V_A.

The static lateral stability requirements of 23.177(a)(2) or early FAR Part 23 or 23.177(b) of current versions is the same as CAR 3 with the following modifications. First, the symmetrical power has an upper limit of 75% MCP where there was no upper limit in CAR 3. Second, for the early FAR Part 23, the sideslip angle should be appropriate to the type airplane, but in no case should be less than the sideslip with 10 deg of bank. The current regulation changes the minimum airspeed to $1.2V_{S1}$ for the takeoff configuration and $1.3V_{S1}$ for all other configurations. It modifies the "10 deg of bank" requirement by saying: "or if less, the maximum bank angle obtainable with full rudder deflection or 150 lb of rudder force."

Current versions of FAR Part 23 contain 23.177(c) that states 23.177(b) does not apply to acrobatic airplanes certified for inverted flight.

FAR 23.177(a)(3) in early versions of the regulation or 23.177(d) in current versions are similar to CAR 3.118(a)(3). Both FAR Part 23 versions read nearly the same and both have changed the required maximum power for which there shall be no rudder force reversal to 50% MCP. CAR 3 does not specify the power setting. All FAR Part 23 versions have added the requirement that rapid entries into, or recoveries from, these sideslips shall not result in uncontrollable flight characteristics.

28.2.3 *Advisory Circular 23-8A (Ref. 6)*

The advisory circular states that the purpose of these tests is to require positive directional and lateral stability and to verify the absence of rudder lock in three-control airplanes. It states that for directional stability the rudder force tests are conducted at airspeeds between $1.2V_{S1}$ and V_A while the directional stability tests are conducted using speeds from $1.2V_{S1}$ to V_{NE}. Remember that the later version of Part 23 changed some of those lower airspeeds to $1.3V_{S1}$.

The advisory circular specifically addresses two-control airplanes with regard to 23.177(b)(2). It states that this test is a check of spiral stability for these airplanes and that the "dangerous attitude" referred to in the regulation is construed to be 60 deg, or more of bank, 30 deg, or more of pitch, while a dangerous airspeed is taken to be V_{NE} or V_{S1}.

Also addressed is the need to test with autopilot or stability augmentation systems installed if these systems increase control system friction levels. Since such systems are seldom installed during prototype airplane development and certification tests, directional and lateral stability should be reevaluated when those systems are installed.

Under test procedures, the advisory circular recommends that the test be conducted at the highest altitude considering engine power and aerodynamic damping. This is particularly true for those directional or lateral stability tests that must be conducted at the never-exceed speed, V_{NE}.

The directional stability test is described as yawing the airplane left and right with the rudder while the wings are held level with the ailerons. The test is considered satisfactory when the airplane returns to straight flight when the rudder is released. It should be noted that straight, steady sideslips where control deflection and force are measured have also been deemed an acceptable method of test, although it is not spelled out in the advisory circular.

The section for test procedures on lateral stability clarifies the 10 deg bank requirement when the aircraft is not capable of generating 10 deg of bank with full rudder deflection. In that case, the advisory circular recommends applying 10 deg of bank with aileron while full rudder is depressed and allowing the aircraft to turn while attempting to raise the wing from the bank.

28.3 Theory

Since sideslip is a factor in both lateral and directional stability, these items are generally measured in steady sideslips. Since in a steady sideslip there is no acceleration we can write the equations of motion as follows:

Rolling moment equation:[1]

$$C_{l_\beta}\beta + C_{l_{\delta_r}}\delta_r + C_{l_{\delta_a}}\delta_a = 0 \tag{28.1}$$

where

C_{l_β} = rolling moment coefficient due to sideslip

$C_{l_{\delta_r}}$ = rolling moment coefficient due to rudder deflection

δ_r = rudder deflection
$C_{l_{\delta_a}}$ = rolling moment coefficient due to aileron deflection
δ_a = aileron deflection

Yawing moment equation:[1]

$$C_{n_\beta}\beta + C_{n_{\delta_r}}\delta_r + C_{n_{\delta_a}}\delta_a = 0 \tag{28.2}$$

where
C_{n_β} = yawing moment coefficient due to sideslip
$C_{n_{\delta_r}}$ = yawing moment coefficient due to rudder deflection
$C_{n_{\delta_a}}$ = yawing moment coefficient due to aileron deflection

Side-force equation:[1]

$$C_{Y_\beta}\beta + C_{Y_{\delta_r}}\delta_r + C_{Y_{\delta_a}}\delta_a + C_L\phi = 0 \tag{28.3}$$

where
C_{Y_β} = side-force coefficient due to sideslip
$C_{Y_{\delta_r}}$ = side-force coefficient due to rudder deflection
$C_{Y_{\delta_a}}$ = side-force coefficient due to aileron deflection
ϕ = bank angle

As can be seen from these three equations, all have a mixture of lateral and directional coefficients, and treatment of lateral-directional stability and control cannot be as simple as for longitudinal stability and control. It is, however, of some benefit to discuss them separately and that will be the approach here.

28.4 Directional Stability

In examining Eq. (28.2), which is essentially the directional stability equation, we can see that the first term in the equation is the sideslip term. This is quite proper since the problem of directional stability is essentially that of insuring that the airplane maintains zero sideslip. Directional stability then is "weathercock" stability and involves the moments generated about the vertical axis.[3] Directional stability also has a closer comparison with longitudinal stability than does lateral stability. If we view the airplane from the top we can see that the fuselage, vertical tail, nacelles, and power all act about the vertical axis much in the same way that the fuselage, horizontal tail, nacelles, and power act about the lateral axis. Only the wing contributions differ. Like longitudinal stability, the static directional stability is a result of the contribution of the components of the airplane.[3] Although the contributions of the components are sometimes related, it is much easier to study them separately.

The vertical tail is the primary contributor to directional stability.[3] When the airplane is placed in a sideslip the vertical tail experiences a change in angle of attack. This change in angle of attack causes the vertical tail to generate a force in the direction, which rotates the airplane so as to reduce the sideslip.[3]

This side-force can be expressed by the nondimensional derivative $C_{Y_{\beta_V}}$ (Ref. 1).

$$C_{Y_{\beta_V}} = -a_V\left(1 - \frac{d\sigma}{d\beta}\right)\eta_V \frac{S_V}{S_W} \tag{28.4}$$

where

a_V = lift curve slope of the vertical tail
$d\sigma/d\beta$ = change in side wash σ with change in sideslip β. This factor is caused by interference of other parts of the airplane
η_V = vertical tail efficiency factor q_V/q

The yawing moment due to the vertical tail is a function of the side-force due to the vertical tail and the length of the vertical from the c.g. or:[2]

$$C_{n_{\beta_V}} = -C_{Y_{\beta_V}} \frac{l_V}{b} \tag{28.5}$$

where

l_V = tail arm, or length of the vertical tail a.c. from the c.g.

In studying the last two equations, we can see that the directional stability contribution of the vertical tail is controlled by three main factors: 1) the tail arm, 2) the lift curve slope, and 3) the vertical tail area.

Once the airplane has progressed to the flight test stage, a modification of the tail arm is not likely, since it would probably require a major redesign of the fuselage. It is, however, probably one of the more powerful parameters in the equation.

The vertical tail lift curve slope is somewhat easier to modify during flight test. For instance, the addition of a dorsal fin (Fig. 28.1)[3] will reduce the vertical tail aspect ratio and cause the vertical tail to stall at a higher sideslip angle. A ventral fin (Fig. 28.2) will increase the vertical tail aspect ratio and steepen the lift curve slope. The ventral fin also lowers the center of pressure of the vertical tail which will reduce the roll with yaw caused by the vertical tail.[7]

The vertical tail area can also be changed with dorsal and ventral fins, so they are handy gimmicks to use during flight test. The contribution of the wing to directional stability is small. Sweeping the wings improves the contribution, but it is still small when compared to other components.

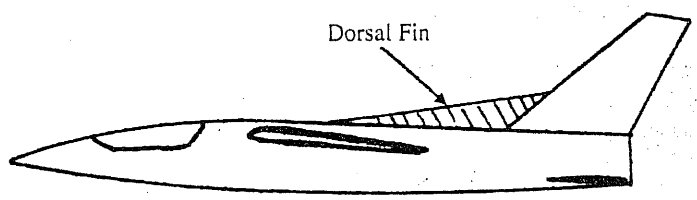


Fig. 28.1 Aircraft with dorsal fin.[3]

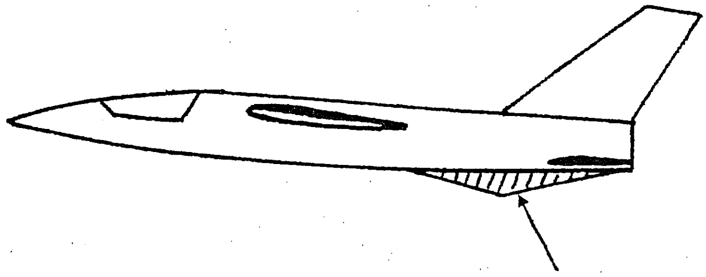


Fig. 28.2 Aircraft with ventral fin.

The fuselage and nacelles provide a large contribution that is generally destabilizing.[3] This destabilizing influence is similar to the longitudinal case except that in this case the up-wash and down-wash of the wing are not a factor. Therefore, the destabilizing influence of the fuselage is not as great in this case as it is in the longitudinal case.

The power and propeller fin effects on directional stability are similar to the effects of these items on longitudinal stability, i.e., generally destabilizing.[3]

Freeing the rudder has the same effects on directional stability as it does on longitudinal stability, i.e., a reduction in stability.[3]

High Mach numbers also reduce directional stability due to the reduction of the vertical tail lift curve slope with increasing Mach number.[3,7] This is one of the reasons that today's high-speed fighters have twin vertical tails.

In the category of minor contributors to directional stability are the roll control devices such as ailerons and spoilers. Because ailerons produce more induced drag on the up-going wing than on the down-going wing, they tend to increase sideslip and have a destabilizing effect. This effect is called adverse yaw. Spoilers, on the other hand, produce additional drag on the down-going wing and, therefore, reduce sideslip or cause proverse yaw.[1]

28.5 Lateral Stability

Static lateral stability is not comparable to either longitudinal or directional stability even though it does have an effect on directional stability. Lateral stability is a study of the effects of sideslip on the rolling moments of an airplane. If positive sideslip provides a negative rolling moment and vice versa, then the airplane is said to possess positive lateral stability. If a sideslip produces no rolling moment then the airplane possesses neutral lateral stability. A positive sideslip producing a positive rolling moment and vice versa is described as negative lateral stability. We can say then that when an airplane tries to roll away from the sideslip it has positive lateral stability, and if it rolls into the sideslip it has negative lateral stability.[3]

It is generally desirable only to have weak positive lateral stability. This is because excessive roll due to sideslip complicates such tasks as crosswind takeoffs and landings where it is necessary to sideslip.[3] Also, strong positive lateral stability increases roll-yaw coupling, which is undesirable for certain

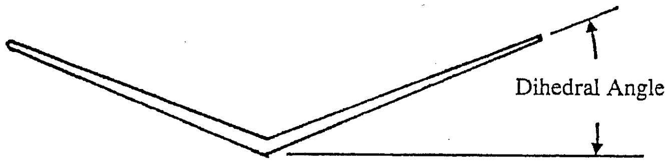


Fig. 28.3 Wing dihedral.[3]

tracking tasks such as instrument approaches and strafing runs.

If the above is the case, what methods do we have for increasing or decreasing lateral stability? Once the aircraft reaches the flight test stage we have very few. The designer, on the other hand, has several options that may be used during the design stage. To evaluate these options let us examine the contributions of the various airplane components to lateral stability.

The wing is the primary contributor to lateral stability.[3] This is accomplished by the use of geometric dihedral (Fig. 28.3).[3] A wing with dihedral will develop stable rolling moments with sideslip because the dihedral causes the "wing into" the sideslip to have an increased angle of attack while the opposite wing has a decreased angle of attack (Fig. 28.4). It is for this reason that lateral stability is sometimes referred to as dihedral effect.

Aft sweep in a wing also creates favorable rolling moments with sideslip.[1,3,7] This is also due to the differences in angle of attack experienced due to an effective reduction in sweep of the wing "into the wind" and an effective increase in sweep of the opposite wing (Fig. 28.5).[3] This effect increases as angle of attack increases and presents a significant roll-yaw coupling problem for high performance, highly swept wing airplanes.[2]

The fuselage, or location of the wing on the fuselage, is a major contributor to lateral stability. This is essentially caused by the effect of the fuselage on

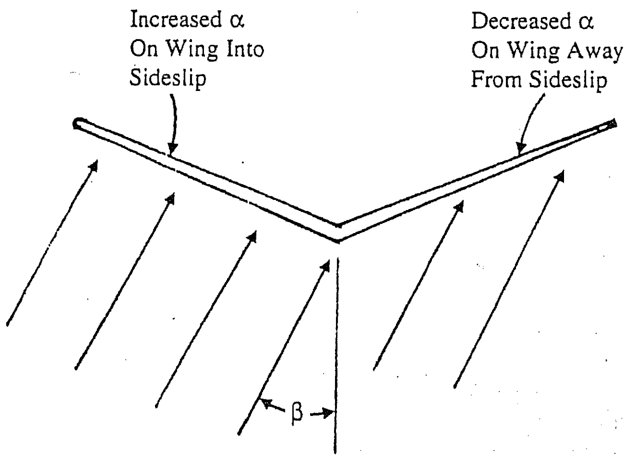


Fig. 28.4 Effects of dihedral during sideslips.

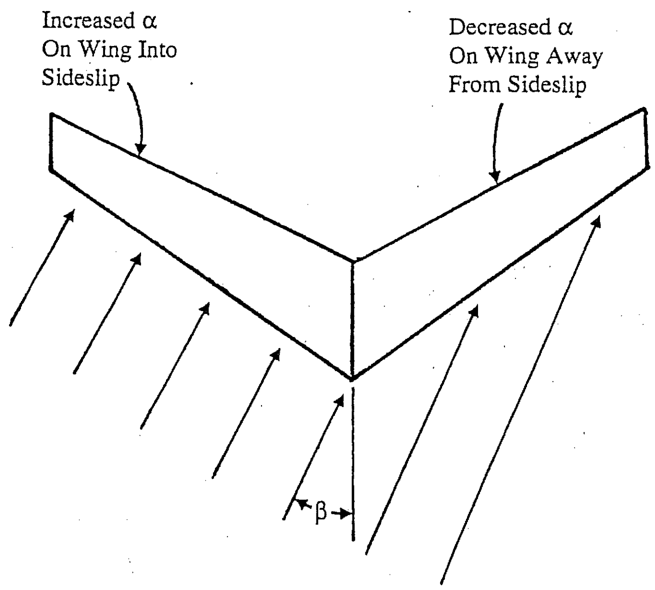


Fig. 28.5 Effects of wing sweep on lateral stability.[3]

the wing angle of attack. A fuselage below the wing creates the positive lateral stability while a fuselage above the wing creates negative lateral stability (Fig. 28.6).[3,7] This is such a strong effect that low-wing airplanes require 3–4 deg more dihedral than do high-wing airplanes.

The vertical tail also contributes to lateral stability.[3] The side-force acting on the vertical tail during a sideslip acts at some distance from the longitudinal axis. This creates a rolling moment that is favorable for conventional tail configurations.[7] It is possible to make this moment zero or even unfavorable by placing part or all of the vertical tail below the longitudinal axis.

The propeller slipstream also contributes to lateral stability. Due to the fact that during a sideslip the slipstream flows mostly over the wing "away from the wind," it creates a destabilizing rolling moment. This moment is greater when the flaps are down (Fig. 28.7)[1] and, as a result, the power approach configuration is generally critical for lateral stability.[1]

28.6 Side Force

Side-force can be generated by many parts of the airplane; however, the two main contributors are the vertical tail and an inclination of the lift vector. In a steady sideslip it is the side-force that causes the lateral translation. This lateral translation is very useful in countering drift due to crosswinds in takeoff, landing, and target tracking situations. If side-force is generated in a sideslip, it creates the roll-yaw coupling problems that make the piloting task difficult. This is one reason that considerable effort is now being expended on direct side-force control through auxiliary surfaces on the wings or fuselage. As can

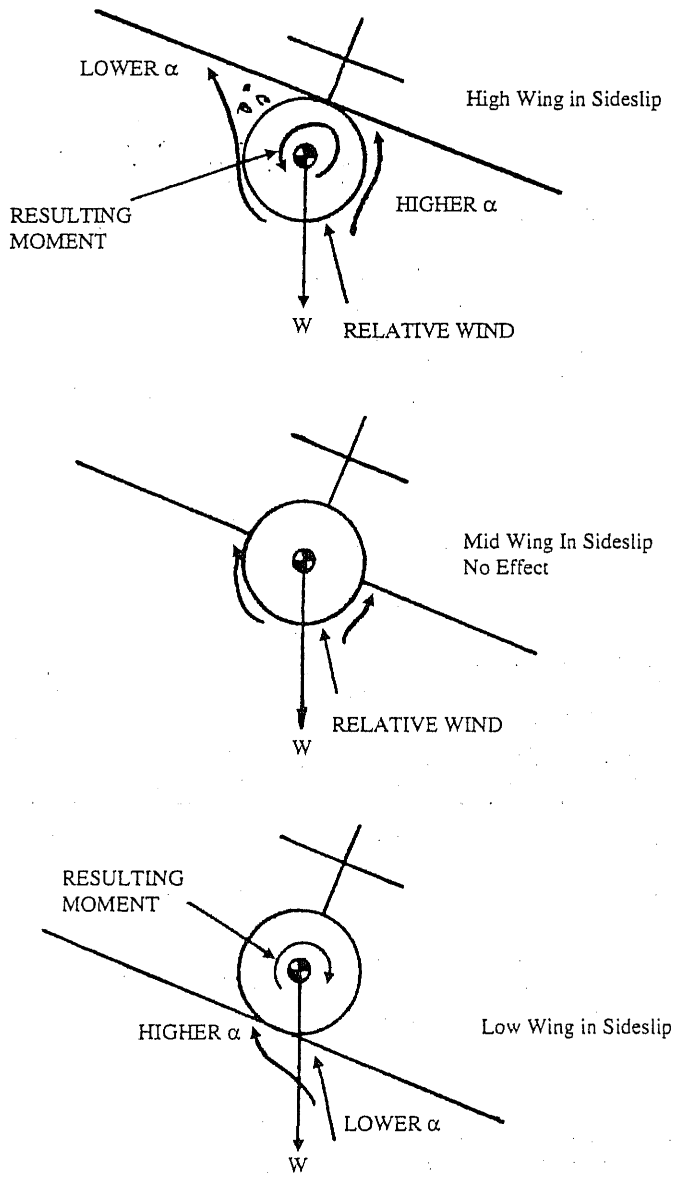


Fig. 28.6 Effects of fuselage locations on lateral stability.

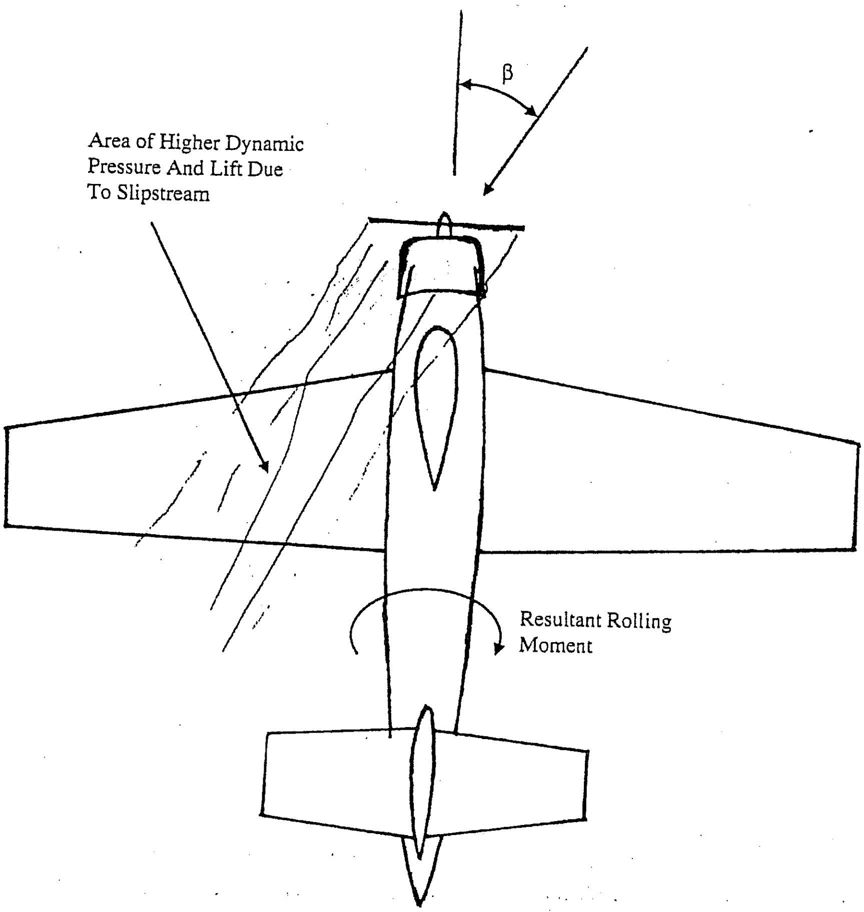


Fig. 28.7 Slipstream effects during sideslip.[1]

be seen from Eq. (28.3), the side-force is very dependent on bank angle due to the inclination of the lift vector. This makes a plot of bank angle vs sideslip an important plot in determining side-force.

28.7 Control-Free, Lateral-Directional Stability

In the past discussions we have been talking about control-fixed, lateral-directional stability or control position, lateral-directional stability. It can be noted that all of the steady side-slip equations contain terms for aileron and rudder deflection. Now let us turn our attention to control-free, lateral-directional stability.

Like longitudinal stick-free stability, control-free, lateral-directional stability is a function of the control force. For an irreversible control system the rudder or aileron forces are simply a function of control position.[1] For instance, for a rudder system the rudder force can be expressed as:[1]

$$F_r = K_1 \Delta\delta_r \tag{28.6}$$

for a linear feel spring system and:[1]

$$F_r = K_2 q \Delta\delta_r \tag{28.7}$$

for a q-feel system. In these equations the constants K_1 and K_2 describe the characteristics of the system.[1]

For a reversible control system the control forces are a function of control surface float. For a lateral control system, aileron floating tendency is generally very small and, as a result, lateral control forces are small. The rudder, on the other hand, may exhibit considerable floating tendency and is more comparable to a longitudinal control system in this respect.

The floating angle of the rudder can be expressed by the equation:[1]

$$\delta_{r_{float}} = -\frac{C_{h_{\beta_V}}}{C_{h_{\delta_r}}}\beta_V \tag{28.8}$$

where

$C_{h_{\beta_V}}$ = rudder hinge moment coefficient variation with sideslip at zero rudder deflection; normally positive

$C_{h_{\delta_r}}$ = rudder hinge moment coefficient variation with rudder deflection at zero sideslip; normally negative

In order for the pilot to move the rudder from its float position to a position of equilibrium the pilot must exert a force on the rudder.[1] This force can be expressed by the equation:[1]

$$F_r = -KC_{h_{\delta_r}} q_V S_r \overline{C_r}(\delta_{r_{equil.}} - \delta_{r_{float}}) \tag{28.9}$$

where

K = the rudder control system gearing constant, in radians per feet

q_V = dynamic pressure at the vertical tail, in pounds per square foot

S_r = area of rudder in square feet

C_r = MAC of the rudder in feet

To obtain the variation of rudder force with sideslip we differentiate Eq. (28.9) and obtain:[1]

$$\frac{dF_r}{d\beta} = -KC_{h_{\delta_r}} q_V S_r \overline{C_r}\left(\frac{d\delta_r}{d\beta} - \frac{d\delta_{r_{float}}}{d\beta}\right) \tag{28.10}$$

We can now obtain a plot of how rudder force varies with sideslip from flight test.

An interesting point to note about Eq. (28.9) is that when the rudder float angle equals the rudder angle required for equilibrium, then the rudder force becomes zero.[1] This condition is called rudder lock and generally occurs at the point where the vertical fin stalls. It is an undesirable situation as we would like for the rudder force to increase with increasing sideslip. One method for reducing rudder lock potential is the addition of a dorsal fin.[3]

28.8 Flight Test Methods

From the lateral-directional stability theory one can see that more information regarding the capabilities of the airplane can be gained from measuring the quantities of control displacements, forces, bank angle, and sideslip, along with aircraft test weight while the aircraft is in a steady heading sideslip. This technique will also allow the extraction of some lateral-directional stability derivatives for use by the stability and control group in addition to providing the information needed for aircraft certification. However, the steady heading sideslip method requires more complex instrumentation than that required by the FARs and the advisory circular. Those methods can be used to show compliance to the regulations with minimum instrumentation.

28.8.1 Steady Heading Sideslips

The steady heading sideslip is performed by first trimming to hands-off flight in all three axis (providing three axis trim is available) at the test airspeed. In order to minimize altitude loss in power-off tests, one should start at the lowest trim airspeed at the highest altitude. The aircrew should approach large sideslip angles at this lowest airspeed with some caution as large sideslip may cause one wing to stall resulting in a snap roll departure. Once trimmed, the steady heading sideslip is entered in one direction using about 0.25 full rudder deflection. Once stabilized, data is recorded. The sideslip is then increased to 0.5 full rudder deflection, stabilized in a steady heading and data again recorded. This process is continued using 0.75 full deflection and full rudder deflection. Once completed in one sideslip direction, the testing is repeated in the opposite direction. Seldom does one find that propeller-driven airplanes are symmetrical left and right. This will be especially true if there is a fuel unbalance. Therefore, initial testing should be conducted with a symmetrical fuel loading. Since AC 23-8A requests testing with a maximum allowed fuel unbalance, the critical sideslips should be repeated with that unbalance. These tests should be repeated in even increments of airspeed throughout the required range and in the specified configurations of landing gear, flaps, and power.

The data to be recorded during these tests consist of sideslip β, bank angle ϕ, aircraft weight at the time of the data point, rudder position δ_r and force F_r, aileron position δ_a and force F_a, and elevator position δ_e and force F_s. Elevator position and force are recorded to determine if the aircraft has a tendency to tuck or pitch up with sideslip. Although this pitching information is not required by the FARs it is worthwhile information. Since there are several parameters to record, it is best to record this data with an automatic recording device. The data may be hand recorded, but accuracy and test time suffer, and the pilot may experience fatigue from stabilizing the airplane in the many sideslips. Once data are recorded, they are corrected for instrument error and plotted as shown in Fig. 28.8. Most lateral-directional stability data are nonlinear. This is due to separation from various parts of the airplane during

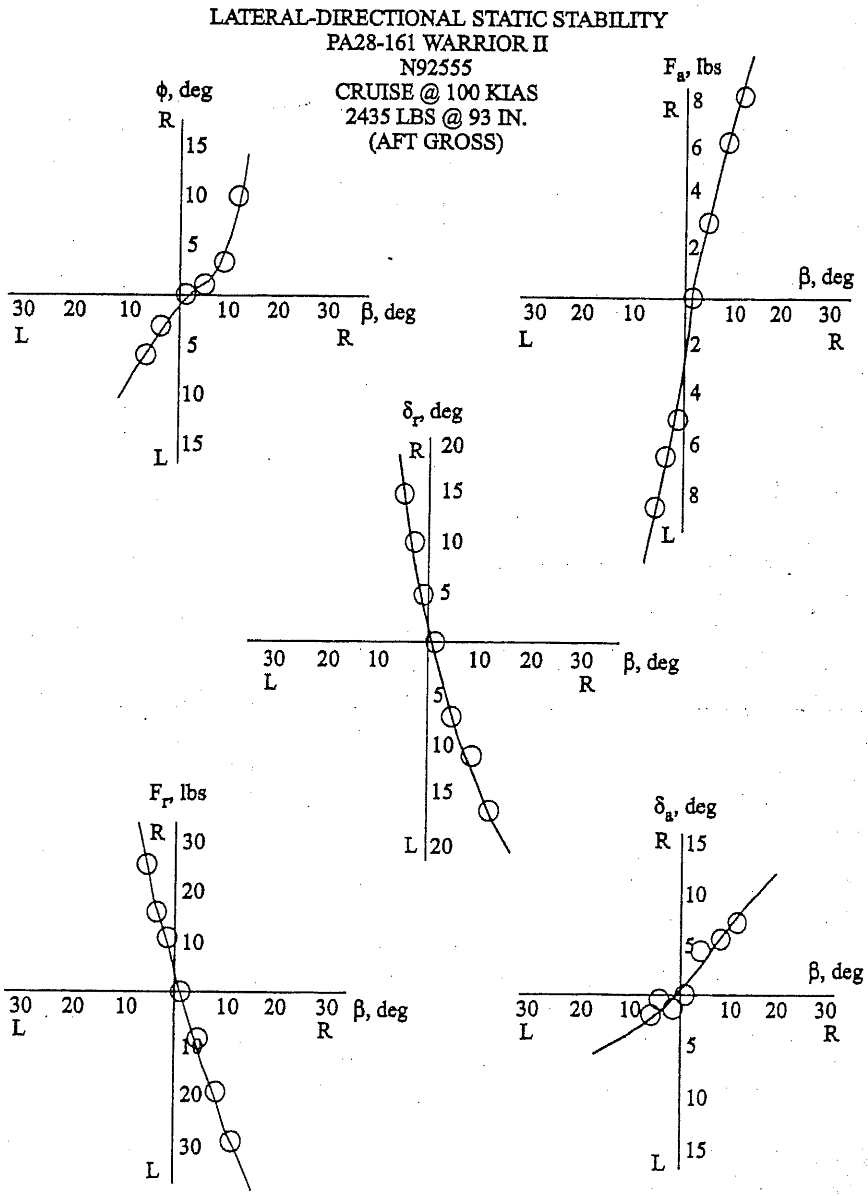


Fig. 28.8 Steady heading sideslip data.

the sideslip. The propeller slipstream will cause this separation to be different depending upon the direction of sideslip resulting in plots that are not symmetrical.

28.8.2 FAA Directional and Lateral Methods[6]

The lateral stability test method prescribed by the FAA in AC 23-8A is essentially the same as the steady heading sideslip described above. The difference comes when the airplane is incapable of performing a steady heading sideslip with 10 deg of bank. In that case, 10 deg of bank is used with full rudder and the aircraft is allowed to turn. Compliance with the regulation is demonstrated in the lateral stability test using the FAA method if when the aileron is released the wings tend to return to level. Note that they only have to have a *tendency* to return to level. As one can see, this test only collects qualitative data and does not provide the data that can be obtained from a fully instrumented steady heading sideslip.

Directional stability tests as prescribed by the FAA consist of wing-level sideslips to determine if, when the rudder is released, the aircraft returns to straight flight. Care should be taken in any of the test methods that full control deflections are not used above the maneuvering speed V_A. The directional stability test using the FAA method also only provides qualitative data. However, that is sufficient to show compliance with the regulation.

References

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

[2] Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

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

[4] Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[5] Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[6] Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

[7] Staff, USAFTPS, "Stability and Control Flight Test Techniques, Vol. II," AFFTC-TIH-77-1, Feb. 1977.

29 Dynamic Lateral-Directional Stability Theory and Flight Test Methods

29.1 Introduction

In our past discussions on lateral-directional stability and control, we have separated the lateral and directional responses. This was convenient for the discussion of the static cases, but in free flight the responses are coupled. For the dynamic case, we have to consider this coupling and the effects that the airplane's inertia have on it.

29.2 Federal Aviation Administration Regulations

The FAA Regulations from their early days have only addressed the motions both longitudinal and lateral-directional that are of such short period as to couple with the pilots reaction time. As a result, motions like the spiral do not appear in the FAA Regulations.

29.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

CAR 3.118(a)(4) for three-control airplanes and (b)(3) for two-control airplanes both say the same thing. They state: "Any short-period oscillation occurring between stalling speed and maximum permissible speed shall be heavily damped with the primary controls (i) free and (ii) in a fixed position."

29.2.2 Federal Aviation Regulation Part 23 (Ref. 2)

FAR 23.177(a)(4) for three-control airplanes and (b)(3) for two-control airplanes in early versions of FAR Part 23 read the same as CAR 3. Later versions of the regulation, beginning with Amendment 21 and changing again with Amendment 45 have placed all of the regulations for dynamic stability under one heading.

This heading is FAR 23.181 Dynamic Stability. FAR 23.181(a) reads the same as the previous regulation, but combines both lateral-directional and longitudinal dynamics under this regulation. FAR 23.181(b) speaks specifically to the Dutch roll motion. It requires that any Dutch roll oscillation that occurs between stalling speed and maximum allowable speed must be damped to 1/10 amplitude in 7 cycles with the primary controls fixed and free.

The current regulation has added 23.181(c) that states if a stability augmentation system is needed to meet the flight characteristics requirements then the aircraft does not have to meet the controls fixed portions of 23.181.

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

AC 23-8A reiterates that the lateral-directional regulations only address the Dutch roll mode of motion and do not consider other modes. It states: "The damping of the Dutch roll motion is probably the most important Dutch roll characteristic to be considered." The advisory circular describes the accepted flight test procedures for testing the Dutch roll. What is described as rudder pulsing is essentially what is described in this chapter as the doublet input, while the steady sideslip as described in the advisory circular is the same maneuver described here as rudder kicks. Otherwise the description of the test methodology is very similar.

29.3 Theory

By use of a transformation the lateral-directional equations of motion for small disturbances can be expressed as a determinant (Fig. 29.1).[4] In this determinant the assumption is made that the effects of lateral-directional cross-coupling are included in each of the terms. This cross-coupling is a result of the product of inertia in roll and yaw, I_{xz} (Refs. 4 and 5).

If we solve the lateral-directional determinant we can obtain a considerable amount of information about the lateral-directional modes of motion. In order to somewhat simplify the solution we will assume that the dihedral effect and the roll due to yaw rate are small enough to be zero:[4,5]

$$L_\beta = 0 \qquad \text{and} \qquad L_r = 0$$

We know that this assumption is not always true, and we will discuss the effects of increasing these variables later. However, by making this assumption the lateral-directional determinant reduces to:

$$S(S - L_p)\begin{vmatrix} S - Y_\beta & 1 \\ -N_\beta & S - N_r \end{vmatrix} = 0$$

By solving this determinant we have:[4]

$$S(S - L_p)[S^2 + (-Y_\beta - N_r)S + (N_\beta + Y_\beta N_r)] = 0 \tag{29.1}$$

This equation describes the three classic modes of lateral-directional motion. These modes of motion are:[4,5]

1) the spiral mode indicated by S
2) the roll mode indicated by $(S - L_p)$
3) the Dutch roll mode indicated by $[S^2 + (-Y_\beta - N_r)S + (N_\beta + Y_\beta N_r)]$

Fig. 29.2 (Ref. 4) shows a plot of these roots as they would plot in the imaginary plane. This plot shows the spiral root at the origin indicating neither a convergence nor divergence when excited. The roll mode is shown as a large negative real root. This indicates, as we might suspect, a nonoscillatory rolling

$$
\begin{array}{l}
\text{Sideforce Characteristics} \rightarrow \\
\text{Rolling Moment Characteristics} \rightarrow \\
\text{Yawing Moment Characteristics} \rightarrow
\end{array}
\begin{vmatrix}
S - Y_\beta & -g/\mu_o & 1 \\
-L_\beta & S\,(S - L_p) & -L_r \\
-N_\beta & -N_p\,S & S - N_r
\end{vmatrix} = 0
$$

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 (μ_o = initial horizontal velocity)

I_{XX} = airplane moment of inertia in roll.

I_{ZZ} = airplane moment of inertia in yaw.

$Y_\beta = \dfrac{\partial Y/\partial\beta}{m\mu_o} = C_{Y_\beta}\dfrac{qS}{m\mu_o}$ = sideforce due to sideslip term.

$L_\beta = \dfrac{\partial L/\partial\beta}{I_{XX}} = C_{\ell_\beta}\dfrac{qSb}{I_{XX}}$ = rolling moment due to sideslip (dihedral effect) term.

$L_p = \dfrac{\partial L/\partial p}{I_{XX}} = C_{\ell_p}\dfrac{qSb^2}{2\mu_o I_{XX}}$ = rolling moment due to roll rate (roll damping) term.

$L_r = \dfrac{\partial L/\partial r}{I_{XX}} = C_{\ell_r}\dfrac{qSb^2}{2\mu_o I_{XX}}$ = rolling moment due to yaw rate term.

$N_\beta = \dfrac{\partial N/\partial\beta}{I_{ZZ}} = C_{n_\beta}\dfrac{qSb}{I_{ZZ}}$ = yawing moment due to sideslip (directional stability) term.

$N_p = \dfrac{\partial N/\partial p}{I_{ZZ}} = C_{n_p}\dfrac{qSb^2}{2\mu_o I_{ZZ}}$ = yawing moment due to roll rate term

$N_r = \dfrac{\partial N/\partial r}{I_{ZZ}} = C_{n_r}\dfrac{qSb^2}{2\mu_o I_{ZZ}}$ = yawing moment due to yaw rate (yaw rate damping) term.

Fig. 29.1 Lateral-directional determinant.[4]

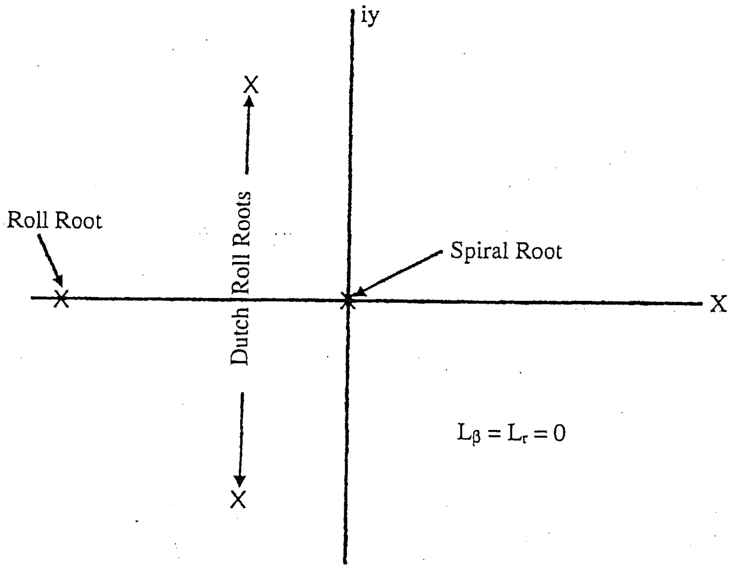


Fig. 29.2 Complex plane representation of classic lateral-directional roots.[4]

motion that is heavily damped. The Dutch roll mode is shown by a pair of complex roots. This indicates a second order oscillatory type of motion.[4,5]

Now let us turn our attention to each of these modes of motion and the effects on them of varying the lateral-directional parameters.

29.4 Spiral Mode

The spiral mode of motion is a very gentle mode even when divergent and is easily controlled by the pilot.[6,7,8] If divergent, the spiral mode may present a problem for the pilot under instrument conditions when his/her attention is diverted from flying the airplane. The spiral mode can be described as a bank angle increase or decrease after a bank angle disturbance from wings level flight. The spiral mode can be either convergent, divergent, or neutral. This is dependent on the sign of the following combination of derivatives:[4,5]

$$L_\beta N_r - N_\beta L_r$$

If this combination is positive then the spiral mode will be convergent. If it is negative the spiral mode will diverge. This indicates that strong directional stability N_β tends to make the spiral mode divergent while strong positive lateral stability tends to make it converge. If the combination is zero then the spiral mode is neutral.[4,5]

The rate at which the spiral mode converges or diverges is a function of airspeed. At low airspeeds the rate is high while at high airspeeds it is low.[4]

29.5 Roll Mode

The roll mode will be discussed in a later chapter in our discussion of lateral control power and we will not belabor it any further here.

