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STABILITY AND CONTROL. VOLUME I. STABILITY AND CONTROL FLIGHT TEST TECHNIQUES

Air Force Flight Test Center
Edwards Air Force Base, California

July 1974

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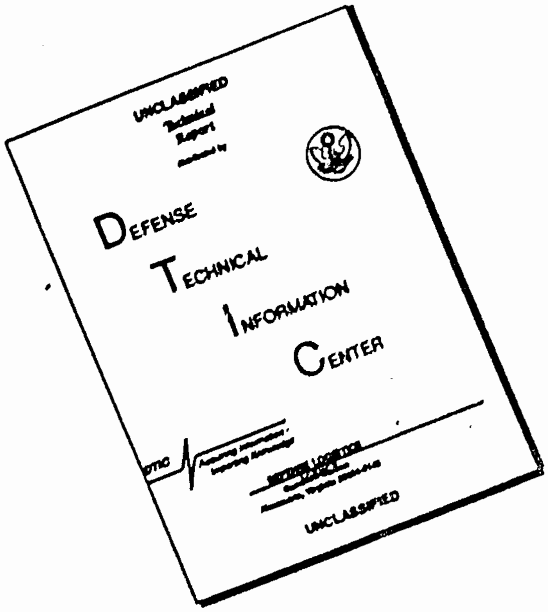

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STABILITY AND CONTROL

Volume I of II

STABILITY AND CONTROL FLIGHT TEST TECHNIQUES

July 1974

Final Report

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USAF TEST PILOT SCHOOL
AIR FORCE FLIGHT TEST CENTER
EDWARDS AFB, CALIFORNIA

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Joseph A. Guthrie Jr.

JOSEPH A. GUTHRIE, JR.
Colonel, USAF
Commandant, USAF Test Pilot School

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This handbook has been compiled by the instructors of the USAF Test Pilot School for use in the Stability and Control portion of the School's course. Most of the material in Volume I of this handbook has been extracted from several reference books and is oriented towards the test pilot. The flight test techniques and data reduction methods in Volume II have been developed at the Air Force Flight Test Center, Edwards Air Force Base, California.

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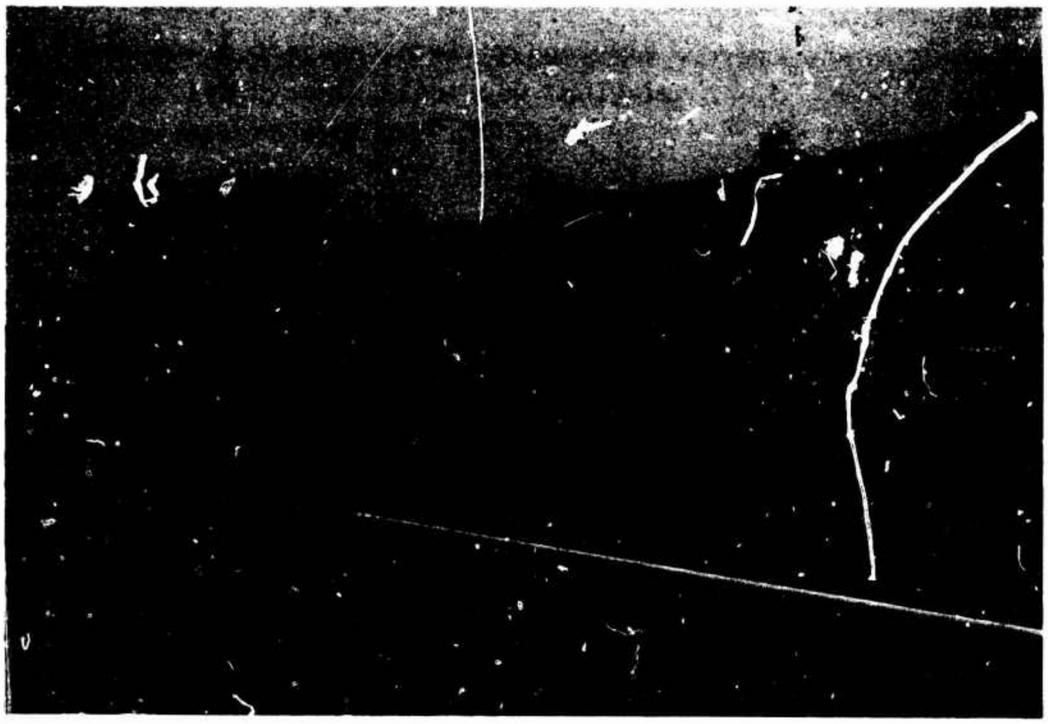

PREFACE

Stability and Control is that branch of the aeronautical sciences that is concerned with giving the pilot an aircraft with good handling qualities. As aircraft have been designed to meet greater performance specifications, new problems in Stability and Control have been encountered. The solving of these problems has advanced the science of Stability and Control to the point it is today.

This handbook has been compiled by the instructors of the USAF Test Pilot School for use in the Stability and Control portion of the School's course. Most of the material in Volume I of this handbook has been extracted from several reference books and is oriented towards the test pilot. The flight test techniques and data reduction methods in Volume II have been developed at the Air Force Flight Test Center, Edwards Air Force Base, California. This handbook is primarily intended to be used as an academic text in our School, but if it can be helpful to anyone in the conduct of Stability and Control testing, be our guest.

JOSEPH A. GUTHRIE, JR.
Colonel, USAF
Commandant, USAF Test Pilot School

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CHAPTER 1

INTRODUCTION TO STABILITY FLIGHT TEST TECHNIQUES

1.1 ATTITUDE FLYING

In stability flight testing, attitude flying is absolutely essential. Under a given set of conditions (altitude, power setting, center of gravity location, etc.) the aircraft's speed is entirely dependent upon the attitude. This being the case, the pilot's ability to fly the aircraft accurately depends upon his ability to see and interpret small attitude changes. This can best be done by reference to the outside horizon. Any change in aircraft attitude will be noticed by reference to the distant horizon long before the aircraft instruments (airspeed, etc.) show a change. Thus, it is often possible to change the attitude of the aircraft from a disturbed position back to the required position before the airspeed has a chance to change. The outside horizon is also very useful as a rate instrument. If a stabilized point is required, hold zero rate of change of pitch; i.e., hold aircraft attitude fixed in relation to some outside reference which calls for one particular speed. If, as in acceleration run, the airspeed is continuously increasing or decreasing, one should look for a steady, smooth, and extremely slow rate of change of the aircraft's attitude.

It is suggested that the method of lining up a particular spot on the aircraft with some outside reference can be useful at times but is often wasteful of time. A general impression is often all that is necessary; i.e., it is possible to see that the aircraft rate of pitch is zero by use of the pilot's peripheral vision while also glancing at the airspeed indicator or some other cockpit instrument. As soon as the pilot notes a rate of change of pitch, he can make proper control movements to correct the attitude of the aircraft. The pilot should always be aware of the outside view even while reading the instruments.

In flight tests involving turning flight, this overall view of the horizon is of utmost importance in order to be able to hold constant velocity or Mach number. If the airspeed is high the nose should be raised and then stabilized at the new position required, using the horizon as a displacement and a rate instrument. If a change of aircraft attitude is necessary, this change should be made relative to the present picture until the rate of change of aircraft attitude again goes to zero at the new stabilized condition.

If it is necessary to stabilize on an airspeed several knots from the existing airspeed, time can be saved by overshooting the required pitch attitude and using the rate of change of airspeed as an indication as to when one should raise or lower the nose to the required position. A little practice will allow the pilot to stabilize on a new airspeed with a minimum amount of airspeed overshoot in the least time.

01.2 TRIM SHOTS

Prior to each stability flight test requiring photopanel or oscillograph data, a trim shot will be taken near the test pressure altitude ($\pm$100 feet). The trim shot will be made using the remote camera and oscillograph switch (not the stick trigger) so that no control forces will be inadvertently fed to the system. The trim shot is used primarily to make any necessary corrections to the force-measuring equipment readings. For example if the stick force gage reads +0.1 with no force applied, it is apparent that 0.1 must be subtracted from all stick force readings for this particular test.

The means of obtaining the different force information will be covered in detail in class; however, it should be kept in mind that it is possible to get erroneous rudder force information if the foot is placed improperly on the rudder bar. The strain gages are located on the lower rectangular pad and the foot should be placed centrally on this pad. Care should be taken not to apply any force to the toe pads since force applied here will not register on the force-measuring equipment. When making force measurements using the instrumented stick grip, it is very important that no extraneous force inputs are made by torquing the stick grip. The force measurements should be taken by using straight fore and aft or left and right force inputs on the center of the stick grip.

The importance of proper trim in stability flight testing cannot be overemphasized. Most of the tests involve force information and therefore it is essential that the aircraft be properly trimmed at the desired speed since one is interested in forces necessary to fly in conditions differing from the trim condition.

In order to stabilize and trim at a particular speed or Mach number at a constant altitude, the speed should first be established by placing the aircraft in the required attitude to give this speed. While obtaining this approximate attitude by reference to the outside horizon, the throttle setting should be changed to give zero rate of climb at the proper test altitude. Minute changes in attitude may be necessary in order to hold the exact airspeed as the power is changed. Once the proper attitude and power setting have been established, the force should be trimmed to zero while holding the required control position to give the required attitude. Release the stick and check for a change in pitch attitude. If the nose starts up or down put the nose back at the trim position with the stick and retrim. Then repeat the procedure. The lateral and directional controls (aileron and rudder) should be used in the proper manner to hold the wings level, maintain a constant heading, and keep the ball in the center of the turn and bank indicator. The necessary forces should be held in order to accomplish this and then the forces should be relieved by proper trim actuation. As in all flying, the pilot who can get the aircraft trimmed most accurately and quickly is the pilot who can do the most things simultaneously. For example, the pilot who can make the required attitude correction while adjusting the power will become trimmed before the pilot who flies strictly by the numbers. The often-used method of moving the trim device and allowing the aircraft to seek a new speed "hands off" is very time-consuming and inaccurate. Hold the aircraft attitude fixed and then relieve the existing control forces. If the aircraft gains or loses a little altitude during the trimming process the parameters of interest in stability testing will not have changed significantly. Therefore if the altitude is within

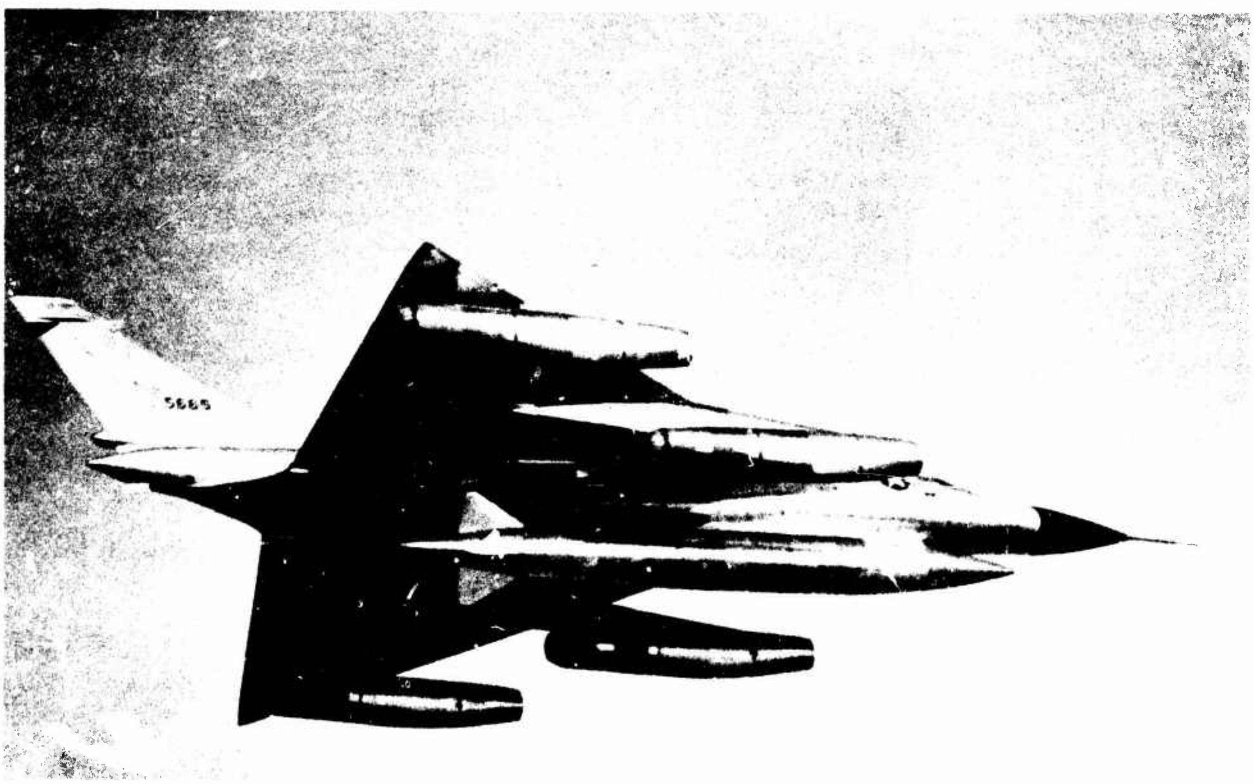
100 feet of the test altitude, the pilot should then take his trim shot.

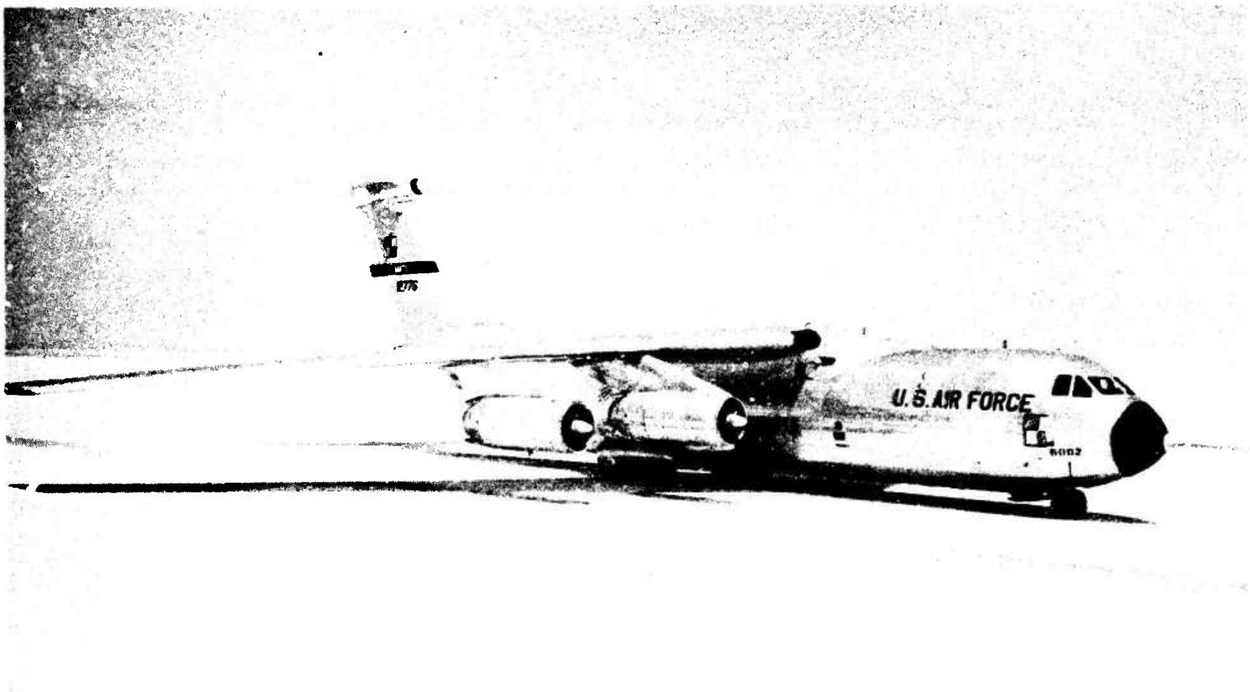
●1.3 TIMING

Proper utilization of time is always of paramount importance in conducting a test program. As a consequence, proper preflight preparation is absolutely essential. The complete test should be well in mind prior to takeoff. The pilot should always be thinking ahead and planning what he is to do next, keeping himself properly positioned in respect to the airfield. A flight that is well planned prior to flight has a very good possibility of working out well.

●1.4 PRIMARY OBJECTIVES

All stability flight tests will be flown with the prime objective of giving the prospective user the most information possible about the particular aircraft. This will be done by noting the aircraft's degree of compliance with the latest specification of flying qualities along with any other information that the test pilot feels essential for safe, effective use of the machine under all conditions.

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CHAPTER 2

STALLS

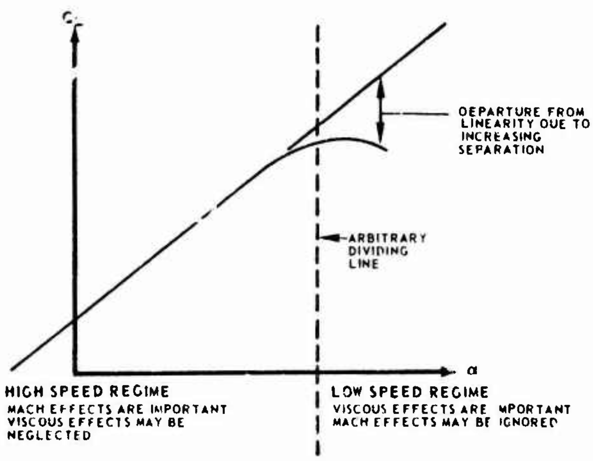
2.1 INTRODUCTION

Stall speed is the minimum steady speed attainable, or usable, in flight. A sudden loss of lift occurring at a speed just below that for maximum lift is considered the "conventional" stall, although it has become increasingly common for the minimum speed to be defined by some other characteristic, such as a high sink rate, an undesirable attitude, loss of control about any axis, or a deterioration of handling qualities.

For rather obvious safety and operational reasons, determination of stall characteristics is a first-order-of-business item in flight testing a new aircraft. Stall speeds are also required early in the test program for the determination of various test speeds.

2.2 SEPARATION

Figure 2.1



Separation, a condition wherein the streamlines fail to follow the body contours, produces a large disturbed wake behind the body and results in a pressure distribution greatly different from the that of attached flow. On an aircraft, these changes in turn produce:

a. A loss of lift

b. An increase in drag

c. Control problems due to:

   1. Control surfaces operating in the disturbed wake

   2. Changes in the aerodynamic pitching moment due to a shift in the center of pressure and an altered downwash angle.

Separation occurs at a point where the boundary layer kinetic energy has been reduced to zero, therefore the position and amount of separation is a function of the transport of energy into and out of the boundary layer and of dissipation of energy within the boundary layer.

Some factors which contribute to energy transport are:

a. Turbulent (non-laminar) flow: Higher energy air from upper stream tubes is mixed into lower stream tubes. This type flow, characterized by a full velocity profile, occurs at high values of Reynolds number (R_e) and involves microscopic turbulence.

b. Vortex generators: These devices produce macroscopic turbulence to circulate high energy air down to lower levels.

c. Slats and slots: These devices inject high energy air from the underside of the leading edge into the upper surface boundary layer.

d. Boundary Layer Control: The blowing type of boundary layer control (BLC) injects high energy air into the boundary layer; while the suction type removes low energy air.

Two examples of energy dissipation functions are:

a. Viscous friction: Energy loss varies with surface roughness and distance traveled.