29.6 Dutch Roll Mode

The Dutch roll mode can be described as a lateral-directional oscillation. The oscillation is generally convergent, but the rate at which it converges is quite significant to the pilot's opinion of the airplane. It is desirable to have the motion heavily damped since a near neutral oscillation would make any tracking task difficult due to the fact that the pilot will excite it with any lateral-directional control input. In addition, if the oscillation is not heavily damped, atmospheric turbulence will excite it and make the ride of the airplane unpleasant. The Dutch roll mode is not a totally nuisance mode, however, since the pilot may use this mode of motion to generate sideslip changes for straight flight in crosswind landings. It is also the mode of motion used to control bank angle with the rudder.[4]

Since Dutch roll is a coupled motion there are no simple methods for determining the frequency and damping ratio of the Dutch roll mode. If we assume that both L_β and L_r are zero then we can arrive at an equation that gives a rough approximation of the undamped natural frequency of the motion.[4]

$$\omega_{n_{DR}} \cong M\sqrt{C_{n_\beta}\frac{\gamma P_a S b}{2I_{zz}}} \tag{29.2}$$

where
M = Mach number
γ = a constant, 1.4 for air
P_a = absolute pressure in pounds per square foot
I_{zz} = moment of inertia in yaw

If we evaluate this equation we can learn several things about the Dutch roll undamped natural frequency. First, we can see that it varies directly with Mach number. The higher the Mach number the higher the frequency. It also varies directly with directional stability C_{n_β}. In this case the directional stability can be equated to the spring in the spring-mass-damper system. Increasing the spring stiffness $+C_{n_\beta}$ increases the frequency. This equation also shows that as altitude increases (lower P_a) the frequency decreases. We can also see that as moment of inertia in yaw increases the frequency decreases. We would then expect airplanes with engines on the wing and tip tanks to have a low Dutch roll frequency if all else is equal.[4]

In all studies of dynamic motions, both the frequency and the damping ratio are important. In order to come up with Dutch roll damping ratio we have to make another simplifying assumption. In addition to the assumptions made to obtain the frequency equation, we must also assume that the side-force due to sideslip is equal to the yawing moment due to yaw rate or $Y_\beta = N_r$. If this is done we can arrive at the following equation for Dutch roll damping:[4]

$$\zeta_{DR} = C_{n_r}\sqrt{\frac{\rho S b^3}{8 C_{n_\beta} I_{zz}}} \tag{29.3}$$

where
C_{n_r} = yaw rate damping coefficient

Again, by a study of this equation we can reach several conclusions about Dutch roll damping. First, we can see that as yaw rate damping C_{n_r} increases, Dutch roll damping increases. The yaw rate damping acts like the damper in the spring-mass-damper system. We can also see that the Dutch roll damping decreases as altitude increases. This is the reason that most high-altitude airplanes have artificial yaw damping. Increasing directional stability decreases Dutch roll damping. Although an increase in yawing moment of inertia reduces the frequency, it also reduces damping. One other item might be noted about Dutch roll damping: neither airspeed nor Mach number appears in this equation and neither affects damping.[4]

The equations we have just evaluated only consider the case where there is no roll present in the motion, just a wagging of the tail. Dutch roll is in fact a coupled motion and the amount of roll present with the yawing, or the ratio of roll to yaw ϕ/β is very important in how the pilot perceives the Dutch roll characteristics of the airplane.[7] Generally, more damping is required when the ϕ/β ratio is increased.

29.7 Effects of Lateral-Directional Parameters on Lateral-Directional Dynamics[4]

To evaluate the coupling effects on the lateral-directional stability, the characteristic equation becomes much more complicated. The result of this complication is that the Dutch roll mode has both numerator and denominator roots. So to fully evaluate the effects of the various lateral-directional parameters on the three modes of lateral-directional dynamic stability, we must consider both the numerator roots indicated by circles and the denominator roots indicated by *x*s.

Fig. 29.3 (Ref. 4) shows the effects of increasing lateral stability or dihedral effect. When lateral stability is increased the spiral root becomes negative and the spiral stabilizes. The roll-to-sideslip ratio ϕ/β of the Dutch roll is increased while the damping ζ_{DR} decreases. In this case the roll due to yaw L_β and yaw due to roll N_p are assumed fixed at zero.

If we again make the same assumption we can evaluate the effects of roll due to yaw rate L_r. In this case the spiral becomes positive and is destabilized. For the Dutch roll case the roll-to-sideslip ratio ϕ/β and the natural frequency $\omega_{n_{DR}}$ increase. The Dutch roll damping δ_{DR} increases slightly. The damping in roll L_p is also increased. These effects are shown in Fig. 29.4 (Ref. 4).

Figs. 29.5a and 29.5b (Ref. 4) describe the effects of directional stability N_β. In Fig. 29.5a we hold L_β and L_r constant and equal to zero. In this case decreasing N_β causes the spiral to become more negative, or stabilize, while

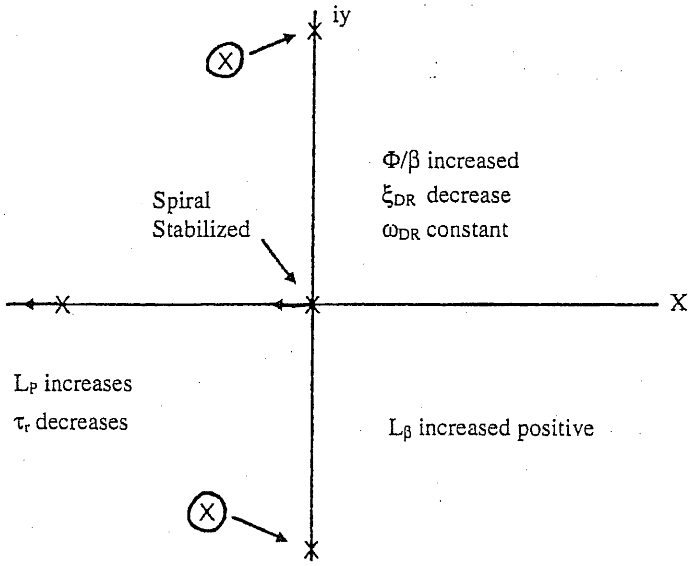


Fig. 29.3 Influence of adding positive dihedral effect.[4]

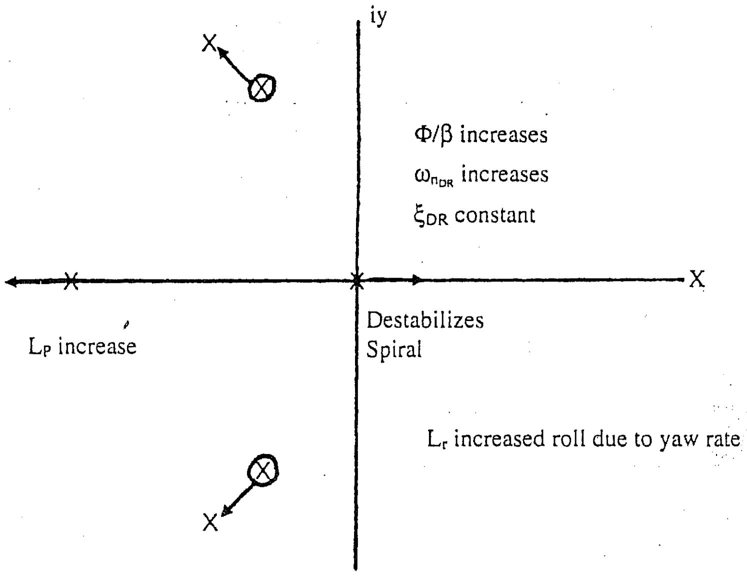


Fig. 29.4 Effects of increasing roll due to yaw rate.[4]

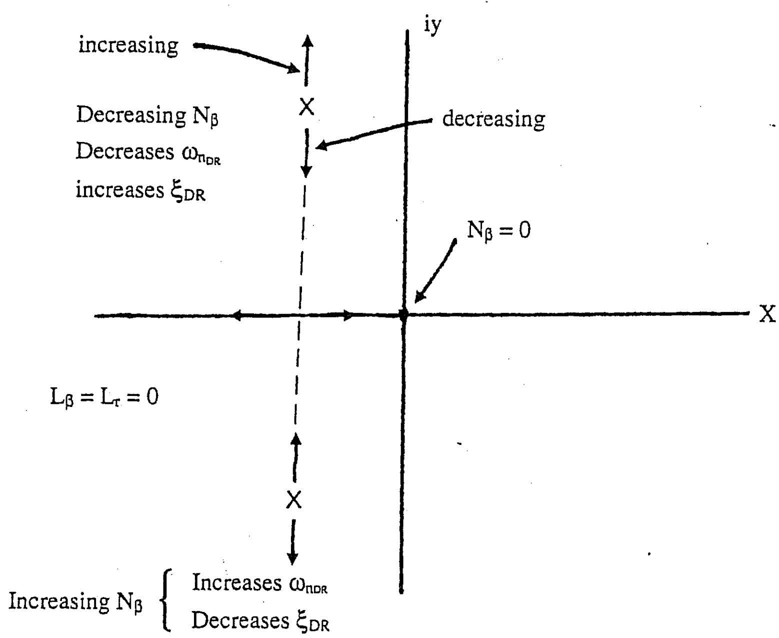


Fig. 29.5a Effects of changing N_β on the 2° of freedom Dutch roll motion.[4]

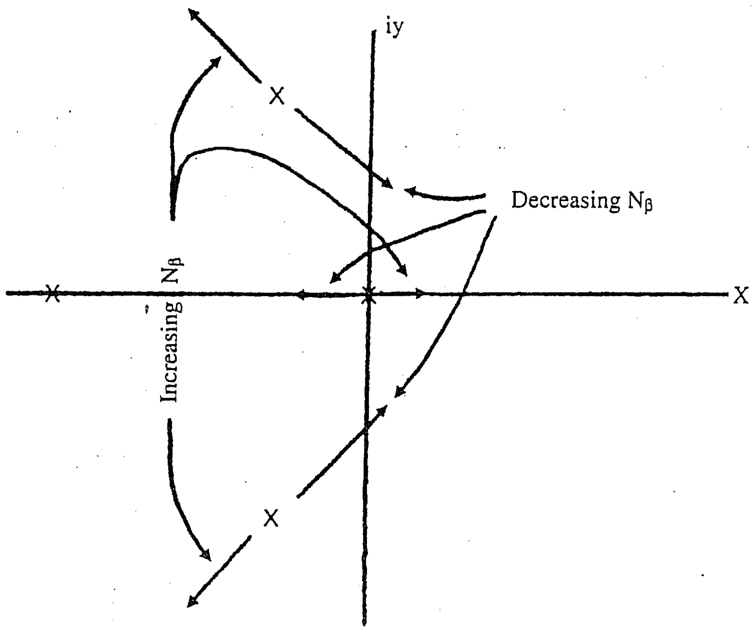


Fig. 29.5b Effects of changing N_β on the "real" airplane.[4]

the Dutch roll frequency $\omega_{n_{DR}}$ decreases and the damping ζ_{DR} increases. Increasing N_β causes the spiral root to move in the positive direction and destabilize, while the Dutch roll frequency $\omega_{n_{DR}}$ increases and the damping ζ_{DR} decreases.

In the real case where L_β and L_r are not held constant at zero, the roots move as is shown in Fig. 29.5b. This figure shows that for the real case, increasing the directional stability actually increases both frequency $\omega_{n_{DR}}$ and damping ζ_{DR} of the Dutch roll mode. It also shows how oversimplification can, on occasion, provide unrealistic answers.

Fig. 29.6 (Ref. 4) shows one of the more powerful derivatives for control of the Dutch roll mode, yaw rate damping N_r. Increasing this derivative causes both Dutch roll roots to move toward the negative real axis which has a powerful effect on damping the motion. An increase in yaw rate damping N_r leaves the roll-to-sideslip ratio ϕ/β and the Dutch roll natural frequency $\omega_{n_{DR}}$ essentially constant. An increase of N_r also causes the spiral mode to tend toward convergency.

Fig. 29.7 (Ref. 4) shows some of the effects of the derivative N_{δ_a}. As we have mentioned earlier, yaw with lateral control deflection can be either proverse or adverse. Proverse yaw is generally thought of as helpful; however, in some cases it can be detrimental to the Dutch roll mode. Therefore, in instances such as pilot in the loop, a reduction in aileron differential may actually improve Dutch roll damping.

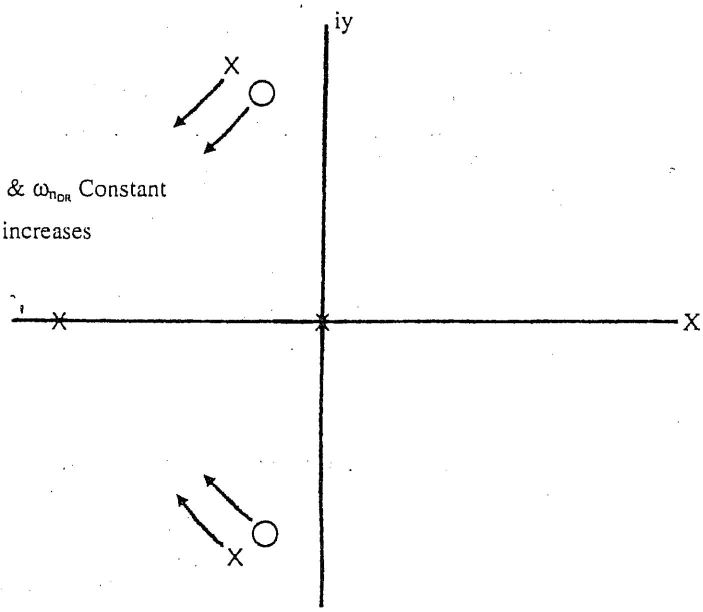


Fig. 29.6 Effects of increasing yaw rate damping (N_r) (Ref. 4).

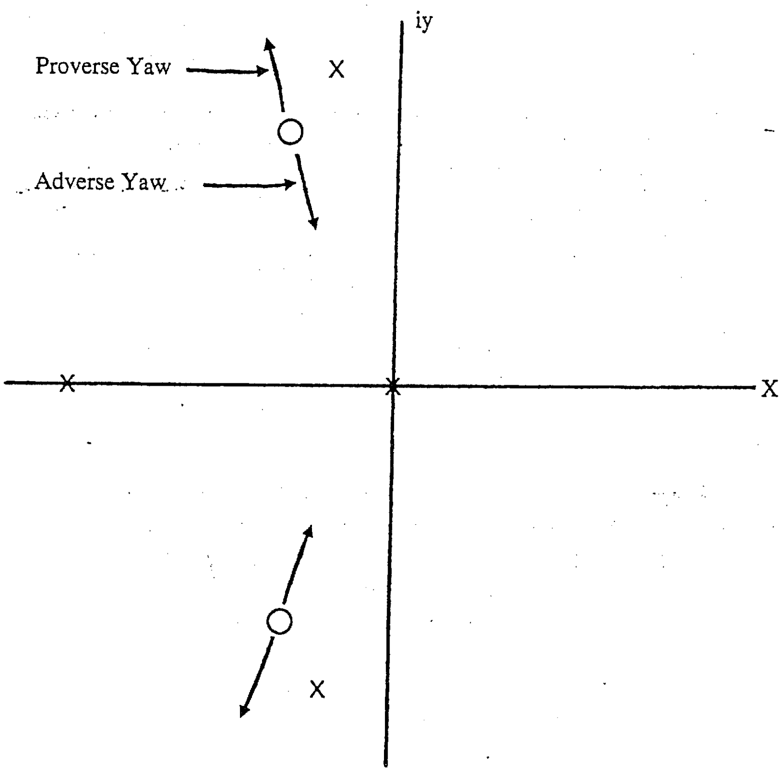


Fig. 29.7 Effects of proverse and adverse yaw.[4]

29.8 Flight Test Methods for Evaluating Dynamic Lateral-Directional Stability

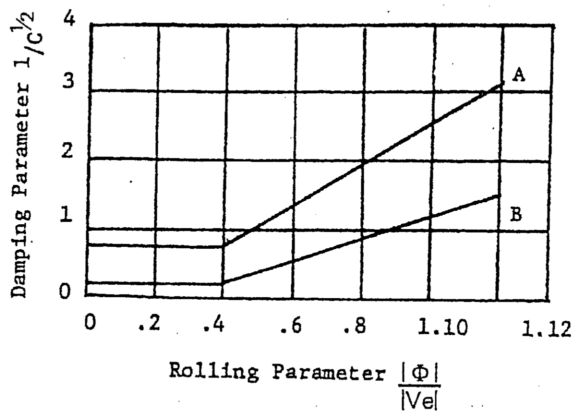
Since the roll mode is evaluated during rolling performance and lateral control power evaluations, we will only discuss methods for evaluating the spiral and Dutch roll modes.

The FARs do not contain any requirements for the spiral; however, they do contain damping requirements for the Dutch roll modes. Military specification MIL-F-8785B does contain requirements for both Dutch roll (in Sec. 3.4.1) and spiral (in Sec. 3.4.2).[8]

The requirements for Dutch roll are stated as functions of the damping parameter $1/C_{1/2}$ and the rolling parameter $|\phi|/|\beta|$; see Fig. 29.8 (Ref. 8).

The damping parameter $1/C_{1/2}$ is a function of the number of cycles for the motion to damp to half-amplitude or $C_{1/2}$. This value is a function of the Dutch roll damping.[5]

$$C_{1/2} = 0.1104 \frac{\sqrt{1 - \zeta_{DR}^2}}{\zeta_{DR}} \tag{29.4}$$



1. All Airplanes - $1/C^{1/2}$ Should be less than curve A, and Residual Motion should not be objectional.

2. Airplanes in firing or bombing configuration
$1/C^{1/2}$ Should be 1.73 or curve A whichever is higher.

3. Airplanes with Yaw Dampers
$1/C^{1/2}$ At least.24 with damper inoperative in all confiqurations. In P.A. Configuration at Least Curve B

Fig. 29.8 Military specification requirements for Dutch roll.[8]

This is approximately equal to:[5,8]

$$C_{1/2} \approx \frac{0.11}{\zeta_{DR}} \tag{29.5}$$

The damping ratio can be found from flight test data using methods described in Refs. 4 and 5. Once the damping ratio is known, we can also determine the undamped natural frequency $\omega_{n_{DR}}$ by using the damped natural frequency ω_D and the following equation:[5]

$$\omega_{n_{DR}} = \frac{\omega_{D_{DR}}}{\sqrt{1 - \zeta_{DR}^2}} \tag{29.6}$$

The rolling parameter $|\phi|/|V_e|$ is a function of the roll-to-sideslip ratio ϕ/β (Refs. 5 and 8).

$$\frac{|\phi|}{|V_e|} = \frac{57.3}{V_e}\left(\frac{|\phi|}{|\beta|}\right) \tag{29.7}$$

Both the number of cycles to damp to half-amplitude and the roll-to-sideslip ratio can be obtained from visual observation of the aircraft wing tip during the oscillation.[4] However, a more accurate method is to record the parameters of sideslip β, bank angle ϕ, and rudder deflection δ_r on an oscillograph or other tracing vs time. We may then determine such items as damping ratio, undamped natural frequency, and rolling parameter very accurately.

To excite the Dutch roll for test purposes there are several methods. The two most common methods are rudder kicks and the doublet input.

In the rudder kicks method the rudder is depressed and released rapidly, and the resultant Dutch roll oscillation observed. The problem with this method is that it also tends to excite the spiral mode causing a wing to drop.[4]

A better method is called the doublet input. In this case the rudder is moved both left and right in phase with the natural motion of the airplane and then returned to neutral or the trimmed condition before being released. This method tends to excite the Dutch roll mode well without exciting the spiral mode.[4]

The tests for the spiral mode along with the requirements are very simple. The military specification for the spiral mode is that after a small disturbance it should take at least 20 s for the bank angle to double.

To perform this test the lateral control is held fixed while the aircraft is banked 5 deg with the rudder. The rudder is then returned to trim and all controls released. Timing is started upon control release and bank angle ϕ vs time is recorded.[4]

References

[1] Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[2] Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[3] Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

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

[5] Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

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

[7] Staff, USAFTPS, "Stability and Control Flight Test Techniques, Vol. II," AFFTC-TIH-77-1, Feb. 1977.

[8] Bunker, R. S., Maj., Hendrix, G. D., Maj., et al., "Stability and Control," AFFTC-TIH-61-2004, Air Force Flight Test Center, Edwards AFB, CA, rev. Nov. 1964.

30
Lateral Control Power (Rolling Performance)

30.1 Introduction

In discussing longitudinal stability and control we were able to easily confine our discussions to motion about one axis. In dealing with lateral-directional motions this becomes much more difficult since a lateral motion will couple and cause a directional motion and vice versa.

The lateral control of an aircraft is achieved primarily by use of the ailerons. However, the rudder is also needed to trim out the yawing moments that result from the rolling moments. For the purposes of this discussion, we will simplify the responses and discuss only the single degree of freedom roll response, keeping in mind that these actions do couple.

In flying an aircraft the pilot is concerned with the lateral-control power for several important tasks. First, the most efficient way to make a heading change is to bank the aircraft. In making large heading changes such as maneuvering, the pilot is concerned with the initial response when he deflects the ailerons and the steady state, roll rate of the aircraft. For making small changes in heading or maintaining wings level after a gust upset, the pilot is concerned with only the initial response, or roll acceleration, since the airplane never reaches a steady state roll rate. We can say then that there are two items of primary importance to the pilot in the area of lateral control power: Is the lateral control sufficient to provide adequate roll acceleration for precise flying tasks, and is the steady state roll rate sufficient for the needs of the mission?

30.2 Federal Aviation Administration Regulations

Prior to FAR Part 23 Amendment 14, there were no FAA requirements for rolling performance. With Amendment 14 came FAR 23.157 Rate of Roll.

30.2.1 Federal Aviation Regulation Part 23 (Ref. 1)

There are two requirements for rate of roll in FAR 23.157, a takeoff requirement and an approach requirement.

FAR 23.157(a) lists the takeoff requirement. It states that for a favorable combination of controls it must be possible to roll the airplane from a 30 deg bank in one direction to a 30 deg bank in the opposite direction (60 deg of roll) within 5 s for airplanes of 6000 lb or less TOGW. For airplanes over 6000 lb TOGW, the time must be equal to, or less than, $(W + 500)/1300$ s, where W is in pounds, or not more than 10 s. The conditions under which this measurement must take place are covered in 23.157(b). The conditions state that: rolls must be performed in each direction with the flaps in the takeoff position and the landing gear retracted. Maximum takeoff power must be used for single engine airplanes. For multiengine airplanes the critical engine must be inoperative, the propeller feathered, and the remaining engines at maximum takeoff power. The trim speed for this test is the greater of $1.2V_{S1}$ or $1.1V_{MC}$.

FAR 23.157(c) gives the approach requirement. For airplanes of 6000 lb or less TOGW, the maximum time to roll through the 60 deg is 4 s. For airplanes above that TOGW, the requirement is (W + 2800)/2200 s, but not more than 7 s. Section (d) covers the required conditions for the test. They are: rolls in each direction, landing gear and flaps extended, all engines operating at the power required for a 3 deg descent, and the airplane trimmed at V_{REF}.

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

The advisory circular has little to say regarding FAR 23.157 that is not covered in the regulation. It does mention that "favorable combination of controls" means that the rudder can be used as necessary to achieve a coordinated maneuver.

30.3 Single Degree of Freedom Roll Response

Now let's examine the factors that affect the rolling performance of the airplane. The equation for the total rolling moment is written as follows:[3]

$$C_{l_\beta}\beta + C_{l_{\delta_r}}\delta_r + C_{l_{\delta_a}}\delta_a + C_{l_{\delta_r}}\frac{rb}{2V} + C_{l_p}\frac{pb}{2V} = \frac{1}{qSb}I_{xx}\dot{p} \tag{30.1}$$

If we consider only a single degree of freedom then the three terms associated with roll due to rudder deflection and sideslip go to zero and we are left with:[3]

$$\dot{p}\frac{I_{XX}}{qSb} - C_{l_p}\frac{pb}{2V} - C_{l_{\delta_a}}\delta_a = 0 \tag{30.2}$$

where

$\dot{p}$ = roll acceleration in radians per second
I_{XX} = moment of inertia in roll
C_{l_p} = roll damping coefficient
$pb/2V$ = helix angle or nondimensional roll rate
$C_{l_{\delta_a}}$ = control power coefficient
δ_a = aileron deflection rad

This equation can be solved to yield a time solution in roll rate that can be expressed as:[3]

$$p(t) = p_{ss}\{1 - e^{-(t/\tau_r)}\} \tag{30.3}$$

where
p_{ss} = steady state roll rate in radians per second
τ_r = roll mode time constant, or the time for the roll rate to reach 63.2% of the steady state roll rate after a step lateral control input

This equation shows that the lateral control system generates a roll rate and that a period of time is required for the bank angle to occur. This is shown in Fig. 30.1.[3]

Now let's turn our attention back to Eq. (30.2). The first term is the roll acceleration term; the second term contains roll damping and steady state roll rate; the third term is the lateral control power. If we consider that at the initial response to a lateral control input the roll damping and steady state roll rates are zero, then we can write the roll acceleration as follows:[4]

$$\dot{p}_{t=0} = \frac{C_{l_{\delta_a}} \delta_a q S b}{I_{XX}} \tag{30.4}$$

This is the slope of the roll rate vs time curve at $t = 0$ (see Fig. 30.2).[3] As was stated earlier, this item is very important as to the pilot's opinion of the lateral control of the airplane.[3] It is, however, quite difficult to measure without a roll rate gyro and time correlating instrumentation.

The second term of Eq. (30.2) is the roll damping and steady state roll rate term. If we consider that when we reach a steady state roll the acceleration is zero, then:[3]

$$\frac{pb}{2V_T} = \frac{-C_{l_{\delta_a}} \delta_a}{C_{l_p}} \tag{30.5}$$

So we can see that the steady state roll rate is a function of the lateral control power and roll damping. What do we mean by roll damping and from what physical phenomena does it occur? If we examine the spanwise lift distribution of an airplane before and after aileron deflection, we will see that the spanwise lift distribution is altered somewhat like that shown in Fig. 30.3. However, as the airplane rolls a new spanwise lift distribution is created which opposes the rolling moment due to aileron deflection. This damping moment is due to the change in effective angle of attack at any wing section due to roll; see Fig. 30.4 (Ref. 4).

This change in angle of attack can be stated as:[4]

$$\Delta\alpha_{oll} = \frac{py}{V_T} \tag{30.6}$$

where
p = roll rate in radians per second
y = the distance from the X axis of the point in question (see Fig. 30.4).

This equation shows that as the roll rate increases the damping increases, and we will eventually reach a point where the damping will equal the control power. When this occurs the steady state roll rate will have been reached.

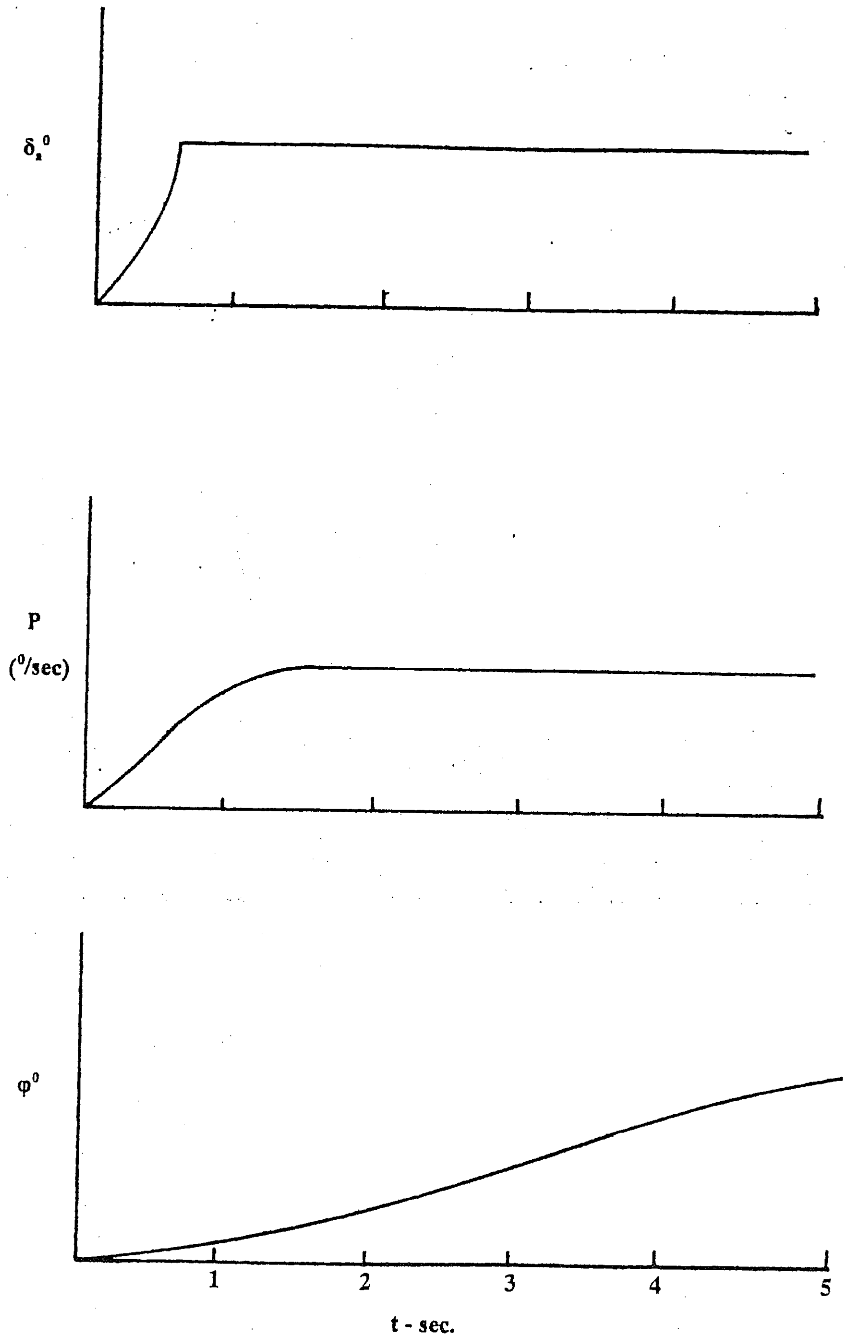


Fig. 30.1 Airplane response in roll to aileron input.[3]

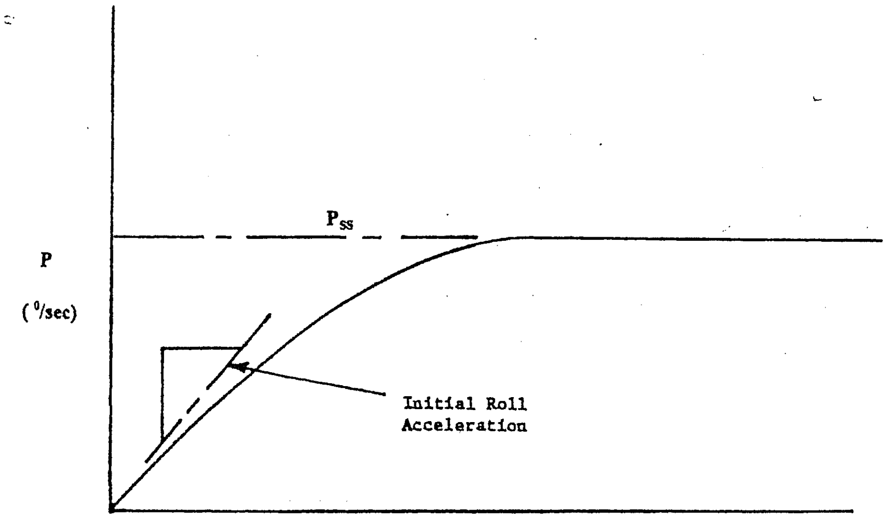


Fig. 30.2 Roll rate response and roll acceleration after a step lateral control input.[3]

The maximum steady state roll rate $pb/2V_T max$ has been used in the past as a criterion for judging lateral control power and rolling performance.[5] This single criterion for rolling performance is quite poor, since it does not take into consideration roll acceleration. Also, on airplanes of high performance and small wing span, the roll rates for acceptable values exceed the value at which a pilot can stay oriented (about 220 deg/s[6]).

In discussing the roll performance we must also discuss the roll mode time constant τ_r. This constant influences the manner in which the roll rate builds and subsides after a lateral control input.[3]

$$\tau_r = -\frac{1}{L_p} \tag{30.7}$$

As can be seen from the equation its value is only dependent on the roll damping.[3] It is generally 1 s or less and has been proposed as a criterion for measuring rolling performance. Like roll acceleration, it is difficult to measure without sophisticated instrumentation.

30.4 Influence of Various Parameters on Rolling Performance

There are a number of parameters that have an effect on the rolling performance of an airplane and the criteria that have been used to measure it. These parameters and their effects on $pb/2V_T$, $\dot{p}$, and τ_r are discussed below.

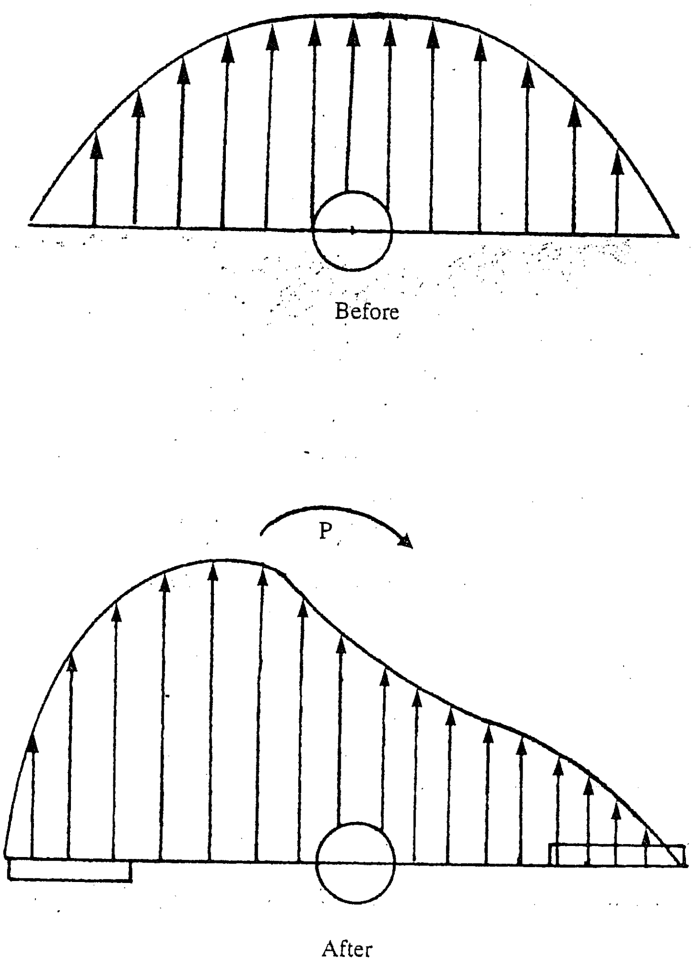


Fig. 30.3 Effects of aileron deflection upon lift distribution.

30.4.1 Lateral Control Deflection[3]

As most pilots realize, the size of the lateral control input has a direct effect on the steady state roll rate. Since steady state roll rate is directly related to the helix angle, the helix angle is also directly affected by the magnitude of lateral control input. The roll mode time constant τ_r is only a function of roll damping and is not affected by lateral control deflection. These items are shown in Fig. 30.5 (Ref. 3).

30.4.2 Moment of Inertia in Roll[3]

The term inertia can be described as a resistance to change. If we increase the inertia in roll we increase the resistance to a change in roll rate. Therefore,

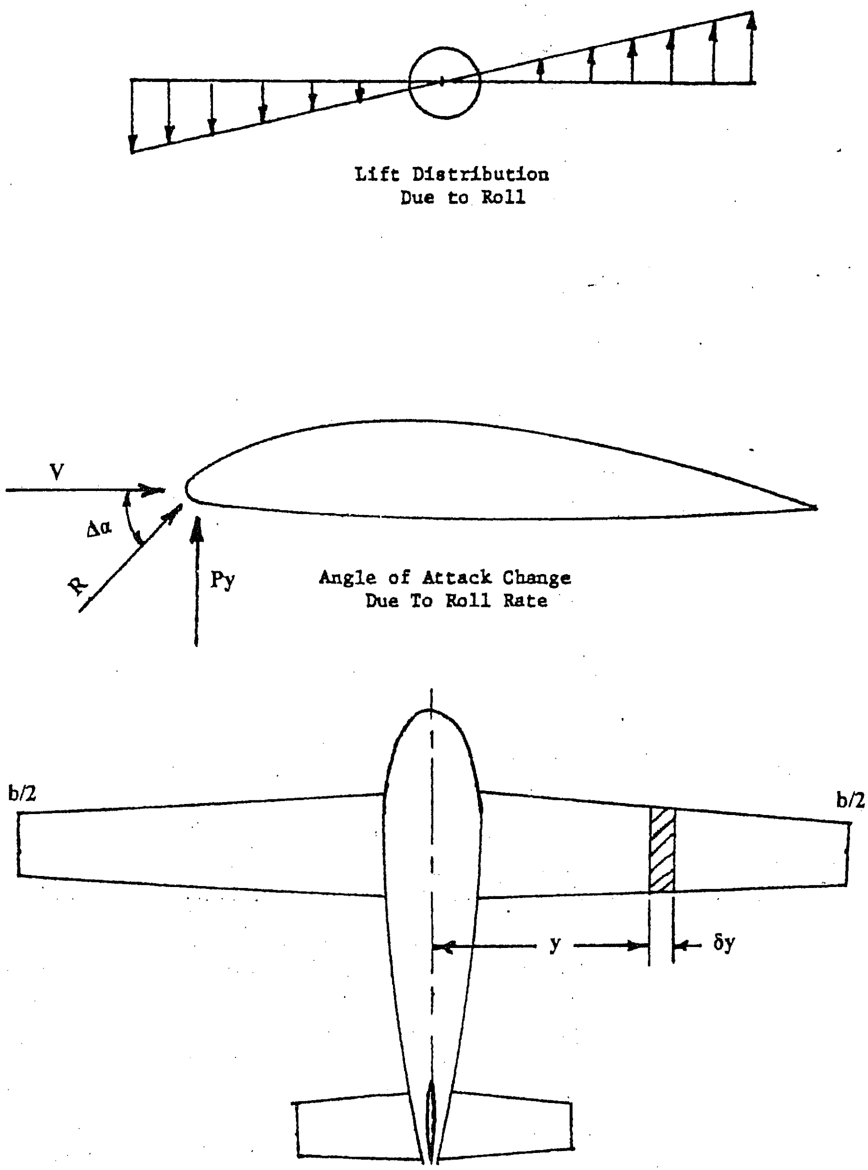


Fig. 30.4 Roll damping.[4]

if the moment of inertia in roll I_{XX} increases we would also expect the roll mode time constant τ_r to increase.

If we recall our basic physics again, we will remember that the inertia does not affect the speed or velocity that an object can achieve, only the time it takes to achieve that speed. We would then expect that the steady state roll rate

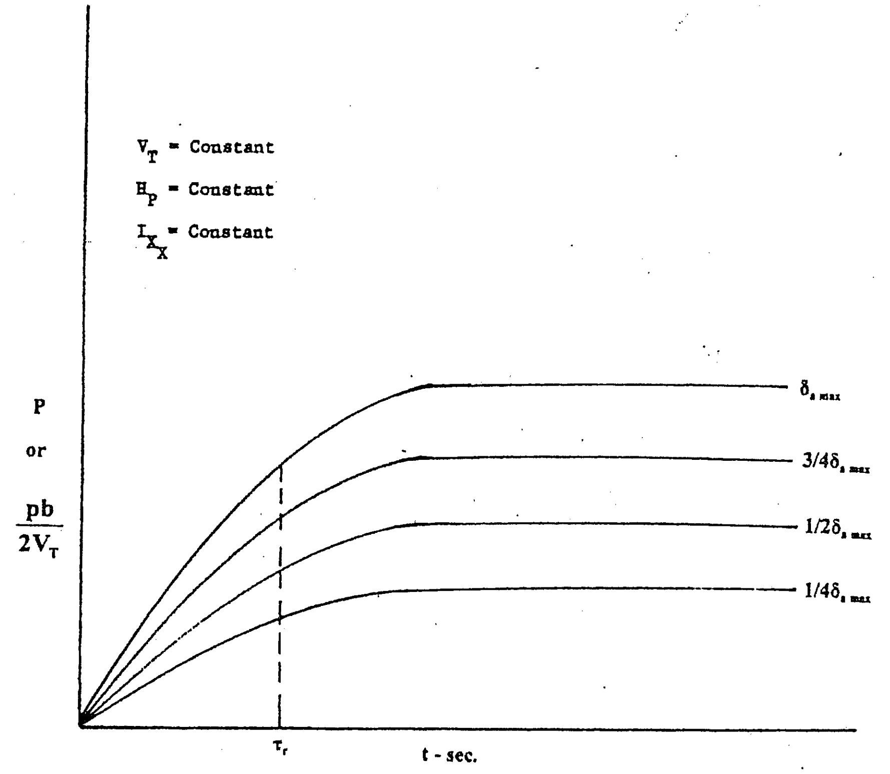


Fig. 30.5 Effects of lateral control deflection on roll response.[3]

p_{ss} would not change with an increase in moment of inertia in roll, all other things being constant.

The fact that an increase in moment of inertia in roll increases the roll mode time constant, while it does not affect the steady state roll rate or roll helix angle, is one of the reasons that helix angle values are not a good single indicator of roll performance. Fig. 30.6 (Ref. 3) shows the effects of a change in moment of inertia in roll.

30.4.3 Altitude[3]

If true airspeed V_T is held constant while altitude is varied, the steady state roll rate remains constant. The roll helix angle also remains constant, but the roll mode time constant increases with increasing altitude. These effects are shown in Fig. 30.7 (Ref. 3).

If we vary altitude while holding equivalent airspeed V_e constant, the effects are somewhat different. The roll mode time constant τ_r still increases with increasing altitude, and the steady state roll rate p_{ss} also increases because of the increase in true airspeed.[3] These effects are shown in Fig. 30.8 (Ref. 3).

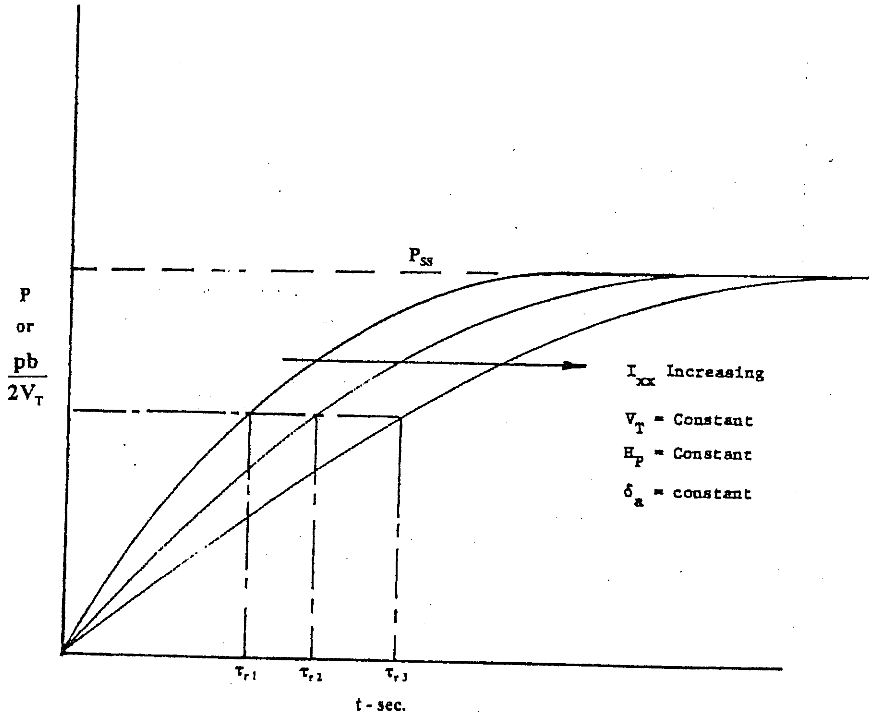


Fig. 30.6 Effects of moment of inertia in roll on roll response.[3]

30.4.4 Airspeed[3]

As we have stated earlier, the steady state roll rate is affected directly by true airspeed. This is shown in Fig. 30.9 (Ref. 3). The helix angle $pb/2V_T$ is not affected by changes in true airspeed since true airspeed appears in the denominator of the helix angle term and cancels out the roll rate increase.

The roll mode time constant varies inversely with true airspeed. These effects are shown in Fig. 30.10 (Ref. 3).

The above effects apply for all control systems up to the limit of pilot effort. Pilot effort limits occur in reversible, unboosted control systems when the pilot can no longer apply enough control force to obtain maximum control deflection. In a lateral control system where a pilot effort limit is reached, the steady state roll rate will not continue to increase with an increase in true airspeed. This is shown in Fig. 30.11 (Ref. 5). The roll helix angle will also decrease after this limit has been reached.

30.5 Lateral Control in the Real Case[3–5]

The lateral control responses we have just discussed were for a rigid airplane with a single degree of freedom. This is a highly idealized case that we used to simplify the roll response. The real airplane has six degrees of free-

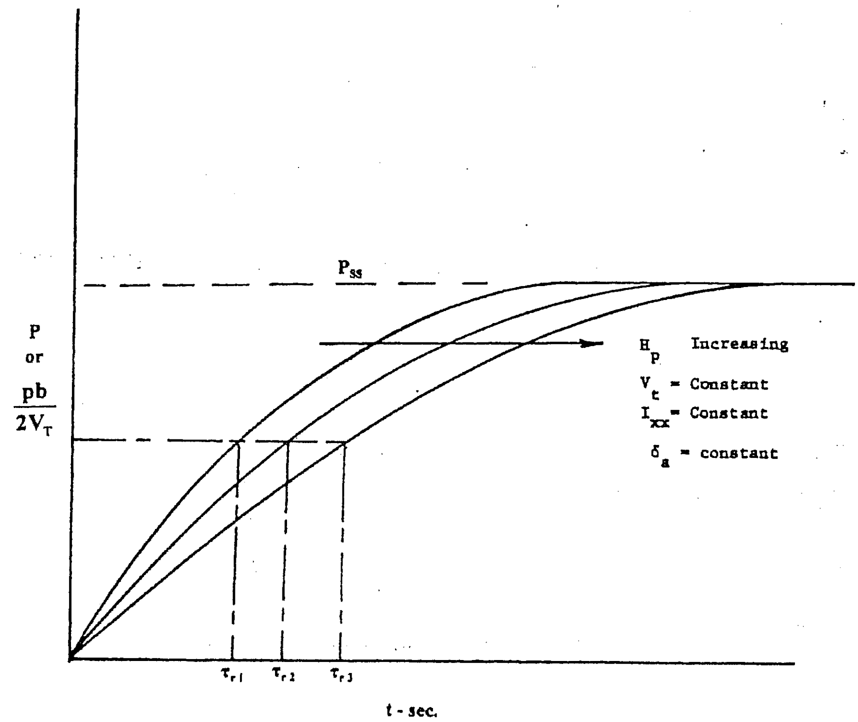


Fig. 30.7 Effects of altitude on roll response at a constant true airspeed.[3]

dom and is not always rigid; therefore, it may respond somewhat differently. The differences of the real airplane's response in roll is primarily caused by three influences: 1) Dutch roll; 2) roll coupling; and 3) aeroelastic effects.

30.5.1 Dutch Roll Influences

The Dutch roll influences on the rolling performance are variable depending upon the nature of the Dutch roll.

If the airplane has a Dutch roll with small roll-to-sideslip ratio $|\phi|/|\beta|$ or a snaking type of Dutch roll, the rolling performance could be affected in one of two ways. First, if the small roll-to-sideslip ratio were caused by strong directional stability with low values of lateral stability, and adverse or proverse yaw, then the effects on rolling performance would be small. This airplane's rolling performance would closely approximate the one degree of freedom case. Second, if the low roll-to-sideslip ratio were caused by high adverse yaw and high yaw due to roll rate C_{n_p} with weak lateral and directional stability, then the roll handling qualities would be degraded. Although the aircraft's steady state roll rate and roll acceleration might be quite acceptable, there would be

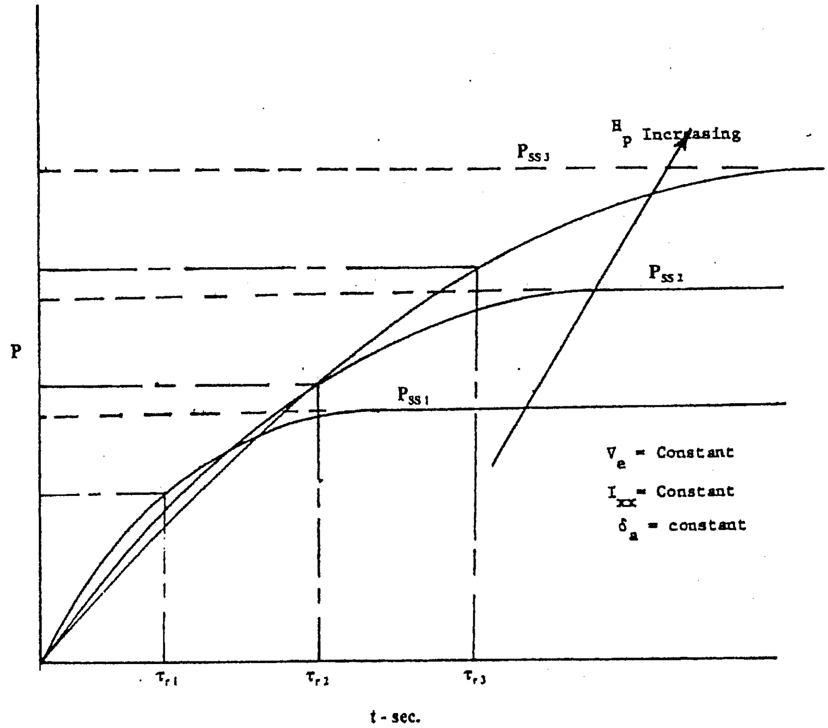


Fig. 30.8 Effects of altitude on roll response at a constant equivalent airspeed.[3]

large sideslip excursions with roll. The pilot would experience lag of the nose in turn entries and other objectionable responses.

If the airplane's Dutch roll motion is the type with a large roll-to-sideslip ratio caused by weak directional stability, strong lateral stability, and strong adverse or proverse yaw, then rolling performance will be degraded. In this case, large sideslip excursions in the direction of roll occur. This effect can cause roll oscillations or, in more severe cases, roll reversals.

Both of the cases of adverse effects on rolling performance can be counteracted by the pilot if he/she precisely coordinates rudder input with the roll to counteract the sideslip. This requirement places another burden upon the pilot in critical tracking tasks where workload is already high. Therefore, it would be desirable not to have such a requirement.

30.5.2 Roll Coupling Influences

Another influence on rolling performance of the real airplane is the pitching and yawing motions during roll caused by roll coupling. There are three items that make up roll coupling. Although these items do not occur singularly

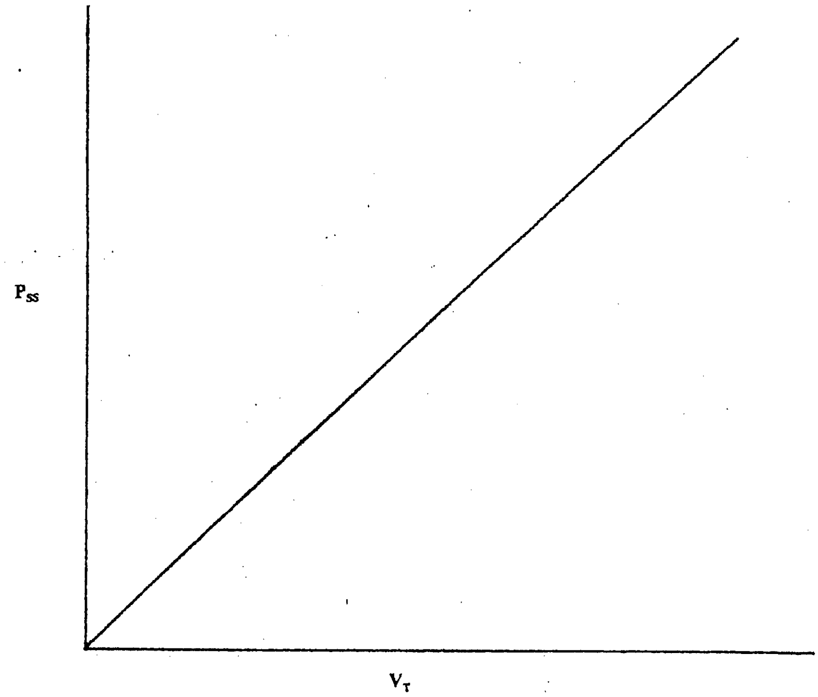


Fig. 30.9 Steady state roll rate vs true airspeed.[3]

during a roll, in order to simplify their study we will consider them separately. They are:

1) inertia coupling
2) kinematic coupling
3) I_{xz} effect

Inertia coupling occurs when an airplane is both rolled and yawed, or rolled and pitched simultaneously. When such motion occurs the resultant angular velocity vector is not aligned with any of the principal aircraft axes, and the aircraft then rotates about this resultant vector. If we represent the aircraft's mass distribution by concentrated masses (dumbbells) as is shown in Fig. 30.12,[3] and our aircraft has most of its mass along the fuselage, then a simultaneous roll and yaw in the same direction will cause the airplane's nose to pitch up. If the airplane rolls and yaws in opposite directions (right roll, left yaw), then the pitching moment will be nose down.

If the airplane is rolled and pitched simultaneously as is shown in the bottom of Fig. 30.12, then a yawing couple is developed. In this case, when the nose is pitched up the yaw develops in the opposite direction to the roll.

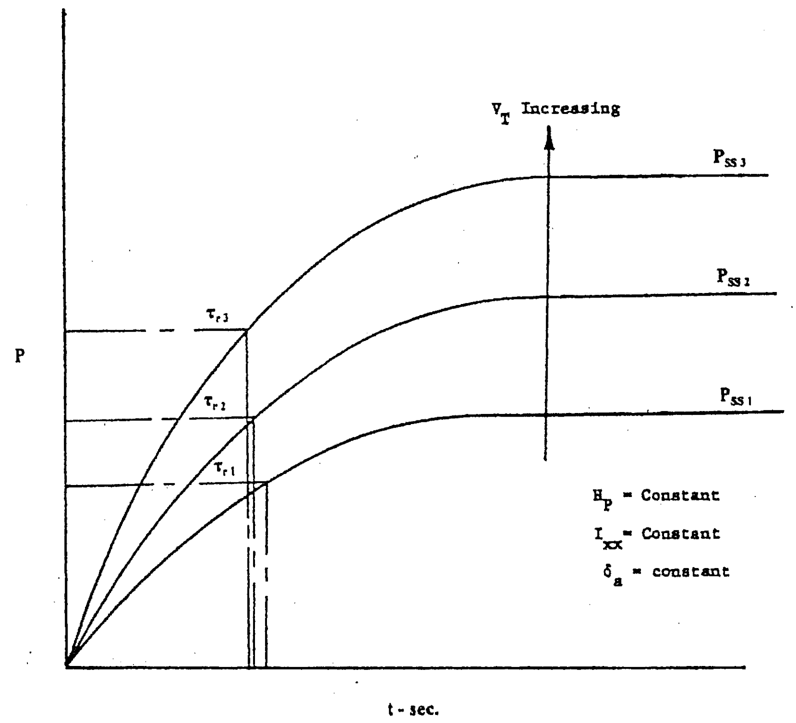


Fig. 30.10 Effects of true airspeed upon roll response.[3]

A close examination of the inertia coupling phenomenon will show that its effects can be reduced to zero if the moments of inertia about all axes are equal. If the yawing and pitching moments of inertia are much greater than the rolling moment of inertia, then the effects of inertia coupling are more pronounced. Light aircraft and multiengine airplanes have their masses well distributed and, as a result, do not suffer severe inertia coupling. Modern jet fighters have most of their mass in the fuselage and do suffer from inertia coupling.

The second effect in roll coupling, kinematic coupling, is essentially the geometric interchange between angle of attack and sideslip angle that occurs when an airplane rolls. This effect is shown in Fig. 30.13 (Ref. 3). The problem that occurs is that the longitudinal and directional stabilities of the airplane oppose this interchange. If the roll rate is high enough that the frequency of this cyclic interchange couples with the natural frequency of the airplane in pitch or yaw, then divergence in pitch or yaw can occur.

For most airplane configurations the products of inertia, I_{XY} and I_{YZ}, are zero. This is not true for the product of inertia, I_{XZ}, which causes the third effect in roll coupling. When I_{XZ} is unequal to zero, the X axis of the airplane is not aligned with the principal inertial axis, or the mass distribution about the

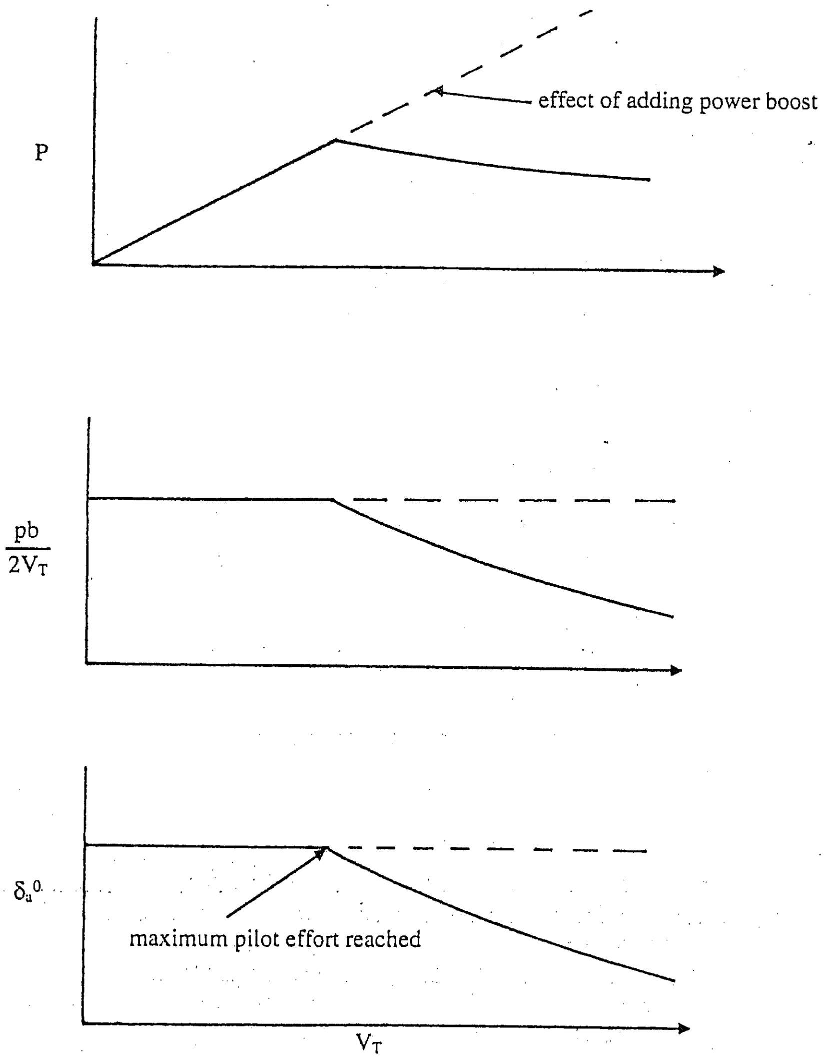


Fig. 30.11 Effects of reaching maximum pilot effort on roll response.[5]

X axis is not symmetrical. This is true for most airplanes. Since the mass is not distributed symmetrically about the X axis, it acts as a flyweight when the airplane is rolled tending to make the airplane diverge. This effect is shown in Fig. 30.14 (Ref. 3). The effect is counteracted by the natural stability of the airplane, but should the stability level be low and the roll rate high the problem is intensified.

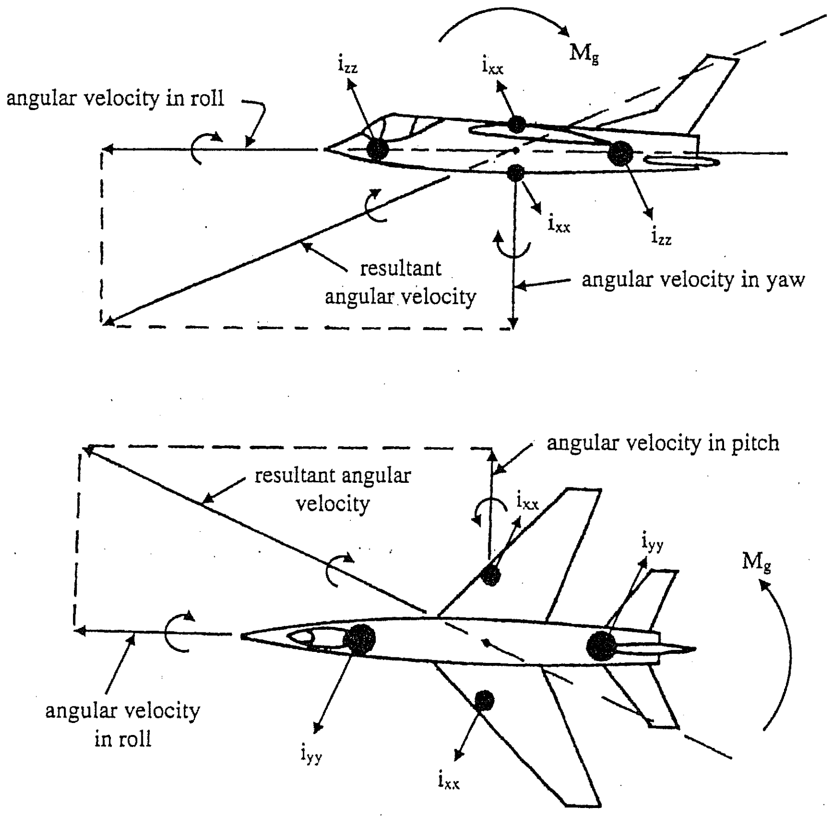


Fig. 30.12 Causes of inertia coupling.[3]

30.5.3 Aeroelastic Effects

The third influence on the real airplane's rolling performance is aeroelastic effects. In their most severe case, aeroelastic effects cause what is known as aileron reversal. The reason that aileron reversal can occur is that the wing is not rigid, as we assumed earlier, but is, in fact, flexible.

When we deflect the aileron as is shown in Fig. 30.15 (Ref. 5), a force is created that tends to twist the wing about its elastic axis. This twisting causes the wing angle of attack to either increase or decrease depending upon the direction of aileron deflection. In either case, this change in angle of attack causes a change in lift that opposes the roll. As airspeed increases the twisting force created by the aileron increases, causing a larger change in wing angle of attack and a larger opposition to the roll. Eventually a point will be reached where rolling moment created by the change in angle of attack will overcome the rolling moment created by the aileron, and aileron reversal will occur. If the airplane has swept-back wings this problem is magnified, since the bending of a swept-back wing causes the effective angle of attack to be reduced on

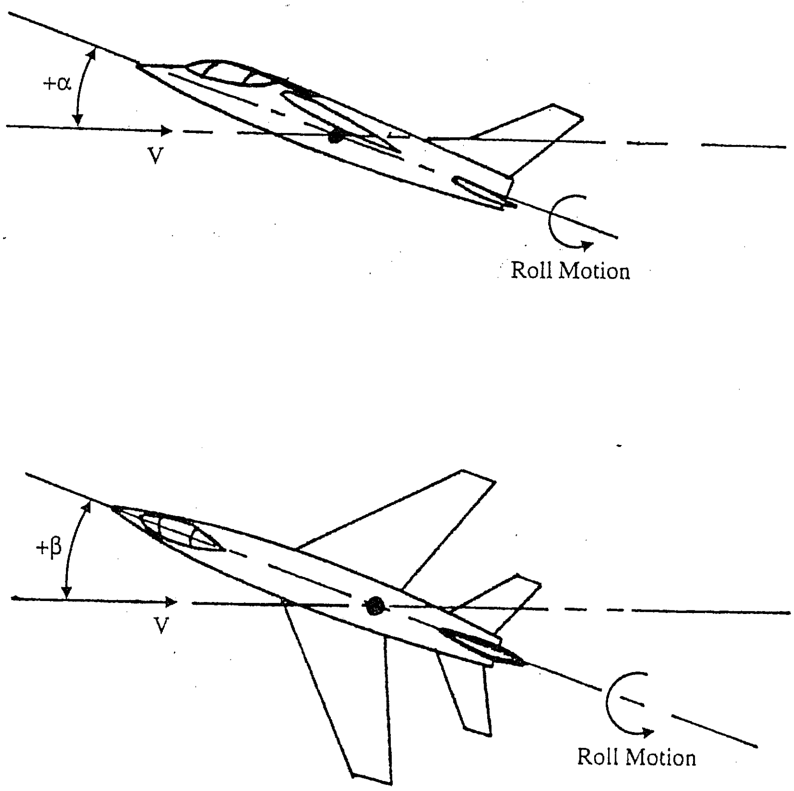


Fig. 30.13 Kinematic coupling during roll.[3]

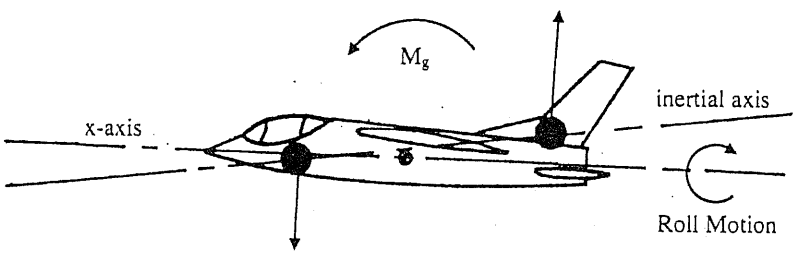


Fig. 30.14 Roll I_{XZ} effects.[3]

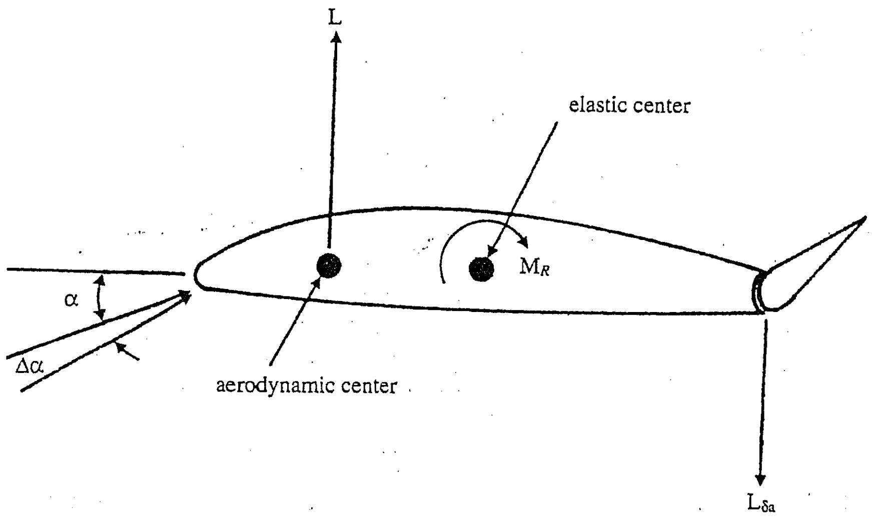


Fig. 30.15 Wing twist and angle of attack change due to aileron deflection.[5]

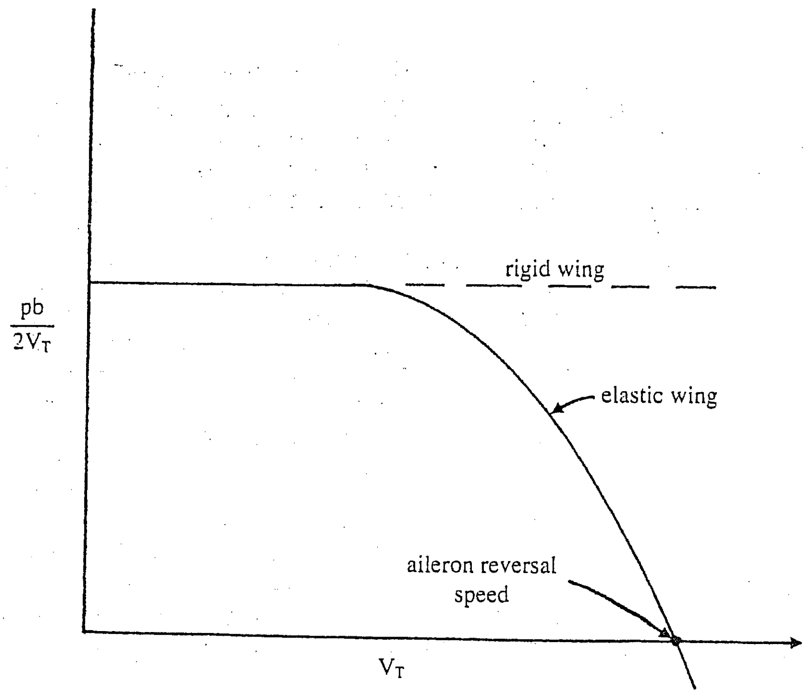


Fig. 30.16 Effects of aeroelasticity on the roll helix angle.[5]

the up-going wing. Aeroelastic effects on the roll helix angle are shown in Fig. 30.16 (Ref. 5).

30.6 Other Methods of Roll Control

Our discussions on lateral control power have so far dealt with the most common form of lateral control, ailerons.

Several other methods of lateral control have been devised, but only one of these has found much use in full-scale aircraft. This is the spoiler. Spoilers provide lateral control by spoiling lift on one wing while the other wing operates at its original C_L. Spoilers have been used on some airplanes in conjunction with ailerons to increase roll rate.

Spoilers suffer from two problems that require careful system design. First, at low angles of deflection the flow tends to reattach behind the spoilers. This causes a lateral control deadband or, in certain cases, roll reversal at low spoiler deflections. The second problem is that spoilers tend to have very low hinge moments, and as a result require some artificial "feel" system such as a centering spring to give the pilot some control force.

It is for the above two problems that spoiler lateral control systems are sometimes combined with small trim ailerons.

References

[1]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[2]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

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

[4]Langdon, S. D., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM-No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

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

[6]Bunker, R. S., Maj., Hendrix, G. D., Maj., et al., "Stability and Control," AFFTC-TIH-64-2004, Air Force Flight Test Center, Edwards AFB, CA, rev. Nov. 1964.

31
Directional Control

31.1 Introduction

The requirements for directional control can essentially be divided into control requirements for single-engine airplanes and those for multiengine airplanes. The reason for this is that the directional control requirement is nearly always set on the multiengine airplane by the engine-out case, while on the single-engine airplane it may be set by some other condition such as cross-wind landing.

Directional control is essentially a function of the rudder; however, since lateral-directional motions cannot be separated, other functions also play a part. But in order to simplify the discussion, we shall confine ourselves to the rudder term $C_{N_{\delta_r}}$.

31.2 Federal Aviation Administration Regulations

31.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

CAR 3.106 is the general controllability requirement for all airplanes. It requires that the airplane shall be satisfactorily controllable in all flight conditions with and without power. It also requires a smooth transition from one flight condition to another without an exceptional degree of skill or strength on the part of the pilot. It gives the strength limits for yaw as 150 lb for temporary application and 20 lb for prolonged application.

Directional control for all airplanes is covered under "Ground and Water Characteristics" in CAR 3.145. It requires that there be no uncontrollable (ground) looping tendency in 90 deg crosswinds up to a velocity equal to $0.2V_{S0}$ at any speed that the aircraft may be expected to operate on ground or water. This regulation also requires that all land planes be satisfactory controllable without exceptional pilot skill in power-off landings at normal landing speeds without the use of brakes or engines to maintain a straight path.

Multiengine airplane directional control requirements are covered in CAR 3.110(b) and CAR 3.111. CAR 3.110(b) requires that with the wings held level within 5 deg it should be possible to execute sudden heading changes up to 15 deg in both directions without dangerous characteristics being encountered. The heading change may be less than 15 deg if the rudder force reaches the limit specified in CAR 3.106 (150 lb). This requirement must be demonstrated at an airspeed of $1.4V_{S1}$ or V_Y with the critical engine inoperative and the remaining engines at MCP. It shall be accomplished both landing gear up and extended with the flaps in the most favorable climb position. The propeller on the inoperative engine shall be in a minimum drag condition. The c.g. shall be at the most aft loading.

CAR 3.111 discusses minimum control speed V_{MC} for multiengine airplanes. CAR 3.111(a) requires that a minimum speed be determined where it shall be possible to recover control of the airplane when any one engine is suddenly made inoperative. It should be possible to maintain straight flight at that speed (with the engine still inoperative) with either zero yaw or with a bank into the good engine not to exceed 5 deg This minimum speed shall not exceed $1.2V_{S1}$ with takeoff or maximum available power on all remaining engines, flaps in the takeoff position, the landing gear retracted and a most aft c.g. The rudder force may not exceed the limits specified in CAR 3.106 (150 lb), and it shall not be necessary to retard the throttle on the remaining engines. During the recovery to straight flight, the airplane may not assume a dangerous attitude, and exceptional piloting skill should not be required to prevent a heading change of more than 20 deg after the failure.

31.2.2 Federal Aviation Regulation Part 23 (Ref. 2)

FAR 23.143 the general controllability requirement of FAR Part 23 reads much the same as CAR 3.106 in both the early versions of FAR Part 23 and the current version. Only the go-around condition of flight was added in the current version of FAR Part 23.

FAR 23.233 covers the directional control under the "Ground and Water Handling Characteristics" of FAR Part 23. This regulation is also essentially the same as CAR 3.145 except for wording changes in the current version.

FAR 23.147(b) gives the directional control requirements that are covered in CAR 3.110(b). The early versions of FAR Part 23 has essentially the same wording as provided in CAR 3. In later versions of FAR Part 23 this requirement has been moved to 23.147(a) and some additional requirements have been added. Included is a requirement to be able to safely control the airplane directionally at any speed or altitude within the operating envelope by use of the rudder only. This requirement is only for all engines operating and is covered in FAR 23.147(c).

FAR 23.149 states the minimum control speed requirements covered in CAR 3.111 of the older regulation. Early versions of FAR 23.149 read essentially the same as CAR 3.111. The current version of the regulation (as amended by Amendment 23-21) requires that the weight and c.g. be in the most unfavorable case while the early version of the regulation only required a most aft c.g. and did not address the weight issue. The current regulation in item (c) also requires that V_{MC} be determined in the approach configuration for all airplanes except those reciprocating engine airplanes of 6000 lb or less maximum weight. For this determination, the airplane should be trimmed at an approach speed of V_{REF} at an approach gradient equal to the steepest gradient used to determine landing distances with the propellers in the recommended approach position and the landing gear and flaps in the landing position.

A minimum speed to intentionally fail the critical engine is required by 23.149(d) of the current FAR Part 23. This speed is called the safe one engine inoperative speed and is abbreviated V_{SSE}.

In the current regulation the applicant has the option of determining a minimum control speed on the ground, V_{MCG}, as part of complying with the takeoff speeds requirement of FAR 23.51. If this is done, it must be accomplished with rudder only (nosewheel steering must not be used) and the airplane must not deviate from the runway centerline more than 30 ft after failure of the engine. This is accomplished in the takeoff configuration, trimmed for takeoff with maximum available power on the operating engines at the most unfavorable condition of weight and c.g.

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

With respect to FAR 23.143, the general controllability requirements the advisory circular discusses what is meant by temporary and prolonged application of the control forces discussed in the regulation. It defines temporary application as the time necessary for the pilot to take action to relieve the control force, such as retrimming the airplane or reducing power. Prolonged application would be for some application that could not be trimmed out.

The advisory circular also discusses spring devices, such as rudder centering springs, in this section. It states that if a spring device is installed in a control system, the airplane must not have any unsafe flight characteristic without the use of the spring device. If the spring device can be shown by test simulating service conditions that it is reliable, then safe flight without the spring device does not have to be demonstrated.

The directional control section of FAR 23.147, according to the advisory circular, is intended as an investigation for dangerous characteristics during sideslip. Rudder lock and loss of directional control due to blockage of airflow over the vertical tail are examples of these characteristics. Compliance with the regulation is shown by rudder stop being reached before 15° of heading change, or reaching the 150-lb force limit. The advisory circular comments that the rudder stop is a more well-defined limit than is the 150-lb force limit.

In the advisory circular, FAR 23.233 the directional stability and control requirement crosswind component is clarified by stating that the required values are a minimum requirement and that higher values may be demonstrated.

Regarding ground loops, the advisory circular states that the regulation does not preclude an airplane from having a tendency to ground loop in crosswinds, provided that the pilot can control this tendency with use of the aerodynamic controls, brakes, engines, or combinations thereof. However, procedures for controlling these tendencies should be included in the airplane flight manual.

Controllability during taxi should allow taxiing at the "maximum demonstrated crosswind" value according to the advisory circular.

For minimum control speed (FAR 23.149), the advisory circular contains over four pages of discussion. Highlights of this discussion are included here. First, the advisory circular states that because different configurations will provide different values of V_{MC}, the highest of these values should be used for instrument markings and the Airplane Flight Manual.

During the flight test for V_{MC}, the propeller on the inoperative engine should be windmilling unless the propeller has an "autofeather system," which feathers the inoperative propeller without pilot action. V_{MC} tests should be conducted with both rudder and aileron travel set to the minimum tolerance. In some cases, aileron travel may determine V_{MC}. If maximum aileron travel determines V_{MC}, the airplane may not be capable of further maneuvering and may not meet the controllability requirements. The flight test should be conducted at aft c.g. since that provides the shortest moment arm for the rudder. Minimum weight is usually critical for these tests, because the helpful effects of banking into the good engine are minimized at light weight. Flight tests for this determination should be conducted with the maximum allowable fuel unbalance. Determining which engine is the critical engine is normally determined prior to actually determining V_{MC}. The requirement to maintain straight flight may be accomplished with wings level or with 5 deg of bank into the good engine. Normally, 5 deg of bank is used for this test since it provides near-zero sideslip and a lower V_{MC}. Tests conducted at altitude should be extrapolated to sea level conditions since power and propeller efficiency are higher at sea level conditions. If deicing equipment is to be offered as an option, the test should be repeated with this equipment installed. The test procedures covered in the advisory circular are essentially those given later in this chapter.