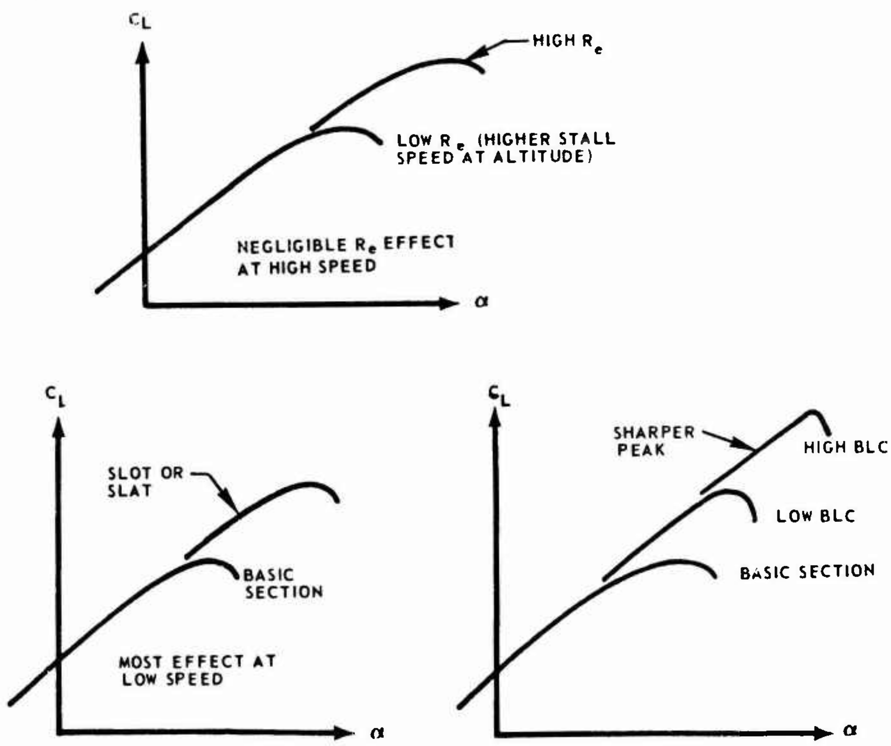
b. Adverse pressure gradient: Boundary layer energy is dissipated as the air moves against the adverse pressure gradient above a cambered airfoil section. The rate of energy loss is a function of:

1. Body contours - such as camber, thickness distribution, and sharp leading edges.

2. Angle of attack - Increased angle of attack steepens the adverse pressure gradient.

Some typical coefficient of lift versus angle of attack (C_L versus α) curves illustrating these effects are shown in figure 2.2.

Figure 2.2



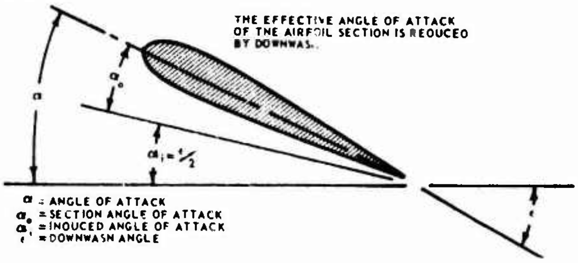
• 2.3 THREE-DIMENSIONAL EFFECTS

2.3 A three-dimensional wing exhibits aerodynamic properties considerably different from those of the two-dimensional airfoil sections of which it is formed. These differences are related to the planform and the aspect ratio of the wing.

Planform:

Downwash, a natural consequence of lift production by a real wing of less than infinite span, reduces the angle of attack at which the individual wing sections are operating.

Figure 2.3



An elliptical wing has a constant value of downwash angle along its entire span. Other planforms, however, have downwash angles that vary with position along the span. As a result, the lift coefficient for a particular wing section may be more or less than that of nearby sections, or that of the overall wing. Airfoil sections in areas of light downwash will be operating at high angles of attack, and will reach stall first. Stall patterns therefore depend on the downwash distribution, and vary predictably with planform as shown in figure 2.4.

Sweptback and delta planforms suffer from an inherent spanwise flow. This is caused by the outboard sections being located to the rear, placing low pressure areas adjacent to relatively high pressure areas.

Figure 2.4

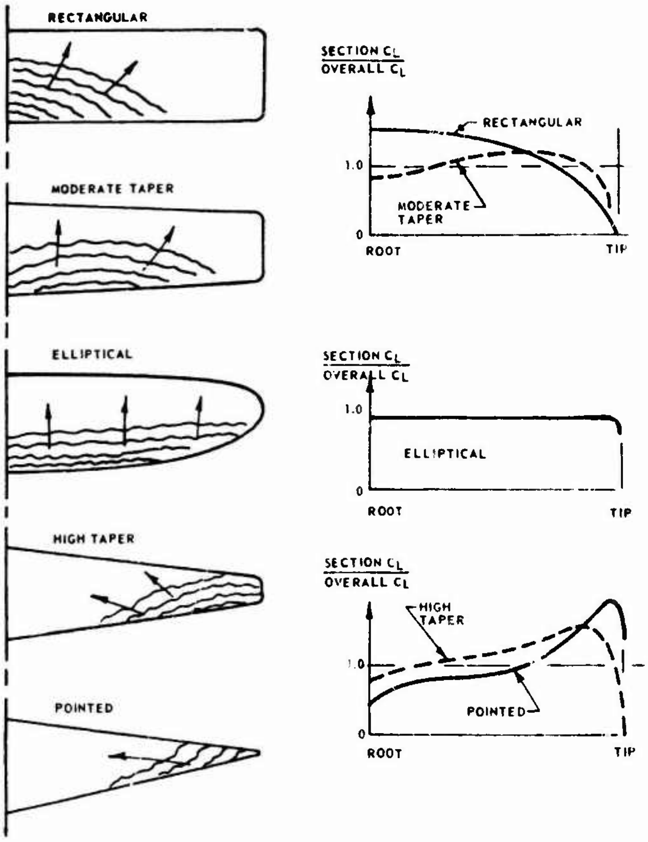
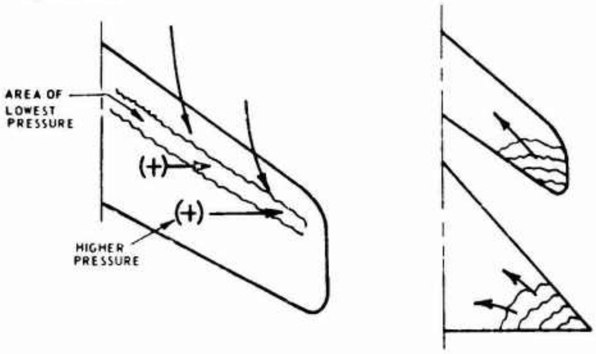


Figure 2.5



This spanwise flow transports low energy air from the wake of the forward sections outboard toward the tips, inviting early separation. Both the sweptback and delta planforms display tip-first stall patterns.

Pointed or low chord wing tips are unable to hold the tip vortex, which moves further inboard with increasing angle of attack.

Figure 2.6

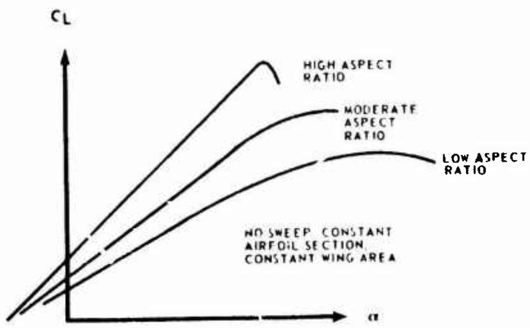
The extreme tips operate in upwash and in the absence of aerodynamic fixes such as twist or droop are completely stalled at most angles of attack.

Aspect Ratio:

Aspect ratio may be considered an inverse measure of how much of the wing is operating near the tips. Wings of low aspect ratio (much of the wing near the tip) require higher angles of attack to produce a given lift.

The curves of figure 2.7 illustrate several generalities important to stall characteristics. High aspect ratio wings have relatively steep lift curve slopes with well defined peaks at $C_{L_{max}}$. These wings have a relatively low angle of attack (and hence pitch angle) at the stall, and are usually characterized by a rather sudden stall break.

Figure 2.7



Low aspect ratio wings display the reverse characteristics; high angle of attack (high pitch angles) at slow speeds and poorly defined stalls. They can frequently be flown in a high sink rate condition to the right of $C_{L_{max}}$ where drag increases rapidly.

Aerodynamic Pitching Moment:

On almost all planforms the center of pressure moves forward as the stall pattern develops, producing a noseup pitching moment about the aircraft center of gravity (cg).

This moment is not great on most straight wing planforms and the characteristic root stall of these wings adds a compensating nosedown moment such that a natural pitchdown tendency exists at high angles of attack. This occurs because the stalled center section produces much less downwash in the vicinity of the horizontal tail, decreasing its download. If the tail actually enters the turbulent wake the nosedown moment may be further intensified due to a decrease in elevator effectiveness. This latter case usually provides a natural stall warning in the form of airframe and control buffet.

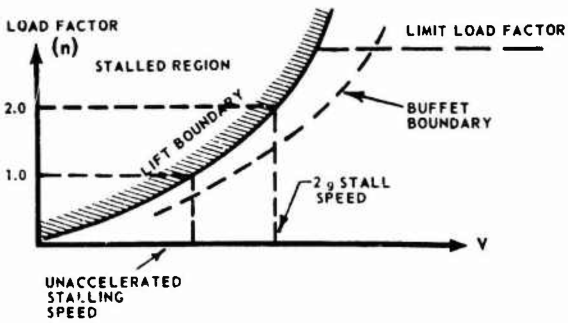
On swept-wing and delta planforms the moment produced by the center of pressure (c.p.) shift is usually more pronounced and the moment contributed by the change in downwash at the tail in this case is noseup. This occurs because the wing root section remains unstalled, producing greater lift and greater downwash as the angle of attack increases. The inboard movement of the tip vortex system also increases the downwash behind the center of the wing. Horizontal tails even in the vicinity of this increased downwash will produce more download. If the tail is mounted such that it actually enters the downwash area at high angles of attack, such as on the F-101, an uncontrollable pitchup may occur.

Many fixes and gimmicks have been used to alter lift distribution and stall patterns. Tip leading edge extensions, tip slots and slats, tip washout and droop, fences and root spoilers are but a few. Horizontal tail position is also subject to much adjustment such as has been necessary on the F-4C.

2.4 LOAD FACTOR CONSIDERATIONS

The relationship between load factor (n) and velocity may be seen on a V-n diagram.

Figure 2.8



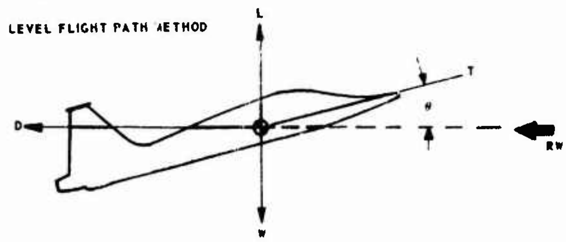
Every point along the lift boundary curve, the position of which is a function of gross weight, altitude, and aircraft configuration, represents a condition of $C_{L_{max}}$ (neglecting cases of insufficient elevator power). It is important to note that for each configuration, $C_{L_{max}}$ occurs at a particular α_{max}, independent of load factor, i.e., an aircraft stalls at the same angle of attack and C_L in accelerated flight, with n = 2.0, as it does in unaccelerated flight, with n = 1.0. The total lift (L) at stall for a given gross weight (W) does however vary with load factor since L = nW. The increased lift at the accelerated stall must be obtained by a higher dynamic pressure (q).

$$q_{stall} = 1/2\rho \, V^2_{stall} = \frac{nW}{C_{L_{max}} S}$$

Thus stall speed is proportional to n, making accurate control of normal acceleration of primary importance during stall tests.

Two flight test methods are described below. The first, involving a level flight path, is an older method that is valid only for unaccelerated stalls. It has several disadvantages that limit its application, but in certain cases such as VSTOL testing or initial envelope extension it might prove useful. It has been largely replaced by the second method that involves a curved flight path and is valid for both accelerated and unaccelerated stalls.

Figure 2.9



L + T sin θ = W and the flight path is straight. In order to slow the aircraft to stall speed, however, an acceleration (a_D) in the drag direction must be obtained by adjustment of thrust or drag such that D is greater than T cos θ. This represents a disadvantage of the method, since a particular trim power or drag configuration cannot be maintained to the stall.

Examination of figure 2.10 shows that the load factor will be at the desired value of 1.0 only if a_D is large enough. The size of a_D will be indicated by the rate of change of airspeed, termed the bleed rate. Experience has shown that undesirable dynamic effects are encountered if bleed rates much in excess of 1 or 2 knots per second are used. In practice, this usually restricts a_D to a value insufficient to close the acceleration diagram to the desired n = 1.0, another disadvantage of this method.

Figure 2.10

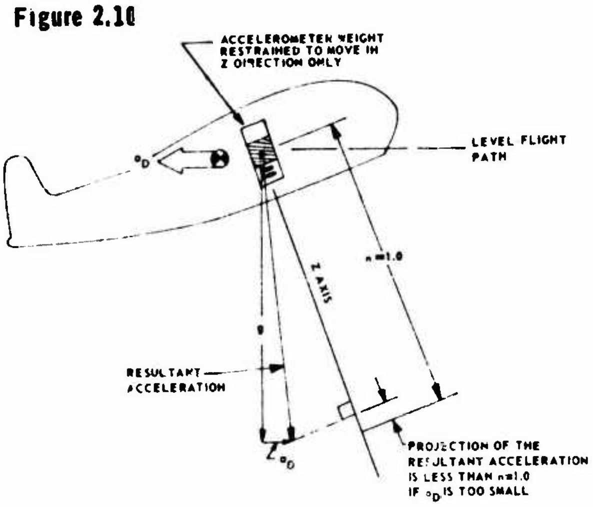
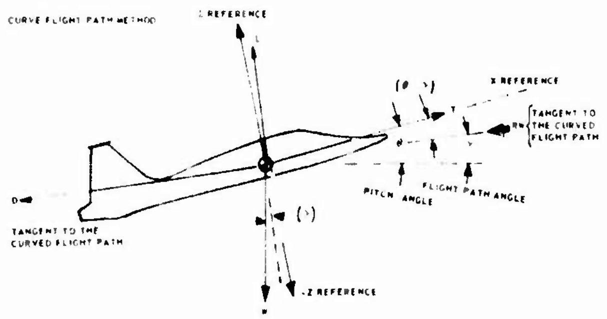


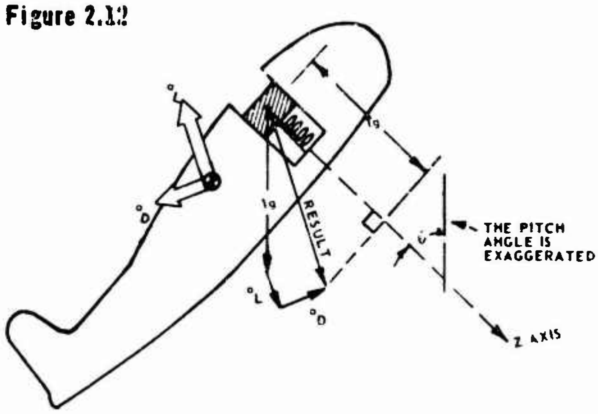
Figure 2.11



Here L + T sin $(\theta - \gamma)$ is greater than W $\cos(\gamma)$ and the aircraft is accelerating (a_L) toward the center of curvature. If trim power was set in level flight, D + W sin γ will be greater than T cos $(\theta - \gamma)$ and there will be an additional acceleration in the drag direction (a_D).

Note that a_L is closely aligned with the vertical reference and that adjustment of the radius of curvature may be used to close the acceleration diagram to the desired value of n. Thus load factor may be easily controlled with the elevator while maintaining trim power and configuration.

Figure 2.12



It is important to realize that the proper value of load factor may be maintained even with large bleed rates (a_D) simply by changing the radius of curvature (a_L), which will of course require a different pitch rate. The steady state diagrams above do not illustrate the need for a small bleed rate. They in fact indicate that the desired load factor may be obtained within a wide range of bleed rates. No rigorous limit for bleed rate can be calculated. In order to remove the possibility of dynamic effects, however, maximums of 1 knot/second for unaccelerated stalls, and 2 knots/second for accelerated stalls have been arbitrarily set on the basis of experience.

2.5 STALL FLIGHT TESTING

General:

Stalls, a familiar maneuver mastered by every pilot when he first learned to fly, must not be taken for granted in a test program. There is a rather large collection of examples from flight test history to document the need for caution. Designs that combine an inherent pitchup tendency with miserable spin characteristics have contributed much to these examples. Stalls are usually first demonstrated by a contractor pilot, but it is possible for a military test pilot to find himself doing the first stalls in a particular configuration, especially on test bed research programs where frequent modifications and changes are made after the vehicle has been delivered by the contractor.

The cautious approach starts with good preplanning. Discuss with the appropriate engineering talent the predicted stall characteristics. Develop with them the most promising recovery technique for each stage of the stall, to include possible post-stall gyrations. In marginal cases, a suggestion for further wind tunnel testing or other alternative investigations might be warranted. Determine the most favorable loading and configuration to be used in the initial stages. Stall and spin practice in trainer aircraft will enhance pilot performance during any out-of-control situations that might develop.

If pitchup or other control problems seem remotely possible, the first runs should terminate early in the approach to the stall and the data carefully examined (on the ground) for trends such as lightening or reversal of control, excessive attitudes or sink rates. Advance this data systematically on subsequent flights - avoid the mistake of suddenly deciding in flight, because things are going well, to take a bigger step than planned.

A stall test point will in general involve three phases; the approach, the stall, and the recovery.

Approach to the Stall:

As will be described later, the aircraft must be flown through this phase in a manner to insure that the stall occurs at the desired altitude and load factor.

Stall warning, if any, will occur during this phase. This requires a subjective judgement by the pilot - only he can tell when he has been warned. This judgement should be extrapolated to the conditions under which the aircraft will be used in service, when distractions such as combat maneuvering may be present. A warning barely discernable during the test program would be of little use under these conditions. Excessive warning is also not desirable; MIL-F-8785 specifies definite upper and lower airspeed limits within which warning should occur. Control shake or airframe buffet is desired although artificial warning devices such as stick and rudder shakers are becoming increasingly common.

The Stall:

Stall has been defined as the minimum steady speed attainable, or usable, in flight. This minimum may be set by a variety of factors, for example:

a. Reaching $C_{L_{max}}$ - the conventional stall.

b. Insufficient longitudinal control to further decrease speed - lack of elevator power.

c. Onset of control problems. (Loss of control about any axis.)

   1. Pitchup

   2. Insufficient lateral-directional control to maintain attitude

   3. Poor dynamic characteristics

d. Back-side problems.

   1. High sink rate

   2. Insufficient wave-off capability

   3. Excessive pitch attitude

Aircraft with lift curves having sharp peaks may be prone to wing drop near the stall if local gusts or control motion cause a high angle of attack to occur on one wing before the other. MIL-F-8785 prescribes definite limits on pitch and bank angle at the stall. The test pilot should describe any other undesirable characteristics that may be evident.

The Recovery:

The recovery is started when the stall or minimum steady speed has been attained. For a conventional stall this is indicated by the inability to maintain the desired load factor - usually a sudden break is apparent on the cockpit accelerometer.

The goal of the recovery must be specified. For example, it might be to keep the altitude lost to a minimum or to obtain the fastest acceleration to a maneuver speed. In a test program all promising recovery procedures consistent with the objectives should be tried. It is important to have the recovery specified in detail before each stall - do not wait until the stall breaks to decide what procedure is to be used. There are no iron-clad rules for recovery - a "standard procedure" such as full military power could be disastrous in certain vehicles. Keep the instrumentation running throughout the recovery until the goal has been attained. In the case of minimum altitude loss this would be when rate of descent is zero and the aircraft is under control (the altimeter is the first indication of R/C = 0).

2.6 DEMONSTRATION MISSION

The student will fly an Attitude and Stall Demonstration Mission from the rear cockpit of the B-57.