31.3 Directional Control—Single-Engine Airplanes

There are four items that may set the directional control requirements for a single-engine airplane. They are:

1) rudder required to reduce sideslip to zero during a turn
2) rudder required to counteract P-factor and slipstream rotation during takeoff
3) rudder required to maintain a constant heading during a crosswind takeoff and landing
4) rudder required for spin recovery

The directional control requirement for turn coordination is created by adverse yaw. Adverse yaw is usually not great on a well-designed aileron system and usually does not set the rudder power requirement on a single-engine airplane. There are cases, however, where adverse yaw may create a critical requirement. An example is a short takeoff and landing (STOL) airplane that droops its ailerons with flap deflection. Adverse yaw increases as lift coefficient increases so at low speeds this type of airplane would require considerable rudder power to coordinate turns.

If we combine items two and three we have a directional control requirement that is quite severe for single-engine airplanes and may design the rudder. P-factor, which is unequal thrust from opposite propeller blades, and slipstream rotational effects (see Fig. 31.1) are strongest at very low airspeeds. When these effects are combined with a strong left crosswind, considerable rudder power is required for directional control. The crosswind landing case is also quite severe since, in this case, we must bank the airplane so as to generate side-force to counteract wind drift while maintaining heading with the

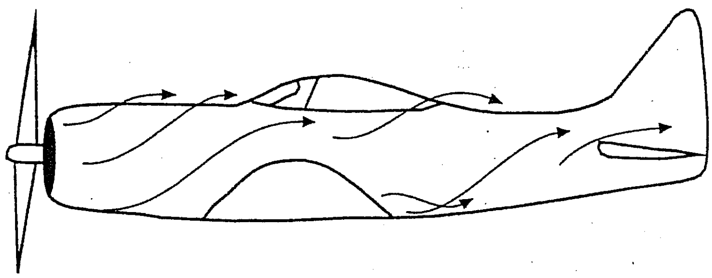


Fig. 31.1 Slipstream rotation of propeller-driven airplane.

rudder. During landing the dynamic pressure at the vertical tail is at one of its lowest values requiring large values of rudder side-force coefficient.

The post-stall gyration and fully developed spin may also design the rudder. Since the dynamic pressure over the rudder is very low and the yaw rate high, the spin is one of the more critical directional control requirements for single-engine airplanes.

31.4 Directional Control—Multiengine Airplanes

Since multiengine airplanes are not spun and requirements for spin recovery of multiengine airplanes are nonexistent, the most critical condition for directional control in these airplanes is the engine-out case. If we examine a multiengine airplane with one engine failed (Fig. 31.2) and the propeller feathered, we can see that the operative engine generates a yawing moment.

This moment (N_T) may be expressed by the equation:

$$N_T = F_N Y_P \tag{31.1}$$

where

F_N = the thrust developed by the operative engine in pounds
Y_P = distance from center of gravity to thrust vector in feet

If we express this equation in coefficient form then:

$$C_{N_T} = F_N Y_P / qSb \tag{31.2}$$

and if for level flight we substitute ($qS = W/C_L$) then:

$$C_{N_T} = C_L F_N Y_P / Wb \tag{31.3}$$

This equation can be used directly if the airplane has jet engines. However, if it is propeller-driven we must first calculate the thrust.

$$F_{N_{prop}} = 550 \eta_P BHP / V_T \tag{31.4}$$

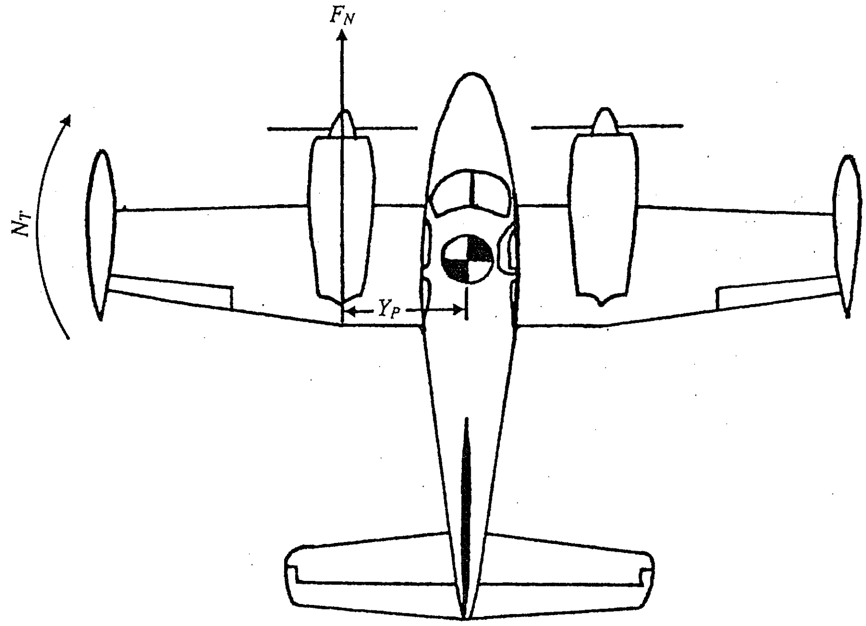


Fig. 31.2 Yawing moment due to asymmetrical power.

where
550 = horsepower constant (ft-lb/s)
η_P = propeller efficiency
BHP = brake horsepower

By substitution of this equation in Eq. (31.3) we have:

$$C_{N_{T_{prop}}} = 550\eta_P(BHP)Y_P/V_TqSb = 550\eta_P(BHP)C_LY_P/V_TWb \qquad (31.5)$$

If we hold horse power and propeller efficiency constant we can obtain a plot of C_{N_T} vs V_T (see Fig. 31.3).

If we superimpose on that plot the C_N available for maximum rudder deflection, we can determine the minimum true airspeed at which we can maintain directional control with one engine feathered. This is not necessarily the minimum speed at which you can control the airplane during an engine failure. The reason for this is that when an engine fails the propeller on that engine is not feathered and creates a negative thrust or drag. Determining this drag will be difficult due to unknown engine failure modes and other variables such as propeller efficiency. Therefore, the engine-out minimum control speed called V_{MC} is determined by flight test. In design the number is usually arrived at by taking the number obtained and adding some constant to it to account for the dynamic condition.

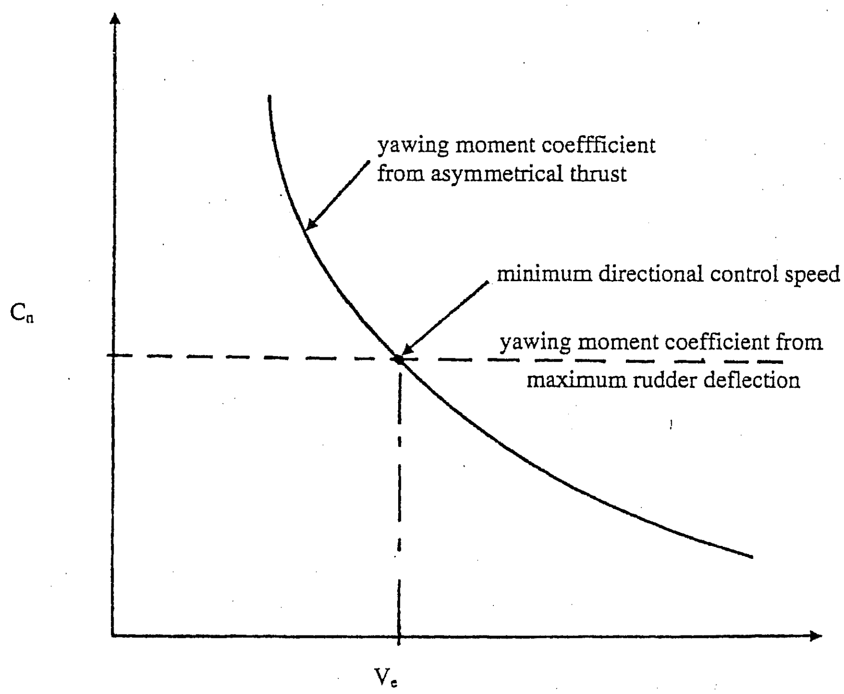


Fig. 31.3 Estimation of minimum control airspeed.

By rewriting the yawing moment equation for steady sideslips to account for the engine-out we have:

$$C_L F_N Y_P / Wb + C_{N_\beta} + C_{N_{\delta_r}} \delta_r = 0 \tag{31.6}$$

We can solve this equation and the other steady sideslip equations for numerous items. A list follows.

1) Rudder required for zero sideslip
2) Bank angle for zero sideslip
3) Rudder required for zero bank angle
4) Sideslip at zero bank angle
5) Sideslip for zero rudder deflection
6) Bank angle for zero rudder deflection

It is interesting to note that a lateral control problem also exists for the engine failure case due to unequal lift on the wings. In some rare instances this case may set V_{MC}.

As can be seen from the previous equations, the engine-out minimum control speed is critical where the thrust is the highest (sea level takeoff). However, we cannot test at this condition due to physical and safety limitations. This means that we must test at some other condition and extrapolate the data to sea level.

In addition, studies such as that conducted in Ref. 10 have shown that the minimum control speed increases as weight decreases and as c.g. moves aft. Then, in order to determine the maximum operational value of V_{MC}, we will want to conduct the test with the aircraft at the most aft c.g. and the minimum takeoff weight. Since we may not always be able to obtain these conditions, it may be necessary to test at several other weight and c.g. combinations and extrapolate the data to the desired conditions.

Since the number of variables involved in the V_{MC} test is large, we will need to hold some constant while we work the extrapolations of the others. Aircraft weight and c.g. can be held nearly constant without much difficulty, since test duration is fairly short and weight change minimal. It is also relatively easy to hold altitude constant.

We then can perform the basic test by failing the engine and varying the airspeed while recording sideslip β, aileron deflection δ_a, and rudder deflection δ_r. Since we know the maximum deflections of the control surfaces, we can find the V_{MC} for that condition by extrapolating the data as shown in Fig. 31.4. If we continue to hold weight and c.g. constant and vary altitude while repeating the basic test, such as is shown in Figs. 31.5 and 31.6, then we can generate a plot to extrapolate the V_{MC} for that weight and c.g. to sea level. This is shown in Fig. 31.7. If we repeat the above outlined procedure for other weights and centers of gravity, we can generate a plot such as is shown in Fig. 31.8, which allows us to determine the V_{MC} at the condition we desire.

It is possible that control forces may set the minimum control speed. If this is suspected then control forces should be recorded along with the above data and a similar extrapolation performed.

The minimum control speed just described is a static V_{MC}. An actual engine failure is a much more dynamic situation, and the dynamic V_{MC} may be considerably higher than the static case, depending upon a number of factors including pilot recognition time and pilot skill. It is also obvious that the dynamic case requires more control than does the static case and should really establish the V_{MC}. Why, then, do we go through the previous exercise if it does not set V_{MC}?

Due to the large number of variables involved in the dynamic V_{MC}, especially those of pilot skill and recognition time, the amount of scatter in dynamic data is too great to make an accurate extrapolation. There is also a great deal of hazard involved in testing the dynamic case without first knowing the static V_{MC}. Therefore, in order to determine the dynamic V_{MC} the test pilot must first find the static V_{MC}, and then through qualitative evaluation determine the increment of airspeed necessary to handle the dynamic engine failure. This value of V_{MC} is then the one published in the flight manual.

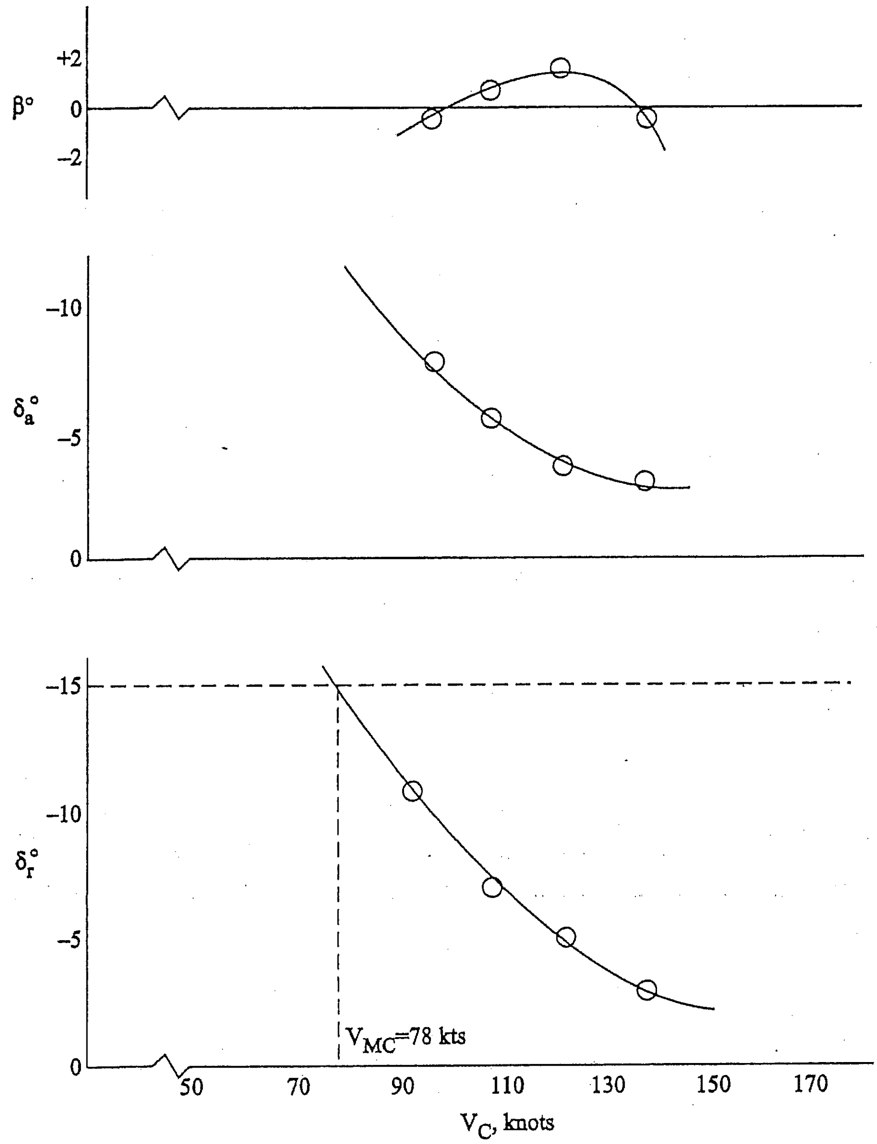


Fig. 31.4 Engine-out minimum control speed, Cessna 310A, 2900 ft altitude.

31.5 Methods for Determining $C_{N_{\delta_r}}$

Since the directional control for single- or multiengine aircraft is a function of the rudder effectiveness coefficient $C_{N_{\delta_r}}$, it is useful to determine its value. Before conducting V_{MC} testing, it may be worthwhile to construct a plot like that of Fig. 31.3 to get a preliminary estimate of V_{MC}. To do this one would

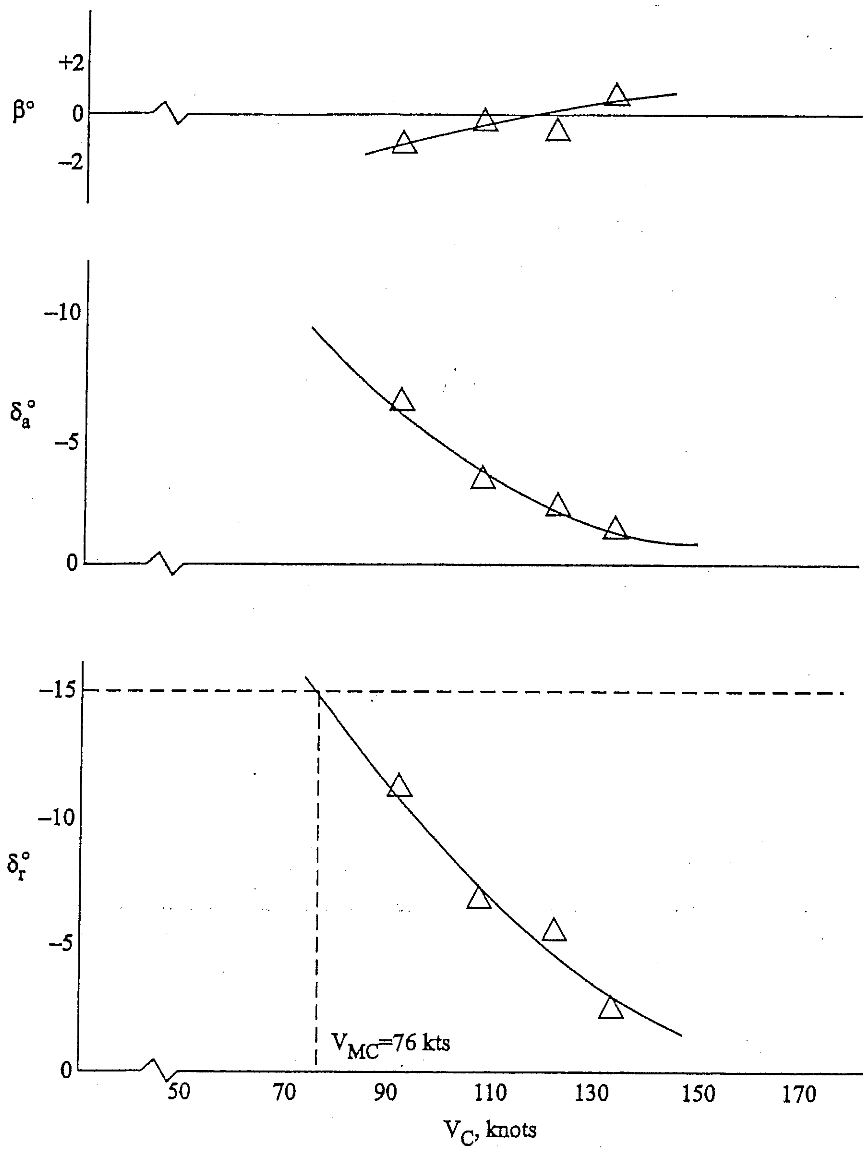


Fig. 31.5 Engine-out minimum control speed, Cessna 310A, 4500 ft altitude.

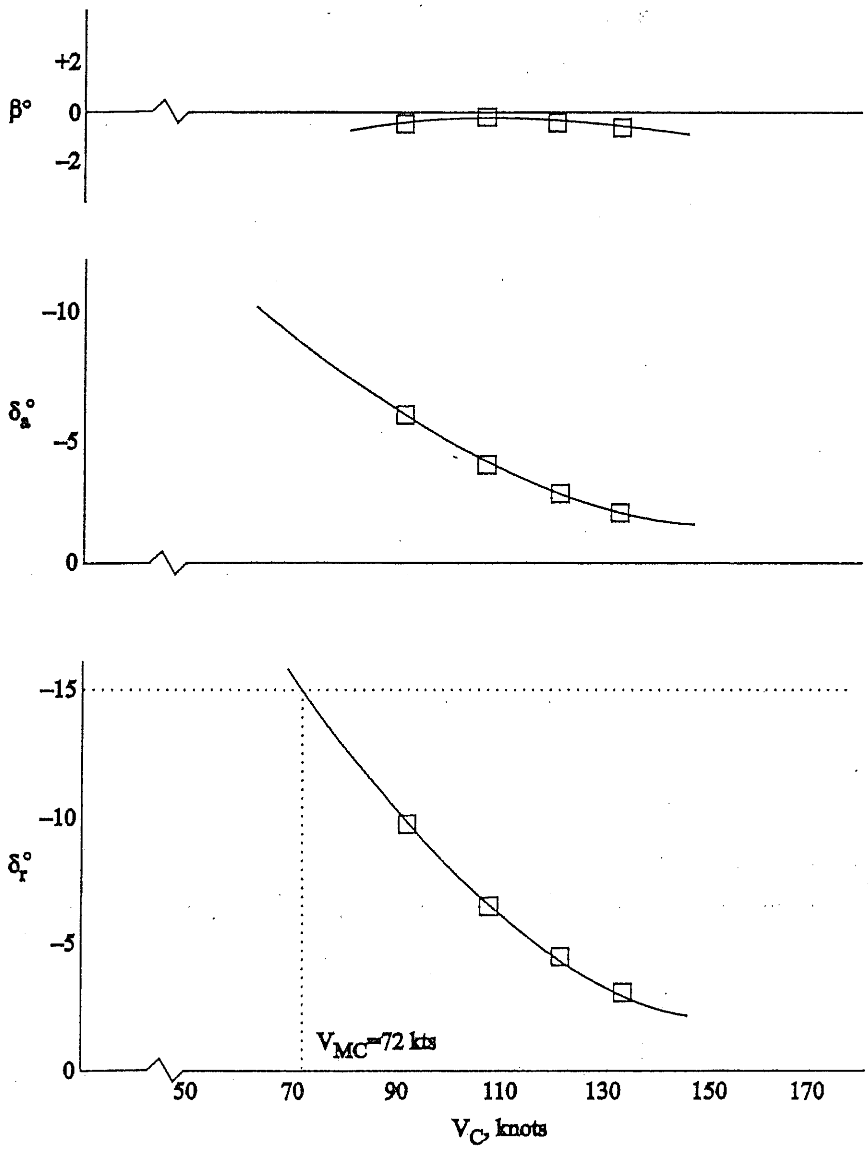


Fig. 31.6 Engine-out minimum control speed, Cessna 310A, 5800 ft altitude.

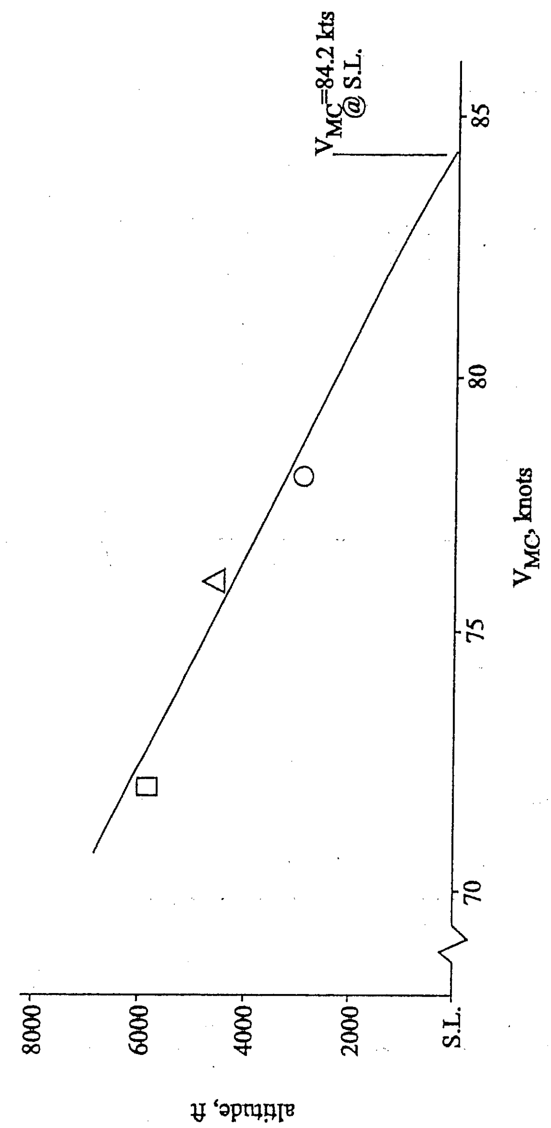


Fig. 31.7 Engine-out minimum control speed, Cessna 310A, altitude vs minimum control speed.

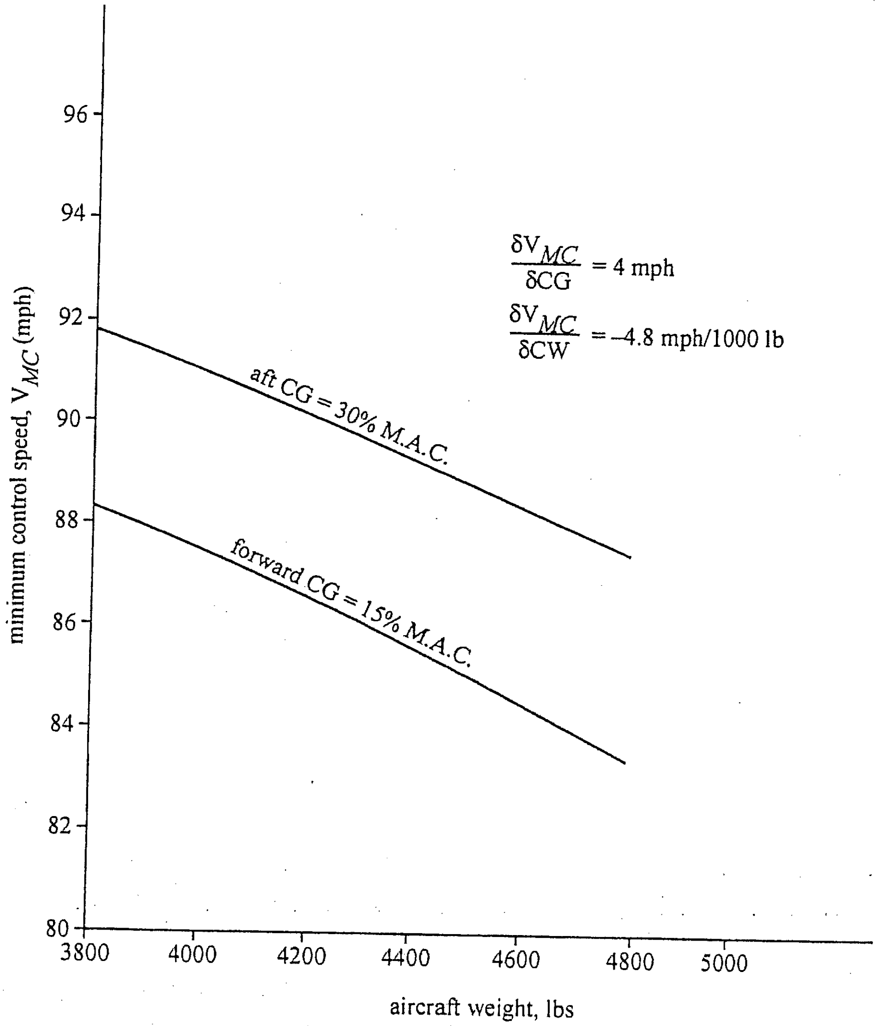


Fig. 31.8 Effects of weight and c.g. upon V_{MC} (Ref. 10).

need to calculate a value for $C_{N_{\delta_r}}$. This may be accomplished by use of the equation:

$$C_{N_{\delta_r}} = -a_v \tau_v \eta_v (S_v/S_w)(l_v/b) \tag{31.7}$$

where
a_v = lift curve slope of the vertical tail
τ_v = control surface effectiveness factor
η_v = vertical tail efficiency factor (q_v/q_∞)

Values for a_v and τ_v for a given configuration may be found in NACA TN 775 (Ref. 11).

If during the flight test directional control problems occur, we may wish to determine $C_{N_{\delta_r}}$ by flight test (Ref. 9). This may be accomplished by using one of the following methods. The first method is based on measuring control positions in straight steady sideslips at two c.g. positions. The second method uses a wing-mounted parachute, or other drag device, to generate a known yawing moment.

The first method is based on several assumptions. If we assume that the bank angle is small in a steady straight sideslip, then we can ignore the rudder side-force and write the lateral-directional equations as:[9]

$$C_L(\mathrm{d}\phi/\mathrm{d}\beta) + C_{Y_\beta} = 0 \tag{31.8}$$

$$C_{l_\beta} + C_{l_{\delta_a}}(\mathrm{d}\delta_a/\mathrm{d}\beta) = 0 \tag{31.9}$$

$$C_{N_\beta} + C_{N_{\delta_a}}(\mathrm{d}\delta_a/\mathrm{d}\beta) + C_{N_{\delta_r}}(\mathrm{d}\delta_r/\mathrm{d}\beta) = 0 \tag{31.10}$$

If we shift the c.g. by some increment ΔX_{cg}, then the only coefficients that will change are C_{N_β} and $C_{N_{\delta_r}}$. Since the tail length is large in comparison to the change in c.g., we may assume $C_{N_{\delta_r}}$ to be independent of c.g. position also. This leaves only C_{N_β} as a variable, and we may find the change in C_{N_β} due to c.g. shift by reference to Fig. 31.9 (Ref. 9) and use of the following equation:[9]

$$\Delta C_{N_\beta} = C_{Y_\beta}(\Delta X_{cg}/b) \tag{31.11}$$

The side-force coefficient due to sideslip C_{Y_β} can be determined by use of Eq. (31.8) since neither $\mathrm{d}\phi/\mathrm{d}\beta$ nor C_L of the airplane are functions of c.g. position.

If we then write Eq. (31.10) for both c.g. positions and subtract one from the other, we have remaining:[9]

$$\Delta C_{N_\beta} + C_{N_{\delta_r}}\left(\Delta \frac{\mathrm{d}\delta_r}{\mathrm{d}\beta}\right) = 0 \tag{31.12}$$

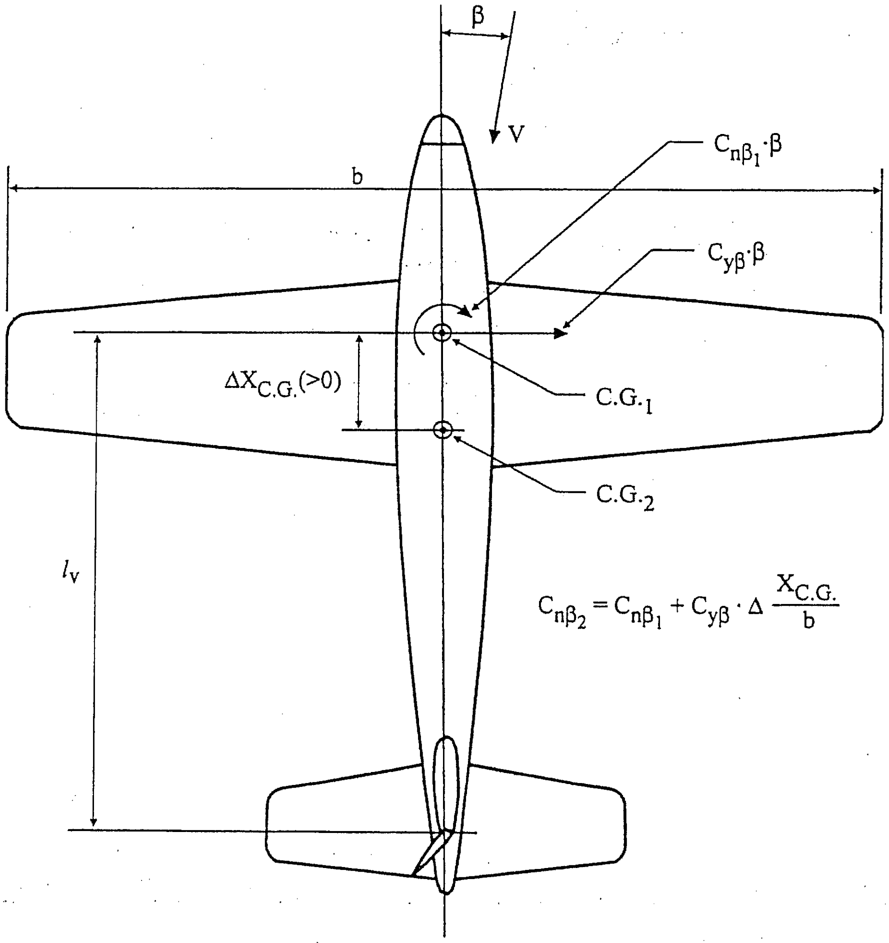


Fig. 31.9 Change in static directional stability, C_{n_β}, due to c.g. displacement.[9] [The original version of this material was first published by the Advisory Group for Aerospace Research and Development, North Atlantic Treaty Organization (AGARD/NATO) in *AGARD Flight Test Manual, Volume II—Stability and Control.*]

If we then substitute Eq. (31.11) into Eq. (31.12) and solve for $C_{N_{\delta_r}}$ we have:[9]

$$C_{N_{\delta_r}} = C_L(\mathrm{d}\phi/\mathrm{d}\beta)\left(\frac{\Delta\dfrac{X_{cg}}{b}}{\Delta\dfrac{\mathrm{d}\delta_r}{\mathrm{d}\beta}}\right) \tag{31.13}$$

All of the information necessary to solve Eq. (31.13) is available from a lateral-directional static stability flight test at two separate c.g. positions. Also, by knowing vertical tail length l_v one can find the side-force coefficient due to rudder deflection $C_{Y_{\delta_r}}$ by use of the $C_{N_{\delta_r}}$ value found by Eq. (31.13) and the following equation:[9]

$$C_{Y_{\delta_r}} = -C_{N_{\delta_r}}(b/l_v) \tag{31.14}$$

The second method for determining $C_{N_{\delta_r}}$ has fewer assumptions involved but is a more complicated flight test. First, one must have a parachute or other

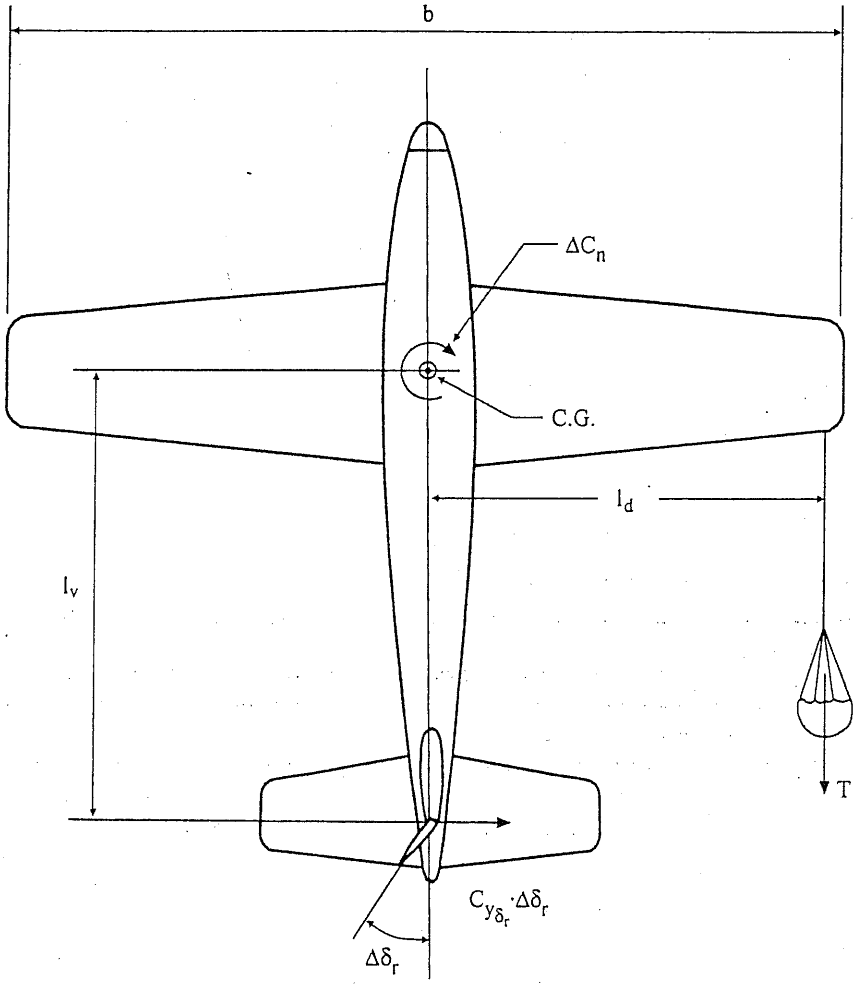


Fig. 31.10 Equilibrium of yawing moments with wing parachute at zero sideslip.[9] [The original version of this material was first published by the Advisory Group for Aerospace Research and Development, North Atlantic Treaty Organization *AGARD Flight Test Manual, Volume II—Stability and Control.*]

object of known drag coefficient. In addition, it must be possible to deploy and jettison this device in flight. It also must be stable in flight and not oscillate.

If we know the distance from the aircraft c.g. to the device (see Fig. 31.10), then we can determine the yawing moment due to it from the equation:[9]

$$\Delta N_d = D_d l_d \tag{31.15}$$

where
ΔN_d = the yawing moment due to the drag device
D_d = the drag due to the device
l_d = the lateral distance of the device from the c.g.

If we maintain a zero sideslip with rudder only while the device is deployed (as shown in Fig. 31.10), then we can express this condition by the equation:[9]

$$\Delta C_{N_d} + C_{N_{\delta_r}} \Delta\delta_r = 0 \tag{31.16}$$

where

$$\Delta C_{N_d} = \Delta N_d / q S_w b$$

$C_{N_{\delta_r}}$ is then found by:[9]

$$C_{N_{\delta_r}} = -\frac{\Delta C_{N_d}}{\Delta\delta_r} \tag{31.17}$$

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[3]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

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

[5]Langdon, S. C., and Cross, W. V., "Fixed-Wing Stability and Control Theory and Flight Test Techniques," USNTPS-FTM No. 103, 1 Jan. 1975, rev. 1 Aug. 1977.

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

[7]Staff, USAFTPS, "Stability and Control Flight Test Theory, Vol. I," AFFTC-TIH-77-1, Feb. 1977.

[8]Staff, USAFTPS, "Stability and Control Flight Test Techniques, Vol. II," AFFTC-TIH-77-1, Feb. 1977.

[9]Perkins, C. D., Dommasch, D. O., and Durbin, E. J., *AGARD Flight Test Manual, Vol. II*, "Stability and Control," Pergamon Press, New York, 1959.

[10]Bervin, L., "Analysis of V_{MC} for Light Twins," lecture given at University of Tennessee Space Institute—Experimental Flight Mechanics—short course, June 1978.

[11]Pass, H. R., "Analysis of Wind-Tunnel Data on Directional Stability and Control," NACA-TN-775, National Advisory Committee for Aeronautics, Langley Field, VA, Sept. 1940.

32
Flying Qualities

32.1 Introduction

Aircraft stability and control is often referred to as flying qualities or handling qualities. However, when we discuss flying qualities we want to make sure that we are talking about the airplane with the pilot "in the loop." We do this because having the pilot in the loop may very well affect many of the stability and control parameters. For instance, if the longitudinal short period frequency is near the pilot's response frequency, then the pilot may "feed" the longitudinal motion and produce pilot-induced oscillations.