Attitude and Stall Demonstration:

It has been found advantageous to devote a portion of the first Stability and Control flight to an exercise in attitude flying and aircraft trim techniques. The purpose of the exercise is to demonstrate and practice the proper techniques for rapidly getting an aircraft on altitude and airspeed in order to obtain an accurate trim shot.

The visual attitude technique for stabilized points will be flown to a zero/zero airspeed and altitude tolerance. Front side and back side trim techniques will be employed; the latter receiving more stress in its application to Stability and Control flight testing.

The technique for stalls will also be demonstrated and practiced. No data will be recorded. The student should however be thoroughly familiar with the B-57 instrumentation operation by the completion of this flight.

TABLE 2.1

DEMONSTRATION FLIGHT DATA

TRIM POINTS		
ALTITUDE (ft)	AIRSPEED (KIAS)	REMARKS
20,000	200	IP Demonstration
24,000	150	Student Practice
20,000	300	Student Practice
STALLS		
CRUISE CONFIGURATION		
20,000		V_{TRIM} 300 KIAS
n = 1.0		n = 2.0
IP Demonstrates each student practices each.		
POWER APPROACH CONFIGURATION		
20,000		V_{TRIM} 140 KIAS
n 1.0		n 1.5
Student Practice		

• 2.7 STALL TEST TECHNIQUES

As on all missions, the student will be expected to keep abreast of the overall progress of the flight, including such considerations as airspace boundaries, restricted areas, turbulence conditions, flight time remaining and proximity to the base. Avoid getting "tunnel vision" on the immediate details of the task to the exclusion of all else.

Trim Point:

a. Set the configuration

b. Check for symmetric engine power and good lateral-directional trim

c. Record a stabilized trim point (±1 knot ±100 feet)

d. Record trim power

Entry Conditions:

Decide on an entry airspeed and altitude (based on previous experience) and do not settle for other values. A smooth well established entry is essential to good results.

Unaccelerated Entries.

a. Get established straight and level on entry airspeed and altitude with trim power reset.

b. Make a slight initial pitch rotation to start the bleed rate, using the outside visual attitude for reference, not the airspeed indicator.

Accelerated Entries.

a. Establish a roughly level turn at the entry airspeed, altitude and load factor. Reset trim power.

b. Using the visual attitude and the accelerometer as reference, substitute an increment of pitch for an increment of bank angle while maintaining the aim load factor. The nose should begin to follow a chandellelike helix above the horizon. The angle of this helix with the horizon determines the bleed rate.

Approach to the Stall:

Pitch control is used primarily in this phase to maintain the aim load factor, although some adjustment of the bleed rate may be made.

Unaccelerated Stall Approach.

a. Check the bleed rate and correct with pitch if necessary.

b. Start the instrumentation at some predetermined speed.

c. Call out the airspeed and actuate the event marker at stall warning. Mentally note the type and adequacy of the warning.

d. Use pitch control to keep n = 1.0. Do not attempt bleed rate corrections after stall warning. Keep wings level. Pitch rotation must increase as the stall is approached to keep the aim load factor, and the accelerometer must be closely monitored to catch the stall break.

Accelerated Stall Approach.

a. Bleed rate is largely determined by the initial helix angle A bank angle increase will slow the bleed rate and a bank angle decrease will speed it up, provided the load factor is maintained. Concentrate on the aim load factor; once it is deviated from it is difficult to salvage a run.

b. Start the instrumentation at a speed well in advance of stall warning.

c. Check the bleed rate again only to evaluate the stall. Corrections late in the approach are useless.

d. Call out airspeed and actuate the event marker at stall warning. Qualitatively evaluate the warning for the type and adequacy.

e. Maintain the load factor until the stall breaks.

The Stall:

The accelerometer must be closely cross checked to catch the break. A good positive pitch rotation will make the stall easy to identify. If the stick is relaxed near the stall (a natural tendency) a pseudo-break will confuse the issue. At the break call out the airspeed and altitude. Initiate recovery controls and configuration change, if any. Qualitatively evaluate the aircraft stall characteristics.

The Recovery:

Keep the instrumentation running and follow the predetermined procedure. Qualitatively evaluate the recovery characteristics. Call out the final recovery altitude.

Clean-Up Phase.

a. Stop the instrumentation.

b. Check the general situation and start the aircraft toward the next point.

c. Hand record warning speed, stall speed, altitude lost, and qualitative comments. The verbal call-outs aid in retaining the numbers until they can be written down.

d. Decide if a correction to the entry conditions is required.

• 2.8 DATA

Data to be Recorded:

A continuous oscillograph recording will be taken from before stall warning until after recovery for each of the stalls on which data are collected. The oscillograph will also be used to record trim points. The following data should be hand recorded for each stall:

a. Indicated speed at stall warning (actuate event marker).

b. Indicated stall speed.

c. Altitude lost during recovery.

d. Qualitative comments on:

   1. Type and adequacy of stall warning.

   2. Stall characteristics such as pitch and roll.

   3. Control characteristics during the three phases of the stall.

   4. Recovery technique and effectiveness.

e. Fuel counter readings.

f. Oscillograph run number.

The format of the flight data cards is not specified. However, the stall mission is a very busy one and it will tax the pilot's concentration and agility. Extra time should be spent to devise data cards that will aid in keeping track of the details. A space for every type of comment desired should be provided beforehand; then the pertinent information may be rapidly entered during the clean-up phase. Do not crowd the cards. Prepare an outline type flight card for the IP with space for remarks but not necessarily a duplicate of the data cards.

Data Presentation:

A table and a time history similar to figures 2.13 and 2.14 will be presented in the report. The time history should be of a particularly well flown stall, and/or of one during which some unusual characteristic was observed. All the parameters for the time history may be obtained from the oscillograph record.

The report should include a discussion of the qualitative findings and an evaluation of the aircraft in comparison to the requirements of MIL-F-8785. (The School flight profile was chosen to demonstrate as much as possible in a reasonable time - it does not necessarily fulfill all the requirements of paragraph 3.4.1.)

Figure 2.14

TYPICAL STALL TIME HISTORY

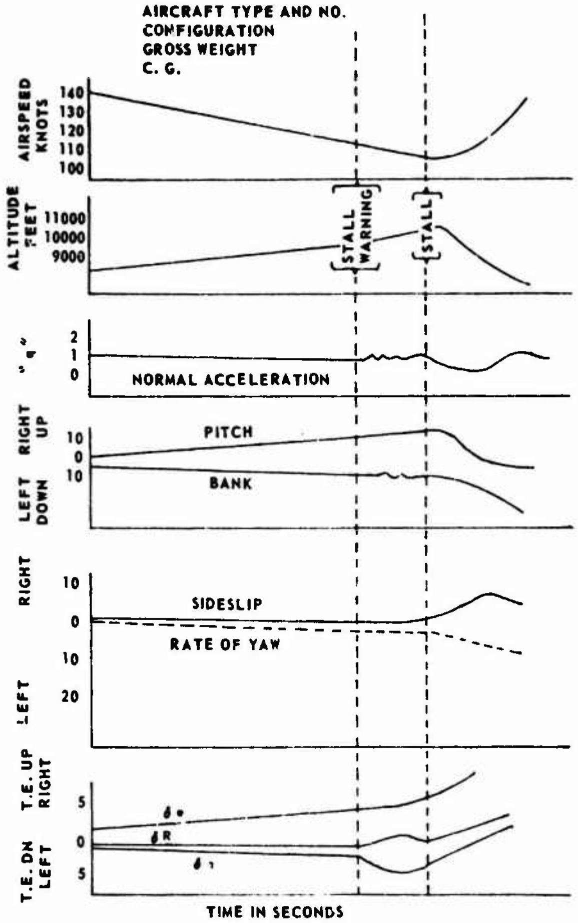


Figure 2.13

Conf	n	Weight	CG	TRIM M	TRIM IAS	ALTITUDE	IAS Warn	IAS Stall	C_{LMax}	V_W/V_S	Remarks
CR											
etc.											

2.9 DETERMINATION OF THE LIFT BOUNDARY

The purpose of this test is to determine the limiting normal acceleration or g's that can be pulled at various speeds and Mach numbers. This may be determined by buffet and/or pitchup. From this data it is possible to determine the best maneuvering Mach number.

Data Recording:

The data for this test will be hand recorded. Normally this test would be flown at several center of gravity positions. However, at the School, to conserve time, the test will be flown while the center of gravity is being moved from the forward to the aft position.

Test Techniques:

Trim the aircraft at 250 KIAS at 20,000 feet. Since the center of gravity is shifting, it will be impossible to maintain trim for an extended period of time. Furthermore, the data obtained on this test is not a function of stabilizer position, therefore, do not take a trim shot. Place the aircraft in a steady level turn and increase power in an attempt to hold constant altitude, trim velocity and Mach number. Continually increase the load factor until initial buffet is reached and note the load factor at this time. Continue to increase the bank and load factor until moderate buffet is reached and again note this load factor.

The same technique will be used at airspeeds of 280, 300, 320, and 350 KIAS. At the higher airspeeds, when it may be impossible to hold constant altitude at full power, plan the entry from a higher altitude so that the aircraft will reach buffet at the required speed and altitude. A tolerance of plus or minus 500 feet will be allowed.

Care should be taken not to increase load factor more than one half g per second in order to minimize dynamic effects.

If any intolerable condition of flight is experienced prior to initial or heavy buffet the run will be discontinued and appropriate mention made of this fact.

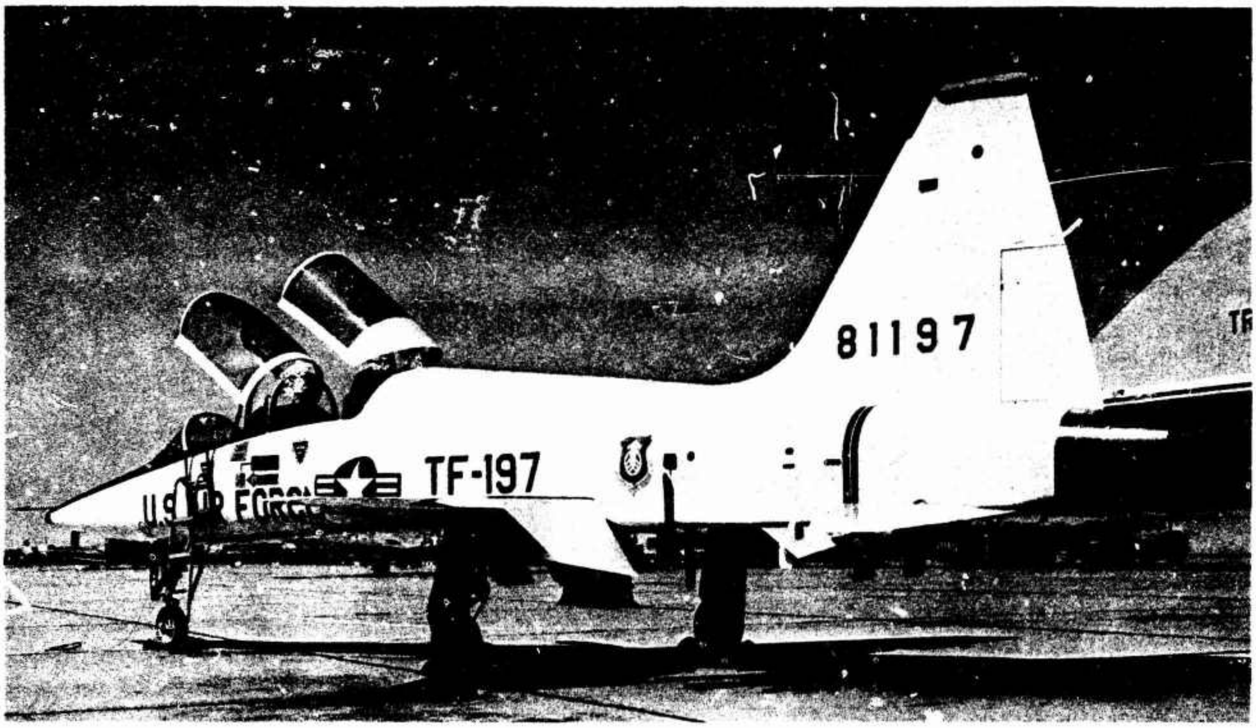
Data Reduction Outline:

The data reduction is as follows:

	Parameter	Source	How Obtained
1	V_i	Instrument Panel	Data Card
2	ΔV_{ic}	Calibration Sheets	
3	V_{ic}		1 + 2
4	ΔV_{pc}	Position error	
5	V_c		3 + 4
6	H_i	Instrument Panel	
7	ΔH_{ic}	Calibration Sheets	
8	H_{ic}		6 + 7

	Parameter	Source	How Obtained
9	ΔH_{pc}	Position error	
10	H_c	Calibration Altitude	8 + 9
11	M_c	Appropriate charts at 10	
12	n (Initial buffet)	Instrument Panel	
13	n (Moderate buffet)	Instrument panel	
14	W_t	Gross Weight	
15	δ	Pressure ratio	Appropriate charts at 10
16	W/δ		14 ÷ 15
17	nW/δ (Initial)		12 x 16
18	nW/δ (Moderate)		13 x 16

Plot 17 and 18 versus 11 showing lines of initial buffet and moderate buffet.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ABBREVIATIONS USED IN THIS CHAPTER

Elevator position (degrees)	δ_e
Pitching moment coefficient	C_m
Elevator power (degrees^{-1})	$C_{m_{\delta_e}}$
Lift coefficient	C_L
Stick force (pounds)	F_s
Dynamic pressure (pounds per square foot)	q
Hinge moment coefficient	C_h
Center of gravity position (pct MAC)	cg
Stick fixed neutral point (pct MAC)	h_n
Stick free neutral point (pct MAC)	h'_n
Recommended find approach airspeed (knots)	$V_{o_{min}}$ (PA)
Flight-path angle (degrees)	γ
Rate of descent (ft per min)	R/D

CHAPTER III
LONGITUDINAL STATIC STABILITY

3.1 INTRODUCTION

The purpose of this flight test is to determine the longitudinal static stability characteristics of an aircraft. These characteristics include gust stability, speed stability, flight-path stability, and the associated terms static margin, neutral point, and friction/breakout.

An aircraft is said to be statically stable longitudinally (positive gust stability) if the moments created when the aircraft is disturbed from trimmed flight tend to return the aircraft to the condition from which it was disturbed. Longitudinal stability theory shows the flight test relationships for stick-fixed and stick-free gust stability, dC_m/dC_L, to be

stick-fixed

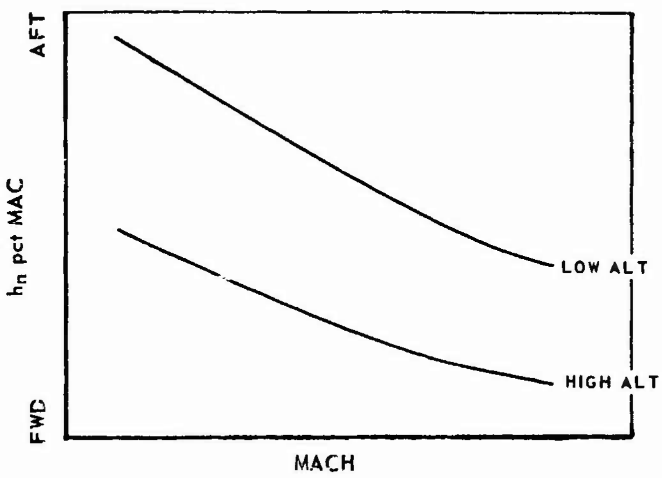
$$\frac{d\delta_e}{dC_L} = -\frac{dC_m/dC_L}{C_{m_{\delta_e}}} \text{ Fixed} \quad (3.0)$$

$$\frac{d(F_s/q)}{dC_L} = -A\frac{C_{h_{\delta_e}}}{C_{m_{\delta_e}}} \frac{dC_m}{dC_{L_{Free}}} \quad (3.1)$$

Stick force (F_s), elevator deflection (δ_e), equivalent velocity (V_e) and gross weight (W) are the parameters measured to solve the above equations. When $\frac{d\delta_e}{dC_L}$ is zero, an aircraft is defined as having neutral stick-fixed longitudinal static stability. As $\frac{d\delta_e}{dC_L}$ increases the stability of the aircraft increases. The same statements about stick-free longitudinal static stability can be made with respect to $\frac{d(F_s/q)}{dC_L}$. The neutral point is the cg location which gives neutral stability, stick-fixed or stick-free. These neutral points are determined by flight testing at two or more cg locations, extrapolating the curves of $\frac{d\delta_e}{dC_L}$ and $\frac{d(F_s/q)}{dC_L}$ versus cg to zero.

The neutral point so determined is valid for the trim altitude and airspeed at which the data were taken and may vary considerably at other trim conditions. A typical variation of neutral point with subsonic Mach number and altitude is shown below.

Figure 3.1 STICK-FIXED NEUTRAL POINT versus MACH AND ALTITUDE



The use of the neutral point theory to define gust stability is therefore time consuming and of limited practical value except for initially predicting aft cg limits. This is especially true for aircraft that have a large airpseed envelope and aeroelastic effects.

Speed stability is the variation in control stick forces with airspeed changes. Positive stability requires that increased aft stick force be required with decreasing airspeed and vice versa. It is related to gust stability, but may be considerably different on aircraft that employ a certain combination artificial feel and stability augmentation. Speed stability is the longitudinal static stability characteristic most apparent to the pilot and it therefore receives the greatest emphasis.

Flight-path stability is defined as the variation in flight-path angle when the airspeed is changed by use of the elevator alone. Flight-path stability generally applies only to the power approach flight phase and is basically determined by aircraft performance characteristics. Positive flight-path stability ensures that the aircraft will not develop large changes in rate of descent when corrections are made to the flight path with the throttle fixed. The exact limits are prescribed in MIL-F-8785B(ASG), paragraph 3.2.1.2. An aircraft likely to encounter difficulty in meeting these limits would be one whose power approach airspeed was far up on the "backside" of the power required curve. A corrective action might be to increase the power approach airspeed, thereby placing it on a flatter portion of the curve or installing an automatic throttle to improve handling qualities.

• 3.2 MILITARY SPECIFICATION REQUIRMENTS

The 1954 version of MIL-F-8785 established longitudinal stability requirements in terms of the neutral point. While the neutral point criteria is still valid for testing certain types of aircraft, this criteria was not optimum for aircraft operating in flight regimes where other factors were more important in determining longitudinal stability. The 1968 version of MIL-F-8785 does not even mention neutral points, instead, section 3.2.1 of MIL-F-8785 specifies longitudinal stability with respect to speed and flight-path. The requirements of this section are relaxed in the transonic speed range except for those aircraft which are designed for prolonged transonic operation.

• 3.3 TEST METHODS

There are two general test methods (stabilized and acceleration/deceleration) used to determine either speed stability or neutral points. There is an additional test method used for determining flight-path stability which is discussed later.

3.3.1 Stabilized Methods

This is used for aircraft with a small airspeed range in the cruise flight phase and virtually all aircraft in the power approach, landing or takeoff flight phases. Propeller type aircraft are normally tested by this method because of the effects on the elevator power changes. It involves data taken at stabilized airspeed at the trim throttle setting with the airspeed maintained constant by a rate of descent or climb. As long as the altitude doesn't vary excessively (typically ±1,000 feet) this method gives good results, but it is time consuming.