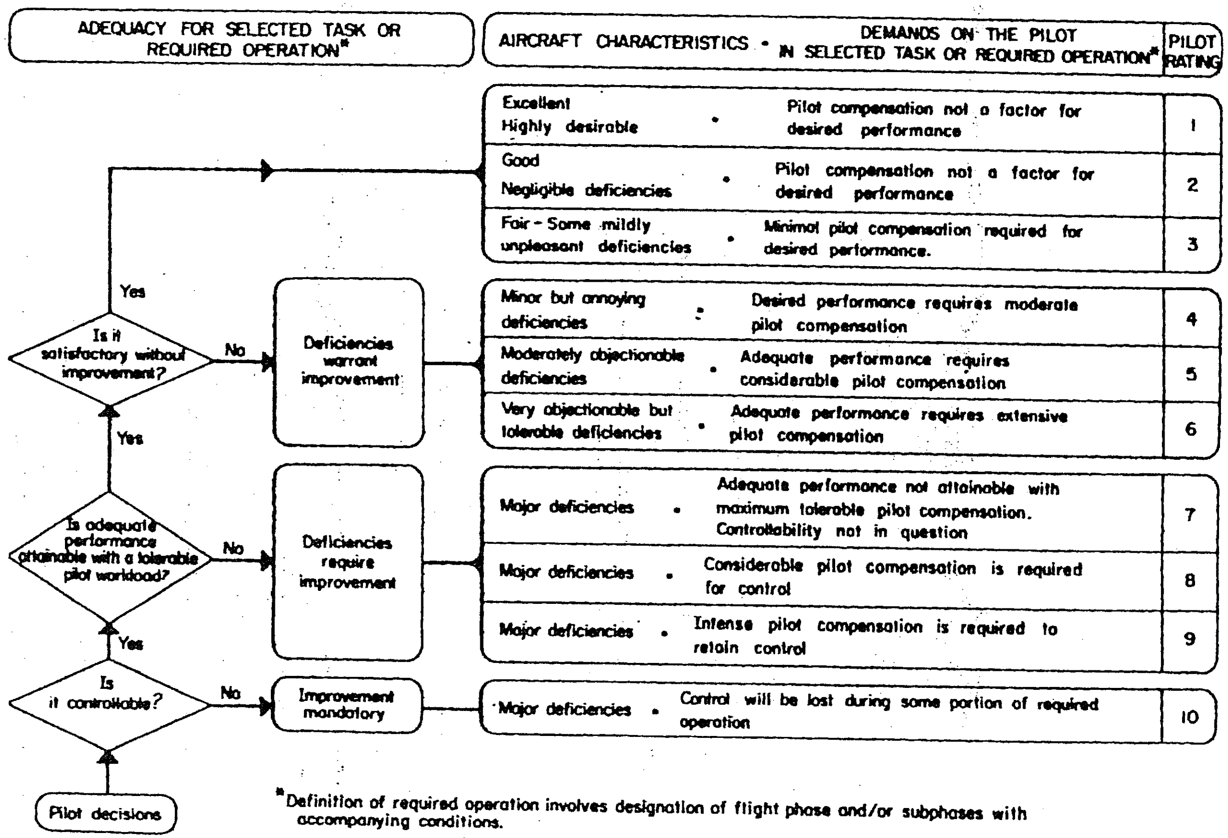
One of the easier ways to evaluate the airplane handling qualities with the pilot in the loop is to use pilot opinion. However, the problem with pilot opinion is that a pilot's opinion of an airplane's handling qualities will be biased based upon past experience and numerous other factors. Therefore, nearly every pilot would have a different opinion, or would stress different factors in their evaluation making consistent evaluation difficult if not impossible.

32.2 Federal Aviation Administration Regulations

Since the FAA regulations are considered minimum standards by the FAA, little effort has been made by the FAA to develop, or use, a flying qualities criteria other than the basic open loop stability and control criteria listed in other chapters of this book. This is especially true for small airplanes, since fly-by-wire and fly-by-light control systems have not reached this category of airplanes due to costs and maintainability issues. Since such systems are now seeing use in transport category aircraft, the FAA has attempted to establish minimum standards for flying qualities criteria[1] for these aircraft.

32.3 Cooper-Harper Pilot Rating Scale[2]

The above problems with pilot opinion as a method to evaluate handling qualities led to several attempts to overcome some of the shortcomings and find some method to standardize and quantify it. This was of particular interest to the military who were developing aircraft with modern control systems and relaxed static stability. The most successful of these methods is the Cooper-Harper pilot rating scale developed by researchers at NASA and the Cornell Aeronautical Labs.[2] This method of quantifying pilot opinion was developed in the mid-1960s and has been revised and refined in succeeding years. The scale, along with the meanings of each rating is shown in Fig. 32.1.



*Definition of required operation involves designation of flight phase and/or subphases with accompanying conditions.

Fig. 32.1 Cooper-Harper handling qualities rating scale.[2]

The Cooper-Harper pilot rating scale works very well. Once a pilot is briefed on how to apply the ratings, his evaluation will generally agree within one number of any other pilot who evaluates the same aircraft. The method becomes much more accurate if a number of pilots of varying backgrounds and experience levels are used to evaluate a given airplane. The resultant average Cooper-Harper pilot rating will give an accurate evaluation of the aircraft or handling qualities item involved.

32.4 Levels of Flying Qualities

The background information for MIL-F-8785B (Ref. 3) simplifies the flying qualities ratings of the Cooper-Harper rating scale into three levels of acceptable flying qualities. These levels of acceptability and their respective definitions are:

Level 1: Flying qualities clearly adequate for the mission flight phase.
Level 2: Flying qualities adequate to accomplish the mission flight phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.
Level 3: Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both.

The above levels would correspond to Cooper-Harper rating scale as follows:

Level 1: 1–3.5 on Cooper-Harper scale
Level 2: 3.5–6.5 on Cooper-Harper scale
Level 3: 6.5–9+ on Cooper-Harper scale

It might also be noted that the handling qualities levels mention pilot workload. Pilot workload is a function of a large number of variables in addition to those related to stability and control. As a result, handling qualities may take on a much larger meaning, which may include:

1) human factors items such as cockpit design
2) airplane performance
3) meteorological conditions during the mission
4) other factors that may affect pilot workload

Therefore, when we conduct handling qualities investigations we need to be sure that the items and the conditions under which we want them evaluated are properly defined.

32.5 Flight Test Procedures

Although the FAA regulations for small airplanes do not address the use of flying qualities rating scales, the use of these scales is an excellent way of evaluating many flying tasks that the pilot may experience. For instance, the regulations do not address the level of difficulty that the pilot may experience in flying an ILS precision approach in a specific airplane. For such an evaluation use of the Cooper-Harper rating scale is a good way to perform the evaluation since it evaluates the flying qualities, in total, required to perform this task in addition to other related factors such as design of the instrument panel and specific instruments.

32.5.1 Designing the Test

The test should be designed so as to not cover too large of an area for evaluation. If possible, keep the evaluation to a specific task. Involve a large enough sample of pilots to identify the pilot population whose gains may match the airplane responses when they are in the loop. Five to ten pilots would provide a reasonable sample; one or two pilots does not.

32.5.2 Pretest Briefing

You should be sure to conduct a very complete preflight briefing, which should include the use of the rating scale. The pilots should be cautioned about deviating from the planned test as this may change the results.

32.5.3 Conducting the Test

The pilot should assign the rating immediately after completing the task. Allowing time to pass prior to assigning the rating may allow the pilot to rationalize and change the rating. Especially avoid having the pilot discuss the rating with other pilots prior to assigning the rating.

32.5.4 Post-Flight Debriefing

This debriefing should also be conducted as quickly as possible after the flight and without any discussions with other evaluation pilots. During this debriefing questions may be asked as to why the pilot assigned the rating, plus any other questions pertinent to the task.

32.5.5 Evaluating the Results

In evaluating the results, one will find that the majority of evaluation pilots assign ratings that are within one number of each other. However, if one pilot does not agree with the majority, particular attention should be paid as to why the rating was assigned. This may be the individual in the pilot population whose response frequencies and gains more nearly match those of the aircraft with the potential for problems. If such an individual exists in the evaluation pilot sample, efforts should be made to change the aircraft to eliminate the problem or provide notes in the Pilot's Operating Handbook to call attention to the potential for a problem.

References

[1]Federal Aviation Administration Advisory Circular No. 25-7A Change 1, "Flight Test Guide for Certification of Transport Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1999.

[2]Cooper, George E., and Harper, Robert P., Jr., "The Use of Pilot Rating in the Evaluation of Aircraft Handling Qualities," NASA TN D-5153, National Aeronautics and Space Administration, Washington, D.C., April 1969.

[3]Chalk, C.R., Neal, T.P., et al., "Military Specification—Flying Qualities of Piloted Airplanes," Background Information and User Guide for MIL-F-8785B (ASG), AFFDL-TR-69-72, Aug. 1969.

PART 3
Hazardous Flight Tests

33
Stall Characteristics

33.1 Introduction

In addition to determining the stalling speed, a knowledge of the aircraft's characteristics as the airplane stalls is important, since if an inadvertent stall should occur we want the airplane to be well behaved without a tendency to enter uncontrolled flight. Therefore, an investigation of the stall characteristics of the airplane is an important part of any airplane certification.

As stated in Chapter 4, defining when the stall occurs is important since several different definitions of that event exist. In fact, there are several different definitions in the Federal Air Regulations. CAR 3 (Ref. 1) and FAR 23 (Ref. 2) define the stall as the airspeed when the nose pitches uncontrollably or the elevator control reaches the up stop. The current FAR 23 further says that for determining stall characteristics the elevator must be held against the up stop for two seconds before recovery can be initiated. However, FAR 25 (Ref. 3) defines the stall as the minimum airspeed seen during the maneuver. In this text we will use the CAR 3/FAR 23 definition.

33.2 Federal Aviation Administration Regulations

33.2.1 Civil Aeronautics Regulation 3 (Ref. 1)

Civil Aeronautics Regulation 3 covers stall characteristics in sections 3.120, 3.121, 3.122, and 3.123.

CAR 3.120 states that stalls must be demonstrated both with power-off and with power used to demonstrate climb performance for airplanes over 6000 lb TOGW, or for airplanes of less than 6000 lb TOGW, 90% of MCP. These stalls shall be demonstrated in the most unfavorable condition of weight and c.g. with the landing gear and flaps both retracted and extended.

For airplanes having independent roll, yaw, and pitch controls it shall be possible to maintain the intended attitude by unreversed use of the controls up until the time the nose pitches downward at the stall.

For two-control airplanes (elevator and interconnected ailerons and rudder) it shall be possible to produce roll by unreversed use of the rolling control without producing excessive yaw up until the time the airplane pitches nose down at the stall.

During the recovery portion of the maneuver, it shall be possible to prevent more than 15 deg of roll or yaw by normal use of the controls and any pitch more than 30 deg below the horizon, or an altitude loss of more than 100 ft, shall be shown in the airplane flight manual.

In straight and turning flight with the landing gear and flaps in any position, a clear and distinct stall warning shall occur at a speed not more than 10 mph or less than 5 mph before the stall occurs. This warning shall continue until the stall occurs.

Trim speeds for the maneuver shall be $1.4V_{S1}$ for airplanes with TOGW in excess of 6000 lb and $1.5V_{S1}$ for airplanes with TOGW less than 6000 lb.

The maneuver is performed by pulling the elevator control back, slowing the airplane until a speed slightly above stalling speed, and then pulling the control at such a rate that a deceleration of 1 mph/s is obtained. This deceleration rate is continued until the stall is produced. The stall is identified by an uncontrollable downward pitching motion of the nose or when the elevator control reaches the up stop. Normal use of the elevator control for recovery may be initiated after the pitching motion has unmistakably developed.

CAR 3.121 covers climbing stalls or more correctly stated as stalls from an excessive climb attitude. It says that it shall be possible to recover from an excessive attitude stall without exceeding the limiting airspeed or allowable acceleration limits. This regulation was implemented to cover airplanes where the elevator control had been limited in up travel so as to prevent a normal stall from occurring, but it would still be possible to stall the airplane if it was zoomed.

CAR 3.122 provides the regulations for turning flight stalls. It requires that when stalled from a 30 deg banked turn with 75% MCP on all engines, flaps and landing gear retracted, it shall be possible to recover to normal level flight without excessive loss of altitude, uncontrollable rolling characteristics, or uncontrollable spinning tendencies. To perform this stall, a steady 30 deg banked turn is established at the proper power setting and the turn is tightened with the elevator control until the airplane stalls or the control reaches the up stop. When the stall has fully developed, recovery to level flight can be accomplished by normal use of the controls. These stalls are to be performed in both left and right turns.

CAR 3.123 is the requirements for one-engine inoperative stalls for multiengine airplanes. This paragraph requires that multiengine airplanes shall not show any undue spinning tendency and shall be safely recoverable without applying power to the inoperative engine when the airplane is stalled with the critical engine inoperative, the landing gear and flaps retracted and the remaining engines operating at 75% MCP. However, the power on the operating engines need not be greater than that at which the use of maximum control travel just holds the wings level in approaching the stall. During recovery from the one-engine inoperative stall the operating engines may be throttled back.

33.2.2 Federal Aviation Regulation Part 23 (Ref. 2)

Stall characteristics are covered in sections 23.201, 23.203, 23.205, and 23.207 of FAR Part 23. This regulation is arranged somewhat differently than

CAR 3 in that the three sections cover wings level stalls, turning flight and accelerated stalls, one-engine inoperative stalls, and stall warning. In addition, FAR 23.205 one-engine inoperative stalls was removed from FAR Part 23 by Amendment 23-45.

FAR 23.201 covers the requirements for wings level stall characteristics. Like CAR 3.120 it requires that it be possible to produce and correct roll and yaw by unreversed use of their respective controls up until the time the aircraft stalls. The method that the aircraft is stalled is also like CAR 3.120 except that FAR 23.201 requires that a deceleration of rate of 1 kn/s be achieved at least 10 kn above the stall. In defining where the stall occurs, FAR 23 adds to the two methods of CAR 3 a downward pitching motion of the airplane that results from the activation of a stall avoidance device (stick pusher). Normal use of the elevator control for recovery is allowed after the downward pitching motion of the nose has unmistakably developed, or after the control has been held against the up stop for a period of 2 s. This 2 s requirement is a recent addition to FAR 23 and was not a part of the earlier regulation.

During entry into and recovery from the stall, it must be possible to prevent, with normal use of the controls, more than 15 deg of roll or yaw. In performing the test the airplane should be trimmed for an airspeed of $1.5V_{S1}$ with the propeller in the full increase rpm position. Stalls must be performed with the landing gear and wing flaps both retracted and extended and the cowl flaps (if installed) in the position appropriate for the test. Stalls must be performed both power off and with 75% MCP. However, if 75% MCP results in extreme nose up attitudes during the stalling maneuver, the test may be carried out with the power for level flight in the landing configuration at maximum landing weight and a speed of $1.4V_{S0}$, except that the power may not be less than 50% MCP.

FAR 23.203 gives the requirements for turning flight and accelerated stalls. The early versions of FAR Part 23 read much like CAR 3.122, however, in later amendments to FAR Part 23 some additional requirements were added to these stalls. First, for normal turning flight stalls the rate of speed reduction must be constant and a deceleration rate of 1 kn/s must be used. Secondly, for accelerated turning flight stalls this rate must be from 3 to 5 kn/s with steadily increasing normal acceleration.

The recovery criteria include, in addition to those requirements stated in CAR 3.122, no undue pitch up or exceeding a bank angle of 60 deg in the direction of turn or 30 deg in the opposite direction. These bank angles increase for accelerated turning stalls to 90 deg in the direction of turn and 60 deg in the direction opposite the turn.

Power for turning stalls must be both power off and 75% power unless 75% gives an excessive nose up attitude, then power for level flight as in the wings level stalls is prescribed. The trim airspeed is specified as $1.5V_{S1}$ and the propeller must be set at full increase rpm.

FAR 23.205 one-engine inoperative stalls has been removed from the later versions of FAR Part 23 but remains in the versions that most of today's multi-engine Part 23 airplanes were certified under. This regulation reads nearly exactly the same as CAR 3.123.

FAR 23.207 provides the requirements for stall warning. In CAR 3 this was covered as a part of CAR 3.120, but in FAR Part 23 it is given its own section.

Early versions of FAR Part 23 read much like CAR 3, except for the miles per hour being changed to knots, but later amendments have added several changes. First, the current version eliminates a maximum speed for the warning to initiate but maintains that it must initiate at a minimum of 5 kn above the stall and continue until the stall occurs. Second, FAR Part 23 does not allow the use of a warning light in the cockpit as a sole warning device. This was allowed under CAR 3. The current FAR Part 23 adds three other requirements not seen in previous regulations. The first of these is that the warning device cannot sound during takeoff or landing while following the takeoff procedures stated in the airplane flight manual. This includes a takeoff continued with one inoperative engine. The second of these new requirements states that the warning must begin enough in advance that the pilot can avert the stall. Therefore, it seems that if 5 kn is insufficient for this to occur, it must be set higher than 5 kn. The third of these new items allows a mutable stall warning device for acrobatic category airplanes provided that it is armed automatically during takeoff and again during the approach configuration.

33.2.3 *Advisory Circular 23-8A (Ref. 4)*

AC 23-8A gives acceptable methods for conducting the stall characteristics testing. Although the advisory circular restates much of what is in the regulation, several items are worth noting. First, with regard to the terms "uncontrollable downward pitching motion," this is taken to be the point at which the pitching motion can no longer be arrested by application of up elevator and not the first indication of nose down pitch. In other words, the elevator must be against the up stop before most airplanes would be pitching uncontrollably downward. Second, the advisory circular says that the use of zero thrust does not apply to evaluating the stall characteristics and that all power-off stalls should be done with the engine idling. Third, for determining altitude loss the advisory circular states that, during recovery, power may only be applied after the aircraft has reached a speed of $1.2V_{S1}$.

With regard to aircraft configuration, the advisory circular states that stall characteristics should be evaluated at the aft c.g. at both maximum and minimum weights as some airplanes with high thrust to weight may be more critical for stall characteristics at light weight. For stall characteristics tests the elevator stop should be set to the maximum allowable deflection. These tests should also be conducted with the maximum allowable fuel unbalance and at an altitude at or near the maximum altitude.

The advisory circular addresses the safety aspect of this testing by discussing crew restraint, emergency egress, and the use of parachutes by the crew. It also recommends a buildup technique starting at the forward center of gravity.

During the tests the test pilot should insure that the elevator stick force curve remains stable up to the stall and that it is possible to produce and correct roll and yaw by unreversed use of the aileron and rudder controls. The deceleration rate into the stall shall be one kn/s.

Data to be collected include the altitude loss, the pitch attitude below level, and the amount of roll and yaw encountered during recovery.

For two-control airplanes, it should be possible to produce and correct roll and yaw up to the stall without producing, in the opinion of the test pilot, excessive yaw.

For data acquisition the advisory circular recommends as a minimum a recently calibrated sensitive altimeter, airspeed indicator, accelerometer, outside air temperature gauge, and appropriate power measurement instruments. It also asks for a means to determine roll, pitch, and yaw angles. If any control forces are in question then a force gauge may be necessary.

The advisory circular also discusses stick pusher systems or stall barrier systems. It relates that there are two conditions where a stick pusher might be required. The first condition is where an airplane was recoverable but would not meet the requirements for roll or yaw after the stall. The second condition is where the airplane is not recoverable from the stall. In the first case, the occurrence of a failure of the system should be evaluated for an unsafe condition. In the second case, the event of a failure would cause an unsafe condition. During the flight test of these systems, the system tolerances should be considered. For instance, during stall characteristics investigations the system should be set for the lowest system tolerance in airspeed above stall, while during stall speed measurement the system should be set for the highest airspeed tolerance. The airspeed margin above stall where the system activates should also be evaluated. For airplanes that will not recover from a stall, that margin should be at least 5 kn but for airplanes that are recoverable the margin may be reduced to not less than 2 kn.

If reliability credit is to be given for a preflight check of a stick pusher system, then the check should include the functioning of the complete system. It also should be easy to perform and require little pilot effort. In addition, a note should be placed in the limitations section of the airplane flight manual requiring the check to be required before flight. Also the flight manual should identify the criticality of the system and the need to accomplish the preflight check.

Inadvertent operation of such systems should be shown to be extremely improbable or investigated and shown not to be hazardous.

For turning and accelerated stalls, the only items that differ from wings level stalls are that there should be no undue pitch up; that the test pilot can complete the recovery with average piloting skills; that, in his/her opinion, the altitude loss is not excessive; and that the roll does not exceed 60 deg incrementally in either direction. For accelerated turning stalls the maximum speed, or limit load factor shall not be exceeded and the entry rate shall be 3 to 5 kn/s with steadily increasing normal acceleration.

Under critical engine inoperative stalls the advisory circular addresses the meaning of undue spinning tendency as a case where other than normal use of the controls is required to recover from the stall. When 75% MCP results in reaching the minimum control speed, V_{MC}, prior to reaching stall, the advisory circular states that the power may be reduced until one can just maintain wings level and the heading constant until the stall occurs.

The advisory circular addresses stall warning by parroting the regulation and saying that an artificial stall warning light by itself is not acceptable. Regarding the warning margin, it states that the warning margin must be between 5 kn and the greater of 10 kn or 15% of the stalling speed. For accelerated stalls it states that the margin should not be less than 5 kn or above a speed where the warning would become objectionable during normal operations.

33.3 Stall Characteristics Theory

Those factors that were discussed in Chapter 4 of this text apply to stall characteristics and will not be repeated here. Refer to that chapter for the stall theory.

33.4 Aircraft Loading

Stall characteristics testing should begin at the most forward c.g. at the maximum gross weight. During aircraft development the stall characteristics at forward c.g. should be examined at the same time that stall speeds are measured and the most critical stall characteristics evaluated as the c.g. is moved aft during stability testing and neutral point determination. Once the desired aft c.g. is reached a complete set of characteristics can be evaluated.

The aircraft ballasting should be well secured so as to not be likely to shift, from either positive or negative accelerations, during the testing. It should also be positioned so as to represent a normal vertical c.g. of the aircraft. This may require ballast boxes at seat height or water tanks elevated from the aircraft floor.

33.5 Safety Considerations

Depending upon aircraft design, stall characteristics testing can be hazardous so it should be approached with caution. A number of safety items should be evaluated before beginning this testing.

33.5.1 Aircraft Egress

The ease with which the crew can egress should be evaluated. This is particularly true for aircraft that have aft cabin doors. For all aircraft, one should consider quick release doors or escape hatches that are easily reached by the crew during post stall gyrations or inadvertent spins. If an aft cabin door is the only option, then methods to get to that door in advent of an unrecoverable deep stall should be devised.

33.5.2 Considerations for a Recovery Parachute

If the aircraft has swept wings and a T-tail, or an unswept wing with swept center section and a T-tail, then one should consider a recovery parachute system as these design features have been known to cause unrecoverably deep stalls. Recovery chute design should follow the same guidelines as that required for spin recovery parachutes which is provided in the following chapter.

33.5.3 Personal Equipment

Personal equipment for the crew during stall characteristics testing should include flying suit, helmet, jump boots, and a personal parachute. The more modern personal parachutes with deployment bag, to reduce opening shock, and a steerable canopy are preferred.

33.5.4 Minimum Test Altitude

A minimum test altitude should be established depending on the type of aircraft. For light single, or multiengine aircraft, this altitude should be at least 5000 ft above ground level (AGL). For heavier aircraft, this altitude AGL should be increased accordingly. These altitudes should be the recovery altitudes and not the altitude where the stall is initiated.

For multiengine aircraft with turbine, or normally aspirated reciprocating engines, care should be taken to ensure that the altitude selected is not the altitude where the minimum control speed and the stalling speed are the same or nearly the same. Selecting such an altitude will result in a rapid spin entry or out of control condition from which there may be no recovery!

33.5.5 Chase Aircraft

During initial, and any critical, stall characteristics testing a chase aircraft is a good idea. The chase aircraft crew should be prebriefed on minimum altitudes and the stalls to be performed so as to be in position to observe, clear the area, and provide assistance for the test aircraft without being in its way.

33.6 Flight Test Method

33.6.1 Aircraft Test Configurations

Good test planning dictates investigation of clean configuration power-off stalls first, followed by the power on clean configuration to regain altitude lost from the power-off tests. These are then followed by turning and accelerated stalls, which will normally result in more altitude gain. For the completion sequence, a power-off gear and flap down stall should be followed by a power-on stall in the same configuration. This approach to mixing the test configurations will result in less time spent climbing to regain altitude lost during a power-off stall sequence and will result in minimum flight time.

33.6.2 Trim Airspeed

Both CAR 3 and FAR Part 23 call for a trim speed of 1.5 times the stalling speed in the test configuration. For initial stalls where the stalling speed is not known, this can be the calculated stalling speed. Once test numbers are obtained the trim should be readjusted to 1.5 times the actual stalling speed. In trimming the aircraft for these speeds, one should insure that the power is in the power setting required for the given stall.

33.6.3 Power Settings

Regarding power settings for evaluating stall characteristics, for power on stalls, CAR 3 requires a power setting of 90% MCP while FAR Part 23 requires a power setting of 75% MCP. Generally, 75% MCP can be obtained to altitudes of 7000 ft MSL or above which does not present a problem from the safety point of view. However, 90% MCP may not be obtained above safe altitudes for many engines while conducting initial stall characteristics investigations. For airplanes that are approved under CAR 3, these initial investigations should be conducted at maximum available power at a safe altitude prior to retesting at a lower altitude where 90% MCP can be obtained. If the test location will not allow achieving 90% of sea level MCP then the maximum power available at the lowest safe altitude should be used.

Power-off stalls for stall characteristics should be performed with the propeller in the high rpm setting (low pitch) and the throttle reduced to the idle setting.

Power may be reapplied during recovery after the aircraft has achieved an airspeed of 1.2 times the stalling speed.

33.6.4 Deceleration Rate

CAR 3 calls for a deceleration rate prior to the stall of 1 mph/s while FAR Part 23 calls for a rate of 1 kn/s. Fig. 4.1 in this text shows a method of determining deceleration rate and stall speed if an automatic recording device that records airspeed vs time is available. Such instrumentation may be available for a commuter category certification or for a heavy twin, however, on most light aircraft programs such instrumentation is not available. Therefore the most common method is to pick up a count "thousand one, thousand two..." and attempt to make the deceleration rate match the count in knots or miles per hour prior to the airplane reaching 10% in excess of the stall speed and then maintaining that rate until the airplane stalls.

33.6.5 Control Usage

The regulations require maintaining wings level flight up until the stall break and no more than 15 deg of roll or yaw after the nose pitches downward uncontrollably. In order to do this there is nothing in the regulation that prohibits use of full control travel as long as that usage of the controls is unreversed and coordinated. Many pilots are hesitant to use full control travel during the stall for fear of an inadvertent spin entry. However, if the pilot is alert and ready to rapidly remove the full travel should the airplane roll in the direction of control deflection an inadvertent spin is unlikely. In any case, the pilot should be ready to rapidly reduce the angle of attack by applying nose down elevator. During stall characteristics testing the test pilot must be somewhat like the boxer in the ring ready to shuffle and jab the controls in response to the airplane's reactions.

The later amendments to FAR Part 23 require that the elevator control be held in the full up position for 2 s after the nose pitches downward uncontrollably. However, if a rapid uncontrollable departure from wings level flight occurs during this time the pilot should consider the characteristics for that stall unsatisfactory and recover immediately.

33.6.6 *Stall Warning*

The airspeed at which the stall warning device activates should be recorded for each stall. If the warning occurs at too high or too low an airspeed then the device should be adjusted and the stall repeated. Care should be taken during stall characteristics testing to insure that the deceleration rate is obtained prior to the activation of the device as higher or lower deceleration rates may affect when the device activates. However, with any deceleration rate the device should give adequate warning for the stall, but not come on so high an airspeed as to be a nuisance or to be ignored.

33.6.7 *Defining the Stall*

On airplanes that do not require a stick pusher or stall barrier device, the stall is usually defined as when the elevator control reaches the up stop, as most light aircraft have sufficient elevator power to prevent the nose from pitching downward uncontrollably until the up stop is reached.

For aircraft with stick pushers or stall barrier devices, the stall is defined when that device activates. Again, Fig. 4.1 provides a graphical depiction of stall definition.

33.6.8 *Recovery Technique*

Normally, recovery from the stall can be effected by relaxing the back pressure on the elevator control. In cases where the airplane does not recover immediately from this action, use of down elevator, including full down elevator, is warranted. If rolling and yawing are occurring during the recovery these should be opposed by rudder and aileron in the direction opposite the roll or yaw, up to and including full rudder and aileron.

33.7 Data Requirements

The following data should be collected during the stall event:

1) warning airspeed
2) stalling airspeed
3) altitude loss during the stall
4) maximum roll, pitch, and yaw during the stall, including direction
5) normal acceleration during recovery

These data may be recorded by a data system or by hand from calibrated panel instruments. If hand recording is used a crew of two is useful since the test pilot needs to be paying attention to the aircraft rather than trying to gather numbers. However, if only a test pilot is used as a crew and a data system is not available then practice in flying the airplane while collecting data in an aircraft with benign stall characteristics may be worthwhile. Also, if hand recording is used a sensitive airspeed indicator graduated in 1 kn increments should be installed. For roll, pitch, and yaw angles, panel mounted attitude indicators have been found acceptable for collecting these data.

33.8 Potential Problems

There are two potential problems that may occur during stall characteristics testing. They are wing drop or roll-off at the stall and deep stall lock-in or pitch-up.

33.8.1 Wing Drop or Roll-Off

There are two possibilities for causing this problem. One is sideslip at the stall. If the test pilot is not attentive during the last stages of the approach to the stall then sideslip may develop. This may be caused by propeller slipstream or insufficient rudder to counter P-factor in the case of power on stalls. If a strong wing drop that exceeds the 15 deg bank limit is encountered, then the test pilot should repeat the stall paying close attention to sideslip as the stall occurs to determine if the wing drop repeats when there is zero sideslip at the stall. A crutch that may be useful in helping the pilot determine this is a yaw string on the windshield marked at zero sideslip during level flight. If wing drop still occurs then one should look at the second possibility.

The second possibility is differences in contour (shape) of the right and left wings. This is a possibility for even aircraft built upon the finest tooling by the finest craftsmen, and the minute differences that cause this will not be visible to the eye or most measurement devices. This is especially true on light aircraft with thin wing skins where the leading edge is not formed by dies. However, it may even occur on aircraft on which leading edge wing skins are formed on dies. The following section on problem fixes discusses some potential fixes for this problem.

33.8.2 Deep Stall Lock-In or Pitch-Up

Deep stall problems or pitch-up are usually associated with a combination of aft swept wings and T-tails. They may also be found on airplanes with straight wings with large swept center sections and T-tails. They are generally not a phenomenon associated with T-tails alone, which is an old wives tale that continues in the industry. Swept wings cause spanwise flow that causes the wing tips to stall first. This causes the tip vortex to move inboard increasing the downwash at the tail. This increase in downwash causes the tail to produce a greater download resulting in pitch-up or insufficient down elevator control to recover from the stall. Something similar occurs with straight wings with large swept center sections. In this case, the outboard straight wing sections stall while the inboard swept center section continues to fly producing wing tip vortices at the end of the swept center section.

33.9 Problem Fixes

There are a number of devices that have been used to successfully fix stall characteristics problems. They can be divided into two categories. They are: 1) aerodynamic fixes and 2) electromechanical fixes. First, let us discuss the aerodynamic fixes.

33.9.1 *Stall Strips*

The stall strip, discussed in Chapter 4, is a very powerful device for fixing stall characteristics problems. It is a triangular strip on which the height depends upon the size of the airfoil and the length depends upon the severity of the stall problem. It is primarily used for fixing problems of roll-off, but can be used for deep stall problems by forcing the wing to stall at the center section first.

It is normally best to use stall strips on both wings so that they may be adjusted up or down to correct roll-off problems. For light airplanes a triangular stall strip of about 0.25 in. in height and about 6 in. long is a good place to start. The strips should be located on the inboard portion of the wing initially but can be added to the outboard portion as necessary. The strips can be increased in length on the wing opposite the roll-off, moved up to hasten the stall, or both. On the wing in the direction of roll-off, the strips may be moved down or shortened. The strips may be used to tailor the stall to the desired characteristics. On some airfoils it may be necessary to tailor the location of the stall strips for each individual airplane. In this case, rubber adhesive stall strips are used during the production flight test in order to tailor the stall characteristics and then once their location is determined the permanent stall strips are installed. For airplanes with deicing boots, the rubber adhesive strips are used on top of the boots. Figs. 33.1, 33.2, and 33.3 show stall strips and their location on several different airplanes.

33.9.2 *Wing Fences*

The wing fence is another device that is sometimes used for improvement of stall characteristics. Fences are chordwise barriers that prevent spanwise flow and are most common on aircraft with swept wings. However, they have also found some use on straight wing airplanes when placed between the flap and aileron. In this position, they slow the stall from progressing from the wing center section to the aileron so aileron control is maintained deeper into the stall.

For swept wing airplanes, the fence stops the spanwise component of the flow over the airfoil and helps to cause the center section to stall first. This reduces the tendencies for deep stall lock-in or pitch-up. A combination of wing fences and stall strips may offer a potential solution for problems associated with deep stall. Figs. 33.4 and 33.5 show the use of wing fences on a straight wing aircraft and a swept wing aircraft.

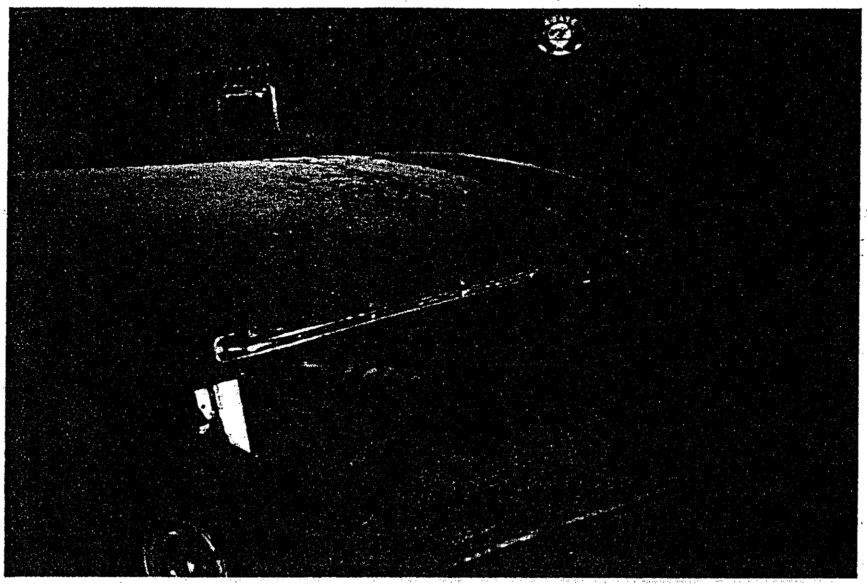


Fig. 33.1 Leading edge stall strip on the Ryan Navion.

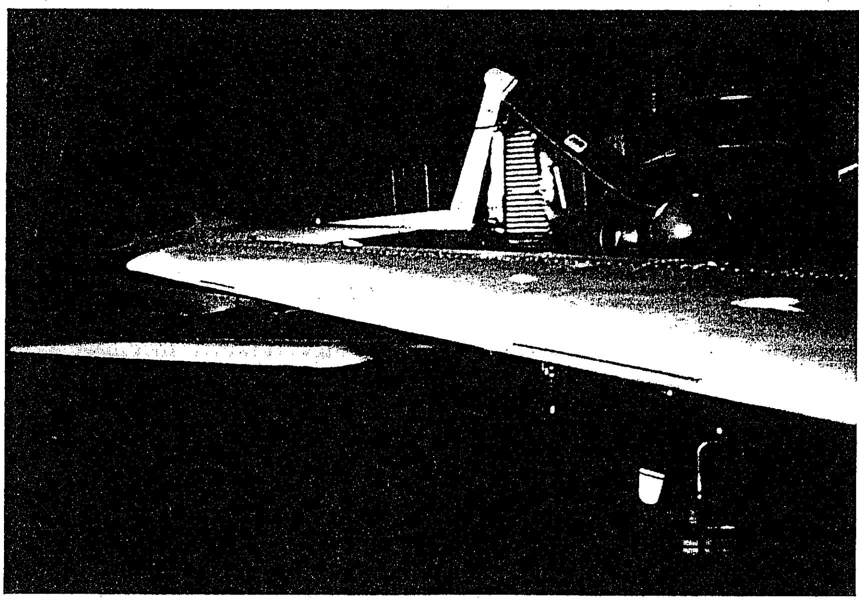


Fig. 33.2 Stall strips on right wing of Piper PA31 Navajo.

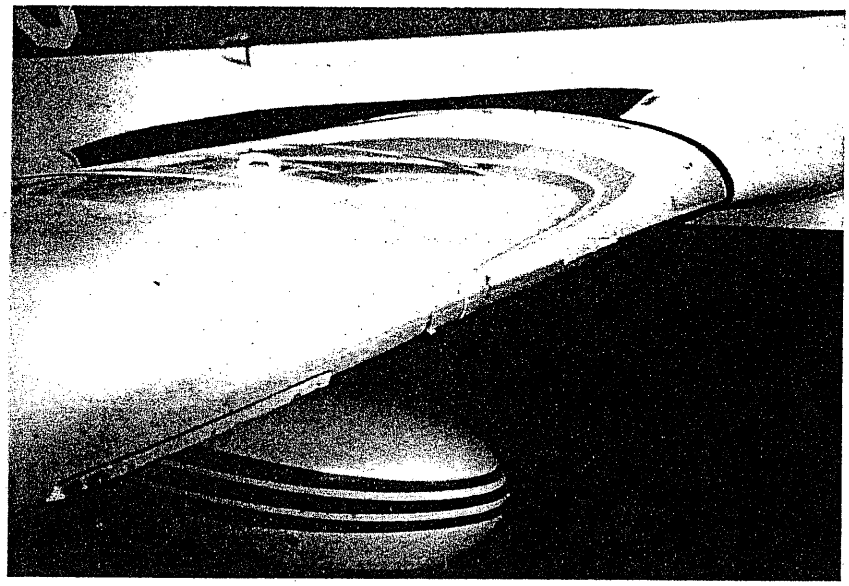


Fig. 33.3 Stall strip on right wing of Grumman Tiger.

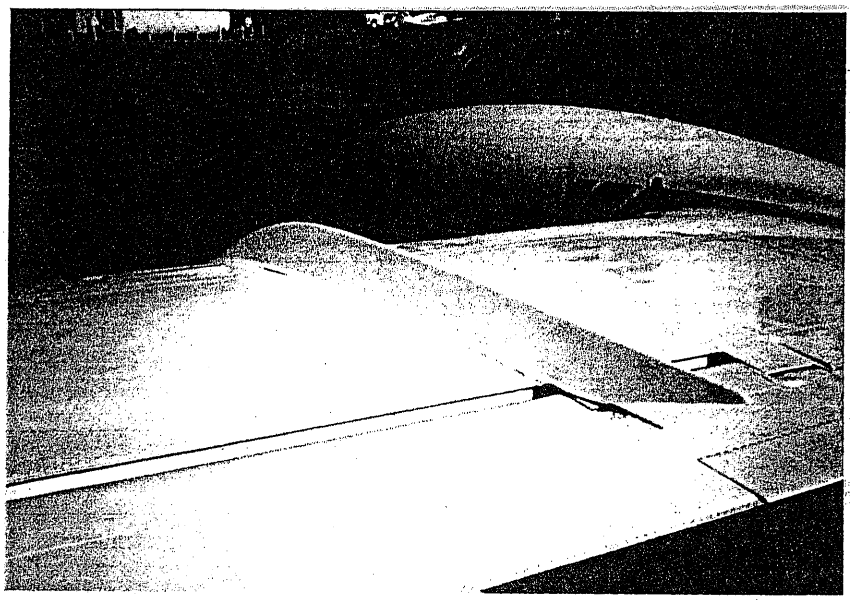


Fig. 33.4 Stall fence on Beechcraft King Air 100.

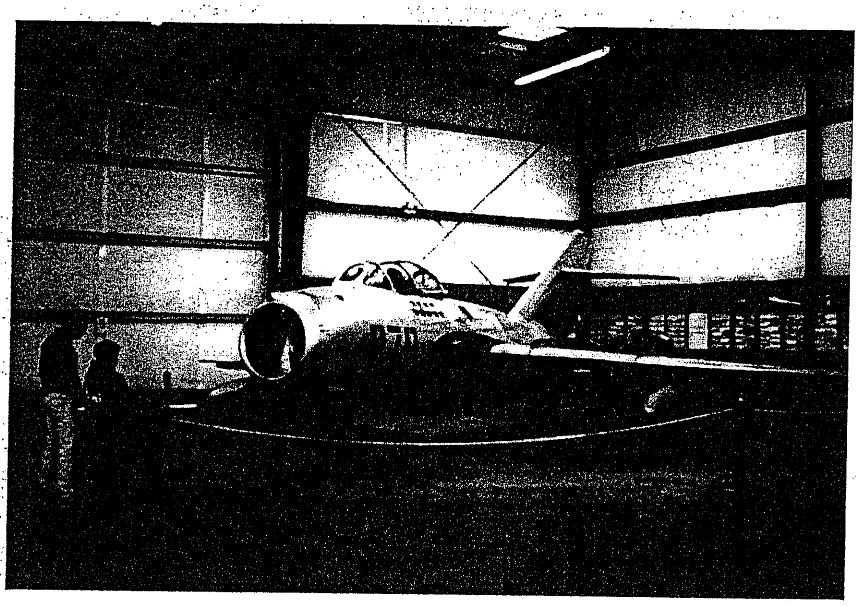


Fig. 33.5 Spanwise flow fences on the MIG-15.