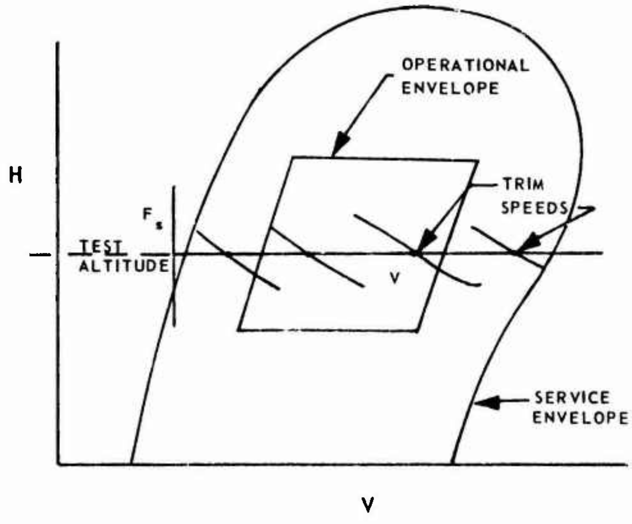
The aircraft is trimmed carefully at the desired altitude and airspeed and a trim shot is recorded. Without moving the throttle or trim setting, the pilot changes aircraft pitch attitude to achieve a lower or high airspeed (typically ±10 knots) and maintains that airspeed. Since the pilot has usually moved the control stick fore and aft through the friction band, he must determine which side of the friction band he is on before recording the test point data. The elevator position for this airspeed will not vary, but stick force varies relative to the instantaneous position within the friction band at the time the data is taken. Therefore, the pilot should (assuming an initial reduction in airspeed from the trim condition) relax the force until the first indication that the nose is beginning to drop and then increase force carefully until the nose starts to rise. This defines the magnitude of the friction band. Since it is generally advisable to record data below trim airspeed points on the backside of the friction band, the pilot should relax the stick force and then increase stick force to a point estimated to be close to, but not on, the backside of the friction band. If the backside of the friction band is reached, there is a good possibility that the elevator will move and the point will no longer be stable. Once this exercise has been completed, the stick is frozen and the data recorded. The same technique should be used for all other airspeed points below trim, although the examination of the friction band may not be required to ensure that the stick force is on the backside. For airspeed points above the trim airspeed, the same technique is employed, although now the front side of the friction band is preferred.

3.3.2 Acceleration/Deceleration Method

This is commonly used for aircraft that have a large airspeed envelope. It is alwasy used for transonic testing. It is less time consuming than the stabilized method but introduces thrust effects. The U.S. Navy uses the acceleration/deceleration method but maintains the throttle setting constant and varies altitude to change airspeed. The Navy method minimizes thrust effects but introduces another consideration because of the change in altitude.

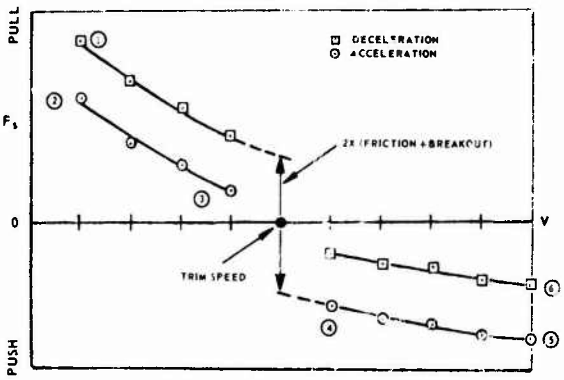
The same trim shot is taken as in the stabilized method to establish the trim conditions. MIL-F-8785B(ASG) requires that the aircraft exhibit positive speed stability only within ±50 knots or ±15 percent of the trim airspeed, whichever is less. This requires very little power change to traverse this band and maintain level flight unless the trim airspeed is near the backside of the thrust required curve. Before the 1968 revision to MIL-F-8785, the flight test technique commonly used to get acceleration/deceleration data was full military power or idle, covering the entire airspeed envelope. Unfortunately this technique cannot be used to conclusively determine the requirements under the current specification with the non-linearities that usually exist in the control system. Therefore a series of trim points must be selected to cover the envelope with a typical plot (friction and breakout excluded) shown in figure 3.2.

Figure 3.2 SPEED STABILITY DATA



The most practical method of taking data is to note the power setting required for trim and then either decrease or increase power to overshoot the data band limits slightly. Then turn on the instrumentation and reset trim power and a slow acceleration or deceleration will occur back towards the trim point. A few percent change in the trim power setting may be required to obtain a reasonable acceleration or deceleration without introducing gross power effects. The points near the trim airspeed point will be difficult to obtain but they are not of great importance since they will probably be obscured by the control system breakout and friction (figure 3.3).

Figure 3.3 ACCELERATION DECELERATION DATA ONE TRIM SPEED, cg, ALTITUDE



Throughout the acceleration or deceleration, the primary parameter to control is stick force. It is important that the friction band not be reversed during the test run. A slight change in altitude is preferable to a reversal of stick force. It is therefore advisable to let the aircraft climb slightly throughout an acceleration to avoid the tendency to reverse the stick force by over-rotating the nose. The opposite is advisable during the deceleration.

There is a relaxation in the requirement for speed stability in the transonic area unless the aircraft is designed for continued transonic operation. The best way to define where the transonic range occurs is to determine the point where the F_s versus V goes unstable. In this area, MIL-F-8785B(ASG) allows a specified maximum of instability in the stick force and a rate of change of instability. The purpose of the transonic longitudinal static stability flight test in the transonic area is to determine the degree of instability.

The transonic area flight test begins with a trim shot at some high subsonic airspeed. The power is increased to maximum thrust and an acceleration is begun. (Note that this applies also to aircraft that are normally termed subsonic, such as the T-33 and B-57.) It is important that a stable gradient be established before entering the transonic area. Once the first sensation of instability is felt by the pilot, his primary control parameter changes from stick force to altitude. From this point until the aircraft is supersonic, the true altitude should be held as closely as possible. This is because the unstable stick force being measured will be in error if a climb or descent occurs. A radar altimeter output on an over-water flight is the most precise way to hold constant altitude, but if this is not feasible the pilot will have to use his pitot-static instruments and outside references to maintain level flight.

Once the aircraft goes supersonic, the test pilot should again concern himself with not reversing the friction band and with establishing a stable gradient. The acceleration should be continued to the limit of the Service Envelope to test for supersonic speed stability. The supersonic data will also have to be shown at ±15 percent of the trim airspeed, so several trim shots may be required.

A deceleration from V_{max} to subsonic speed should be made with a careful reduction in power to decelerate supersonically and transonically. The criteria for decelerating through the transonic region are the same as for the acceleration. Power reductions during this deceleration will have to be figured carefully to minimize thrust effects and still decelerate past the Mach drag point to a stable subsonic gradient.

● 3.4 DATA REDUCTION

Longitudinal stability stability flight tests serve the same two purposes as all other flight tests: to verify compliance with military specifications and gather data to determine the aircraft's flying qualities. These two purposes are equally important, but unfortunately require slightly different approaches and quite different data reduction techniques.

3.4.1 Speed Stability

3.4.1.1 subsonic and supersonic

MIL-F-8785B(ASG) requirements for speed stability are relatively easy to examine. The stick force versus equivalent airspeed data from either test method is plotted. Since all that is required is for the gradient to be stable, it may save time to plot indicated velocity versus stick force; if this is obviously stable, then conversion to equivalent velocity is not necessary. It is essential, however, to identify the breakout and friction forces to separate them from the gradient of stick force versus velocity. Example data are shown in figure 3.3 for an acceleration and deceleration on each side of the trim airspeed. The acceleration from the trim airspeed to the high side of the band and the deceleration from the trim airspeed to the low side would both introduce power effects and would not normally be required to show compliance. They are shown on this example to illustrate breakout and friction. Power effects are assumed to be negligible.

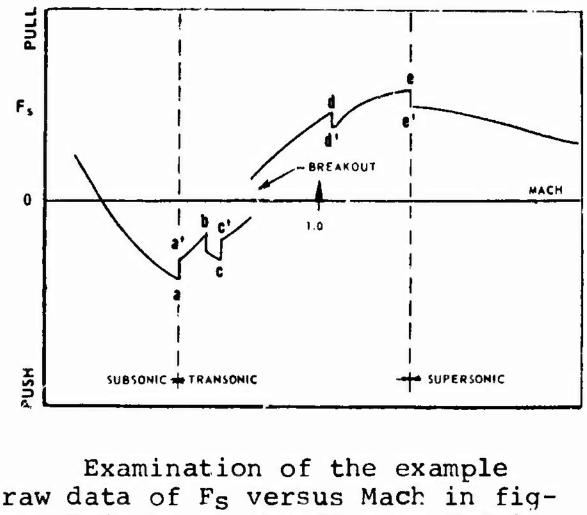
As the deceleration from trim airspeed is begun, the data points recorded are on the backside of the friction band. Upon reaching point 1 power is added to begin an acceleration. At the instant the airspeed starts to increase, the pilot senses the need to lower the nose. As he releases back stick pressure the stick traverses the friction band to point 2. It is not until this point is reached that the elevator starts to move to lower the nose of the aircraft. As the aircraft accelerates the pilot continues to release back stick pressure staying on the front side of the friction band. Somewhere between points 3 and 4 releasing back stick will not lower the nose as the point of hands-off trim is reached. The pilot now pushes forward on the stick to overcome the breakout force again and stay on the forward side of the friction band. As power is reduced to decelerate, the friction band is traversed again from points 5 to 6. If the outside curves are extrapolated to the trim point, the vertical distance between the two represents twice the breakout plus friction force. This is the best way to determine friction plus breakout force since there is no way to determine the location in the friction band where the control stick is at the time the trim shot is taken. This value can be used later in maneuvering flight data. It must be recognized that the friction force is not the same at all airspeeds because the mechanical parts of the longitudinal control system (that cause friction) are in different orientations as stick positions are changed.

For presentation purposes the friction and breakout values are removed from the data. Data for a series of tests at one altitude are shown above in figure 3.2. The trim speeds were selected by MIL-F-8785B(ASG) standards with some overlap.

3.4.1.2 transonic

Speed stability in the transonic regime is generally unstable. The task is to see if the degree of instability exceeds the allowable limits. This involves investigation of the effects of friction that is not required in subsonic and supersonic speed stability if the tests are flown properly. Obviously, friction must be taken out of the measurements of unstable gradients since the gradient can theoreticaly have an infinite slope within the friction band. The value of the friction force must be known to determine whether the gradient of stick force has changed sign or the pilot has merely moved within the friction band. For an acceleration, the total instability should be made from the forward side of the friction band to avoid accounting for friction twice. An example measurement is shown below.

Figure 3.4 TRANSONIC STABILITY



Examination of the example raw data of F_S versus Mach in figure 3.4 shows the effects of friction in the transonic data. Speed instability occurs at point (a) which defines the beginning of the transonic regime. The pilot starts to release forward pressure until reaching the backside of the friction band at point (a'). He continues to release pressure until reaching point (b). Here speed stability is again present until reaching point (c) when the unstable stick forces occur again. The excursions from points (d) to (d') are less than the value of the friction band and are therefore not considered as a change in stability. When the aircraft passes out of the transonic regime - in this example supersonic speed stability occurs at the same point (e) - the pilot begins to release back pressure until the front side of the friction band is reached at point (e'). The unstable gradient may be measured between points (a') and (b) or points (c') and (e) with the excursions at point (d) excluded. The total instability is measured from points (a) to (e').

3.4.2 Neutral Point Determination

Data from acceleration/deceleration or stabilized methods are used to compute the stick-fixed and stick-free neutral points. The minimum requirements are two different cg positions flown at the same test point and trim airspeed. Table 3.1 contains the data reduction outline.

3.4.2.1 stick fixed

The following plots are made from the data reduction outline:

FIGURE 3.5

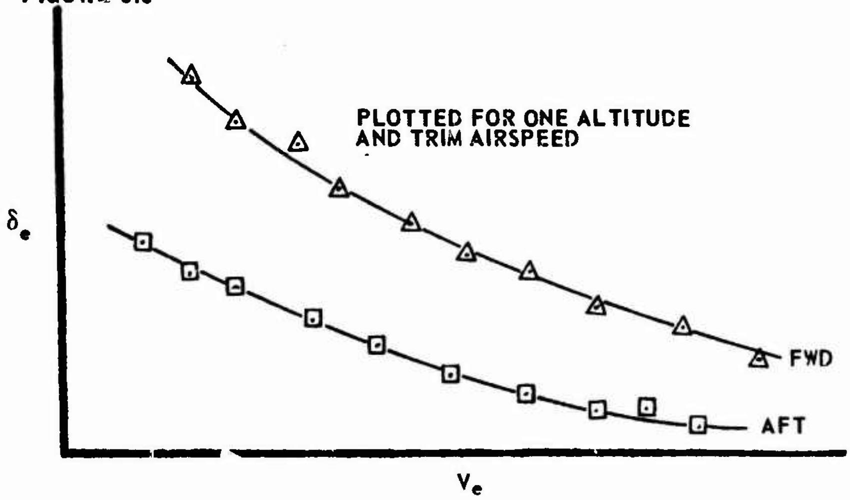
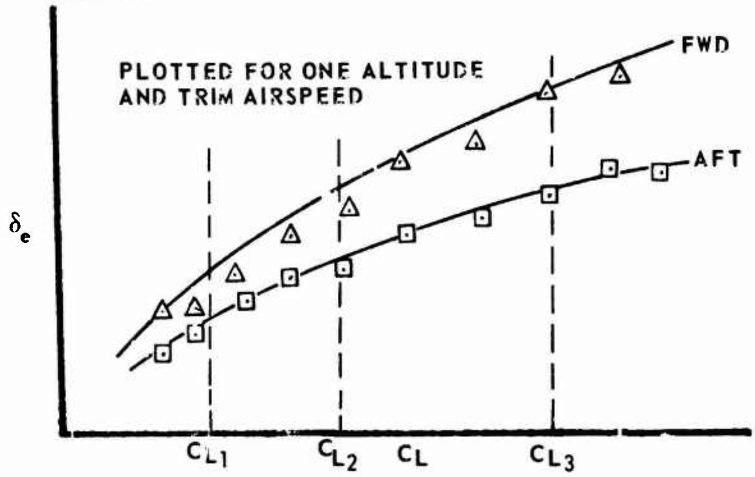


FIGURE 3.6

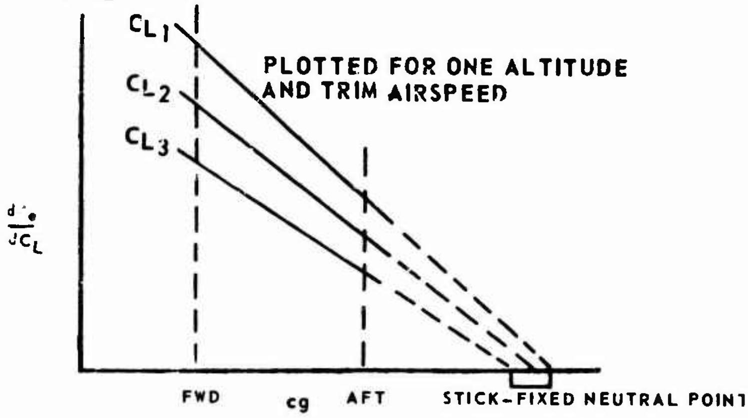


If elevator position plots linearly with lift coefficient, only one stick-fixed neutral point exists. Otherwise, the neutral point varies with angle of attack (or lift coefficient). The derivative $d\delta_e/dV_e$ (at one V_e and cg) does not change with weight unless the neutral point varies with angle of attack. The derivative $d\delta_e/dC_L$ serves better than $d\delta_e/dV_e$ as a plotting variable in locating neutral points because nonlinear weight effects are included.

The rate of change of elevator deflection with respect to lift force coefficient is measured from the plot of δ_e versus C_L. The slope is taken at three or more C_L's over the airspeed range for all cg loadings.

FIGURE 3.7

$d\delta_e/dC_L$ vs cg (WORKING PLOT)

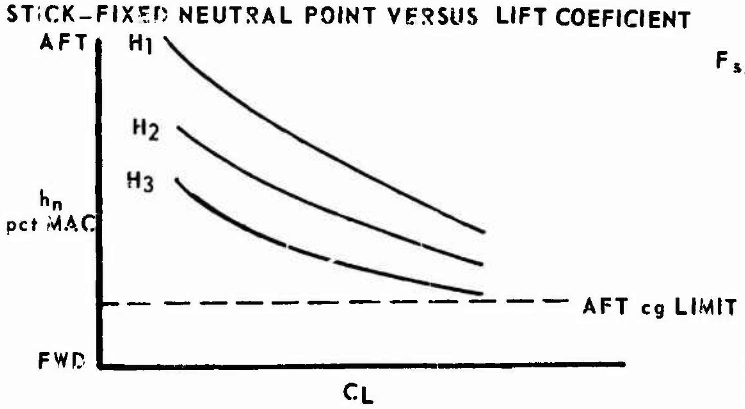


The point where $d\delta_e/dC_L = 0$, is the stick-fixed neutral point for that particular C_L. These neutral points for this one altitude are plotted versus C_L as the curve H_1 in figure 3.8 Additional altitude data would plot as curves H_2 and H_3.

Figure 3.8 indicates the change in stick-fixed stability as the aircraft traverses the speed range at three representative altitudes.

FIGURE 3.8

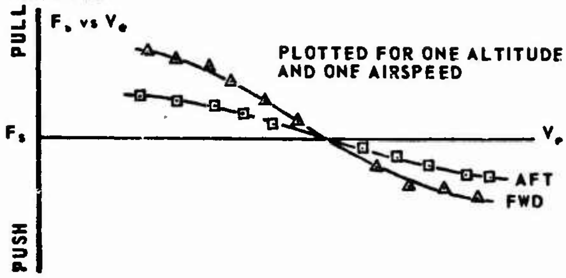
STICK-FIXED NEUTRAL POINT VERSUS LIFT COEFICIENT



3.4.2.2.2 stick free

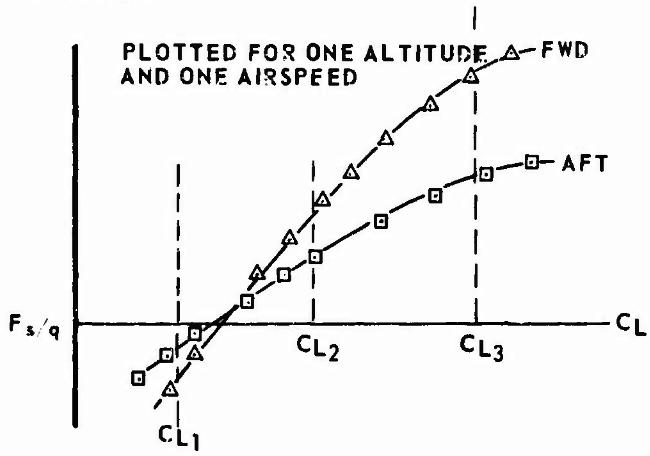
Stick-free neutral points are determined from data in the following manner with friction and breakout removed.

FIGURE 3.9



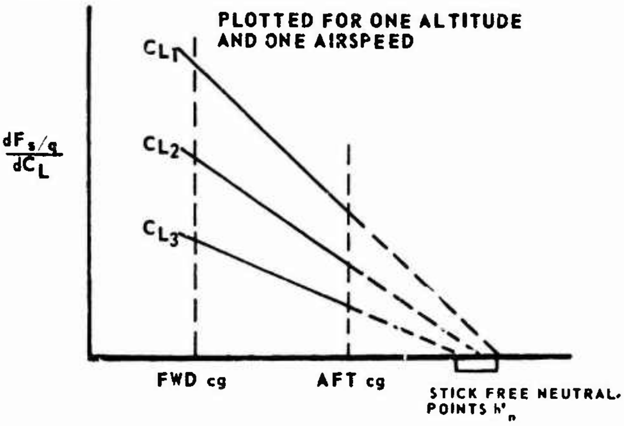
The derivative dF_s/dV_e is a function of aircraft trim as well as stability. This reduces the value of an extracted neutral point. When the stick force is divided by dynamic pressure, the derivative of this quantity, $\frac{dF_s/q}{dC_L}$, is a function of stability only and produces a more valid stick-free neutral point. (See Vol II, chapter 2, page 2.25.)

FIGURE 3.10



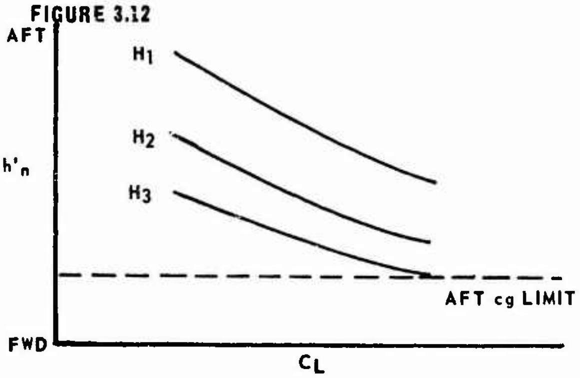
From the plot of F_s/q versus C_L, the rate of change of stick force with respect to lift coefficient is measured and the slope determined at three or more C_L's over the airspeed range for both cg loadings.

FIGURE 3.11



The point where $\frac{dF_s/q}{dC_L} = 0$ is the stick-free neutral point, h'_n, at that particular C_L. The neutral point movement with C_L for one altitude is curve H_1 in figure 3.8. Additional altitudes would plot as H_2 and H_3.

FIGURE 3.12



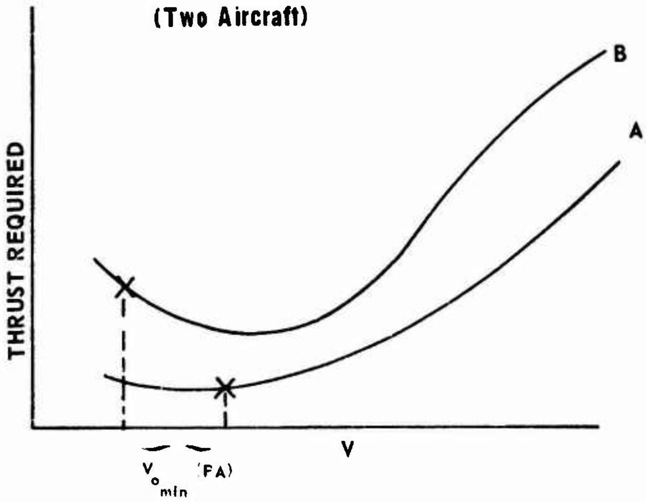
Neutral points vary with configuration, angle of attack, Mach number, and static elastic airframe distortion for a constant weight. No extension attempts should be made to locate transonic stick-fixed or stick-free neutral points because the cg would never be shifted to correct for transonic speed instability in any case. Also the dynamic stability is highly nonlinear with Mach number in this region and the neutral point concept has little utility other than to qualitatively specify an instability.

3.5 FLIGHT-PATH STABILITY

3.5.1 Definition

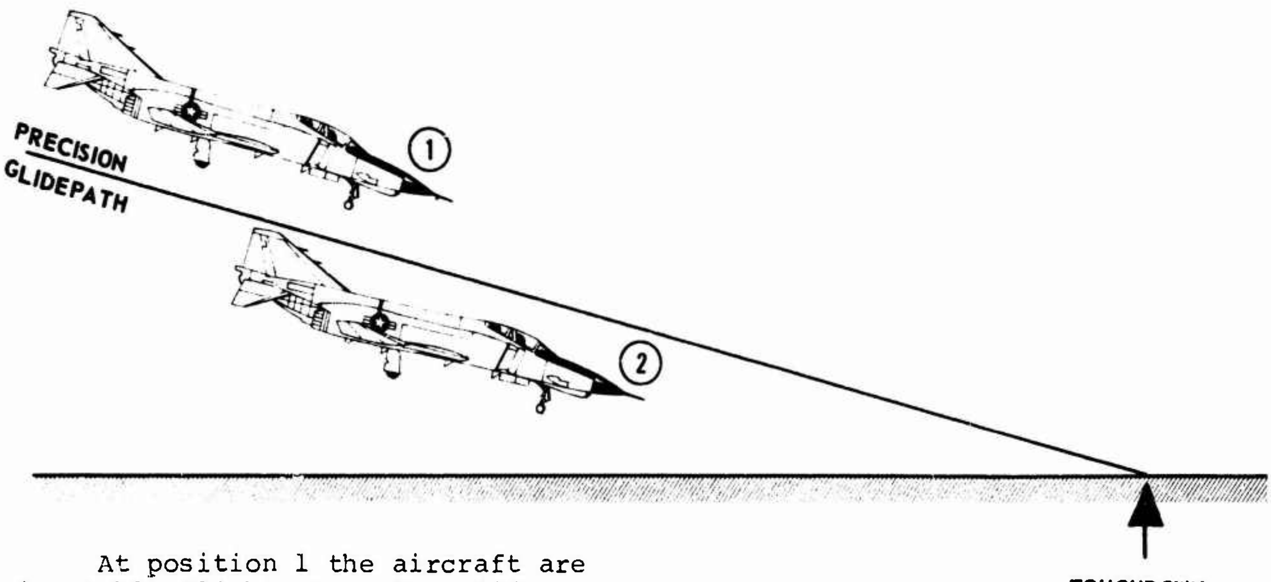
As stated earlier in this chapter, flight-path stability is a criterion applied to power approach handling qualities. It is primarily determined by the performance characteristics of the aircraft and related to stability and control only because it places another requirement on handling qualities. The following is one way to look at flight-path stability. Thrust required curves are shown for two aircraft with the recommended final approach speed marked.

Figure 3.13 THRUST REQUIRED vs VELOCITY (Two Aircraft)



If both aircraft A and B are located on the glidepath shown below their relative flight-path stability can be shown.

Figure 3.14 AIRCRAFT ON PRECISION APPROACH



At position 1 the aircraft are in stable flight above the glidepath, but below the recommended final approach speed. If aircraft A is in this position the pilot can nose the aircraft over and descent to glidepath while the airspeed increases. Because the thrust required curve is flat at this point, the rate of descent at this higher airspeed is about the same as before the correction, so he does not need to change throttle setting to maintain the glidepath. Aircraft B, under the same conditions, will have to be flown differently. If the pilot noses the aircraft over, the airspeed will increase to the recommended airspeed as the glidepath is reached. The rate of descent at this power setting is less than it was before so the pilot will go above glidepath if he maintains this airspeed.

At position 2 the aircraft are in stable flight below the glidepath but above the recommended airspeed. Aircraft A can be pulled up to the glidepath and maintained on the glidepath with little or no throttle change. Aircraft B will develop a greater rate of descent once the airspeed decreases while coming up to glidepath and will fall below the glidepath again. If the aircraft are in position 1 with the airspeed higher than recommended instead of lower, the same situation will develop when correcting back to flight-path, but the required pilot compensation is increased. In all cases aircraft A has better flight-path stability than aircraft B. As mentioned earlier in this chapter, aircraft which have unsatisfactory flight-path stability can be improved by increasing the recommended final approach airspeed or by adding an automatic throttle.

Another way of looking at flight-path stability is by investigating the difficulty that a pilot has in maintaining glidepath even when using the throttles. This problem is seen in large aircraft for which the time lag in pitching the aircraft to a new pitch attitude is quite long. In these instances, incorporation of direct lift allows the pilot to correct the glidepath without pitching the aircraft. Direct lift control will also affect the influence of performance on flight-path stability.

3.5.2 Test Method

Paragraph 3.2.1.3 of MIL-F-8785B(ASG) specifies the slope limits at two points in the flight-path angle (γ) versus true airspeed (V_T) plot. This plot is for the power approach flight phase at the normal glidepath with the throttle set at Vo_{min} (PA). The steeper the glidepath, the more severe the tests, so 3 degrees is chosen as the steepest approach reasonable for a conventional aircraft precision approach. A steeper glidepath is appropriate for aircraft designed to approach in the STOL mode. The preflight planning involves selecting a test altitude so that Vo_{min} (PA) indicated can be converted to V_T and the rate of descent (R/D) can be calculated to give the 3 degree glidepath. An altitude of approximately 10,000 feet MSL is usually selected as the mean altitude for the test. The pilot or engineer must compute the possible Vo_{min} (PA) airspeeds for the aircraft gross weights that he will test. He converts these to true airspeeds and calculates the approximate R/D required to get a 3 degree glidepath in each case.

The pilot begins the test by configuring the aircraft for the power approach configuration at Vo_{min} (PA) at about 12,000 feet MSL. Using a modified back side trim technique, the pilot reduces the power, maintaining airspeed, until the R/D stabilizes at the aim R/D $\pm$100 feet per minute. He hand records R/D, altitude, and airspeed for this point.

The pilot then raises the nose of the aircraft to slow it down about 5 knots and lowers the nose to an attitude that will hold this new airspeed. It is imperative that this airspeed be held within 1/2 knot so that the R/D will settle quickly. Once the R/D stabilizes, the pilot records the R/D, altitude, and airspeed again and repeats the procedure for another decrease of 5 knots. He then noses over to get a point at Vo_{min} (PA) +5 knots and finally Vo_{min} (PA) again.

At least four points are required to define the slope at Vo_{min} (PA) and Vo_{min} (PA) -5 knots. These points should be obtained quickly to avoid excessive altitude loss because changes in altitude affect the R/D for a given airspeed with the throttle fixed. Repeating the Vo_{min} (PA) point at the bottom of the band allows a correction factor to be applied. In any case the data band should not exceed 2,000 - 3,000 feet.

If an aircraft appears to have marginal or unsatisfactory flight-path stability, the test method listed above, using hand recordings of standard aircraft instruments, will not be sufficiently accurate. More sophisticated types of instrumentation must be employed. Ground-based measurements are not practical because winds would affect the data and it would be difficult to account for this effect. Although untried as of this writing, the two most promising methods would be to fly the test over water and use the differentiated output or a radar altimeter to compute flight-path angle or to fly the test anywhere and integrate the output of a vertically mounted accelerometer.

3.5.3 Data Reduction

The first step in data reduction is to convert the airspeeds from indicated to true airspeed. The temperature at the flight level can be obtained from meterological data. Before using the R/D and V_T to compute the flightpath angle, a correction must be applied to account for the changes in R/D with altitude. This correction assumes a linear variation of R/D with altitude. Apply the following relationships for this correction:

R/D_1 = R/D at $V_{O_{min}}$ initial

R/D_2 = R/D at $V_{O_{min}}$ final

h_1 = altitude at $V_{O_{min}}$ initial

h_2 = altitude at $V_{O_{min}}$ final

$\Delta h = h_1 - h_x$ where x is the altitude at a test point

$$\Delta R/D_x = \frac{\Delta h}{h_1 - h_2} \quad (R/D_1 - R/D_2)$$

$$R/D_{x_{corr}} = R/D_x + \Delta R/D_x$$

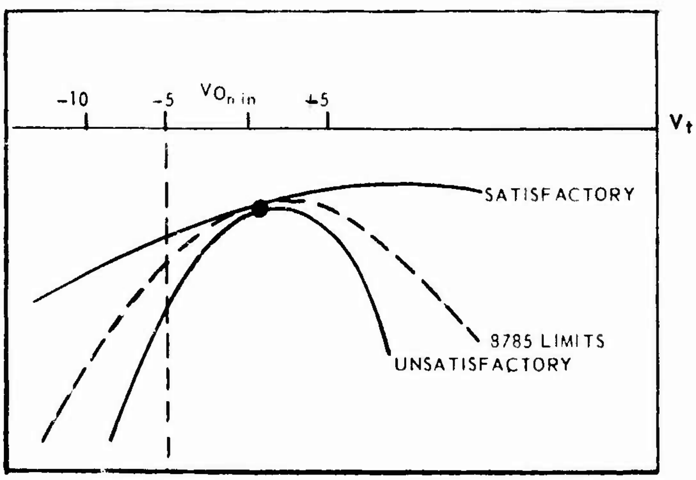
This was used in the following relationship:

$$\gamma = \text{flight path angle} = \sin^{-1} \frac{R/D_{corr}}{V_{True}}$$

Example Data Reduction

Temp (deg C)	CAS (kts)	TAS (kts)	R/D (ft/min)	ALT (ft)	ΔR/D (ft/min)	R/D_C (ft/min)	TAS (ft/min)	γ (deg)
-5	170	202	1,000	11,000	0	1,000	20,000	2.80
-4	164	192	800	10,300	+70	870	19,500	2.56
-3	160	186	700	9,900	+110	810	18,900	2.48
+0	170	199	1,000	8,800	+220	1,220	20,200	3.08
+1	170	194	750	8,500	-	-	-	-

Figure 3.15 SAMPLE DATA PLOT FLIGHT PATH STABILITY



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CHAPTER IV
POST-STALL/SPIN FLIGHT TEST TECHNIQUES

(REVISED DECEMBER 1972)

4.1 INTRODUCTION

This chapter summarizes the preparation for post-stall/spin investigations which test pilots assigned to such projects must undergo. The discussion is purposely general in nature: it will not specifically address the tests flown in the curriculum at the USAF Test Pilot School. These flights are described in detail by reference 1.

At this writing there are drastic philosophical and procedural changes being made in stall, post-stall, and spin testing within the USAF. The thrust of these changes is to provide simpler, more effective techniques for avoiding out-of-control conditions for all classes of aircraft and to force design of better high angle of attack capability into new weapons systems. References 2 and 3 provided some of the impetus for these changes and reference 4 has recently been revised to reflect this emphasis on spin prevention as opposed to spin recovery. Reference 5 has been replaced by reference 6. Clearly, such a shift in emphasis dictates careful attention to planning and execution of flight tests in an obviously hazardous area of flight testing.

4.2 SPIN PROJECT PILOT'S BACKGROUND REQUIREMENTS

4.2.1 PRELIMINARY DATA STUDY:

In previous spin test programs, the contractor was required to demonstrate satisfactory spin recoveries prior to the tests conducted by military project pilots. Under the current stall/post stall/spin demonstration specification (reference 6, paragraph 3.3) military project pilots will participate in high angle of attack investigations concurrently with the contractor's pilots. With this change in philosophy, it is imperative that the military test pilot assigned to a high angle of attack investigation thoroughly study all available background information.

Literature research should begin with the best and most current wind tunnel data available. Take careful note of any configuration or mass changes which were made since the available wind tunnel data were obtained. Look questioningly at the angle of attack and angle of sideslip ranges tested in the tunnel. Go over this data very carefully with the flight test engineers and try to ascertain the probable spin modes and optimum recovery techniques for each of them, as well as the optimum recovery procedure for post-stall gyrations if one is known. Start looking, even at this stage, for the simplest recovery technique possible. If possible, obtain analytical data to confirm or deny the possibility of using a common recovery procedure for both post-stall gyrations and spins (reference 6, paragraph 3.4.3). Spin test reports of similar aircraft should be reviewed thoroughly, but care must be exercised in extrapolating results. The spin characteristics of aircraft which are quite similar in appearance can vary drastically. Attempt to predict the effect that various loadings and configurations will have on post-stall/spin characteristics so that initial tests can be planned conservatively. As examples, the A-1 has loadings from which recovery is not acceptable (reference 7, page 6); and highly asymmetric loadings in the A-7D may prolong recovery to an unacceptable degree. Flight tests of the A-7D were not performed with loadings of greater than 13,000 foot-pounds of asymmetry (reference 8, page 11).

4.2.2 PILOT PROFICIENCY:

It is imperative that the test pilot engaged in a post-stall/spin test program have recent experience in stalls and in spinning aircraft as similar as possible to the test aircraft. Obviously, such aircraft should be those cleared for intentional spins. Coupled departures in a mildly spinning aircraft may be helpful in simulating the post-stall gyrations of an aircraft not cleared for intentional departures. Lack of spin practice for as little as three months will reduce the powers of observation of even the most skilled test pilot. Therefore, he should practice until he is at ease in the post-stall/spin environment immediately prior to commencing the data program. Centrifuge rides, with simulated instrumentation procedures and required data observations, can also be useful.

4.2.3 CHASE PILOT/AIRCRAFT REQUIREMENTS:

A highly qualified chase pilot in an aircraft compatible with the test aircraft increases the safety factor and adds another observer. The chase pilot should participate fully in the preparation phase. In fact it is preferable that more than one pilot be assigned to a given project. Not only does such an arrangement permit more than one qualitative opinion, but by alternating between post-stall/spin and chase assignments, each pilot gets at least two viewpoints. He can evaluate the post-stall/spin characteristics both as an in-the-cockpit observer and from the somewhat more detached chase position. Of course, from a flying safety viewpoint the benefits a competent chase pilot offer are obvious and immeasurable. In order to be useful, the chase pilot should be in an airplane with performance compatible with that of the test aircraft. His responsibilities include: staying close enough to observe and photograph departures, post-stall gyrations, and any spins; staying out of the way of an uncontrollable test aircraft; and being immediately in position to check any unusual circumstances, such as lost p[illegible] malfunctioning drag/spin chutes, or control surface positions. A[illegible]ourse, if necessary, he can call out canopy jettison/ejection a[illegible]ll these responsibilities point up the importance of a well-prep[illegible] servant chase pilot in a similar aircraft.

4.3 DATA REQUIREMENTS

It would be presumptuous to suggest that a comprehensive set of required data can (or even should) realistically be set down within the space limitations in a manual intended as an instructional text for experimental test pilots. However, it is essential that a test pilot have at least a general idea of what parameters must be recorded. Hence, the following two paragraphs are intended to provide only general guidance. Naturally, the test plan for the specific project must be consulted for more detailed and specific requirements.

.3.1 DATA TO BE COLLECTED:

The flight test engineer will be highly concerned primarily with the required quantitative data. Rates of pitch, roll, and yaw, angular accelerations about each axis, control surface positions, angle of attack, indicated airspeed, and altitude are but a few of the typical time histories plotted meticulously by engineers. For the pilot, these data are not the main concern; they are available through automatic recording devices. His most important data gathering lies in a more qualitative area. Can all the necessary controls and switches be reached easily? What are the cockpit indications on production instruments of loss of control warning, departure, post-stall gyration, and spins? Can these indications be readily interpreted, or is the pilot so disoriented that he could not determine what action to take? What visual cues are available at critical stages of the recovery? Reference 8 gives an appropriate example of such a critical stage in the A-7D recovery sequence:

> On several occasions during recovery from fully developed spins, yaw rotation slowed, AOA decreased below 22 units, and roll rotation increased prior to release of anti-spin controls. Pilots found it easy to confuse roll rate for yaw rate leading to the "Auger" maneuver defined as rolling at unstalled AOA with anti-spin controls.

This sort of qualitative finding can be and usually is the most important kind of result from a spin test program. Hence, it is poor practice to ask the pilot to neglect cockpit observations to gather quantitative data which should be recorded by telemetry or on-board recording devices. Project pilots must guard against this pilot overload by looking carefully at the available instrumentation, both airborne and ground-based.

4.3.2 FLIGHT TEST INSTRUMENTATION:

The scope of the post-stall/spin test program will determine the extent of the instrumentation carried on board the aircraft. A qualitative program with a limited objective may require virtually no special instrumentation (reference 9, page 1); while extensive instrumentation may be mandatory for a full-blown stall/post-stall/spin investigation. Table I shows typical on-board instrumentation for a complete evaluation. Of course, this instrumentation is not appropriate for every investigation; each program is a special case.

The prospective test pilot should particularly note the kinds of parameters to be displayed in the cockpit. In this area he must protect his own interests by assuring that the indicators and controls available to him are complete, but that they do not overload his capacity to observe and to safely recover the aircraft. Simulations, preferable under stress of some kind (in a centrifuge, for example), may help the pilot decide whether or not the cockpit displays and controls are adequate.