33.9.3 Drooped Leading Edge

Another method that has been used to improve stall characteristics is the drooped leading edge. Although this method may be used as a fix during the flight test, it will cause less disruption to the test program if it is a part of the basic design. It may be only used on the outboard portion of the wing with a sharp discontinuity at mid-span, as in the NASA Safe Wing, discussed in the following chapter, or it may be gradually introduced along the span with the droop becoming more predominant as the tip is approached. In either case, its idea is to keep the outboard portion of the wing flying while the inboard portion is stalled. Therefore, it may be used for both roll-off and deep stall problems.

33.9.4 Tail-let

The tail-let is an additional small horizontal surface on a T-tail airplane that is placed low on the fuselage so that in the stalled condition it is below the wing wake and provides a nose down moment to the airplane for stall recovery. An example of its use is on the Beechcraft Model 1900 commuter airliner.

33.9.5 Electromechanical Fixes

Electromechanical fixes include stick shakers, stick pushers, and stall barrier devices. They are used as a last resort when an aerodynamic fix cannot be found or when the use of the aircraft is for pure transportation rather than multipurpose and when the cost of an aerodynamic fix might be prohibitive.

Their use is for both stall warning and stall prevention. In most systems of this type, the system is set to an angle of attack where first a stick shaker activates warning the pilot of the approaching stall. If the pilot continues to increase the angle of attack, the shaking of the control column becomes more severe and at a certain angle of attack takes control of the aircraft and moves the control column forward reducing the angle of attack of the aircraft. Flight test of these devices should ensure that varying deceleration rates will not fool the system and that the system is reliable, functioning every time as it should. If the system provides other than warning, then it becomes a "no go" item for the airplane flight manual. Advisory Circular 23-8A or its successor should be consulted prior to beginning testing of such a device.

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness, Normal, Utility, and Acrobatic Categories," Federal Aviation Agency, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, Washington, D.C., June 1974.

[3]Federal Aviation Regulation Part 25, "Airworthiness Standards: Transport Category Airplanes," Department of Transportation, Federal Aviation Administration, Washington, D.C., June 1974.

[4]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, Washington, D.C., 1986.

34
Airplane Spin Testing

34.1 Introduction

In the nearly 100 years that airplanes have existed, the spin has remained a persistent problem. The investigation of the airplane's spin characteristics are among the most hazardous of flight tests. One of the reasons for this is that the dynamics of the spin are not well understood and a good deal of misinformation exists regarding spins. The spin is a disorienting maneuver that may cause even the experienced pilot to make the wrong control inputs to effect recovery. Additionally, modern aircraft designs contain design features that are adverse to good spin and recovery characteristics.

Although much effort has been made to make airplanes unspinnable, nearly all airplanes will spin under some condition. Early attempts at unspinnable airplanes limited c.g. travel and up elevator in an attempt to prevent the airplane from stalling and, therefore, spinning. However, this could be overcome by a zoom climb with resulting stall and spin when the airplane ran out of energy at the top of the climb.

From the flight test standpoint, the spin test is only performed on single-engine airplanes. Multiengine airplanes are exempt from spin testing by the FAA. There are two reasons for this: First, the single-engine airplane is more likely to be inadvertently spun due to the less experienced pilots that generally fly those airplanes. Second, multiengine airplanes are more likely to have unrecoverable spin modes than are single-engine airplanes due to their higher inertia in a spin.

34.1.1 Spin Definitions

Before we discuss the spin further, we should first define it. A spin is an out-of-control maneuver at angles of attack beyond the stall during which the airplane rotates about its c.g. and an axis perpendicular to the Earth while descending vertically at high rates of descent.

Another term that is often used in spin literature and discussions is the incipient spin. The incipient spin is a transition maneuver between the stall and the fully developed spin where the airplane begins its autorotation. The incipient phase of the spin may also be accompanied by large changes in pitch as the airplane rotates.

34.1.2 Spin Types

There are two types of spins. The most common type, and in most cases the most difficult type for recovery, is the upright or erect spin. In this spin the rolling and yawing motion are in the same direction. This type can occur from normal flying maneuvers and is the most likely cause of spin accidents. If this spin occurs at traffic pattern altitudes, a safe recovery is unlikely. The second type of spin is the inverted spin. In this spin the yawing and rolling motion are in opposite directions and although both types of spins are disorienting to the pilot, this opposition of rolling and yawing makes the inverted spin the most disorienting.

34.1.3 Spin Mode

The spin mode refers to the angle of attack at which the spin stabilizes. In general categories, the mode is either steep (40 deg angle of attack and below) or flat (60 deg angle of attack and above). The spin mode reached by the airplane is a function of the aerodynamic energy available to overcome the airplane inertia.

Some spins never stabilize at a constant angle of attack. These spins are described as oscillatory spins.

34.2 Federal Aviation Administration Regulations

The FAA Regulations for spin certification apply only to single-engine airplanes and have not changed significantly for a number of years. However, in recent years the FAA has adopted some regulations on spin resistance as a result of NASA research. This spin resistant regulation is discussed in a later paragraph. Multiengine airplanes are not required to be certified for spins. There are at least two reasons why these airplanes are not required to demonstrate spins. First, these aircraft are not as likely to be involved in inadvertent spins as the pilots of such aircraft are likely to be more experienced than are pilots of single-engine airplanes. Second, these airplanes have higher wing loadings and larger inertias and are much more difficult to recover from a spin.

34.2.1 Civil Aeronautics Regulations 3.124 Spinning (Ref. 1)

Spins in the CAR are required for all airplanes of 4000 lb or less maximum weight (this includes light twins) that are in the normal category. Aircraft in this category must recover from one-turn spins in one additional turn by use of normal spin recovery controls. The airplane may not exceed any airspeed or load factor limit during recovery and shall not exhibit any excessive control back pressure either during the spin or its recovery. It shall not be possible to obtain an uncontrollable spin by any possible use of the controls. Spins must be demonstrated for all approved combinations of weight and c.g. All normal category airplanes must be placarded "spins prohibited."

For utility category airplane spins must be demonstrated using either the normal category or acrobatic category rules. If the normal category rules are used these airplanes must also be placarded "spins prohibited."

The aerobatic category requires that the airplane demonstrate recovery from a six-turn spin, or at any time during such a spin, in one and one-half additional turns by normal use of the controls. Again, it must be possible to recover from the spin without exceeding airspeed or structural limits and any possible use of the controls should not cause an uncontrollable spin. A placard is required in the cockpit providing the proper control movements for spin recovery.

For airplanes that are described as "characteristically incapable of spinning," the flight test required to demonstrate this characteristic shall be conducted with: 1) a weight 5% in excess of the maximum weight requested for approval, 2) a c.g. at least 3% aft of the requested aft c.g. limit, 3) an up elevator travel that is 4 deg greater than that of the type design, and 4) a rudder travel that is 7 deg in each direction greater than that called for in the type design.

For either normal or acrobatic category spin tests, if during the recovery from the one-turn, flaps-down spin the airplane exceeds the flap speed or limit load factor, it is permissible to retract the flaps during recovery.

34.2.2 Federal Aviation Regulation 23.221 Spinning (Ref. 2)

Normal category airplanes must recover from a one-turn, or 3-s spin, whichever requires longer, in one additional turn after the initiation of the first control action for recovery, or the applicant may show compliance with the optional spin resistance requirements.

34.2.2.1 Normal category airplanes FAR 23.221(a)(1) one turn spin requirements. If it is elected to perform the one turn (or 3-s) spins, these must be demonstrated with both gear and flaps up and down and power on and off. The flaps may be retracted and the power may be reduced to idle during the recovery after the rotation has stopped. No speed or normal acceleration limit may be exceeded during the recovery. It must not be possible to achieve an unrecoverable spin through any use of the controls including the power control during any part of the spin. Control forces shall not prevent a prompt recovery.

34.2.2.2 Normal category airplanes FAR 23.221 (a)(2) optional spin resistance requirements. Spin resistance may be demonstrated by using a 1 kn/s deceleration rate until the airplane stalls, or full aft elevator control is reached. Then the elevator is held back against the aft stop and full pro spin rudder is introduced for 7 s or 360 deg of heading change, whichever occurs first. If the 360 deg of heading change occurs first, it must not have taken place in less than 4 s. This must be demonstrated with ailerons neutral and against the direction of rotation. It also must be demonstrated with power off and with 75% MCP. When prospin controls are removed, the airplane should recover immediately. If the aircraft meets this requirement, then in a stall with the elevator control held full aft it must demonstrate the ability to maintain wings level within 15 deg of bank and be able to roll from one 30 deg bank to the opposite 30 deg bank with the proper use of the controls. Stall characteristics must be demonstrated with the airplane in uncoordinated flight up to one ball-width displacement on the slip-skid indicator, or full rudder, whichever comes first.

34.2.2.3 Utility category airplanes FAR 23.221(b) spin requirements. Utility category airplanes may be certified for intentional spins using the requirements for acrobatic airplanes (FAR 23.221(c)) or the requirements for normal category airplanes given above. If the requirements for acrobatic category airplanes are met there must also be provision for the crew members to escape at the highest speed likely to be achieved in the maneuvers for which the airplane is certified (FAR 23.807(b)(6)).

34.2.2.4 Acrobatic category airplanes FAR 23.221(c) spin requirements. Acrobatic category airplanes must recover at any point in a spin up to and including six-turn spins, or a greater number of turns for which certification is requested, in one and one-half additional turns after initiation of the first control action for spin recovery. However, beyond three turns of spinning the spin may be discontinued if spiral characteristics develop. Airspeed and normal acceleration limits may not be exceeded during the spin, and if spinning with flaps is requested then the flaps may not be retracted during the recovery. It must be impossible to obtain an unrecoverable spin with any use of the aerodynamic or power controls on entry or during the spin. There also must be no characteristics during the spin that might prevent successful recovery due to disorientation or incapacitation of the pilot. In addition, the airplane must be so equipped as to allow the crew to abandon the airplane at any speed between the flaps down stalling speed, V_{S0}, and the dive speed, V_D.

34.2.3 Advisory Circular 23-8A, Section 8, Paragraph 100, Spinning

The Flight Test Guide for the Certification of Part 23 Airplanes[3] (AC 23-8A) provides additional guidance for the conduct of spin tests. Spins should be explored throughout the weight and c.g. envelope, including cases of asymmetrical fuel up to the maximum asymmetry requested. These loadings have usually been interpreted as the most forward and most aft c.g. at maximum gross weight. However, it may be wise to check some points at light weight.

Control travels should be set to the most critical side of the travel tolerances. This is easy to understand for the elevator, which should be to the high side of the tolerance with the elevator up and on the minimum of the deflection tolerance with the elevator down. The rudder, on the other hand, is not quite so simple, since the high side of the tolerance would provide more input as the spin is initiated while the minimum deflection appears to be critical for recovery, and on most airplanes we cannot have it both ways without significantly increasing the test matrix. Since recovery is critical, setting the rudder on the minimum tolerance would be the preferred method, although the advisory circular does not specify which tolerance is to be used, only that it be the critical tolerance.

The advisory circular suggests a spin recovery parachute and provides several NASA references for sizing the parachute. It also recommends structurally and functionally testing the spin recovery parachute. The advisory circular states that final certification of the spin characteristics should be conducted with the spin recovery parachute removed. For airplanes that are to be certified for intentional spinning this makes sense. However, it does not make good safety sense for normal category airplanes for which intentional spins are prohibited, since unrecoverable spins have been obtained after several satisfactory tests of the same type. It also conflicts with other FAA safety guidance such as FAA Order 8110.10. The advisory circular does not recommend jettisonable ballast since this has been found to not always affect spin recovery.

A buildup technique is recommended when any doubt exists regarding the recovery. One should keep in mind that the advisory circular is primarily aimed at FAA pilots who expect the applicant's (company's) test pilots to have previously conducted the tests. A company test pilot conducting the tests for the first time should *always* use a buildup technique.

The advisory circular provides a recovery technique of throttle reduced to idle, ailerons neutral, full rudder to oppose the spin followed by forward elevator to break the stall and a recovery to level flight. However, the advisory circular states, "unless the manufacturer determines the need for another technique." Therefore, recovery techniques that work best for a specific airplane type may be used provided they are given in the airplane flight manual or Pilot's Operating Handbook.

For airplanes with trimmable stabilizers, the advisory circular states that critical positions should be investigated, but does not specify the position. That position is most likely to be the full nose-up position since it would drive the aircraft to a higher angle of attack and provide less nose down pitch for recovery.

The advisory circular recommends that spin testing should be conducted at high altitudes with a hard altitude set for emergency egress. This will be further discussed in a later part of this chapter.

The advisory circular provides a recommended test matrix for the various control configurations in spin investigations. This matrix is Fig. 100-1 in the advisory circular and is shown here as Fig. 34.1. As one can see from the figure this test configuration matrix is extensive suggesting over 240 spins, not including buildup, for a normal category spin certification. If this matrix is followed, a normal category spin certification program will approach 1000 spins when the developmental spins and safety buildup are included. This magnitude of a spin program resulted in a complaint from industry and a much more reasonable matrix was given in AC 23-15 (Ref. 4). However, if an applicant wants an airplane to be certified for spins in either the utility or acrobatic category the matrix of Fig. 100-1 is still the recommended matrix.

The advisory circular discusses the spiral characteristics referred to in FAR 23.221 for acrobatic category spins. The advisory circular cautions that only an individual spin may be discontinued after a spiral develops and not the entire spin program. It states that the airplane can be certified as acrobatic even if it cannot spin a minimum of six turns.

The use of maximum available power is clarified as only being required for one turn of the spin even for acrobatic airplanes.

Critical spin tests should be repeated with instrumentation removed if the test airplane contains complex instrumentation such as wing tip booms or telemetry systems.

Minimum instrumentation for the tests consists of a calibrated airspeed indicator, accelerometer, and altimeter. However, additional instrumentation, such as control position indicators and fuel counters for accurate determination of aircraft loading, are also considered essential.

Regarding optional equipment that may affect the spin, the advisory circular suggests that sufficient tests should be conducted with the equipment installed to ensure compliance with the regulation in all configurations. Optional equipment listed includes deicing boots, asymmetric radar pods, outer wing fuel tanks, and winglets.

For acrobatic category airplanes or utility category airplanes certified for acrobatic spins, a placard is required that lists the use of controls necessary to recover from the spin. Recovery control inputs should be conventional, or at least not create a recovery problem.

34.2.4 Advisory Circular 23-15 (Ref. 4)

Due to the complexity that developed in AC 23-8A in its attempt to cover both commuter and small airplanes, the small airplane industry petitioned the FAA for an advisory circular that more closely addressed the needs of the small airplane. That resulted in AC 23-15 which makes some modifications to the provisions of AC 23-8A, Change 1, in order to simplify spin testing for normal category airplanes.

First, the aircraft loading was clarified to specify that only the corners of the aircraft's weight and c.g. envelope be tested.

Next, the advisory circular discusses control deflections. For airplanes with control rigging tolerances ± 1 deg or less, the controls should be set to nominal tolerances. For airplanes with control rigging tolerances greater than ± 1 deg the tolerances should be set to nominal and if any spin configuration is found critical (requires near one turn to recover) then the effects of control rigging should be explored.

Advisory circular 23-15 also clarifies the settings for trimmable stabilizers. In the case of airplanes equipped as such, the spin tests should be conducted at the conditions of the stabilizer set at a trim airspeed of $1.5V_{S1}$ in the cruise configuration, $1.2V_{S1}$ in the takeoff configuration and $1.3V_{S0}$ in the landing configuration.

For the test configuration matrix, the advisory circular precedes the matrix with the following statement: "The airplane configuration to be evaluated should correspond to the configuration expected to be used for takeoff, cruise, and landing. For a normal category airplane, the spin test matrix should cover those abnormal control inputs reasonably expected from a pilot experiencing an inadvertent spin entry such as forgetting to reduce power, reflexively applying antispin aileron (against the turn) in response to roll-off, or inadvertent reversing recovery rudder and elevator sequencing. The "normal category spin test matrix" is an acceptable method of compliance, unless unusual response requires additional exploration." This matrix is given as Fig. 12 in the advisory circular and is reproduced here as Fig. 34.2. This test matrix is much more reasonable for a normal category airplane, since it explores the most likely actions an inexperienced pilot would take for an inadvertent stall and resulting spin entry.

SPIN EVALUATION
CONFIGURATION

Flight Condition Spins from Wing Level Attitude	Spin Number	Flaps Up	Flaps Appch.(As Approp.)	Flaps Landing	Gear Up	Gear Down	Cowl Flaps Closed	Cowl Flaps As Required	Power Off	Power On	Forward C.G.	Aft C.G.	Lateral C.G.
Test with Normal Spin Controls	1	X			X		X		X		X	X	X
Left Spin 1 thru 6	2		X			X	X		X		X	X	X
	3			X		X	X		X		X	X	X
	4	X			X			X		X	X	X	X
	5		X			X		X		X	X	X	X
	6			X		X		X		X	X	X	X
Repeat 1 Through 6 from a right spin. Repeat 1 through 6 from left and right turning flight.													
Tests with Abnormal Spin Controls	7	X			X		X		X		X	X	X
	8		X			X	X		X		X	X	X
Left Spin Aileron Against 7 Thru 12	9			X		X	X		X		X	X	X
	10	X			X			X		X	X	X	X
	11		X			X		X		X	X	X	X
	12			X		X		X		X	X	X	X
Repeat 7 Through 12 From a Right Spin													
Left Spin Aileron with 13 Thru 18	13	X			X		X		X		X	X	X
	14		X			X	X		X		X	X	X
	15			X		X	X		X		X	X	X
	16	X			X			X		X	X	X	X
	17		X			X		X		X	X	X	X
	18			X		X		X		X	X	X	X
Repeat 13 Through 18 From a Right Spin Repeat 7 Through 18 From Left & Right Turning Flight													

Fig. 34.1 AC 23-8A spin evaluation configuration matrix.[3] © QinetiQ Limited 1953, reproduced by permission of QinetiQ Ltd.

NORMAL CATEGORY SPIN TEST MATRIX

Configuration*

Normal Spins	Level Entry	Left (Lt) Turn	Right (Rt) Turn
Clean, Power Off	1 Lt, 1 Rt	1 Lt, 1 Rt	1 Lt, 1 Rt
Takeoff, Power On	1 Lt, 1 Rt	1 Lt, 1 Rt	1 Lt, 1 Rt
Landing, Power Off	1 Lt, 1 Rt	1 Lt, 1 Rt	1 Lt, 1 Rt
Abnormal Spins	**Power On Ailerons Against**	**Power Off Ailerons Against**	**Power Off Elevator 1st Recovery**
Clean	1 Lt, 1 Rt	1 Lt, 1 Rt	1 Lt, 1 Rt
Takeoff	1 Lt, 1 Rt	1 Lt, 1 Rt	1 Lt, 1 Rt
Landing	1 Lt, 1 Rt	1 Lt, 1 Rt	1 Lt, 1 Rt

*For normal category airplanes, the configurations to be examined are defined as follows:

Clean	Flaps up Gear up Cowl flaps closed
Takeoff	Flaps at maximum approved to takeoff setting Gear down Cowl flaps open
Landing	Flaps full down Gear down Cowl flaps closed

Each of these configurations should include testing at the combination of weight and CG determined to be most critical for the proposed loading envelope.

Fig. 34.2 AC 23-15 normal category spin test matrix.[4]

34.3 Spin Theory

As discussed earlier, the spin is an out-of-control maneuver, above the stall angle of attack, where the airplane is rotating about its c.g. and an axis perpendicular to the Earth while descending at a high rate of descent (see Fig. 34.3). The rotation is self-sustaining and can be described as autorotation.

34.3.1 Factors That Cause Autorotation

There are a number of factors that cause, or contribute to, the autorotation. The primary factor is the unequal lift and drag on the wing. However, other

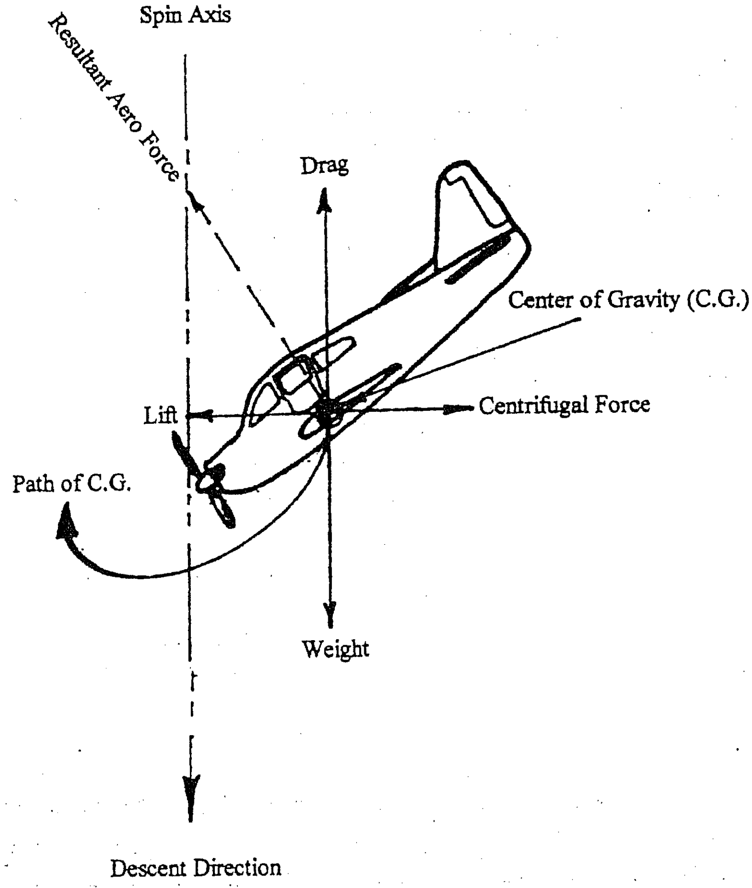


Fig. 34.3 Forces and dynamics of a spin.

factors such as the cross sectional shape of the fuselage and control movements contribute to the autorotation and may determine the difference between a spin that is recoverable and one that is not.

34.3.1.1 Unequal lift and drag on the wings. If both wings were to reach $C_{L\max}$ at the same time in a stalling maneuver and remained at the same angle of attack during the stall a spin could not occur. This is what is being attempted

to achieve during stall characteristics testing. It is represented by point A in Fig. 34.4. This shows that if both wings are at the same angle of attack, then the drag on each wing will be the same. However, if due to a gust, slight differences in their manufacturer, or some other reason, a sideslip exists the wings do not reach $C_{L\max}$ at the same time. Then one wing will stall before the other and we will have the case represented by points B and C in Fig. 34.4. We can then see

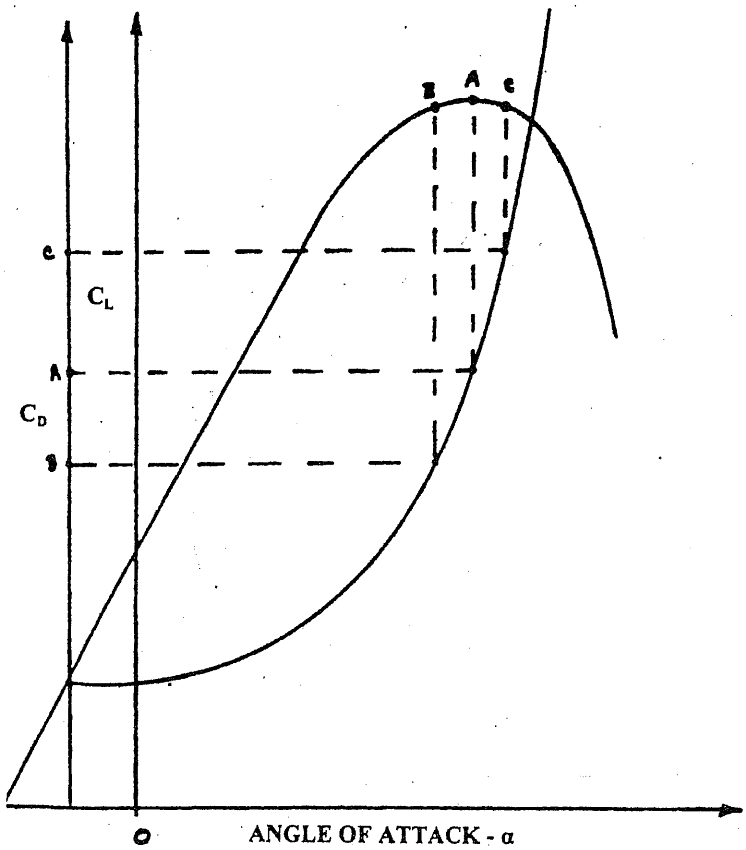


Fig. 34.4 Lift and drag coefficient curves showing the effects of small changes in angle of attack on the wings potential for autorotation.

that the drag on each wing is no longer equal but tends to yaw the aircraft into the stalled wing. This causes the stalled wing to drop while the near-stalled wing rises resulting in an increase in the angle of attack of the stalled wing while the angle of attack on the rising wing is reduced. This motion causes the difference in drag between the wings to increase setting up the autorotation. The rolling moment created by the unequal lift has been shown to be essentially a linear function of the wing thickness to chord ratio t/c (Ref. 5). This is shown in Fig. 34.5 (Ref. 5).

34.3.1.2 Control inputs. If surprised by the sudden drop of the wing, the pilot is likely to move the controls so as to raise the wing (i.e., ailerons against the roll). Fig. 34.6 shows that this control input causes the down-going, or fully stalled, wing to increase its angle of attack even further while the up-going wing's angle of attack is further reduced. As you can see, this makes the autorotative forces even greater.

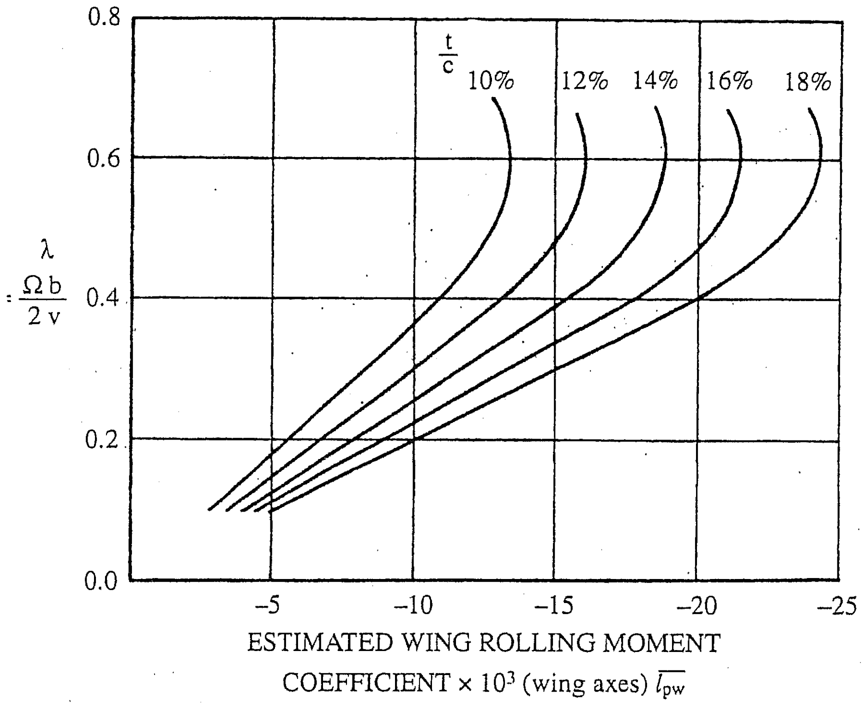


Fig. 34.5 Wing rolling moment coefficient as a function of t/c^5.

34.3.2 Effects of Airplane Mass Moments of Inertia

As the rotation rate of the spin increases, the airplane mass distribution begins to play a significant role in the spin. If the mass is concentrated about the c.g. as shown in Fig. 34.7, then the spin angle of attack will remain steep (below 40 deg). If the airplane mass concentrations are located away from the c.g. as shown in Fig. 34.8 then the spin will flatten. The final angle of attack at which the airplane reaches a trim point will then be a function of the balance between the aerodynamic damping provided to slow the rotation rate and the rotational inertia. Recent test data of two general aviation aircraft have shown that the rolling moment of inertia may have a larger adverse effect upon spin characteristics than previously thought. Since modern general aviation aircraft have tended to higher rolling moments of inertia due to increased fuel in the wings, certification of these airplanes for spins may become more difficult.

34.3.3 Effects of the Tail

During intentional spins, the horizontal and vertical tails provide the input to cause the airplane to spin. They also provide the input for spin recovery for all spins. However, once in the spin these surfaces may have a significant effect upon the aerodynamic damping. The location of the horizontal tail with respect to the vertical tail can change the damping provided by the vertical tail

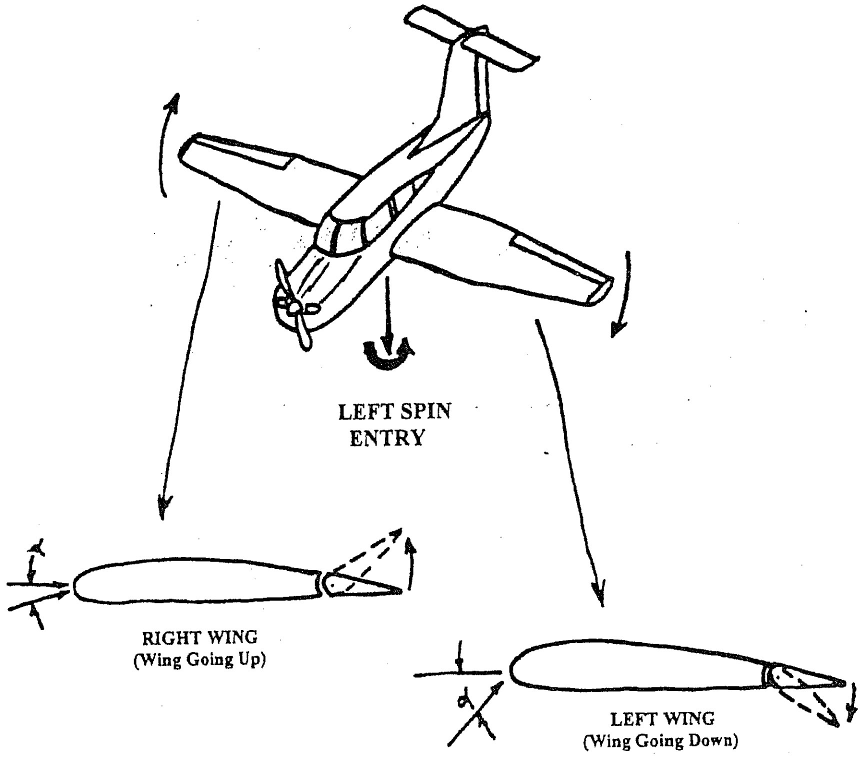


Fig. 34.6 Pilot reaction to surprise stall/spin entry.

and rudder. Generally, for upright spins, a horizontal tail mounted low on the vertical tail reduces this damping while a high-mounted tail, or T-tail, improves the damping provided by the vertical stabilizer and rudder. The damping provided by the vertical tail was so powerful on aircraft tested in the 1930s, 1940s, and 1950s that the National Advisory Committee for Aeronautics (NACA) believed that it was the prime factor in the spin behavior of aircraft. As a result they developed a spin recovery criteria called tail damping power factor (TDPF). This criteria was based upon the testing in the NACA vertical wind tunnel at Langley Air Force Base in southern Virginia. From these tests criteria were developed along with charts for predicting spin behavior of various combinations of airplane mass distribution, TDPF, and aircraft relative density (μ, density of the aircraft with respect to the air surrounding it). The method of calculating TDPF is shown in Fig. 34.9 taken from NACA TN No. 1045.[6] The mass distribution was accounted for in a parameter called the inertia yawing moment parameter (IYMP). The results were plotted on charts such as the one shown in Fig. 34.10 (Ref. 6) with values above the line representing satisfactory spin characteristics and those below the line representing unsatisfactory spin characteristics. However, if one looks at the actual test results used in obtaining Fig. 34.10 as shown in Fig. 34.11 (Ref. 6) one will find that a number of unsatisfactory spins were obtained in the good spin region. The

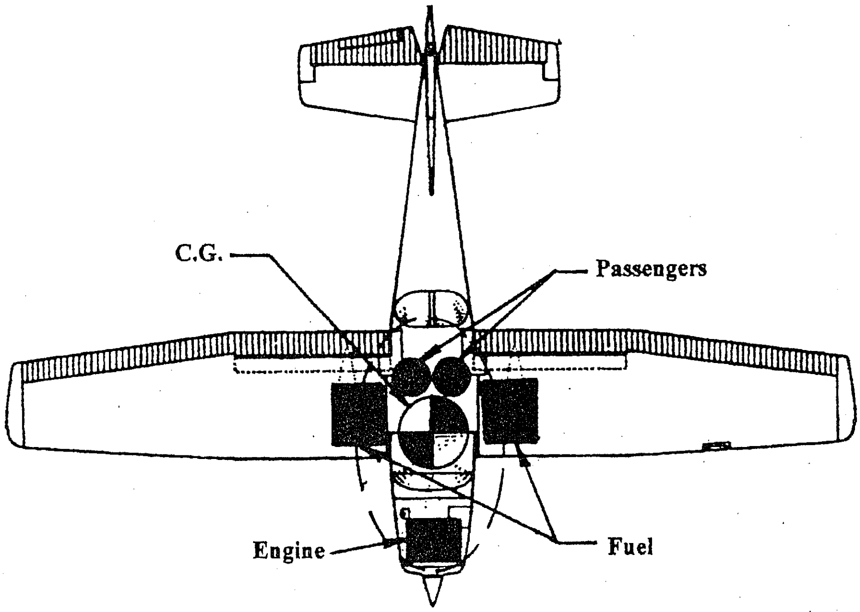


Fig. 34.7 Typical general aviation airplane with mass concentrations near the c.g. and along the fuselage.

NACA followed up this report, which was to apply to all airplanes, with another report that applied to personal owner type airplanes. This report was NACA TN No. 1329 (Ref. 7). The NACA, followed by their successor the National Aeronautics and Space Administration (NASA), continued to promote TDPF as a spin recovery criteria through the mid-1970s, including the publication of NASA TN D 6575 (Ref. 8) "Summary of Spin Technology as Related to Light General Aviation Airplanes." They also continued research in the general aviation stall/spin field.[9,10] However, the loss of a test aircraft (which had the author as pilot) with a T-tail and a TDPF that was off the chart in the good direction (see Fig. 34.12) resulted in NASA beginning to back away from TDPF as a sole criteria.[11,12]

The aircraft involved in the accident had a longer nose than its predecessor and just prior to the accident had strakes added to the nose. This caused the author to believe that the fuselage, particularly the nose and its length with respect to the tail length, had a larger effect upon spins than had been previously believed by the folks at NACA and NASA. This will be discussed further in this chapter.

34.3.4 Effects of the Fuselage

As the angle of attack increases and the airplane begins to rotate, the fuselage begins to play a larger role in the airplane's aerodynamics. Studies conducted in Great Britain[13] and France[14] have shown that the fuselage shape

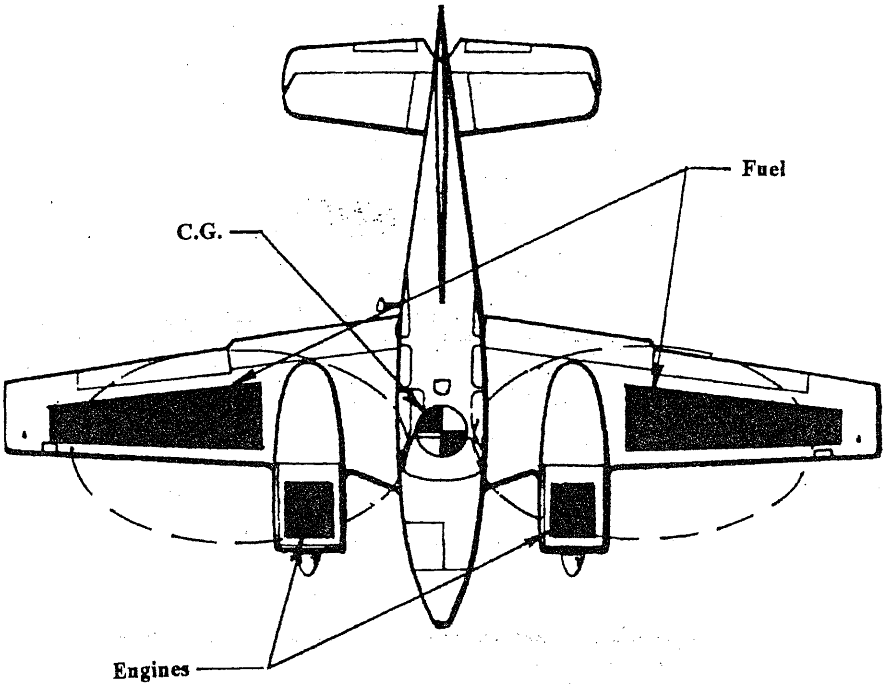


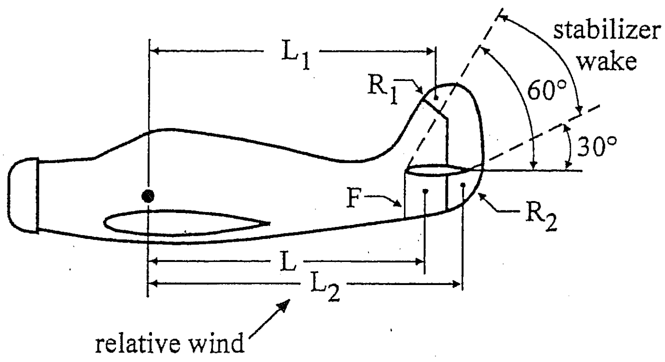
Fig. 34.8 Typical light twin aircraft with mass concentrations along the wings and away from the c.g.

plays a significant role during the spin and that certain shapes are much better than others. Table 34.1 taken from the work of Kerr[13] shows the relative merits of various shapes.

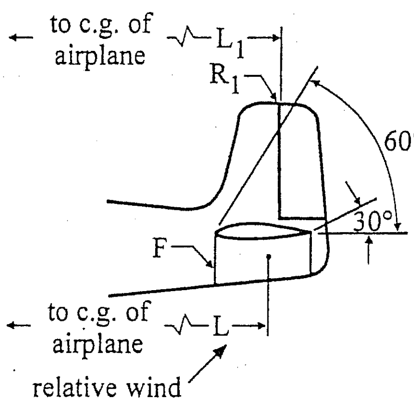
From this table one may also see the powerful effect of aft fuselage strakes, which helps one understand why they appear on many spinnable airplanes. One may also see that the round top, flat bottom of many general aviation light airplanes is less than ideal for spins.