Finally, reference 6, paragraph 6.4.2.3, directs preparation of a technical briefing film and suggests that an aircrew training film may be produced at the option of the procuring activity. Usually, it is advisable to have one or more movie cameras mounted on or in the test

Table I

TYPICAL FLIGHT TEST INSTRUMENTATION

Parameter	Time History[1]	Photopanel[2]	Pilot's Panel
Angle of attack	X		X
Production angle of attack	X	X	X
Angle of sideslip	X		
Swivel boom airspeed	X	X	X
Swivel boom altitude	X	X	X (coarse altimeter)
Production airspeed	X	X	
Production altitude	X	X	
Bank angle	X		
Pitch angle	X		
Pitch rate	X		
Roll rate	X		
Yaw rate	X		
Normal acceleration	X		X (sensitive indicator)
Accelerations at all crew stations	X		
All control surface positions	X		
Stick and rudder positions	X		
Stick and rudder forces	X		
All trim tab positions	X		
SAS input signals	X		
Engine(s) oil pressures	X	X	X
Hydraulic pressures	X	X	X
Fuel used (each tank)	X	X	X
Film, oscillograph, or tape-correlation and amount remaining	X	X	X
Event marker	X	X	X
Spin turn counter	X	X	X
Elapsed time	X	X	
Critical structural loads	X		X
Pilot warning signal(s)[3]			X
Emergency recovery device indicators	X	X	X

[1] Oscillographs, magnetic tape, telemetry.

[2] May not be necessary if time histories are complete and reliable.

[3] Pilot warning signals may include maximum yaw rate indicators, spin direction indicators, minimum altitude indicators, and other such devices to help lower the pilot's workload. They may take the form of flashing lights, horns, oversized indicators, etc.

aircraft to provide portions of this photographic coverage. Motion pictures taken over the pilot's shoulder may provide visualization of the departure motion, readability of production instruments, information about the adequacy of the restraint system, and other similar data. A movie camera taking pictures of the control surface positions can produce dramatic evidence of the effectiveness or lack of effectiveness of recovery controls. These cameras and recording devices should be made as "crash-proof" or at least as "crash-recoverable" as possible.

Further information on flight test instrumentation, cockpit displays, and cameras may be found in paragraphs 3.2.2, 3.2.3, and 3.2.4 of reference 6.

4.4 SAFETY PRECAUTIONS

Stall/post-stall/spin test programs are usually regarded with suspicion by program managers and flying supervisors. This suspicion is not altogether unreasonable since many such investigations have resulted in the loss of expensive, highly instrumented test aircraft and crew fatalities. This is a fact that awakens the prospective post-stall/spin test pilot's sensibilities, and the fact that such testing is hazardous is not questioned. However, careful attention to detail in several areas will at least minimize the dangers involved.

4.4.1 CONSERVATIVE APPROACH:

One of the most important ways to minimize hazards in such a program is to incrementally expand the areas of investigation, choosing safe increments until the aircraft's uncontrolled motions are better understood. Such a conservative approach is suggested in paragraph 3.4. of reference 6, but how can the test pilot help plan to assure that such an approach is actually followed?

First, the entire program is usually broken down into phases. Even the terms now in use - stall/post-stall/spin - suggest the basic phases of such an investigation, although in practice the phases are generally broken down in more detail. Table II (from reference 6, page 4) lists the recommended phases for such investigations.

Within these phases there are several smaller steps to be taken with successive departures, post-stall gyrations, or spins. For example, aircraft loadings are normally changed gradually from clean to symmetric store loadings to asymmetric store loadings. The effects of these loading changes must be evaluated both for the aerodynamic effects and the changes in mass distribution. Unfortunately, it is not often obvious which effect is most damaging until after the tests are completed. One would also be ill-advised to use full-pro-spin controls on the very first departure in phases B, C, or D. Delayed recoveries should be approached by sustaining the desired misapplication of controls in increments in each successive departure up to the maximum of 15 seconds as indicated in phase D. Such conservatism in flying these tests is essential and must be adhered to scrupulously. However, it is also necessary to consider aircraft systems in order to plan a safe post-stall/spin program.

Table II

TEST PHASES

Phase	Control Application
A - Stalls	Pitch control applied to achieve the specified AOA rate, lateral-directional controls neutral or small lateral-directional control inputs as normally required for the maneuver task.
	Recovery initiated after the pilot has a positive indications of: (a) a definite g-break or (b) a rapid angular divergence, or (c) the aft stick stop has been reached and AOA is not increasing.
B - Stalls with aggravated control inputs	Pitch control applied to achieve the specified AOA rate, lateral-directional controls as required for the maneuver task. When condition (a), (b), or (c) from above has been attained, controls briefly misapplied, intentionally or in response to unscheduled aircraft motions before recovery attempt is initiated.
C - Stalls with aggravated and sustained control inputs	Pitch control applied to achieve the specified AOA rate, lateral-directional controls as required for the maneuver task. When condition (a), (b), or (c) has been attained, controls are misapplied, intentionally or in response to unscheduled aircraft motions, and held for 3 seconds before recovery attempt is initiated.
D - Spin attempts (this phase required only for training aircraft which may be intentionally spun and for Class I and IV aircraft in which sufficient departures or spins did not result in Test Phase A, B, or C to define characteristics.)	Pitch control applied abruptly, lateral-directional controls as required for the maneuver task, when condition (a), (b), or (c) has been attained, controls applied in the most critical positions to attain the expected spin modes of the aircraft and held up to 15 seconds before recovery attempt is initiated, unless the pilot definitely recognizes a spin mode.

4.4.2 DEGRADED AIRCRAFT SYSTEMS:

All systems are under an often unknown amount of strain during high angle of attack maneuvering. If the aircraft goes out of control in this flight regime, system design limits may well be exceeded. The propulsion/inlet system is often not designed to allow reliable operation of the engines during extreme angles of attack and sideslip. Of course, if the engine(s) flame out, this failure may result in loss of control in modern aircraft with hydraulic flight control systems. Obviously, the test vehicle is a post-stall/spin program must have an alternate source of hydraulic power for the flight controls if there is even a hint that engine flameouts are likely to occur. However, do not overlook the behavior of the production hydraulic system: loss of production hydraulic pressure may be all that is necessary to prohibit intentional spins. In propeller-driven aircraft the hydraulic power used to govern the propeller pitch can also be a limiting factor, particularly during inverted spins. The electrical system may also be affected by engine flameout, and such a failure can render instrumentation inoperative at a critical time. In fact, even a momentary distruption of electrical power can destroy invaluable data. Hence, a reliable back-up electrical power source may be necessary. Other systems, such as the ejection system, pilot restraint system, or communications/nevigation system, may cause special problems during the post-stall/spin test program. The test pilot and test engineer must think through these special problems and where necessary add back-up systems to the test aircraft to assure safe completion of the program. Furthermore, any back-up systems that are required must not limit the range and scope of the tests; otherwise, they defeat their purpose.

4.4.3 EMERGENCY RECOVERY DEVICE:

The ultimate back-up system, some sort of emergency recovery device, is so important that it deserves a paragraph all its own. Failure of this "last-ditch" system has in the past contributed to the discomfort of test pilot, engineer, and SPO director all to often. Reference 3 suggests that more attention must be given to the design of this system, perhaps to the extreme of making emergency recovery system components government-furnished equipment (GFE). While the feasibility of this rather drastic suggestion is questionable, it is imperative that more reliable systems be designed. Some of the things that must be scrutinized by the test pilot are:

1. Has the deployment/actuation mechanism demonstrated reliability through the expected envelope of dynamic pressures?

2. Are the moments generated large enough for all predicted spin rates?

3. Has the jettison mechanism demonstrated reliability throughout the expected envelope?

4. Are maintenance inspection procedures adequate for this sytem? (This system should be checked just prior to takeoff.)

5. Does the emergency recovery system grossly alter the aerodynamic and/or inertia characteristics of the test aircraft?

Obviously, no such list is complete, but the test pilot must carefully evaluate every component of the emergency recovery system: spin chute, spin rockets, or any other device.

4.5 SPECIAL POST-STALL/SPIN TEST FLYING TECHNIQUES

Stall flight test techniques have been thoroughly discussed in Chapter II and in reference 6, consequently, the remainder of this chapter is devoted exclusively to flying techniques peculiar to post-stall phenomonea. In general, the test pilot must have indelibly fixed in mind what control actions he will take when the first departure occurs. An inadvertent departure can give just as meaningful (perhaps more meaningful) data as an intentional one - if the test pilot overcomes his surprise quickly enough to make preplanned and precise control inputs. The keys to avoiding confusion in the cockpit have already been mentioned, but they bear repeating. The test pilot must be recently proficient in post-stall gyrations and in spinning, and he must be so familiar with the desired recovery controls that they are second nature. Apart from overcoming the surprise factor through adequate preparation, the test pilot may need some other tricks in this highly specialized trade. For instance, entry to a desired out-of-control maneuver can be a very hit-and-miss proposition.

4.5.1 ENTRY TECHNIQUES:

4.5.1.1 Upright Entries.

For aircraft susceptible or extremely susceptible to spins, an upright spin may be easy to attain. In this case the test pilot's main concern may be how to produce repeatable characteristics: that is, he may seek to achieve the same entry g-loading, attitude, airspeed, and altitude in successive spins so that correlation between spins is easier. Of course, if the aircraft is resistant to spins, it may still be susceptible to departure and entry into a post-stall gyration. In this case, correlation of the data may be even more difficult since the random motions of a PSG seldom are repeatable. Again, the attempt usually is to achieve repeatable entry conditions so that over a large statistical sample the characteristics of the PSG become clear. Achieving several departures with repeatable entry conditions is one of the more demanding piloting tasks. Considerable proficiency is required to achieve the AOA bleed rates or airspeed bleed rates specified in reference 6, page 5. Once the baseline characteristics for a given configuration are relatively well known, the test pilot is called on to simulate entries appropriate to the operational use of the aircraft.

4.5.1.2 Tactical Entries.

These entry maneuvers must be carefully thought out in light of the expected role of the aircraft. It is often wise to consult directly with the using command, particularly if the aircraft has already entered operational service. Of course, reference 6 does suggest the types of tactical entries listed in table III, but past experience is no substitute for foresight in planning such tests. By carefully examining the tactics envisioned by operational planners, the test pilot should be able to recognize other possible tactical entries which may cause difficulty in the high angle of attack flight regime.

Table III

TACTICAL ENTRIES

1.	Normal inverted stalls (see paragraph 4.5.1.3).
2.	Aborted maneuvers in the vertical plane (vertical reversals, loops, or Immelmanns).
3.	High pitch attitudes (above 45 degrees).
4.	Hard turns and breaks as used in air combat maneuvering.
5.	Overshot roll-ins as for ground attack maneuvering.
6.	High-g supersonic turns and/or transonic accelerations/decelerations.
7.	Sudden idle power and/or speed brake decelerations.
8.	Sudden asymmetric thrust transients prior to stall.

4.5.1.3 Inverted Entries.

Obtaining entries into inverted post-stall gyrations or spins can be very difficult simply because aircraft often lack the longitudinal control authority to achieve a stall at negative angles of attack. The most straightforward way to depart the aircraft in an inverted attitude is to roll inverted and push forward on the stick until stall occurs at the desired g-loading. But, many aircraft have marginal elevator authority and it is necessary to misapply the controls to obtain an inverted departure. Pulsing the rudder or applying other pro-spin controls as the nose drops can help precipitate departure. In the OV-10, for example, the direction of applied aileron determines the direction of the inverted spin - provided full aileron deflection is used. However, if aerodynamic controls lack authority, the test pilot can also use inertial moments to precipitate inverted departures.

How the inertial terms can aid entry into a spin can best be seen by examining the $\dot{q}$ equation.

$$\dot{q} = \frac{M_{aero}}{I_y} + pr\,\frac{(I_z - I_x)}{I_y}$$

If the negative pitching acceleration generated by $\frac{M_{aero}}{I_y}$ was too small to produce a stalled negative angle of attack, and additional negative pitching acceleration can be produced from: $pr\,\frac{I_z - I_x}{I_y}$. All that is necessary is for p and r to have opposite signs. Typically, the roll momentum is built up by rolling for at least 180 degrees opposite to the desired direction of the inverted spin and then applying full pro-spin controls at the inverted position. Obviously, these control manipulations must be made at an angle of attack near the stall. Sometimes it is even advisable to apply a slight amount of rudder opposite to the roll during the roll momentum buildup period. A typical procedure designed to produce a left inverted spin is given below:

1. Establish a nose high pitch attitude.

2. Apply full right aileron and a slight amount of left rudder.

3. After a minimum of 180 degrees of roll (360 degrees or more may be advantageous in some aircraft), apply full left rudder, maintain full right aileron, and full forward stick (on some aircraft full aft stick may be used).

4. Recover using predicted or recommended recovery procedures.

Of course, this procedure must be modified to fit the characteristics of a particular aircraft, but it does illustrate the kind of control manipulation sometimes required in post-stall/spin investigations. Some aircraft will not enter an inverted spin using this sort of exaggerated technique, but using the inertial moments to augment aerodynamic controls has uncovered spin modes not obtained by other means. Reference 10 provides further information on the subject of inverted spinning.

4.5.2 RECOVERY TECHNIQUES:

4.5.2.1 Out-of-control Recoveries.

The underlying principle of all recovery techniques is simplicity (refer to paragraph 3.4.2 of reference 6). The procedure to be used must not require the pilot to determine the nature or direction of the post-stall gyration. In fact paragraph 3.4.2.2.2 of reference 4 requires recovery from both post-stall gyrations and incipient spins using only the elevator control. Engine deceleration effects must be tested. Any part of the flight control system (the SAS, for example) which hinders desired control surface placement must be identified and carefully evaluated. Care must be taken to ensure that the recovery controls recommended to recover from a post-stall gyration will not precipitate a spin. The test pilot is primarily responsible for identifying reliable visual and cockpit cues to distinguish between post-recovery angular motions (steep spirals, rolling dives) and the post-stall gyration. Taken together, these requirements demand that the test pilot be a careful observer of the motion. In fact, he is likely to become so adept at making these observations that he must guard against complacency. His familiarity with the motions may cause him to over-estimate the operational pilot's ability to cope with the out-of-control motions. Paragraph 3.4.2.2.2 of reference 4 specifies that the start of the recovery shall be apparent to the pilot within 3 seconds after initiation of recovery. This requirement is very stringent and will require very fine judgement on the part of the test pilot.

4.5.2.2 Spin Recoveries.

The criteria for recovery from a spin are specified in paragraph 3.4.2.2.2 of reference 4 and outlined in table IV. These criteria are applicable to any spin modes resulting from any control misapplication specified in reference 6. Timing of control movements should not be critical to avoiding spin reversals or an adverse mode change.

Table V outlines the so-called NASA Standard and NASA Modified recovery procedures. These recoveries are by no means optimum for all aircraft and they must not be construed to be. In contrast, the F-4E recovery technique now includes forward stick, which reflects the philosophy of simplifying out-of-control recovery procedures. Generally, forward stick is desirable for recovery immediately following a departure. The reason for retaining the forward stick is to keep the out-of-control recovery procedure like the spin recovery procedure. However, individual aircraft characteristics may dictate that out-of-control recovery procedures differ from spin recovery procedures. Such characteristics violate the specifications of reference 4 and 6, but the test pilot must evaluate the need for two recovery procedures. He cannot assume that any "canned" recovery procedure will work nor that the design meets the specifications. In summary, the test pilot's job is to assure that the operational pilot has a simple, reliable recovery procedure which will consistently regain controlled flight.

Table IV

RECOVERY CRITERIA

Class	Flight Phase	Turns for Recovery	Altitude Loss in Recovery[1]
I	Category A, B	1-1/2	1,000 ft
I	PA	1	800 ft
IV	Category A, B	2-1/2	5,000 ft

[1]Not including dive pullout.

Table V

RECOVERY TECHNIQUES

NASA Standard	NASA Modified
(If ailerons were held during spin, neutralize)	Same
A. Full opposite rudder	A. Full opposite rudder and at the same time ease stick forward to neutral.
B. Stick full aft	B. Neutralize rudder when rotation stops.
C. When rotation stops - neutralize rudder (immediately)	
D. <u>EASE</u> stick forward to approximately neutral position.	

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4. MIL-F-8785B(ASG), Flying Qualities of Piloted Airplanes, 7 August 1969 (including Interim Amendment 1 (USAF), dated 31 March 1971).

5. MIL-S-25015(USAF), Spinning Requirements for Airplanes, 18 August 1954 (including Amendement 2, dated 22 May 1957).

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CHAPTER 5

MANEUVERABILITY

5.1 INTRODUCTION

The purpose of maneuvering flight is to determine the stick force versus load factor gradients and the forward and aft center of gravity limits for an aircraft in accelerated flight conditions.

To maneuver an aircraft longitudinally from its equilibrium condition, the pilot must apply a force, F_s, on the stick to deflect the elevator an increment, $\Delta\delta_e$. The requirements that must be met during longitudinal maneuvering are covered in MIL-F-8785, section 3.2.2.

5.2 MIL-F-8785

MIL-F-8785 specifies the allowable stick/wheel force per "g" gradient during maneuvering flight. It also specifies that the stick/wheel force gradients be approximately linear with pull forces on the stick/wheel required to maintain or increase normal acceleration. The pilot must also have sufficient aircraft response without excessive cockpit control movement. These requirements and associated requirements of lesser importance provide the legitimate background for good aircraft handling qualities in maneuvering flight.

The backbone of any discussion of maneuvering flight is stick/wheel force per "g". The amount of stick/wheel force that the pilot must apply to maneuver his aircraft is an important parameter. If the force per "g" is very light, a pilot could overstress or overcontrol his aircraft with very little resistance from the aircraft. The T-38, for instance, has a 5 lb/g gradient at 25,000 feet, Mach 0.9, and 20 percent MAC cg position. With this condition, a ham-fisted pilot could pull 10 g's with only 50 lbs of force and bend or destroy the aircraft. The designer could prevent this possibility by making the pilot exert 100 lb/g to maneuver. This would be highly unsatisfactory for a fighter type aircraft, but perhaps about right for a cargo type aircraft. The mission and type of aircraft must therefore be considered in deciding upon acceptable stick/wheel force per "g". Furthermore, the gradient of stick/wheel force per "g" at any normal load factor must be within 50% of the average gradient over the limit load factor. If it took 10 lb/g to achieve a 4 g turn, it would be unacceptable for the pilot to reach the limit load factor of 7.33 g's with only a little additional force.

The position of the aircraft's cg is a critical factor in stick/wheel force per "g" consideration. The fore and aft limits of cg position may therefore be established by maneuvering requirements.

5.3 EXAMPLE TEST METHODS

5.3 Generally speaking, there are four flight test methods for determining maneuvering flight characteristics such as stick force gradients, maneuver points, and permissible cg locations. The names given to these different methods may vary among test organizations. Therefore, care should be exercised when discussing a particular test method to make certain that everyone involved is speaking the same language.

Stabilized g Method:

This method requires holding a constant airspeed and varying the load factor. The aircraft is trimmed at the test altitude for hands-off flight, and a trim shot is taken. The power setting is then noted, and the aircraft is climbed to the upper limit of the altitude band (±2,000 feet). The power is then reset to trim power and the aircraft is slowly rolled into a 15-degree bank while the nose is lowered slowly. Data is recorded when the aircraft has been stabilized on an airspeed and bank angle with no stick movements. The attitude indicator should be used to establish the bank angle. The bank angle is then increased to 30 degrees and data is again recorded when the aircraft has been stabilized. Stabilized data points are also obtained at bank angles of 45 and 60 degrees.