Research performed by Clarkson, et al.,[15,16] and Polhamus[17] on fuselage shape in the NASA Ames Research Center high-pressure wind tunnel has shown that the effects of the fuselage are subject to large changes with Reynolds number. In fact, this research has shown that the fuselage contribution may change from antispin to prospin, or vice versa between small-scale and full-scale Reynolds numbers. This may explain why the vertical wind tunnel or spin tunnel is sometimes not a good predictor of full-scale spin characteristics. Clarkson's work also showed that these effects were a function of the fuselage corner radii.

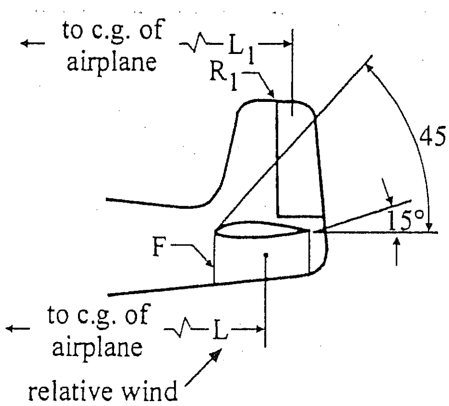
34.3.4.1 Effects of the nose of the aircraft. For a number of years it has been known that the nose of the aircraft plays a role in its spin characteristics.



(a) Full-length rudders α assumed to be 45 deg



(b) Partial-length rudders α assumed to be 45 deg TDR > 0.019



(c) Partial-length rudders α assumed to be 30 deg TDR > 70.019

Fig. 34.9 Method for determining tail damping power factor.[6]

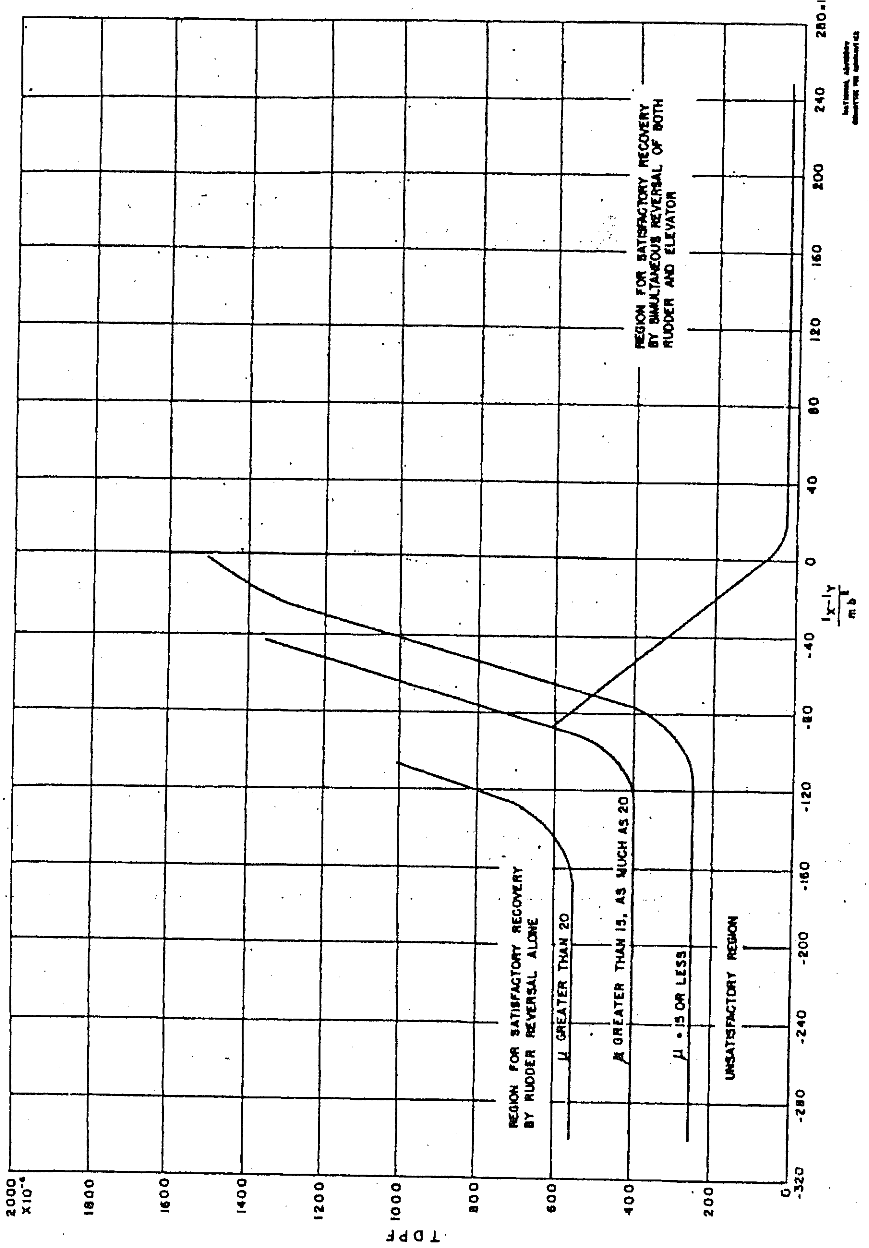


Fig. 34.10 Composite of spin recovery design requirements for airplanes with relative densities between 0 and 35 (Ref. 6).

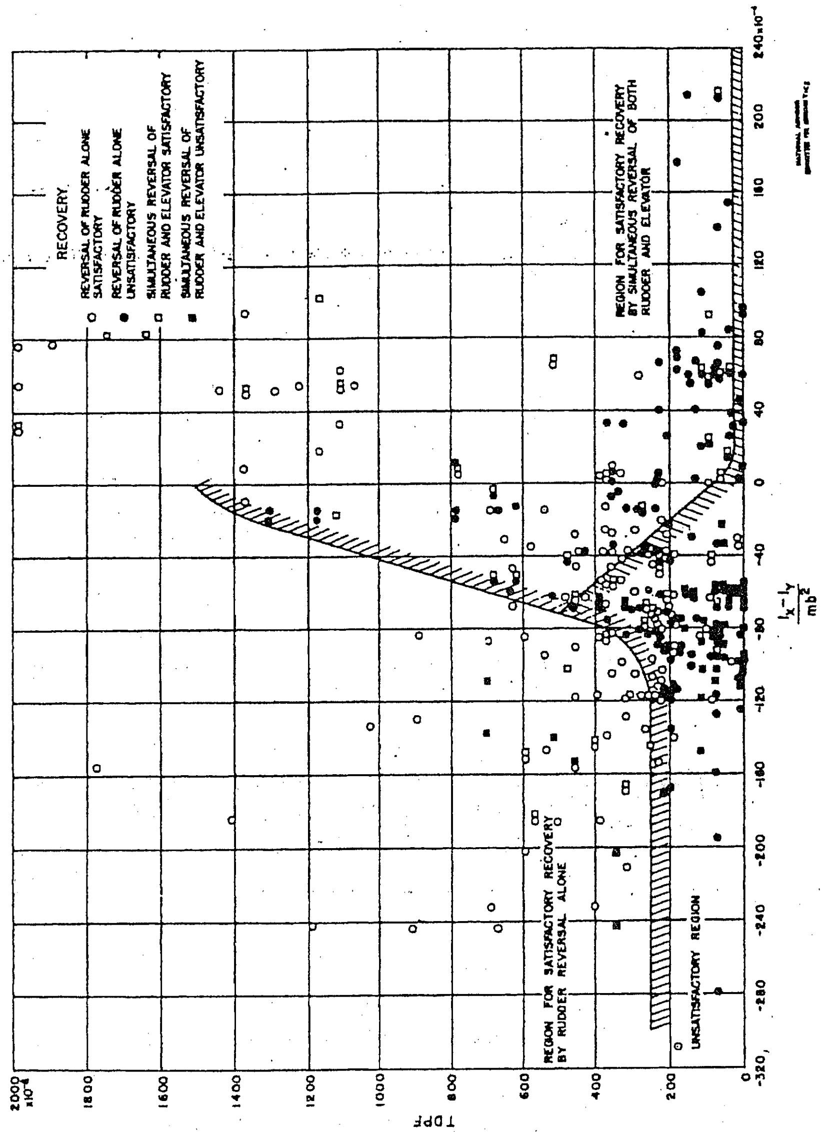


Fig. 34.11 Example data plot used in developing TDPF criteria for airplanes with relative densities of 15 or less.[6]

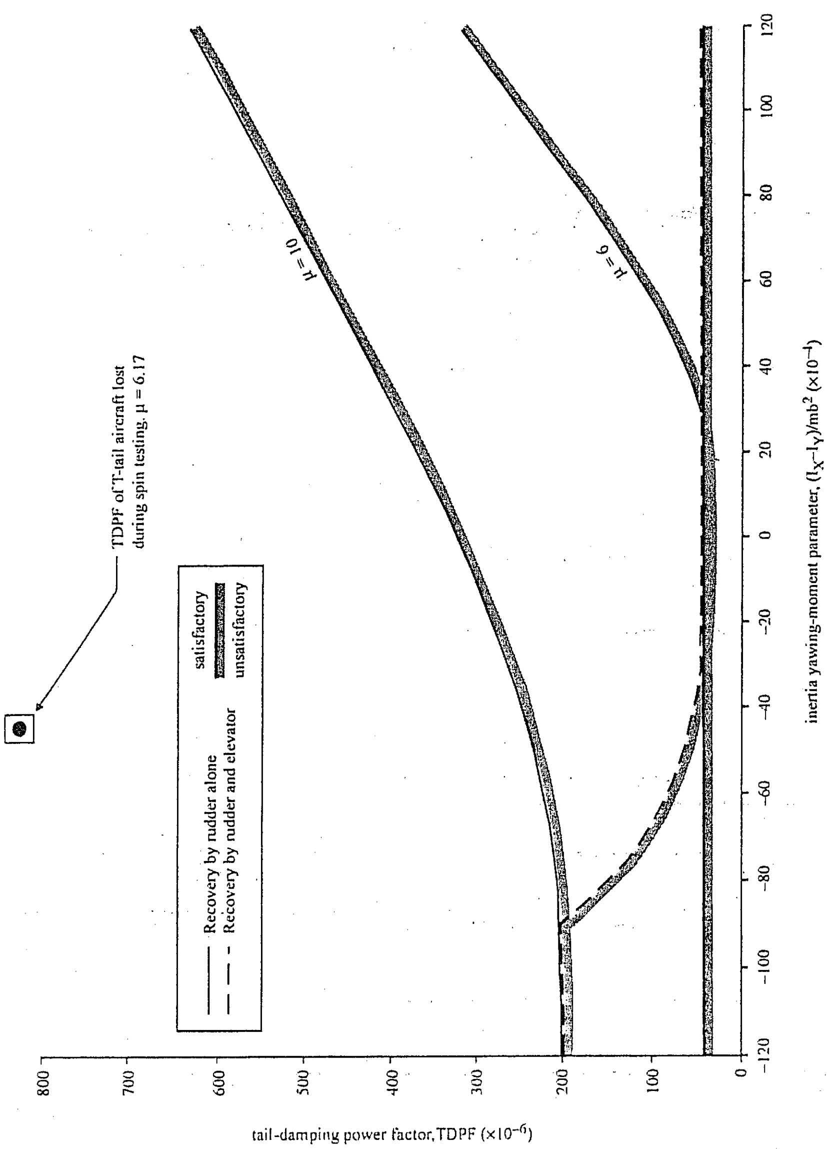


Fig. 34.12 Plot of TDPF of T-tail aircraft lost during spin testing. TDPF diagram taken from Ref. 8.

Table 34.1 Relative merits of aft fuselage cross-sectional shape on spins[13]

Body cross-section		ε at $\alpha = 45°$
Circular (pointed profile)		+0.6
Rectangular		+1.5
Elliptical		+2.1
Round top, flat bottom		+1.1
Round top, flat bottom + strakes		+1.7[a]
Round bottom, flat top		+2.5
Round bottom, flat top + strakes		+3.5[a]
Fin	Free	+1.5
	under tailplane	+3.0
	above tailplane	−0.4
Rudder under tailplane		+2.0

[a] Depending upon width of strakes. This is for 0.014 "l," where "l" is the distance from c.g. to rudder post; ε is the weighting factor applied to various parts of side area.

Aircraft such as the Cessna T-37 Jet Trainer required modification to the nose in order to achieve satisfactory spin characteristics. The British spin criteria of Ref. 13 accounted for both the length and cross-sectional shape of the nose in addition to the damping provided by the vertical tail. The author's own bad experience with an aircraft that had length and shape modifications to its nose resulted in a private historical study based upon personal experience and knowledge of a number of spin flight test programs, plus an evaluation of several aircraft that were certified for spins. This study resulted in the development of a simple criteria for spinnability based upon the ratio of nose length to tail length.

34.3.4.2 Nose length/tail length ratio. The nose-length-to-tail-length ratio is determined as follows. The lengths are measured from the wing 25% MAC point to the propeller hub for the nose length and from the wing 25% MAC point to the 25% MAC point of the vertical tail for the tail length. Wingspan is used as the other variable since the wing provides the autorotation characteristics and much of the energy for the spin. Fig. 34.13 shows the results of the study. From the figure, one can see that for a ratio of 0.4 or less none of the airplanes in the study had spin problems. For ratios of 0.4 to 0.5 the results were mixed and for ratios above 0.5 all the airplanes in the study had problems even with one turn spins. As of this writing, the author has not seen a single airplane that does not confirm this simple criteria. These results in essence say to avoid long noses during design of airplanes that must be certified for spins.

34.3.4.3 Nose shape. Based upon the author's experience, the nose cross-sectional shape also plays a role in the spin. Both the British criteria[13]

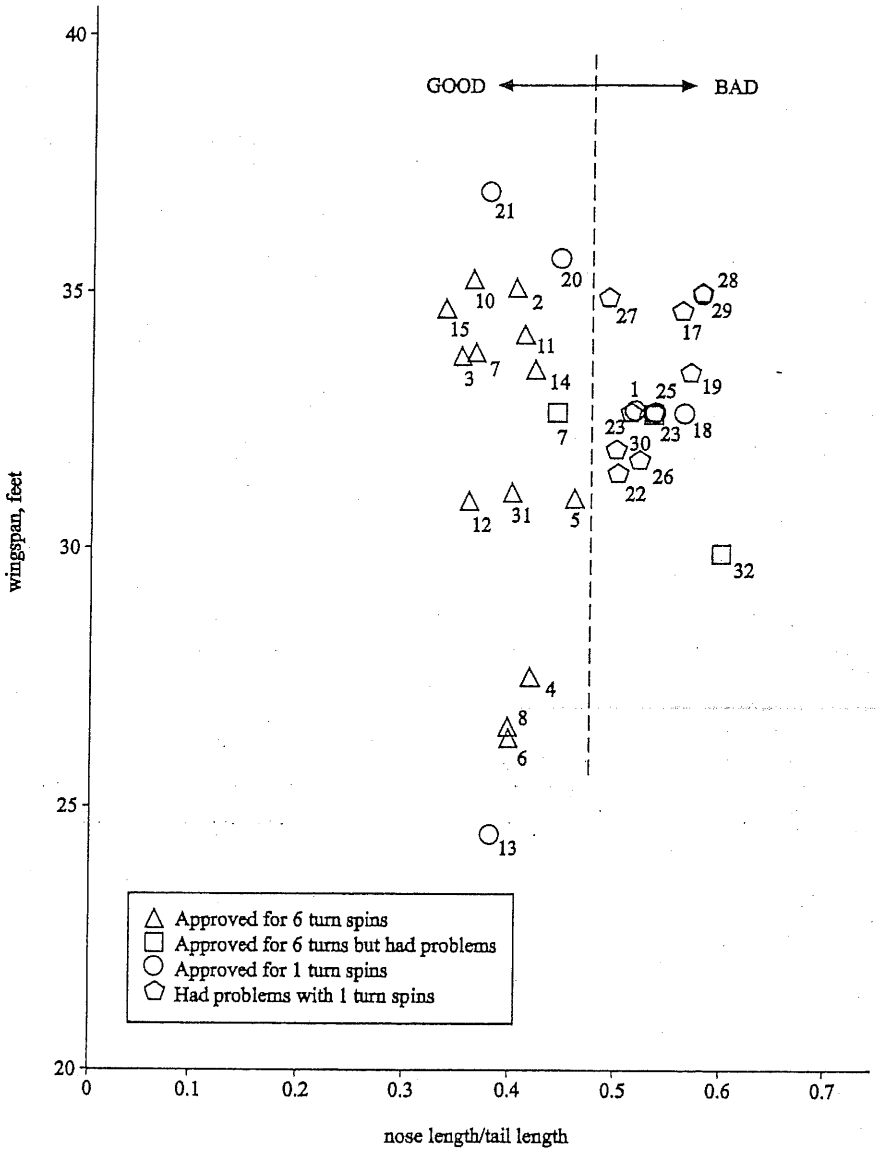


Fig. 34.13 Spin certification results for 32 general aviation aircraft showing nose length/tail length vs wingspan.

and the French study[14] seem to give some credibility to this thesis. From experience it would seem that the nose should have a round top and a flat bottom for good spinnability, which is the opposite of the aft fuselage. Current shapes seen on most general aviation airplanes violate this theory. The British criteria[13] does account for shape by making the length of fuselage ahead of the wing 25% chord a negative number in their spin equation. The corner radii also play a role as was demonstrated in one full-scale test program of which the author has knowledge.

34.3.5 Contribution of the Wing

As discussed earlier the unequal lift and drag on the opposite wings provide the aerodynamic forces for the autorotation. The rolling moments generated by the wing during the spin are nearly linear and are a function of maximum wing thickness to chord ratio (t/c_{max}) as shown in Fig. 34.5 (Ref. 5). The wing provides the energy to initiate spinning and provides energy to continue the spin.

During their General Aviation Stall-Spin Research Program in the 1970s and 1980s NASA developed some spin resistance techniques using a sharp discontinuity in the wing leading edge at about mid-span just in front of the inboard edge of the aileron with the outer portion of the leading edge drooped. This modification later became known as the NASA Safe Wing. The purpose of the modification was to make the aircraft spin resistant. Results of flight tests of one such airplane modified with this leading edge droop are given in NASA TP 2691 (Ref. 18).

Some recent flight tests have shown that fuel located in the wings can have a significant effect upon the spin. In at least two instances, removing the fuel from the wings caused airplanes, which would not recover from a spin of more than two to three turns, to be recoverable from spins up to six turns. Therefore, it appears that fuel in the wings, which contributes to the inertias about both the X and Z axes, has a large effect upon the spin.

34.3.6 Effects of Altitude

The air density has a direct relation to the aerodynamic damping in the spin and nearly all spin researchers have considered it. In most cases[8] it is used as a factor called the relative density factor μ.

$$\mu = m/\rho S b \tag{34.1}$$

where
μ = the relative density factor
m = the airplane mass
ρ = the air density
S = the wing area
b = the wing span

Airplanes with high relative density factors generally require more antispin control effectiveness than airplanes with low relative density factors all else being equal.

34.3.7 Effects of Power, Landing Gear, and Flaps

Both CAR 3 and FAR Part 23 only allow airplanes to be intentionally spun in the clean configuration with the power at idle. However, airplanes can spin during any flight phase, so the regulations allow for this by requiring one turn spins in all configurations of power, landing gear, and flaps.

34.3.7.1 Power effects. It has been the author's experience in 10 spin programs where power for one turn was required that power has a favorable effect on recovery. This is likely due to the slipstream causing the flow along the fuselage to reattach increasing the fuselage damping of the spin. The author has knowledge of only one spin program where power was detrimental to recovery. However, if the spin is allowed to progress for much more than one turn, the fuel being thrown to the outside of the fuel tanks, airplanes with tanks in the wings will likely cause the engine to stop. For years the recovery procedure for inadvertent spins has been published in airplane handbooks with the first action being to reduce the throttle to idle before taking other actions. This author recommends that the power setting remain at the initial setting until rotation stops unless, after application of antispin aerodynamic control inputs, the spin shows no intention of stopping. In fact, in a recent program, power was used to initiate recovery from an out-of-control spin.

34.3.7.2 Effects of landing gear. Again, the author's experience has been that the position of the landing gear has not had a significant effect upon spin characteristics. However, landing gear position does effect airplane inertias and lowering of the landing gear, for gear that retract inboard, may have a detrimental effect upon recovery. Likewise, if the gear are extended for such a configuration, raising the landing gear may assist in recovery.

34.3.8 Effects of c.g. Location

34.3.8.1 Longitudinal c.g. location. Aft movement of the longitudinal c.g. reduces the effective control moment of both the rudder and the elevator, has the effect of making the nose longer, and increases the fuselage moment of inertia due to increasing the arm of the engine mass. In most instances, aft movement of the c.g. causes spin recovery characteristics to degrade. However, the author has experienced aircraft in which aft movement of the longitudinal c.g. did not cause a significant degradation of spin recovery characteristics, and it should be noted that a bad spin mode can be obtained at forward longitudinal c.g. locations.

Even with this caution, it is normal to start spin evaluations at the most forward longitudinal c.g. location and conduct the evaluation in increments of aft c.g. movement.

34.3.8.2 Lateral c.g. location. Very little research has been conducted into the effects of lateral c.g. location on spin characteristics. Therefore, conduct of spin evaluations with the lateral c.g. displaced from the aircraft centerline should be approached with caution. A displacement of the lateral c.g. due to a large fuel unbalance or other cause certainly changes the rolling and yawing moments of inertia and is likely to have an adverse effect upon spin characteristics.

34.3.8.3 Vertical c.g. location. At high angles of attack, a high vertical c.g., through geometry, gives the appearance of a further aft longitudinal c.g. As a result, the FAA has required the consideration of vertical c.g. location during stall characteristics and spin testing. The vertical c.g. of passengers and payload may be simulated during testing by use of elevated ballast boxes.

34.4 Planning for Spin Testing

Spin testing should be approached with considerable caution. In spite of the fact that airplanes have been around for nearly 100 years, we still know relatively little about the complex aerodynamics of the spinning airplane. The result is that many simple theories have been proven invalid, and we continue to have airplanes that have trouble recovering from a spin. All too often these problems are laid at the feet of the engineering test pilot and that person's competence. The "right stuff" syndrome has led to ignoring real problems with the airplane design until the aircraft, and all too often the test pilot, is lost in an accident. As a result of this potential hazard, spin testing should be planned in detail.

Since the lead time is quite long for their development, spin recovery devices should be considered very early in the planning. There are several potential spin recovery devices and more than one should be considered in the planning. Such devices include spin recovery parachutes, antispin rockets, and jettisonable ballast.

In addition, plans for aircraft egress and equipment, the use of chase aircraft, installed instrumentation, and test methodology should also be planned.

34.4.1 Spin Recovery Parachutes and Other Recovery Devices

The design and development of the spin recovery devices should not be taken lightly. These devices may save the test aircraft and the life of the test pilot as emergency egress during spin testing is difficult, at best. First, consideration should be given as to what devices shall be used and if a backup system is to be included. The author recommends a backup system as even the best designed systems have been known to fail. One should also realize that the only true test of a recovery system is in the actual event and simulating that event for a test is very difficult if not impossible.

34.4.1.1 Spin recovery parachutes. The spin recovery parachute has been used as a recovery device for spins that will not recover by use of the aerodynamic controls since at least the 1940s.[19–21] Information learned through the 1970s on spin recovery parachute design is summarized in NASA TN D-6866 (Ref. 22). More recent information obtained on NASA's General Aviation Spin Research Program provides a spin recovery parachute design that was extensively tested before and during actual unrecoverable spins experienced in that program.[23] Care should be taken in the cockpit location of the deployment and jettison controls. These controls should not be located in close proximity to one another, but separated so that the pilot would not inadvertently operate the wrong control in the heat of an uncontrollable spin. The deployment control should be located close to the throttle so that it is easy to find. The jettison control should be within easy reach, but placed so that it could not be accidentally operated when the pilot is trying to deploy the recovery parachute.

Prior to the actual conduct of initial spin tests, the spin recovery parachute system should be tested on the aircraft. This testing, in itself, can be hazardous, so a buildup technique to try to simulate the conditions of a spin as nearly as possible should be attempted. Initial deployments of the recovery parachute should be conducted on the runway during high-speed taxi test just below lift-off speed. These tests confirm the operation of the deployment and release systems. Several of these tests should be performed before taking the system and aircraft to flight to insure the operation of these mechanisms. Once satisfied that these mechanisms work as intended, deployments and releases in-flight can be attempted. Initial in-flight deployments should be conducted at airspeeds just above stalling speed and at a high enough altitude to insure time for an egress in case the recovery parachute will not jettison from the aircraft. A chase aircraft whose crew is briefed with instructions for the test pilot should the recovery parachute not jettison should be used on all tests of the spin recovery parachute system. This chase aircraft would also be responsible for following the parachute after jettison to insure that it is not lost during the tests. As testing progresses, deployments during stalls and initial departures can be evaluated to insure deployment during an actual spin. These deployments will insure that the chute does not foul on parts of the aircraft and that the riser length is correct. The spin recovery parachute when properly designed and tested is a reliable and powerful device for spin recovery.

34.4.1.2 Jettisonable ballast. A backup method for the spin recovery parachute system is jettisonable ballast. Its purpose is to rapidly shift the longitudinal c.g. forward providing a nose down pitching moment. Jettisonable ballast is not recommended as a primary method of recovery since a shift of the longitudinal c.g. may not be sufficient to effect recovery. Examples of jettisonable ballast include water tanks and lead shot tanks. These devices may be tested on the ground to insure operation of the mechanism for jettisoning.

34.4.1.3 Antispin rockets or thrusters. Antispin rockets or thrusters located on the wing tips are very effective spin recovery devices. They should be designed so as to allow recoveries from right or left spins and attention should be paid to the human-factor aspects so they are not fired in the wrong direction causing the spin to progress to a higher mode. However, NASA Langley Research Center in their spin research program of the late 1970s and early 1980s did use these devices to spin the airplane to a high rotation rate to look for hidden spin modes that might not appear from more normal spin entries.

34.4.2 *Aircraft Egress Planning*

Since spin testing by its nature is a hazardous test, the aircraft and the pilot should be configured for rapid egress. Military fighters and attack aircraft that may require spin testing come equipped for such egress without special effort. However, general aviation aircraft do not come equipped with such systems and require special provisions to be made. To equip such aircraft with ejection seats or extraction systems would likely be prohibitive so other provisions must be made. To date these systems have taken three forms:

1) quick release doors
2) pyrotechnic escape hatches
3) ballistic parachutes that recover the entire aircraft

34.4.2.1 Quick release doors. Quick release doors usually involve a method to remove the hinge pins and open latches simultaneously to insure that the door leaves the aircraft. One method of achieving this is to attach the hinge pins and opening mechanism to a T-handle wherein a single pull of the T-handle removes the hinge pins and opens the latches with one motion. Care should be taken to insure that the door does not remain attached to the aircraft where it might prevent egress or flail the pilot during egress. The jettison T-handle should be so located for easy access even when the pilot is under positive, negative, and side *g* loadings. These doors should be tested on the ground for function prior to starting spin testing.

34.4.2.2 Pyrotechnic escape hatches. On aircraft where the door is located some distance from the pilot's seat, it may be desirable to create an emergency opening next to the pilot. In this case pyrotechnic cord may be fashioned to cut a hole in the correct location. Considerable care should be taken in the design of such a system to insure that the system may not be activated accidentally. If an unuseable airframe exists, such as a completed static test article, pyrotechnic systems may also be tested on the ground for proper functioning.

34.4.2.3 Ballistic parachutes. The ballistic parachute is a system of last resort like other egress methods, since it is not likely that the test aircraft will survive the parachute landing in a reparable condition. However, for light aircraft such a device may be more reliable in saving the test pilot's life than an egress from the aircraft. Also, since these systems are now being offered on some production aircraft, the cost may be reduced to the level where they may be employed as a backup system for flight testing. It should also be noted that the FAA has allowed such a system to be used in place of the normal category spin flight tests.

34.4.2.4 Personal equipment. The pilot's personal equipment for spin testing should be the same as for other forms of hazardous testing. It should include a minimum of the following:

1) Personal Parachute: One of the newer steerable parachutes may be desirable, since one falls faster than the aircraft and may be directly below the descending aircraft when the parachute opens. Such parachutes also allow landing into the wind. The parachute should have been recently repacked.
2) Crash Helmet: A military-style pilot's helmet is desirable for such testing and may prevent the pilot from becoming incapacitated during egress from the aircraft. Such a helmet will also allow the pilot to communicate during the testing.
3) Jump Boots: Since a pilot may have to egress during this testing, being well prepared with well-fitting jump boots will reduce the possibility of injury during a bail-out.
4) Flying Suit: A military flying suit is also beneficial during such hazardous testing since it is likely to also reduce the likelihood of injury during egress and bail-out.

Aircraft-related personal equipment would include the restraint system which should be of the type that a single action by the pilot will remove all of the harness. The type with seat belt and single diagonal shoulder strap is *not recommended,* as it has been the author's experience that this type will cause entanglement on egress. Also, the two-shoulder strap-type will keep the pilot more secure in the seat during side loading from spins that go uncontrollable.

34.4.2.5 Egress drills. After the egress system has been fabricated, the test pilot should familiarize himself with the system and conduct several egress drills to confirm functioning and correct procedures. Considerable thought should be put into the correct procedures for egress for the particular system and the various types of spins to be tested. A well-thought out egress procedure may save the life of the test pilot.

34.4.3 Chase Aircraft

The use of a safety chase aircraft is a must for spin testing. It is also worthwhile if this aircraft serves as a photo chase aircraft since video or movie photos of the spinning aircraft taken with a telephoto lens may provide much useful information and documentation of the spin testing. The chase aircraft should have a stalling speed equal to, but preferably slower, than the test aircraft so it can track the test aircraft during the spin by using descending spiral turns. The chase aircraft pilot should assist the test aircraft pilot by counting spin turns and making safety call-outs to the test pilot.

34.4.4 Instrumentation

The spin is a complex maneuver involving motion about all aircraft axes at large angles of attack and sideslip. Therefore, a rigorous analysis requires a number of parameters for which instrumentation must be provided. It is also desirable to have the information in real time for the purposes of safety which would require a telemetry system. However, many trade-offs are involved including costs, safety, and schedule time. The minimum instrumentation suite should include chase video of the spin maneuver and a record of pilot comments regarding control positions for the spin entry, spin, and recovery. This minimal instrumentation is likely to be insufficient if any problems are encountered. Therefore, a more moderate instrumentation package that will record control positions, pitch, roll and yaw rates, plus angle of attack and angle of sideslip in addition to the minimal package, is more likely to assist in analysis of spin problems. If problems do occur, it may be worthwhile to tuft the aircraft and record spin tuft photos through some of the pencil sized video cameras now available. A cockpit video may also be useful in addition to the chase videos. The author does not recommend the minimalist approach to instrumentation for spin testing programs and a prudent instrumentation package will more than pay for itself in this high-risk testing.

34.4.5 Control Deflections and Critical Tolerances

Prior to beginning testing the control deflections should be set to the critical tolerances as specified in FAA AC 23-8A. Generally, these deflections would be set to the maximum tolerance for spin entry and minimum tolerance for the recovery. So, controls like the elevator would be set to the maximum trailing edge up deflection tolerance and the minimum trailing edge down deflection tolerance.

34.4.6 Weight and c.g. Position

Testing should begin at the most forward c.g. position and move to the aft c.g. in small increments. The complete spin matrix should be covered at the forward c.g. and critical spins discovered there should be tested at each increment during the aft movement until the desired aft c.g. limit is reached. At that point, the complete matrix should be repeated.

34.5 Flight Test Method

34.5.1 Preflight Briefing

The preflight briefing should include all members of the test team and a free exchange of ideas should be allowed during the briefing. The briefing should include:

1) Spins to be performed during the flight
2) Buildup technique to be used
3) Actions of the chase crew
4) Geographic location of the test and positioning of the ground crew
5) Communications discipline
6) Procedures in case of a mishap

34.5.2 Normal Category Spin Matrix

Figure 34.1 shows the normal category matrix of spins that must be accomplished for certification in the FAA normal category as recommended by AC

23-8A (Ref. 3). In this case, the regulation requires one turn spinning and no more than one turn during the recovery. As can be seen from this figure, the matrix for an all-up spin certification program requires a large number of spins. In recent years industry complaints have led to the issuance of AC 23-15 (Ref. 4) which provides some relief from this large matrix.

Fig. 34.2 provides the reduced spin matrix given in AC 23-15. As can be seen, the number of one-turn spins required by this matrix is considerably reduced from that recommended by AC 23-8A.

34.5.3 Acrobatic Category Spin Matrix

Fig. 34.1 also provides the recommended spins for an acrobatic category certification. This matrix is essentially the same as the normal category matrix except that the gear and flaps up spins are carried out to at least six turns spinning and one and one-half recovering. Since a buildup to six turns will be required for the purposes of safety, this matrix could well exceed 1000 spins. No relief, such as AC 23-15, is given for this matrix.

34.5.4 Test Conduct

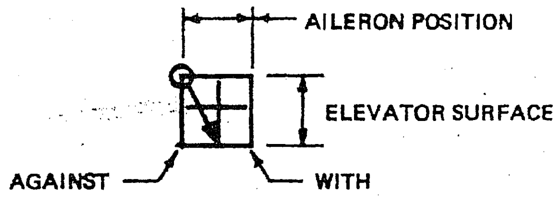
Since the spin matrices shown above are quite large, the testing should be divided into manageable portions. Due to the high-stress nature of this testing, the pilot should not be subjected to more than a dozen spins per flight and no more than two flights per day. Once the decision is made on the type and number of spins to be performed on a given flight, the test crew should construct test cards.

34.5.4.1 Stone spin shorthand. One efficient way to construct the test card is with the use of Stone Spin Shorthand.[24] This shorthand method was developed by the late Robert R. Stone, a friend of the author's, during the spin program on the Beechcraft T-34C U.S. Navy Trainer. The shorthand box was adopted from a box used by the NASA Langley Spin Tunnel to note model control positions and modified to make sense in the cockpit. Fig. 34.14 (Ref. 24) shows the shorthand box and several representative spins and recoveries. Figure 34.15 (Ref. 24) is a sample test card. With this system, the complete control positions for various spins can be described without notes. It can also be used to assist the pilot in both planning the spin flight test and conducting the actual tests.

34.5.4.2 Buildup technique. A buildup technique should be used to reach the maximum number of turns for a given type of spin. The first turn should be approached by starting first with a one-half turn spin, followed by a three-quarter-turn spin and then a one-turn spin. The number of turns to recover from each of these spins should be evaluated before progressing to the subsequent spin. If the program is for acrobatic spins then spins beyond one turn should be approached in one-quarter- or one-half-turn increments depending upon recovery turns from the preceding spin.

The testing should begin with the aircraft loaded to forward c.g. at maximum takeoff weight and after the matrix at that loading is complete the loading

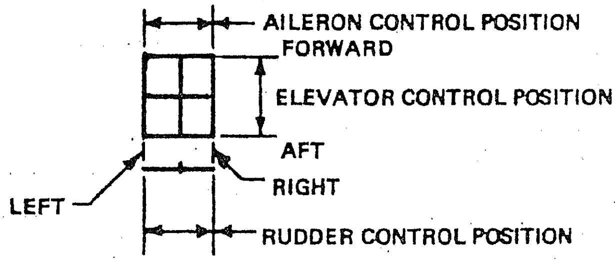
The basic box of the shorthand was adopted from a box used by the NASA Langely Spin Tunnel to note model control positions. The NASA notation indicates control surface positions from an engineers' point of view. For instance, a normal upright left spin with ailerons against would be described as follows:



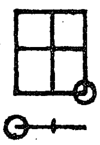
NOTE FOR: Left Rudder in the Spin.

Recovery control noted by the arrow: Ailerons neutral and down elevator. An additional NOTE required for rudder position during recovery.

After an evolutionary process involving pilot perspective and plagiarism, the following cockpit control position plan view emerged:



With this system, the complete control positions for the upright left spin with ailerons against can be described without notes:



And include a recovery investigation of elevator neutral, while holding aileron and rudder, to be applied after four turns in the original position:

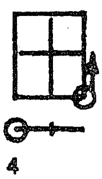
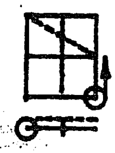


Fig. 34.14 Spin Shorthand developed by Robert R. Stone.[24]

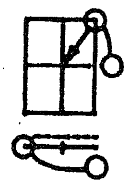
recovery control action can also be included to cover the possibility that the recovery nvestigation does not bring about recovery. In this case, stick full forward, ailerons-with nd rudder against to be applied after an additional four turns.



4 + 4

To illustrate the versatility of the system, consider the following: An innertia couplin entry from a 70° nose high attitude to an inverted left spin with ailerons against. Attemp recovery investigation after four turns by neutralizing ailerons and elevator. If the recover investigation does not bring about recovery in an additional four turns, reverse the rudder t effect recovery:

70° θ



4 + 4

With this basic system, the majority of out-of-control flight investigations can be describe in a form facilitating glance reference in the cockpit and can be used to condense data fo permanent record or communication.

Fig. 34.14 Spin Shorthand developed by Robert R. Stone[24] (continued).

should be changed in small increments of c.g. until aft c.g. is reached. During this aft movement of the c.g. it is not necessary to do the complete spin matrix, but only those spins that took the greatest number of turns to recover at the forward c.g. At aft c.g. the complete matrix should be completed.

Although this technique is quite flight time intensive, it reduces the possibility of encountering an unrecoverable spin and the use of the spin recovery parachute.

34.5.4.3 Control trim positions. The FARs state that the aircraft should be trimmed to $1.5V_{S1}$ as in stall characteristics. AC 23-3A states that if the airplane has a trimable stabilizer the extremes of trim deflection should be

SPIN DATA SHEET

FLIGHT NO.________ S/N________ DATE________ LAND:________

T.O. GROSS WT.:________LB CG:________% MAC T.O.________

CONFIG.:________ IYMP:________ TOTAL:________

SPIN NO.								
T.O.D.								
FUEL QTY.								
Hp START/REC.	/	/	/	/	/	/	/	/
KIAS START/REC.	/	/	/	/	/	/	/	/
THRUST								
ENTRY								
TRIMS: E = R = A = SPIN CONTROLS								
TURNS PLAN/REC.	/	/	/	/	/	/	/	/
INSTR. R/N								

SPIN NO.	INSTRUMENTATION: PRE-FLT R/N:	POST-FLT R/N:

Fig. 34.15 Spin data card.[24]

investigated. Again, a buildup technique might be advised as, at full deflection of a trimable stabilizer, the control forces during recovery might be quite high and the possibility for over stressing the aircraft might exist.

34.5.4.4 Recovery technique. The recovery technique used for recoveries should be rudder opposite the direction of spin, followed immediately by full down elevator. This is known as the NACA recovery. However, other recovery techniques should be evaluated. Advisory Circulars 23-8A and 23-15 both require the investigation of other recoveries such as neutralizing controls, elevator first recoveries, rudder only recoveries, and so forth, to see if such recoveries will produce an unrecoverable spin. In the case of spins that enter the flat mode, it has been found that sometimes aileron into the spin will effect a recovery. It should be noted that the FAA will not accept such recovery techniques for certificated airplanes.

34.5.4.5 Power-on spins. Advisory Circulars 23-8A and 23-15 both recommend the investigation of power on the spin characteristics. Both ACs ask for takeoff power to be used on some entries and to be held for one complete turn on some spins (see Figs. 34.1 and 34.2). Both permit the power to be reduced to idle for recovery. It has been the author's experience that leaving the power at takeoff power until the rotation stops will hasten recovery on many airplanes. However, it is then wise to reduce it to idle to prevent a large speed buildup in the ensuing dive.