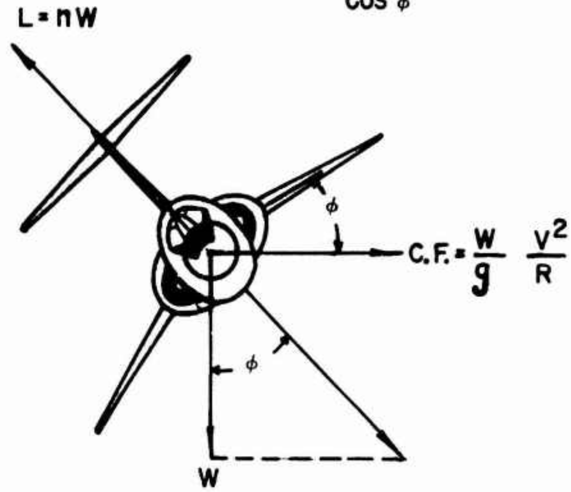
After taking data at the 60-degree bank angle, the bank angle should be increased so as to obtain 0.5 g-increments in load factor. At each 0.5 g-increment, the aircraft is stabilized and data recorded. The test should be terminated when heavy buffet or the limit load factor is reached. Above a load factor of 2.0, only slight increases in bank angle are needed to obtain 0.5-g increments. An idea of the approximate bank angle required can be reached by exploiting the relationship between load factor and bank angle for a constant altitude (figure 5.1).

Figure 5.1 LOAD FACTOR vs BANK ANGLE RELATIONSHIP

W = GROSS WEIGHT
n = LOAD FACTOR
ϕ = BANK ANGLE

$$\cos\phi = \frac{W}{nW} = \frac{1}{n}$$

$$\text{OR } n = \frac{1}{\cos\phi}$$



Little altitude is lost at the lower bank angles up to approximately 60 degrees, and thus more time may be spent stabilizing the aircraft. At 60 degrees of bank angle and beyond, altitude is being lost rapidly; therefore every effort should be made to be on speed and well stabilized as rapidly as possible in order to stay within the allowable altitude block (test altitude ±2,000 feet). If the lower altitude band is approached before reaching limit load factor, the aircraft should be climbed to the upper limit and the test continued. No attempt should be made to obtain data at exact values of g since a good spread is all that is necessary.

The method of holding airspeed constant within a specified altitude band is recommended where Mach number is not of great importance. In regions where Mach number may be a primary consideration, every effort should be made to hold Mach number and airspeed constant.

If power has only a minor effect on the maneuvering stability and trim, altitude loss and the resulting Mach number change may be minimized by adding power as load factor is increased. At times, constant Mach number is held at the sacrifice of varying airspeed and altitude. For constant Mach number tests, a sensitive Mach meter is required or a programmed airspeed/altitude schedule is flown. The stabilized g method is usually used for testing bomber and cargo aircraft and fighters in the power approach configuration.

Slowly Varying g Method:

The aircraft is trimmed as before at the desired altitude. The power is noted and the aircraft is climbed to the upper limit of the altitude band (±2,000 feet). Power is reset at the trimmed value. The data recording switch is activated and the aircraft is slowly banked into a turn. With the airspeed held constant, load factor is slowly increased by increasing bank angle and descending. The slow increase of bank angle and the resulting load factor increase continue until heavy buffet or limit load factor is reached. The rate of g onset should be approximately 0.1 g per second. Again airspeed is of primary importance and should be held to within ±1 knot of aim airspeed. Care should also be taken not to reverse stick forces during the maneuver.

If the airspeed varies excessively, or if the lower limit of the altitude band is approached, the data recorder should be turned off. The aircraft should then be restabilized at the upper altitude limit at a lower g loading. The maneuver should then be continued until heavy buffet onset or limit load factor is reached.

The greatest error made in this method is overbanking beyond 60 degrees of bank. Overbanking, causes the aircraft to traverse the g increments too quickly to be able to accurately hold airspeed. Good bank control is important to obtain the proper g rate of 0.1 g per second.

The slowly varying g method is more applicable to fighter aircraft. Often a combination of the two methods is used in which the stabilized g method is followed until a 60-degree bank angle is reached. The slowly varying g method is then followed from the 60-degree bank angle until heavy buffet or limit load factor is reached.

Constant g Method (Wind-Up Turn):

The aircraft is stabilized and roughly trimmed at the desired altitude and at maximum airspeed for the test. The aircraft is then placed in a constant g turn. Data recording is started and the aircraft is climbed or descended to obtain a 2 to 5 knot per second airspeed bleed rate at the desired constant load factor. Normally the aircraft is climbed to obtain a bleed rate at low load factors and descended to obtain a bleed rate at high load factors. For high thrust-to-weight ratio aircraft at low altitudes, the maneuver may have to be initiated at reduced power to avoid a too rapid traverse of the altitude band. Maintaining the aim load factor is the primary requirement while establishing the bleed rate is secondary. During the maneuver the aircraft should be kept within the altitude band of ±2,000 feet. The airspeed should be noted as the aircraft flies out of the altitude band. When the aircraft is returned to the altitude band, the maneuver is started at an airspeed above the just previously noted airspeed so that continuity of g and airspeed can be maintained for data purposes. Airspeed is again noted at buffet onset and the g break (when aim load factor can no longer be maintained). The buffet and stall flight envelope is determined or verified by this test method. The maneuver is then repeated at 0.5 g increments at high altitudes and 1-g increments at low altitudes.

Symmetrical Pull-Up Method:

The aircraft is trimmed at the desired test altitude and airspeed. The aircraft is then climbed to an altitude above the test altitude using power as required. Trim power is reset and the aircraft is pushed over into a dive. The dive angle is a function of the load factor to be applied (steeper angle for higher g values).

The aircraft is then maneuvered to reach a point, above the test altitude at a lead airspeed below the test airspeed, such that a "g pull" can be established that will place the aircraft at a given constant load factor while passing through the test altitude at the test airspeed. The lead airspeed is determined by the desired load factor – higher load factors and their resultant steeper dive angles require greater leads (lower lead airspeeds). The aircraft should pass through level flight (±15 degrees from horizontal) just as the airspeed reaches the trim airspeed with aim g loading and steady stick forces. Achieving the trim airspeed through level flight, ±15 degrees, and holding steady stick forces to give a steady pitch rate are of primary importance. The variation in altitude (±1,000 feet) at the pull-up is less important. The g loading need not be exact, but should be steady. Data is recorded as the aircraft passes through level flight ±15 degrees. The aircraft is then climbed to an altitude above the test altitude and the maneuver is repeated at another load factor at the same trim airspeed.

5.4 DATA REDUCTION METHODS

The maneuvering flight test is conducted at high, medium, and low altitudes at three different airspeeds or Mach numbers throughout the flight envelope. The aircraft is flown with a forward and an aft cg. Oscillograph recordings give a readout of stick force, elevator deflection, angle of attack, and load factor obtained with the constant airspeed, varying g, flight test method. Data will normally be recorded by use of the camera in both the T-33 and B-57 aircraft. The oscillograph will be used in the T-38. A sample data card is shown.

STUDENT FLIGHT RECORD		DATE 1 APR 68	
NAME OF STUDENT DENT		NAME OF INSTRUCTOR	
AIRCRAFT NUMBER T-38 596		TEST MAN FLT	
CONFIGURATION CRUISE - COMBAT AFT CG			
PRESS ALTITUDE	RUNWAY TEMP	TAKEOFF ROLL	TAKEOFF V_1

GROUND BLOCK AT TAKEOFF	ALTIMETER (at 29.92)	TEMP	TOD	CAM NR	OSC NR

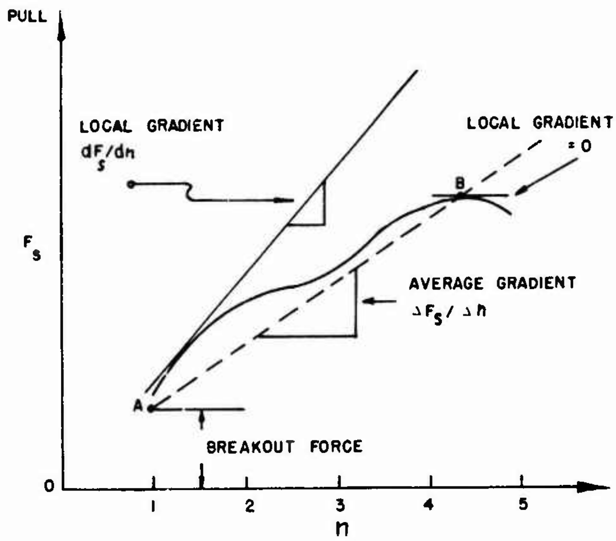
U_i	H_i	TRIM C/N	START C/N	END C/N	LEFT FUEL	RIGHT FUEL
.45	15M					
.60	'					
.75	"					
.60	"					
.75	25M					
.85	"					
.75	35M					
.85	35M					
1.1	"					

GROUND BLOCK AT LANDING	ALTIMETER (at 29.92)	TEMP	TOE	CAM NR	OSC NR

AFFTC FORM 0-112 PREVIOUS EDITION OF THIS FORM WILL BE USED UNTIL STOCK IS EXHAUSTED.

The first step in data reduction is to plot stick force, F_s, against load factor for each test point. A sample plot is shown in figure 5.2. In this figure, the breakout force is determined and labeled Point A. Point B is located where the stick force curve becomes erratic. This point (approximately 85 percent of the limit load factor) may be defined by heavy buffet or change of sign of stick force gradient. The line connecting points A and B is the average gradient. The local gradient is the slope at any point along the curve.

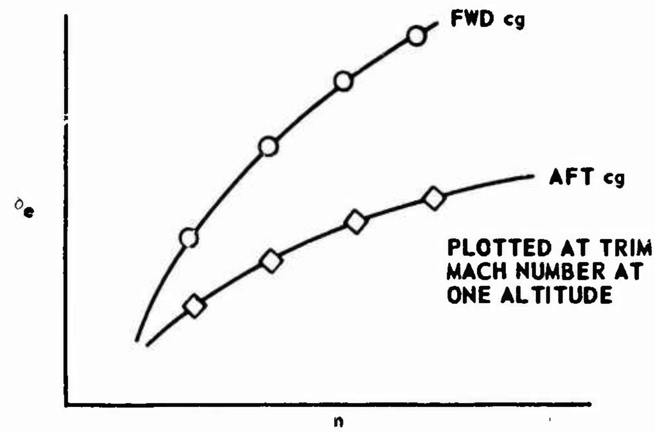
Figure 5.2 DETERMINATION OF AVERAGE GRADIENT FOR IRREGULAR CURVE OF F_s vs n



Stick-Fixed Maneuvering Flight:

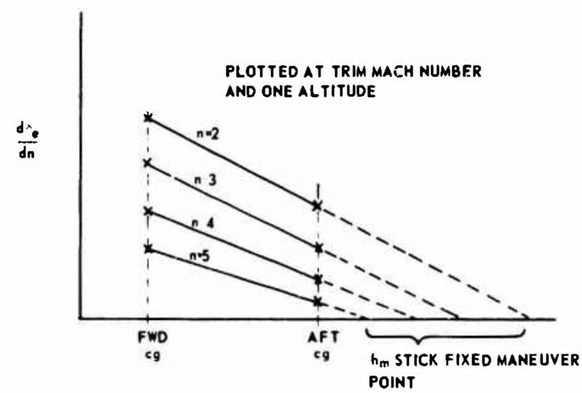
Elevator deflections, δ_e, are plotted versus load factor, n, for each trim airspeed at a particular altitude.

FIGURE 5.3 ELEVATOR DEFLECTION vs LOAD FACTOR



The slope, $d\delta_e/dn$, is determined for several load factors at the two cg positions and is plotted as shown in figure 5.4.

FIGURE 5.4 SLOPE OF ELEVATOR DEFLECTION PER LOAD FACTOR vs cg POSITION (WORKING PLOT)



The cg position where the slope $d\delta_e/dn$ is zero, is the maneuver point location for that load factor at the designated trim Mach and airspeed. Nine plots of δ_e vs n and nine plots of $d\delta_e/dn$ vs cg yield the following summary plot.

Stick-Free Maneuvering Flight:

Stick force, F_s, is plotted versus load factor, n, for each of nine trim Mach numbers (three at

FIGURE 5.5
STICK-FIXED MANEUVER POINT VARIATION WITH LOAD FACTOR

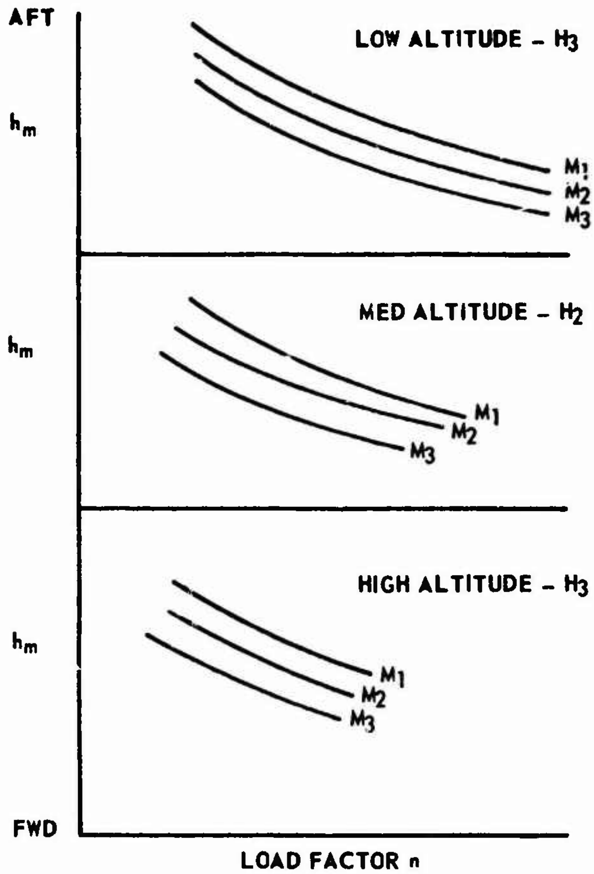
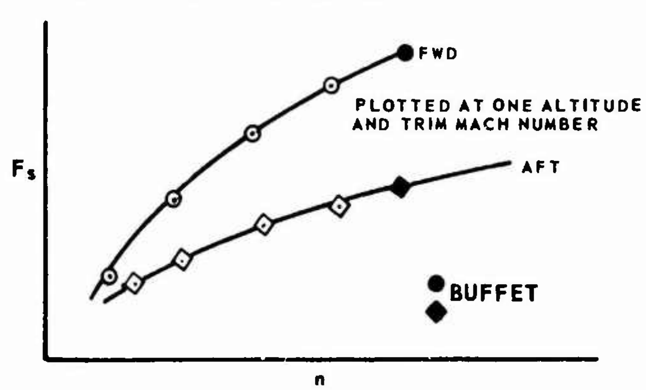


FIGURE 5.6
ST'CK-FORCE VERSUS LOAD FACTOR



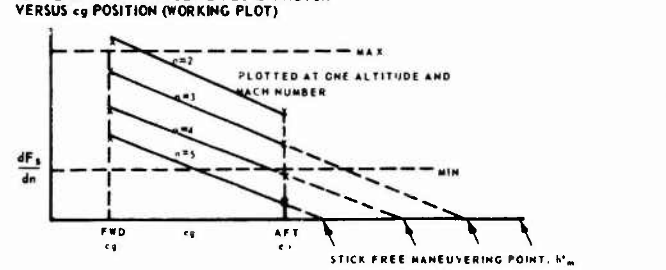
each of three altitudes). One plot for a single trim Mach, M_1, at a particular altitude, H_1, is shown in figure 5.6. The load factor at buffet onset should also be indicated.

It should be noted that the above curves do not necessarily go through zero stick force at one g. This is because there will usually be some breakout force required to allow movement of the longitudinal control.

The average s[illegible] force gradient should be found and the local gradient examined to determine whether or not the local gradient is within 50 percent of the average gradient.

The plot of dF_s/dn versus cg position is derived from the stick force versus load factor curves to determine the stick-free maneuver points.

FIGURE 5.7
SLOPE OF STICK-FORCE PER LOAD FACTOR VERSUS cg POSITION (WORKING PLOT)

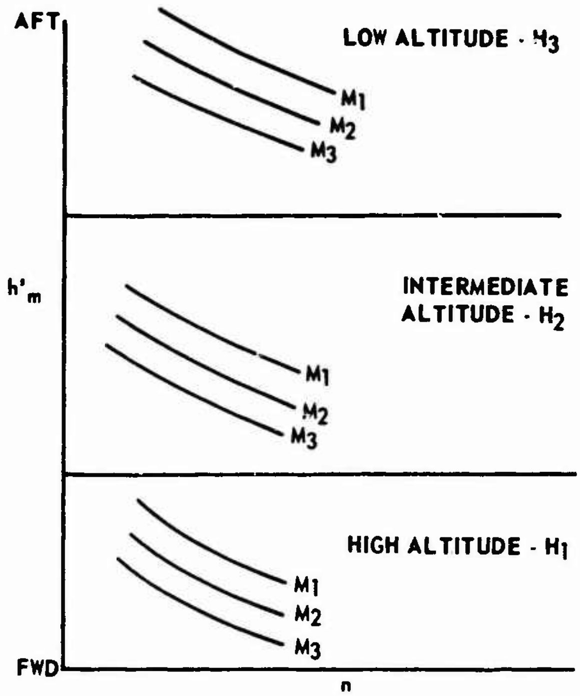


The cg location where dF_s/dn equals zero is the stick-free maneuver point for that particular load factor at the given trim Mach and altitude.

Nine plots for the three altitudes and three trim Mach numbers yield the following summary plots of stick free maneuver point variation with load factor.

A cross plot of figure 5.8 can also be made to show how stick-free maneuver points vary with Mach number.

FIGURE 5.8 STICK FREE MANEUVER POINTS vs LOAD FACTOR



Stick Force Gradients:

It is not sufficient to say that local stick force per "g" gradients do not differ from the average gradient by more than 50%. MIL-F-8785 specifies minimum and maximum values of the local gradient which must be met. These gradients are specified in absolute numbers, $x/n_L - 1$, or in $x/n_Z/\alpha$. Requirements for local gradients specified in terms of $x/n_L - 1$ or in absolute numbers can be determined from the data used to plot figure 5.2. To satisfy the requirements in terms of n_Z/α, another type of plot is required. First form the ratio of n_Z/α in lb/radian (where $\alpha = \alpha_{point} - \alpha_{trim}$). From figure 5.2 determine the F_S/g gradient at the corresponding test point. Finally plot F_S/g vs n_Z/α on a log-log plot as seen below to see if it lies within the MIL-F-8785 requirements.

5.5 DEMONSTRATION MISSION

A demonstration mission will be flown in the T-38 to demonstrate the methods and techniques used to determine stick force gradients and forward and aft center of gravity limits for an aircraft in accelerated flight. No data reduction or plots are required. The procedures to be followed are outlined below.

Procedures:

1. The student will perform an afterburner takeoff and climb to 30,000 feet.
2. He will then stabilize at 300 KIAS and 30,000 feet.
3. The instructor will demonstrate the stabilized g method at 300 KIAS and 30,000 feet.
4. The student pilot will then perform using the stabilized g method under the same trim, altitude, and airspeed conditions.
5. The instructor will demonstrate the slowly varying g method at 300 KIAS and 30,000 feet.
6. The student pilot will next perform the slowly varying g method at 300 KIAS and 30,000 feet.
7. The student will then stabilize at 350 KIAS and 30,000 feet.
8. The student will perform the stabilized g method at 350 KIAS and 30,000 feet.
9. Next, he will perform the slowly varying g method at 350 KIAS and 30,000 feet.