34.5.4.6 Uncontrollable spin. Ailerons placed opposite the spin direction, such as might occur in an inadvertent spin entry, have been known to produce unrecoverable, or flat, spins. Also, in multiturn spins removing this opposite aileron at some point in the spin has produced unrecoverable spins. On some aircraft, power application has also had a similar effect.

If an unrecoverable, or flat, spin is encountered the chase aircraft crew should be briefed to count turns for the test pilot and to call out various recovery controls including landing gear and flap deployment. The emergency recovery controls should be briefed in the preflight briefing along with the number of spin turns these controls will be held before trying another combination. When none of the prebriefed control combinations work, then the spin recovery device should be employed. If this device fails, the test pilot should immediately egress the aircraft at the urging of the chase aircraft crew.

34.5.4.7 Data acquisition. The type of data acquisition system used will depend upon the financial resources of the flight test organization involved. However, the better the system and the more related parameters that it can collect, the more likely spin problems are to be solved rapidly. Also, should an accident occur, data may be available to clear the test pilot from wrongdoing or vice versa.

Chase video. As a minimum, the spin test should be recorded on video from the chase aircraft. A video camera with a 12×1 zoom capability is necessary in order to usefully track the aircraft during the departure and ensuing spin. It is worthwhile to let the chase crew practice tracking an aircraft during practice spins in an aircraft with known spin characteristics.

Cockpit video. It is also extremely worthwhile to have video from the cockpit of the test aircraft. This can assist in counting turns spinning and recovering along with recording other required data. It can be quite helpful in offloading many of the test pilot's data collection duties. If this video can be telemetered to the ground, the test crew can play a much larger role in the overall safety of the test.

Long lens ground tracking camera. If the aircraft can be tracked by long lens tracking camera from the ground this will also provide much useful data.

Onboard data collection package. If funds are available, a data collection system capable of collecting angle of attack and angle of sideslip at both wingtips, roll, pitch, and yaw rates and X, Y, and Z accelerations at the aircraft c.g., along with airspeed, altitude, control positions, and control forces will make spin testing and data analysis much more productive. Such available data will allow analysis of spin problems and improve recovery procedures.

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., March 1959.

[2]Federal Aviation Regulations Part 23, "Airworthiness Standards: Normal, Utility, Acrobatic, and Commuter Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 1998.

[3]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for the Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 9 Feb. 1989, and Change No. 1, 30 Sept. 1993.

[4]Federal Aviation Administration Advisory Circular No. 23-15, "Small Airplane Certification Compliance Program," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., 2 Jan. 1997.

[5]Bamber, M. J., and Zimmerman, C. H., "Spinning Characteristics of Wings," NACA Report 519, National Advisory Committee for Aeronautics, Washington, D.C., 1935.

[6]Neihouse, A. I., Lichtenstein, J. H., and Papoon, P. W., "Tail Design Requirements for Satisfactory Spin Recovery," NACA TN No. 1045 National Advisory Committee for Aeronautics, Washington, D.C., April 1946.

[7]Neihouse, A. I., "Tail Design Requirements for Satisfactory Spin Recovery for Personal-owner-type Light Airplanes," NACA TN No. 1329, National Advisory Committee for Aeronautics, Washington, D.C., June 1947.

[8]Bowman, J. S., Jr., "Summary of Spin Technology as Related to Light General Aviation Airplanes," NASA TN D-6575, National Aeronautics and Space Administration, Washington, D.C., Dec. 1971.

[9]Bowman, J. S., Jr., and Burk, S. M., Jr., "Stall Spin Studies Relating to Light General Aviation Aircraft," SAE paper no. 730320, presented at the SAE Business Aircraft Meeting, Wichita, KS, Society of Automotive Engineers, Inc., New York, April 1973.

[10]Bowman, J. S., Jr., and Burk, S. M., Jr., "Stall/Spin Research Status Report," SAE paper no. 740354, presented at the SAE Business Aircraft Meeting, Wichita, KS, Society of Automotive Engineers, Inc., New York, April 1974.

[11]Burk, S. M., Jr., Bowman, J. S., Jr., and White, W. L., "Spin Tunnel Investigation of the Spinning Characteristics of Typical Single-Engine General Aviation Airplane Designs. I—Low-Wing Model A: Effects of Tail Configurations." NASA TP-1009, 1977.

[12]Stough, H. P., III, Patton, J. M., Jr., and Silwa, S. M., "Flight Investigation of the Effect of Tail Configuration on Stall, Spin and Recovery Characteristics of a Low-Wing General Aviation Research Airplane," NASA TP-2644, 1987.

[13]Kerr, T. H., "A Criterion for the Prediction of the Recovery Characteristics of Spinning Aircraft," TN No. Aero 2251, Royal Aircraft Establishment, Farnborough, U.K., Aug. 1953.

[14]Beaurain, L., "General Study of Light Plane Spin, Aft Fuselage Geometry, Part I, Report No. 77/03, Institute for Fluid Mechanics at Lille, Lille, France, Feb. 1977, NASA Technical Translation, Washington, D.C., June 1977.

[15]Clarkson, M. H., Malcolm, G. N., and Chapman, G. T., "Experimental Poststall Rotary Aerodynamic Coefficients for Airplane-like Configurations," *Journal of Aircraft*, Vol. 13, No. 8, AIAA, New York, Aug. 1976.

[16]Clarkson, M. H., Malcolm, G. N., and Chapman, G. T., "A Subsonic Flow Investigation at High Angles of Attack on a Research Body at Several Reynolds Numbers," Paper No. 77-180 Presented at the AIAA 15th Aerospace Sciences Meeting, Los Angeles, AIAA, Jan. 1977.

[17]Polhamus, E. C., "Effect of Flow Incidence and Reynolds Number on Low-Speed Aerodynamic Characteristics of Several Noncircular Cylinders with Applications to Directional Stability and Spinning," NASA Technical Report R-29, National Aeronautics and Space Administration, Washington, D.C., 1959.

[18]Stough, H. P., III, DiCarlo, D. J., and Patton, J. M., Jr., "Flight Investigation of the Effects of an Outboard Wing Leading Edge Modification on Stall/Spin Characteristics of a Low-Wing, Single-Engine, T-Tail Light Airplane," NASA TP-2691, 1987.

[19]Siedman, O., and Kamm, R. W., "Antispin-Tail-Parachute Installations, NACA RB, Feb. 1943.

[20]Kamm, R. W., and Malvestuto, F. S., Jr., "Comparison of Tail and Wing-Tip Spin Recovery Parachutes as Determined by Tests in the Langley 20-foot, Free-Spinning Tunnel," NACA ARR L5G19a, 1946.

[21]Malvestuto, F. S., Jr., "Method of Estimating the Minimum Size of Tail or Wing Tip Parachute for Emergency Spin Recovery of an Airplane," NACA RM L8D27, 1948.

[22]Burk, S. M., "Summary of Design Considerations for Airplane Spin Recovery Parachute Systems," NASA TN D-6866, Aug. 1972.

[23]Bradshaw, C., "A Spin-Recovery Parachute System for Light General Aviation Airplanes," NASA TM-80237, April 1980.

[24]Stone, R. R., "Stone Spin Shorthand, Including Spin Data Card and Sample Criteria Spin Set," Beech Aircraft Corporation Engineering Flight Test Report No. E22560F, Wichita, KS, Aug. 1974.

35
Dive Testing for Flutter, Vibration, and Buffeting

35.1 Introduction

Determining that the aircraft is free from flutter, vibration, and buffeting for airspeeds from stall speed up to and including the design dive speed V_D is the most hazardous testing in aircraft certification and should be approached with utmost caution. It usually comes at or near the end of the aircraft development program for at least two reasons. First, is that the analysis of structural dynamic phenomena is difficult and time consuming even with modern computers and only begins in earnest after the aircraft is completed and a ground vibration survey has been conducted. Second, it obeys the test pilot's longevity rule, which says that if you put the really hazardous tests off until last you live longer!

35.2 Federal Aviation Administration Regulations

CAR 3 (Ref. 1) has requirements for flutter and vibration while FAR Part 23 (Ref. 2) has requirements for vibration and buffeting and has separated flutter into a stand-alone regulation. There have been numerous changes between CAR 3 and FAR Part 23, and FAR Part 23 has seen several changes on these items since its issuance.

35.2.1 Civil Aviation Regulation 3 (Ref. 1)

Civil Aviation Regulation 3.311 (Ref. 1) flutter and vibration prevention measures, discusses the requirements for freedom from flutter and vibration for airplanes certified under CAR 3. It states that wings, tail, and control surfaces shall be free of flutter, airfoil divergence, and control reversal for all operating conditions within the V–n envelope. In addition, the wing shall have adequate torsional rigidity as demonstrated by test or other suitable methods, the control surfaces shall be mass balanced to preclude flutter and the natural frequencies of main structural components shall be determined by vibration tests or other suitable methods.

The regulation accepts flight flutter tests as one of the acceptable methods for showing compliance with the regulation, however, the written policy associated with the regulation cautions against its use as the sole means of compliance.

Although related to flutter but not flutter testing, CAR 3.330 mass balance weights incorporated on control surfaces discusses the load factors that the supporting structure must be able to withstand.

CAR 3.347 spring devices requires that any spring device used in a control system must have established reliability or be shown to not have any effect upon flutter should it fail.

35.2.2 Federal Aviation Regulation Part 23 (Ref. 2)

In Part 23, flutter has been separated from vibration and buffeting and Amendment 23-7 added a new section on high-speed characteristics.

FAR 23.251 covers vibration and buffeting. This regulation states that there should be no vibration or buffeting severe enough to cause structural damage to the airplane, interfere with control of the airplane, or cause excessive fatigue to the flight crew throughout the speed range up to the design dive speed of the airplane.

FAR 23.253 discusses high-speed characteristics. It states that if a maximum operating speed is established under 23.1505(c) that certain speed increase and recovery characteristics must be met. These include investigating conditions and characteristics likely to cause inadvertent speed increases. After the speed warning has sounded, and allowing for a pilot's reaction time, it must be shown that the aircraft can be recovered to the maximum operating speed without exceeding the dive speed, encountering buffeting that would impair the pilot's ability to control the airplane, or experiencing control reversal. In addition, any tendency to pitch, roll, or yaw should be mild and controllable.

Regarding flutter, FAR Part 23.629 has changed considerably since its initial issuance. Early versions of the regulation called for freedom from flutter up to the lesser of $1.4V_C$ or $1.2V_D$ and did not specify that in-flight flutter tests were required, but allowed them as one method of compliance. More recent versions of the regulation require that freedom from flutter be demonstrated in-flight up to airspeeds of V_D in addition to showing freedom from flutter up to $1.2V_D$ by one of two other methods. In addition to the requirements for airfoil divergence and control reversal discussed in CAR 3, Amendment 23-7 required that for multiengine turboprop airplanes the issues of whirl mode flutter be addressed. Amendment 23-31 extended that to include both single and multiengine turboprop airplanes.

The regulation also addresses the mounting of control surface balance weights in FAR 23.659 and control system springs in 23.687. As can be seen from this summary the flutter regulations have changed significantly, even over the life of FAR Part 23, and any certification test should refer to the specific amendment to the regulation under which the airplane is being certified.

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

AC 23-8A only addresses FAR 23.251 and 23.253 while referring the reader to AC 23.629-1A (Ref. 4) for the acceptable methods for flutter compliance.

In discussing vibration and buffeting AC 23-8A states that this test should not be confused with a test for flutter and that no attempt should be made to excite it. However, this does not ensure that one might not encounter it. The advisory circular discusses the speeds to which the tests should be conducted including accounting for instrument and airspeed position correction. It also discusses that if springs are in the control system that the test should be conducted with and without the springs connected unless the spring system meets the reliability criteria. If the spring meets the reliability criteria then the spring disconnected test does not have to be performed. It also states that for airplanes that have Mach limits the tests need to be repeated at the dive Mach number in addition to the design dive speed. The AC cautions that the airplane need only be held at the maximum speed long enough to determine the absence of excessive vibration or buffeting.

The advisory circular discusses that there is a distinction made in the regulation for airplanes that accelerate quickly when upset and that these high-speed characteristics should be evaluated by flight demonstration. It also recommends that the same maneuvers be accomplished in both the dynamic pressure and Mach critical ranges. It lists a number of factors that are to be evaluated in the flight test of high-speed characteristics from controllability, trim changes, speed control, and speed warning. These items are to be evaluated from various potential upsets. Included are c.g. shift due to passenger movement, inadvertent control movement, gust upsets about lateral, longitudinal axis, and in combination. All of these upsets are to include a 3 s delay before action is taken by the pilot.

35.2.4 Advisory Circular 23.629-1A (Ref. 4)

AC 23.629-1A provides guidance to comply with FAR 23.629 the regulation on flutter. It provides information on airplane categories (with regard to flutter), methods of substantiation, reevaluation of modified aircraft, a discussion of control surfaces and tabs (including balance, vibratory modes, and fail safe requirements), and discussions of divergence and control reversal. It contains four appendices that include topics of ground testing, flutter analysis, flutter flight testing, and references. The section on flutter flight testing is particularly worthwhile to the test pilot in planning a flutter flight test.

As can be seen from the foregoing discussion, the topic of dive testing involves a number of facets and is under regular change. Therefore, it behooves the test crew to consult with the latest version of these regulations and advisory circulars prior to beginning any test program involving high-speed testing.

35.3 Theory

The phenomena discussed in this chapter comes under what is called aeroelastic effects. Such effects include control reversal, divergence, and flutter. All are normally caused by a lack of structural stiffness in one or more of the airplanes components when these components are under aerodynamic load. The author is not an expert on these subjects; therefore, the following should be seen as a test pilot's understanding of the subject and a more thorough knowledge should be obtained from structural dynamic references.

35.3.1 Control Reversal

Control reversal occurs when the deflection of a control surface in flight causes the aircraft component to which it is attached to deflect to a point where its deflection overpowers the movement of the control surface and causes a reaction in the opposite direction to what the pilot intended. This is discussed for aileron controls in Chapter 30, and the structural mechanics discussed in this chapter apply for other control surfaces as well.

35.3.2 Divergence

Structural divergence is like aileron reversal in that it is an aeroelastic effect. However, structural divergence nearly always leads to an immediate failure of the structure. Since the lift of an airfoil acts at the 1/4 chord of an aerodynamic surface and the elastic axis of the structure normally occurs near mid-chord a twisting moment is introduced by the production of lift. However, the lift is increased as a function of velocity squared while the structural stiffness in torsion is near linear. Since the twisting of the surface increases its angle of attack, there will be an airspeed where the lift created will overcome the structural stiffness and the surface will diverge and fail. Therefore, this speed should be well above the airspeed range of the airplane.

35.3.3 Flutter

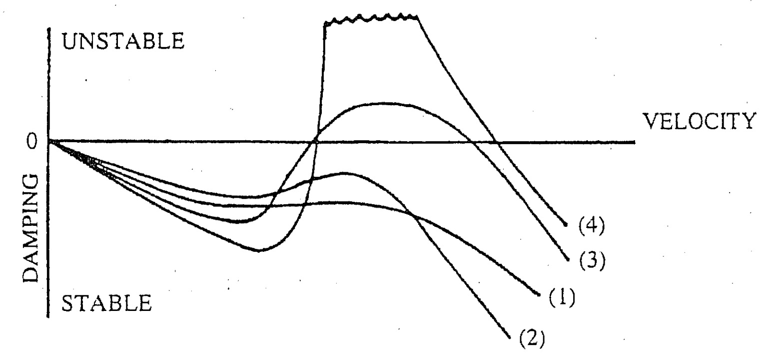
Aerodynamic flutter is a resonance of the airplane's structure when influenced by the atmosphere or control inputs. It is a function of the structural stiffness, inertia properties, and aerodynamic forces. For flutter to occur at subsonic speeds the part, or parts, in question need at least two degrees of freedom and there has to be a coupling between them. This will depend upon the natural frequencies of the part or parts. Depending upon the type of flutter, it can occur at any airspeed. It also can be destructive in a very short period. During the author's experience with flutter in the prototype Aero Commander 112, the complete horizontal tail, upper vertical tail, and rudder departed the aircraft in 0.3 s which was followed by a very wild ride! This occurred when five flutter modes all came together at the dive airspeed.

35.3.3.1 Natural frequencies. Every structure has a natural frequency. This is particularly true of metal structures. In general, it can be said that the stiffer the structure, the higher the frequency. Since most of the frequencies associated with the atmosphere and aviation occur below 50 Hz it is desirable to have the structural frequencies of the aircraft above that range. However, in most cases this is not possible to achieve. It then behooves the airplane designer to keep the natural frequencies of the various components separated by sufficient margin that they will not come together under load. In addition, most structures have several harmonics of the resonant frequency. The in-flight frequencies are a function of the air loads on the airplane which are functions of airspeed.

35.3.3.2 Flutter modes. The flutter modes refer to the degrees of freedom of the structure. Some examples are wing bending, wing torsion, aileron rotation, elevator rotation, horizontal tail bending, horizontal tail torsion, fuselage bending, and so on. The coupling of the resonant frequencies of these modes will create flutter and it can generally be said that the more modes that couple the greater, and quicker, the destruction with the torsional modes being among the more destructive.

35.3.3.3 Nodal lines. A nodal line in the discussion of flutter is a line on the structure that does not deflect during the flutter oscillation. If one considers the elevator rotation mode by itself, one can see that this line would coincide with the hinge line. However, if we combine the elevator rotation mode with the stabilizer bending mode, we can see that this nodal line will have a location other than along the hinge line. It is important in the ground vibration survey to identify the modes, their frequencies, and the location of the nodal lines. The reason for this is that if instrumentation, such as accelerometers or strain gauges, are placed upon a nodal line there will not be any indication that you are approaching a flutter speed. In the author's accident with flutter, the nodal lines had been misidentified and the accelerometers placed on the actual ones. It is worthwhile for the test crew to ask the flutter engineers to show the sketch of the nodal lines on the airplane and ensure that the instrumentation accelerometers are placed as far away from these lines as possible to insure they see the maximum accelerations from the excitation.

35.3.3.4 Coupling. For flutter to occur there must be coupling. In the simple modes (control surface rotation) the coupling frequency must come from the atmosphere. However, if control surfaces are not properly balanced this may occur. For the more complex modes, the coupling will come from other modes such as aileron rotation and wing bending, or other combinations. After the ground vibration survey and the flutter analysis, the flutter engineers should be able to tell which modes are most likely to couple and the airspeed at which that may occur. Fig. 35.1 (Ref. 4) shows a plot of damping vs airspeed for a representative coupling. From this plot one can see that there is an airspeed where the damping becomes unstable and the modes couple causing flutter. That airspeed should be above the dive speed of the airplane.



Curves 1 and 2 show slight trends towards instability, but do not approach actual instability.

Fig. 35.1 Structural damping vs airspeed.[4]

35.3.3.5 Aerodynamic damping and excitation. In-flight the atmosphere provides damping for the motion of the airplane and its parts. However, at the same time the atmosphere imparts energy to the airframe. As speed increases the aerodynamic forces may excite the natural frequencies of the structure and cause the structural damping to go unstable resulting in flutter. As the energy imparted is a function of airspeed squared, the faster the flight, the higher this forcing energy.

35.3.3.6 Mass distribution. The location of large masses such as fuel or engines will affect the structural frequencies. For instance, fuel in tip tanks, or located in the outboard portion of the wing, will lower the structural frequency of wing bending. Unbalanced control surfaces generally have their center of mass behind the hinge line which will cause rotation when subjected to a gust. This rotation provides excitation to the airframe and can induce flutter. As a result, having control surfaces balanced to near 100% is an effective measure to reduce the possibility of flutter. However, having this balance concentrated at the end of the surface may not be the best approach since it is likely to lower the bending and torsional frequencies. Having the balance weight distributed along the surface or located in the center of the surface is the best approach from the flutter standpoint. It may, however, result in a heavier airplane.

35.4 Safety Considerations

Since any one of the three structural dynamics problems discussed in the last section is more likely to occur at high airspeed, considerable care should be taken in the planning of high-speed and flutter testing. Among the first considerations to be addressed in this planning is personal emergency equipment and aircraft egress.

35.4.1 Personal Emergency Equipment

The aircraft crew for this testing should be a minimum crew. The crew should be outfitted with flying suit, helmet, jump boots, and a personal parachute capable of opening at high speed. The restraint system in the aircraft should include military style, lap belt with shoulder harness. Current general aviation restraint systems are insufficient to keep the pilot in the seat during the wild thrashing that occurs after a structural failure at high speed. Even the military system described may stretch to the point of the pilot being able to place his feet in the seat, in the squatting position, after the thrashing from a high speed failure. Such was the author's experience! One can see then that the helmet is a necessity to keep the pilot, or crew member, from being rendered unconscious during this violent after effect.

35.4.2 Aircraft Egress

Since an encounter with any of the high-speed phenomena is likely to result in structural failure, aircraft egress after such failure is going to be difficult. The aircraft should be fitted with quick release doors that depart the aircraft with a single action by the pilot. Methods to remove the doors even when under air load should be considered. The flight testing appendix to Advisory Circular 23.629-1A (Ref. 4) discusses that airplanes with large cabins should have a knotted rope installed the length of the cabin since the doors on most of these aircraft are at the rear. It is obvious to this author that the person who wrote this appendix had never survived an in-flight structural failure at high speed. The initial thrashing after a structural failure at high speed is so violent and at such a high load factor that the pilot is not likely to be able to move and, until this thrashing stabilizes, would be thrown about the cabin and incapacitated. Such an aircraft should have an egress route that is adjacent to the pilot or the pilot is not likely to survive. For some of their hazardous programs NASA developed pyrotechnic devices that cut a hole in the aircraft adjacent to the pilot. Such devices should be considered if the egress path does not have easy access.

35.4.3 Minimum Safe Test Altitude

Advisory Circular 23.629-1A (Ref. 4) recommends that flutter test be conducted at at least two altitudes. One at 75% of the service ceiling, or the altitude at the V_D/M_D knee, and one at high dynamic pressure. The AC calls for an altitude of 9000 ft for emergency egress protection for the high dynamic pressure test point. This is a minimum altitude for the possibility of a safe egress and should be the altitude at which the maximum airspeed is reached, not where the dive is started. However, prior to doing this maximum dynamic pressure point, the high altitude point should be obtained. This approach provides some confidence that the low altitude point can be successfully completed since the atmospheric damping is higher at the lower altitude. From the author's experience with explosive flutter, these altitudes would apply to any airplane that had a dive airspeed in excess of 175 kn calibrated airspeed. For light airplanes with lower dive airspeeds and low service ceilings this 9000 ft altitude might be reduced. However, if possible, it should be maintained as a minimum.

35.4.4 Emergency Procedures

The test planning should include a set of emergency procedures that include a checklist with step-by-step procedures of what actions are to be taken in the event of an accident. This checklist should include phone numbers of hospitals, the FAA, NTSB, and local law enforcement in addition to those within the company or organization who need to be notified. As a part of those procedures, provisions should be made for a chase aircraft and a ground crew who would be positioned within the test area.

35.4.5 Chase Aircraft

A chase aircraft with equivalent or higher speed is a necessity for this type of testing. If the chase aircraft has a greater speed capability, it will be able to pace the test aircraft during the test, observe any abnormalities, and report those to the crew and test conductor. After each dive the chase should join on the test aircraft and look it over for any signs of loose panels or other signs of problems.

35.4.6 Ground Crew

The test ground crew should be positioned in the test area and should consist of persons with a knowledge of first aid and of the test aircraft. Their purpose is to be the first on the scene in case of an accident, to care for any injured including rapid transportation to hospitals or EMS stations, and to secure the crash site to prevent pilfering by spectators. They should be equipped with crash equipment including fire suppression and all-terrain vehicles.

35.4.7 Ballistic Parachutes

Ballistic parachutes for the in-flight recovery of light aircraft have come into larger use in recent years, although they have been around since the late 1960s. Even though they might not be intended for the production aircraft, they may be useful in recovering the test aircraft after a structural failure and saving the lives of the crew. Egress after a structural failure is iffy even under the best of circumstances. In addition, the use of such a parachute may save the aircraft from total destruction and allow a less expensive rebuild. However, such systems are not yet available for larger, higher-speed aircraft.

35.5 Instrumentation

For aircraft who have dive speeds in excess of 200 kn, this is the test where it is worthwhile to spend the money for good instrumentation. The two types of sensors used for these tests are sensitive accelerometers and strain gauges. For data collection, a telemetry system or a magnetic tape recorder capable of a high sample rate are desired, although recording oscillographs can provide adequate data if they have sufficient channels. Telemetry will reduce the test schedule time and number of flights since it will allow near real time data analysis.

35.5.1 Sensor Location

As discussed in the theory section, accelerometers or strain gauges should be located as far as possible from the nodal lines. This will ensure adequate signal strength and help prevent the overlooking of flutter onset. The flutter engineer should be able to provide a sketch of the nodal lines and their location on the aircraft. The engineer should also be involved in the location of the sensors.

35.6 Flight Test Method

Prior to commencing the test the airspeed indicator should have a fresh calibration and the pitot-static system should be leak checked. This assumes that the position correction calibration is recent and linear at high airspeed. If this is not the case then a calibrated chase aircraft with a calibrated speed range greater than that of the test aircraft should be used to determine the speed of the test.

Control surfaces should be balanced to the most under balanced (tail heavy) condition and the trim tabs set to the maximum allowable free play. The control system damping should be minimized to simulate wear. These values should be obtained from the flutter analysis and become part of the type design if the tests are successful.[4]

Since the airplane mass locations may affect the flutter modes, consideration should be given to the fuel loading. If the fuel quantity affects the flutter modes then at least a maximum and minimum fuel load should be tested.[4] The c.g. should be at the aft c.g. location to reduce the down load on the horizontal tail to help prevent overstressing should tail flutter be encountered. The most critical condition of weight should be tested.

Prior to beginning the dives, make sure that the pilot restraint system is secure and so tight that it is uncomfortable.

35.6.1 Buildup Technique

Like any critical or hazardous flight test a buildup technique should be used in reaching the dive airspeed.

35.6.1.1 Altitude. For the test altitude, the technique is to build down as was discussed briefly earlier. In other words, the high-altitude airspeeds should be obtained before the low-altitude airspeeds are obtained.

35.6.1.2 Airspeed. The buildup in airspeed should begin at the initial flutter clearance airspeed (usually 130 kn, or 150 mph for light aircraft) and progress in 10-kn increments until the flutter analysis shows signs of reduced damping. At that point speed increments should be reduced to 5-kn increments. The number of airspeed increments per flight is a function of the instrumentation system (telemetry or on-board recording) and the confidence in the flutter analysis. If telemetry is available and the actual results are as expected from the flutter analysis, then a large number of airspeeds can be obtained per flight. However, if only on-board recording is available, then two to three airspeed increments may be all that should be obtained per flight with that number decreasing as speed increases. During the airspeed increments near the dive speed it is best to be cautious.

35.6.2 In Case of a Problem

Should flutter be encountered or an unusual vibration be felt in the airframe, reduce power to idle and recover from the dive as rapidly as possible without overloading the airplane or introducing large control movements. If no damage to the airframe is observed, *do not* repeat the test point until the cause has been determined, a fix installed, and the airframe inspected thoroughly. If the flutter is destructive before the above action can be taken, hang on, pray, and wait for the thrashing to subside before trying to exit the aircraft. This will require considerable altitude, so be ready to deploy your personal parachute as soon as you are free of the aircraft and expect the ground to appear very rapidly.

35.7 Data Analysis

Since the test aircrews lives are at stake, they should be involved in the data analysis. The degree to which they participate is a function of their knowledge of the subject and the experience and capability of the individual, or individuals, doing the flutter analysis. This becomes more difficult if telemetry is used for data collection. Therefore, if telemetry is used, the aircrew should stop the buildup at a point of their choosing and review the data with the flutter engineer(s). Particular attention should be paid to how the actual data compares with the analysis.

At least two potential data presentations are worth examining as the test progresses. One is how rapid the oscillations from the excitation damp. If this changes significantly as the test progresses and does not agree with the analysis then the test should be suspended until an explanation can be found. The second presentation is a plot of acceleration of the various structural modes vs the frequency of the oscillations. This second presentation is more valuable than the first in evaluating changes in the modes as the airspeed increases. If the modes show a significant change in frequency or acceleration between airspeeds, and this is not shown by the analysis, then again testing should be suspended and explanations found and if necessary fixes applied. Flutter is a very mathematically complex subject that requires considerable experience to master. Therefore, aircrew members should retain a healthy scepticism toward the flutter engineers and their analysis.

References

[1]Civil Aeronautics Manual 3, "Airplane Airworthiness; Normal, Utility, and Acrobatic Categories," U.S. Department of Transportation, Federal Aviation Agency, U.S. Government Printing Office, Washington, D.C., 1959.

[2]Federal Aviation Regulation Part 23, "Airworthiness Standards: Normal, Utility, and Acrobatic Category Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., June 1974.

[3]Federal Aviation Administration Advisory Circular No. 23-8A, "Flight Test Guide for Certification of Part 23 Airplanes," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Feb. 1989.

[4]Federal Aviation Administration Advisory Circular No. 23.629-1A, "Means of Compliance with Section 23.629, Flutter," U.S. Department of Transportation, Federal Aviation Administration, U.S. Government Printing Office, Washington, D.C., Oct. 1985.

Index

TEXTS PUBLISHED IN THE AIAA EDUCATION SERIES

Flight Testing of Fixed-Wing Aircraft
Ralph D. Kimberlin 2003
ISBN 1-56347-564-2

The Fundamentals of Aircraft Combat Survivability Analysis and Design, Second Edition
Robert E. Ball 2003
ISBN 1-56347-582-0

Analytical Mechanics of Space Systems
Hanspeter Schaub and John L. Junkins 2003
ISBN 1-56347-563-4

Introduction to Aircraft Flight Mechanics
Thomas R. Yechout with Steven L. Morris, David E. Bossert, and Wayne F. Hallgren 2003
ISBN 1-56347-577-4

Aircraft Design Projects for Engineering Students
Lloyd Jenkinson and James Marchman 2003
ISBN 1-56347-619-3

Elements of Spacecraft Design
Charles D. Brown 2002
ISBN 1-56347-524-3

Civil Avionics Systems
Ian Moir and Allan Seabridge 2002
ISBN 1-56347-589-8

Helicopter Test and Evaluation
Alastair K. Cooke and Eric W. H. Fitzpatrick 2002
ISBN 1-56347-578-2

Aircraft Engine Design, Second Edition
Jack D. Mattingly, William H. Heiser, and David T. Pratt 2002
ISBN 1-56347-538-3

Dynamics, Control, and Flying Qualities of V/STOL Aircraft
James A. Franklin 2002
ISBN 1-56347-575-8

Orbital Mechanics, Third Edition
Vladimir A. Chobotov, Editor 2002
ISBN 1-56347-537-5

Basic Helicopter Aerodynamics, Second Edition
John Seddon and Simon Newman 2001
ISBN 1-56347-510-3

Aircraft Systems: Mechanical, Electrical, and Avionics Subsystems Integration
Ian Moir and Allan Seabridge 2001
ISBN 1-56347-506-5

Design Methodologies for Space Transportation Systems
Walter E. Hammond 2001
ISBN 1-56347-472-7

Tactical Missile Design
Eugene L. Fleeman 2001
ISBN 1-56347-494-8

Flight Vehicle Performance and Aerodynamic Control
Frederick O. Smetana 2001
ISBN 1-56347-463-8

Modeling and Simulation of Aerospace Vehicle Dynamics
Peter H. Zipfel 2000
ISBN 1-56347-456-6

Applied Mathematics in Integrated Navigation Systems
Robert M. Rogers 2000
ISBN 1-56347-445-X

Mathematical Methods in Defense Analyses, Third Edition
J. S. Przemieniecki 2000
ISBN 1-56347-396-6

Finite Element Multidisciplinary Analysis
Kajal K. Gupta and John L. Meek 2000
ISBN 1-56347-393-3

Aircraft Performance: Theory and Practice
M. E. Eshelby 1999
ISBN 1-56347-398-4

Space Transportation: A Systems Approach to Analysis and Design
Walter E. Hammond 1999
ISBN 1-56347-032-2

Civil Jet Aircraft Design
Lloyd R. Jenkinson, Paul Simpkin, and Darren Rhodes 1999
ISBN 1-56347-350-X

Structural Dynamics in Aeronautical Engineering
Maher N. Bismarck–Nasr 1999
ISBN 1-56347-323-2

Intake Aerodynamics, Second Edition
E. L. Goldsmith and J. Seddon 1999
ISBN 1-56347-361-5

Integrated Navigation and Guidance Systems
Daniel J. Biezad 1999
ISBN 1-56347-291-0

Aircraft Handling Qualities
John Hodgkinson 1999
ISBN 1-56347-331-3

Performance, Stability, Dynamics, and Control of Airplanes
Bandu N. Pamadi 1998
ISBN 1-56347-222-8

Spacecraft Mission Design, Second Edition
Charles D. Brown 1998
ISBN 1-56347-262-7

Computational Flight Dynamics
Malcolm J. Abzug 1998
ISBN 1-56347-259-7

Space Vehicle Dynamics and Control
Bong Wie 1998
ISBN 1-56347-261-9

Introduction to Aircraft Flight Dynamics
Louis V. Schmidt 1998
ISBN 1-56347-226-0

Aerothermodynamics of Gas Turbine and Rocket Propulsion, Third Edition
Gordon C. Oates 1997
ISBN 1-56347-241-4

Advanced Dynamics
Shuh-Jing Ying 1997
ISBN 1-56347-224-4

Introduction to Aeronautics: A Design Perspective
Steven A. Brandt, Randall J. Stiles, John J. Bertin, and Ray Whitford 1997
ISBN 1-56347-250-3

Introductory Aerodynamics and Hydrodynamics of Wings and Bodies: A Software-Based Approach
Frederick O. Smetana 1997
ISBN 1-56347-242-2

An Introduction to Aircraft Performance
Mario Asselin 1997
ISBN 1-56347-221-X

Orbital Mechanics, Second Edition
Vladimir A. Chobotov, Editor 1996
ISBN 1-56347-179-5

Thermal Structures for Aerospace Applications
Earl A. Thornton 1996
ISBN 1-56347-190-6

Structural Loads Analysis for Commercial Transport Aircraft: Theory and Practice
Ted L. Lomax 1996
ISBN 1-56347-114-0

Spacecraft Propulsion
Charles D. Brown 1996
ISBN 1-56347-128-0

Helicopter Flight Dynamics: The Theory and Application of Flying Qualities and Simulation Modeling
Gareth D. Padfield 1996
ISBN 1-56347-205-8

Flying Qualities and Flight Testing of the Airplane
Darrol Stinton 1996
ISBN 1-56347-117-5

Flight Performance of Aircraft
S. K. Ojha 1995
ISBN 1-56347-113-2

Operations Research Analysis in Test and Evaluation
Donald L. Giadrosich 1995
ISBN 1-56347-112-4

Radar and Laser Cross Section Engineering
David C. Jenn 1995
ISBN 1-56347-105-1

Introduction to the Control of Dynamic Systems
Frederick O. Smetana 1994
ISBN 1-56347-083-7

Tailless Aircraft in Theory and Practice
Karl Nickel and Michael Wohlfahrt 1994
ISBN 1-56347-094-2

Mathematical Methods in Defense Analyses, Second Edition
J. S. Przemieniecki 1994
ISBN 1-56347-092-6

Hypersonic Aerothermodynamics
John J. Bertin 1994
ISBN 1-56347-036-5

Hypersonic Airbreathing Propulsion
William H. Heiser and David T. Pratt 1994
ISBN 1-56347-035-7

Practical Intake Aerodynamic Design
E. L. Goldsmith and J. Seddon 1993
ISBN 1-56347-064-0

Acquisition of Defense Systems
J. S. Przemieniecki, Editor 1993
ISBN 1-56347-069-1

Dynamics of Atmospheric Re-Entry
Frank J. Regan and Satya M. Anandakrishnan 1993
ISBN 1-56347-048-9

Introduction to Dynamics and Control of Flexible Structures
John L. Junkins and Youdan Kim 1993
ISBN 1-56347-054-3

Spacecraft Mission Design
Charles D. Brown 1992
ISBN 1-56347-041-1

Rotary Wing Structural Dynamics and Aeroelasticity
Richard L. Bielawa 1992
ISBN 1-56347-031-4

Aircraft Design: A Conceptual Approach, Second Edition
Daniel P. Raymer 1992
ISBN 0-930403-51-7

Nonlinear Analysis of Shell Structures
Anthony N. Palazotto and Scott T. Dennis 1992
ISBN 1-56347-033-0

Orbital Mechanics
Vladimir A. Chobotov, Editor 1991
ISBN 1-56347-007-1

Critical Technologies for National Defense
Air Force Institute of Technology 1991
ISBN 1-56347-009-8

Space Vehicle Design
Michael D. Griffin and James R. French 1991
ISBN 0-930403-90-8

Defense Analyses Software
J. S. Przemieniecki 1990
ISBN 0-930403-91-6

Inlets for Supersonic Missiles
John J. Mahoney 1990
ISBN 0-930403-79-7

Introduction to Mathematical Methods in Defense Analyses
J. S. Przemieniecki 1990
ISBN 0-930403-71-1

Basic Helicopter Aerodynamics
J. Seddon 1990
ISBN 0-930403-67-3

Aircraft Propulsion Systems Technology and Design
Gordon C. Oates, Editor 1989
ISBN 0-930403-24-X

Boundary Layers
A. D. Young 1989
ISBN 0-930403-57-6

Aircraft Design: A Conceptual Approach
Daniel P. Raymer 1989
ISBN 0-930403-51-7

Gust Loads on Aircraft: Concepts and Applications
Frederic M. Hoblit 1988
ISBN 0-930403-45-2

Aircraft Landing Gear Design: Principles and Practices
Norman S. Currey 1988
ISBN 0-930403-41-X

Mechanical Reliability: Theory, Models and Applications
B. S. Dhillon 1988
ISBN 0-930403-38-X

Re-Entry Aerodynamics
Wilbur L. Hankey 1988
ISBN 0-930403-33-9

Aerothermodynamics of Gas Turbine and Rocket Propulsion, Revised and Enlarged
Gordon C. Oates 1988
ISBN 0-930403-34-7

Advanced Classical Thermodynamics
George Emanuel 1987
ISBN 0-930403-28-2

Radar Electronic Warfare
August Golden Jr. 1987
ISBN 0-930403-22-3

An Introduction to the Mathematics and Methods of Astrodynamics
Richard H. Battin 1987
ISBN 0-930403-25-8

Aircraft Engine Design
Jack D. Mattingly, William H. Heiser, and Daniel H. Daley 1987
ISBN 0-930403-23-1

Gasdynamics: Theory and Applications
George Emanuel 1986
ISBN 0-930403-12-6

Composite Materials for Aircraft Structures
Brian C. Hoskin and Alan A. Baker, Editors 1986
ISBN 0-930403-11-8

Intake Aerodynamics
J. Seddon and E. L. Goldsmith 1985
ISBN 0-930403-03-7

The Fundamentals of Aircraft Combat Survivability Analysis and Design
Robert E. Ball 1985
ISBN 0-930403-02-9

Aerothermodynamics of Aircraft Engine Components
Gordon C. Oates, Editor 1985
ISBN 0-915928-97-3

Aerothermodynamics of Gas Turbine and Rocket Propulsion
Gordon C. Oates *1984*
ISBN 0-915928-87-6

Re-Entry Vehicle Dynamics
Frank J. Regan *1984*
ISBN 0-915928-78-7