10. Then the student will perform the constant g, slowly varying airspeed method - pulling 3 g's at 20,000 feet.

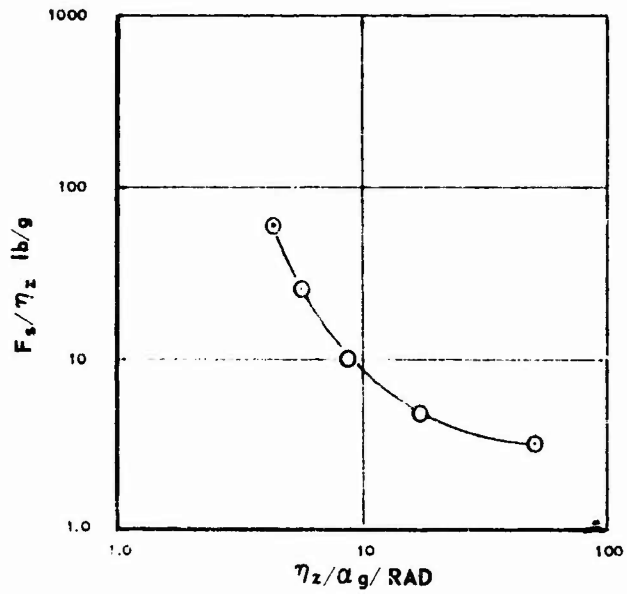
11. Finally, the student will perform two pull-up maneuvers: one of 2 g's at 300 KIAS and 20,000 feet; and the other of 3 g's at 300 KIAS and 20,000 feet.

Instrumentation:

1. None.

2. Simulate picture taking at stabilized g points by calling out "picture."

Figure 5.9 STICK FORCE PER "g" vs n_z/α



CHAPTER 6

TRIM CHANGES

6.1 PURPOSE

The purpose of this test is to ascertain the magnitude of control force changes associated with normal configuration changes, trim system failure, or transfer to alternate control systems in relation to specified limits. It must also be determined that no undesirable flight characteristics accompany these configuration changes.

6.2 TEST CONDITIONS

Pitching moments on aircraft are normally associated with changes in the condition of any of the following: landing gear, flaps, speed brakes, power, bomb bay doors, rocket and missile doors, or any jettisonable device. The magnitude of the change in control forces resulting from these pitching moments is limited by Military Specification F-8785, and it is the responsibility of the testing organization to determine if the actual forces are within the specified limits.

The pitching moment resulting from a given configuration change will normally vary with airspeed, altitude, cg loading, and initial configuration of the aircraft. The control forces resulting from the pitching moment will further depend on the aircraft parameter being held constant during the configuration change. These factors should be kept in mind when determining the conditions under which the given configuration change should be tested. Even though the specification lists the altitude, airspeed, initial conditions, and parameter to be held constant for most normal configuration changes, some variations may be necessary on a specific aircraft to provide information on the most adverse conditions encountered in operational use of the aircraft. The altitude and airspeed should be selected as indicated in the specifications or for the most adverse conditions. In general, the trim change will be greatest at the highest airspeed and the lowest altitude. The effect of cg location is not so readily apparent and usually has a different effect for each configuration change. A forward loading may cause the greatest trim change for one configuration change, and an aft loading may be most severe for another. For this reason, a mid cg loading is normally selected since rapid movement of the cg in flight will probably not be possible. If a specific trim change appears critical at this loading, it may be necessary to test it at other cg loadings to ascertain its acceptability.

Selection of the initial aircraft configurations will be dependent on the anticipated normal operational use of the aircraft. The conditions given in the specifications will normally be sufficient and can always be used as a guide, but again variations may be necessary for specific aircraft. The same holds true for selection of the aircraft parameter to hold constant during the change. The parameter that the pilot would normally want to hold constant in operational use of the aircraft is the one that should be selected. Therefore, if the requirements of MIL-F-8785 do not appear logical or complete, then a more appropriate test should be added or substituted.

In addition to the conditions outlined above, it may be necessary to test for some configuration changes that could logically be accomplished simultaneously. The force changes might be additive and could conceivably be objectionably large. For example, on a go-around, power may be applied and the landing gear retracted at the same time. If the trim changes associated with each configuration change are appreciable and in the same direction, the combined changes should definitely be investigated.

The specifications require that no objectional buffet or undesirable flight characteristics be associated with normal trim changes. Some buffet is normal with some configuration changes, e.g., gear extension; however, it would be considered objectionable in this case if this buffet tended to mask the buffet associated with stall warning. The judgment of the pilot is the best measure of what actually constitutes "objectionable" but anything that would interfere with normal use of the aircraft would certainly be considered objectionable. The same is true for "undesirable flight characteristics." An example would be a strong nose-down pitching moment associated with gear or flap retraction after takeoff.

The specification also sets limits on the trim changes resulting from transfer to an alternate control system. The limits vary with the type of alternate system and the configuration and speed at the time of transfer, but in no case may a dangerous flight condition result. It will probably be necessary for the pilot to study the operation of the control systems and methods of effecting transfer in order to determine the conditions most likely to cause an unacceptable trim change upon transferring from one system to the other. As in all flight testing, a thorough knowledge of the aircraft and the objectives of the test will improve the quality and increase the value of the test results.

6.3 EXAMPLE TEST METHODS

The pre-flight preparation for the trim change test should start with a study of the applicable paragraphs of Military Specification F-8785. By comparing the specification requirements with the expected operational use of the aircraft, it will be possible to determine all the configuration changes and the conditions under which they should be tested.

When all the required changes have been determined, some time should be spent in laying out the sequence in which to test the various items in order to conserve flight time. Most of the configuration changes can be planned so that at the end of one test, the aircraft will be ready for the condition desired on the next test. This will result in a minimum delay. Table X in the specification can be used as an excellent guide in establishing the most advantageous sequence, but this sequence may vary depending on the specific aircraft. In an actual test program, much of the trim change information would probably be obtained during other tests as the aircraft was placed in the required test conditions. For example, the trim change with gear extension might be tested in the landing pattern after completion of another test.

After the required configuration changes and the sequence has been determined, a data card or cards should be prepared. Many forms are possible but the card should include all the information required by the pilot in flight and should be clearly legible. This test does not lend itself to the standard data card format, therefore a card will have to be made up from a blank one. A suggested form follows:

TRIM CHANGES

STUDENT ________________ INSTRUCTOR __________

DATE ______ A/C No ________ CAMERA No __________

GRND SHOT ______________ TEST ________________

TOTAL FUEL _____ CONFIGURATION _________ CG ____

RUN NO	ALT	INITIAL TRIM CONDITIONS				CONFIG CHANGE	HOLD CONSTANT	C.N.	REMARKS
		SPEED	GEARS	FLAPS	POWER				
1	5M	140	UP	UP	PLF	GEAR DN	ALT.		
2	5M	140	ON	UP	PLF	FLAPS DN	ALT.		

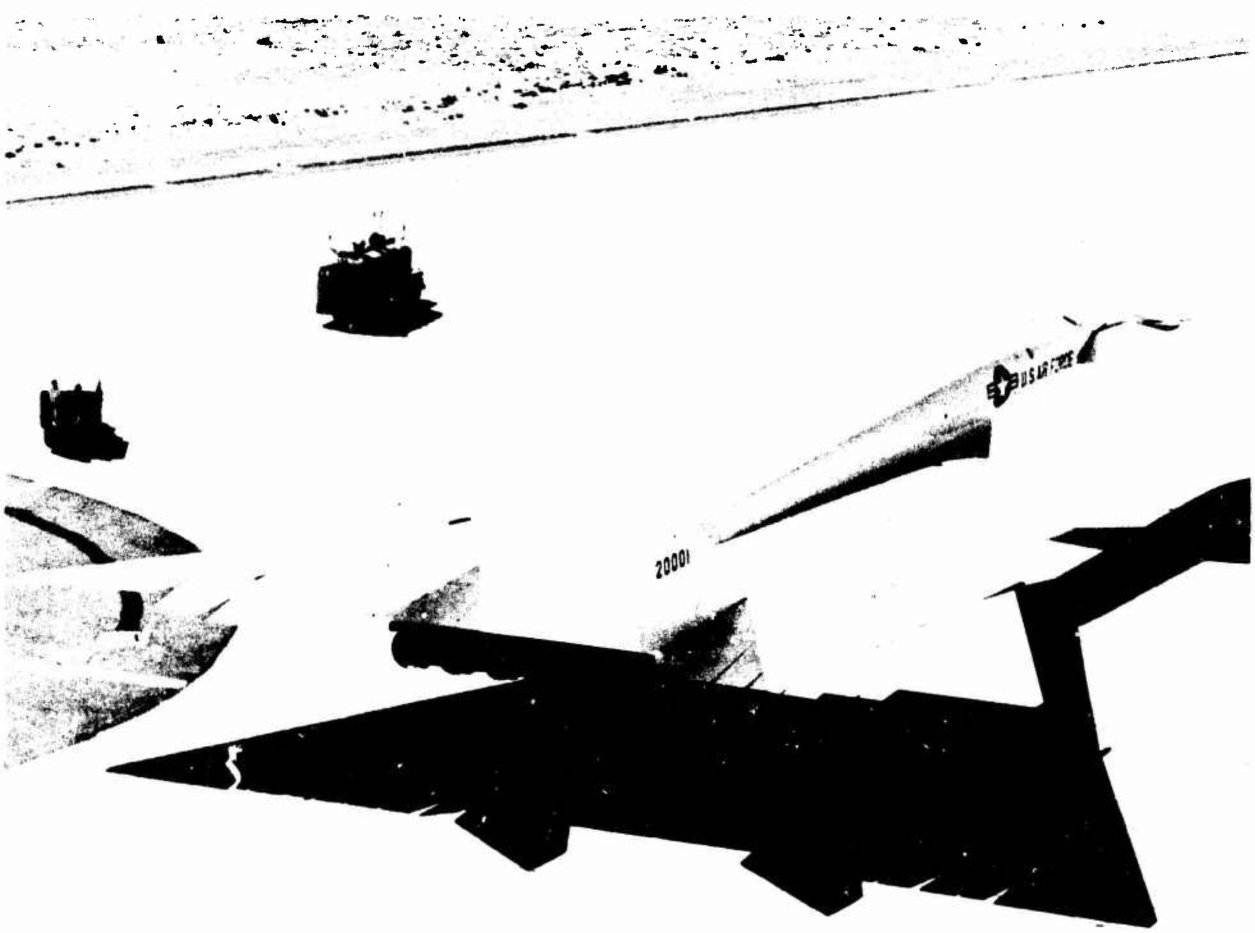
Prior to leaving the line a normal ground photo will be taken to insure that all force and position measuring equipment is functioning properly. Following the flight data card, the aircraft should be placed in the desired condition and trimmed for pilot comfort. Excessive time should not be wasted on being exactly trimmed since a change in control forces is the objective rather than the magnitude of the total force. When trimmed and stabilized the oscillograph should be started, (speed 3 is recommended) and then the desired configuration change is made. Actuating the event marker at the same time the configuration change is initiated will facilitate interpretation of the recorded data. The desired parameter (attitude, altitude, rate-of-climb, etc.) should be held constant with force alone for approximately 5 seconds and then normal trimming action employed to relieve the forces. It is important to pick out the most logical parameter to hold constant, and then smoothly hold that parameter constant. The control force changes which occur are often strongly affected by overcontrolling or allowing the supposedly constant parameter to vary. Both extremes must be avoided. While trimming, a qualitative evaluation of the adequacy of the aircraft trim should be made. If it is too fast or too slow or there is insufficient trim available to relieve the forces, a notation should be made on the data card. It should also be possible to trim the control forces to zero throughout the operational envelope of the aircraft. While there is no specific test for this requirement, the pilot should be alerted to note non compliance during other tests. When approximately trimmed, the oscillograph should be turned "OFF" and the aircraft readied for the next test condition.

This same general procedure will be repeated until all the points listed on the data card have been accomplished. The test for transfer to alternate control modes will have to be done twice. The first one done with the controls free and the second test done with the pilot maintaining attitude and zero sideslip throughout. As always, fuel, time, and orientation should be kept in mind.

The oscillograph traces will be used to determine the changes in forces and control deflections associated with each configuration change. If all the force changes are within the specification limits, all that is necessary is a table of values similar to the example below. However, for one configuration change a time history will be included. This should be for a configuration change that exceeded the limits of the specification. If none exceeded the specification, then a time history of a typical change will be made. The time history will include, but not be limited to, airspeed, altitude, load factor, angle of pitch, angle of bank, elevator force, aileron position, rudder position, and rudder force plotted against time. The table will be as shown below.

TABULAR SUMMARY OF RESULTS

RUN No.	ALT	INITIAL TRIM CONDITIONS				CONFIG CHANGE	HELD CONSTANT	F_s Lbs	δ_e Deg
		A/S	GEAR	FLAPS	POWER				
1	5M	195	UP	UP	PLF	GEAR DN	ALTITUDE	-7	-1.5
2	5M	165	DN	UP	PLF	FLAPS DN	ALTITUDE	+16	-1.0
ETC									



CHAPTER 7
LATERAL-DIRECTIONAL FLIGHT TESTS

7.1 INTRODUCTION

7.1 With lateral-directional theory as a background, it is possible to look at the flight test techniques used to investigate the actual lateral-directional static stability of an aircraft.

Basically, the lateral-directional characteristics of an aircraft are determined by two different flight tests; the Sideslip Test and the Aileron Roll Test. The tests do not measure lateral and directional characteristics independently. Rather, each test yields information concerning both the lateral and the directional characteristics of the aircraft.

In this chapter, both the Sideslip Test and the Aileron Roll Test will be covered. As each test is discussed, the required results, as determined by MIL-F-8785 will also be covered. Finally, an example test mission in the T-33 aircraft will be discussed.

7.2 SIDESLIP FLIGHT TEST TECHNIQUE

7.2 The purpose of the sideslip test is to investigate the static lateral and directional stability characteristics of a particular aircraft in each of several configurations. Since the static lateral and directional stabilities are functions of Mach number, angle of attack, elasticity and configuration, it is important to check the aircraft in various configurations at several altitude-airspeed combinations. In so doing, the sense (+) of the lateral and directional stabilities and the characteristics of the side force can be determined throughout the flight envelope.

All equations relating to the static directional stability of an aircraft were developed under the assumption that the aircraft was in a "steady straight sideslip." This is the maneuver used in the Sideslip Test. To develop a ground for discussion, it is appropriate to discuss the basic mechanics the pilot must perform to establish a "steady straight sideslip." First, the aircraft is trimmed at the desired altitude-airspeed combination. The rudder is then depressed and an increment of sideslip is developed. In order to maintain "straight" or constant heading flight, it will then be necessary for the pilot to bank the aircraft in a direction opposite that of the applied rudder. The aircraft is then stabilized in this condition. Thus, the pilot establishes a "steady straight sideslip." To understand the forces and moments at play in this condition, consider figure 7.1. In this figure the aircraft is in a steady sideslip. Thus, the moment created by the rudder, η_{δ_r}, must equal the moment created by the aerodynamic forces acting on the aircraft, η_β. It can be seen, however, that in this condition the side forces are unbalanced and that this will cause an acceleration. The force, F_{y_β}, will always be greater than $F_{y_{\delta_r}}$. Thus, in the case depicted, the aircraft will accelerate, or turn, to the right. In order to stop this turn, it is necessary to bank the aircraft; in this case to the left (figure 7.2). The bank allows a component of aircraft weight, $W\sin\phi$, to act in the y direction and thus balance the previously

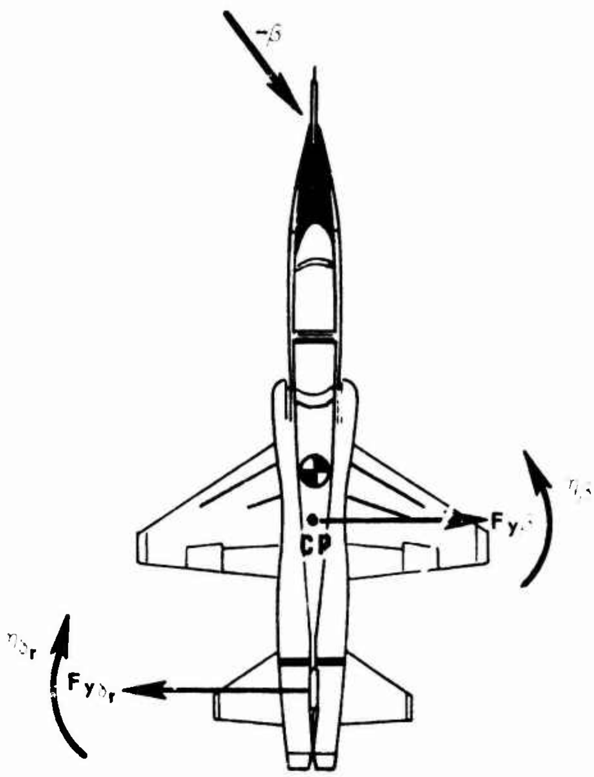


FIGURE 7.1 STEADY SIDESLIP

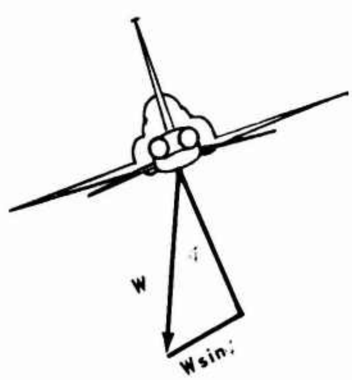


FIGURE 7.2 STEADY STRAIGHT SIDESLIP

unbalanced side forces. Thus, the pilot establishes a "straight sideslip." By holding this condition constant with respect to time, or varying it so slowly in a continuously stabilized condition that rate effects are negligible, he establishes a "steady straight sideslip" - the condition that was used to derive the flight test relationships in static directional stability theory.

By simply establishing a steady straight sideslip, the side force characteristics of the aircraft can be investigated. MIL-F-8785, paragraph 3.3.6.2 states that "an increase in right bank angle must accompany an increase in right sideslip," where right sideslip is defined by the incident airflow approaching from the right side of the plane of symmetry.

One property of basic importance in the sideslip test is the directional stiffness of an aircraft or its static directional stability. To review, the static directional stability of an aircraft is defined by the initial tendency of the aircraft to return to or depart from its equilibrium angle of sideslip when disturbed from the equilibrium condition. In order to determine if the aircraft possesses static directional stability, it is necessary to determine how the yawing moments change as the sideslip angle is changed. Thus, the slope of a line in a plot such as figure 7.3 is of interest. For positive directional stability a plot of Cn_β must have a positive slope.

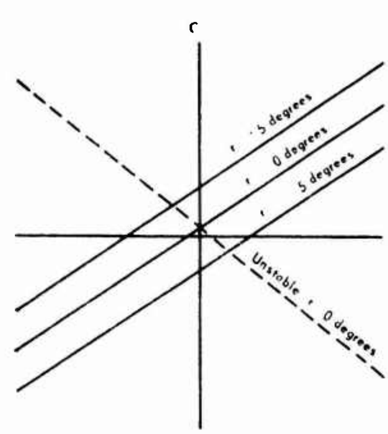


Figure 7.3 WIND TUNNEL RESULTS OF YAWING MOMENT COEFFICIENT vs SIDESLIP ANGLE

Plots like those presented in figure 7.3 would be obtained from wind tunnel data. The aircraft model would be placed at various angles of sideslip with various angles of rudder deflection, and the unbalanced moments would be measured. However, it is impossible to determine from flight tests the unbalanced moments at varying angles of sideslip. It was shown in static directional theory, however, that the amount of rudder deflection required to fly in a steady straight sideslip is considered to be an indication of the amount of yawing moment present tending to return the aircraft to or remove it from its original trimmed angle of sideslip. Thus, in order to determine the sign of the rudder fixed static directional stability, C_{n_β}, a plot is made of rudder deflection required versus sideslip angle.

The apparent stability parameter, $\partial\delta_r/\partial\beta$, for a directionally stable aircraft is shown in figure 7.4. For a stable aircraft, this plot has a negative slope.

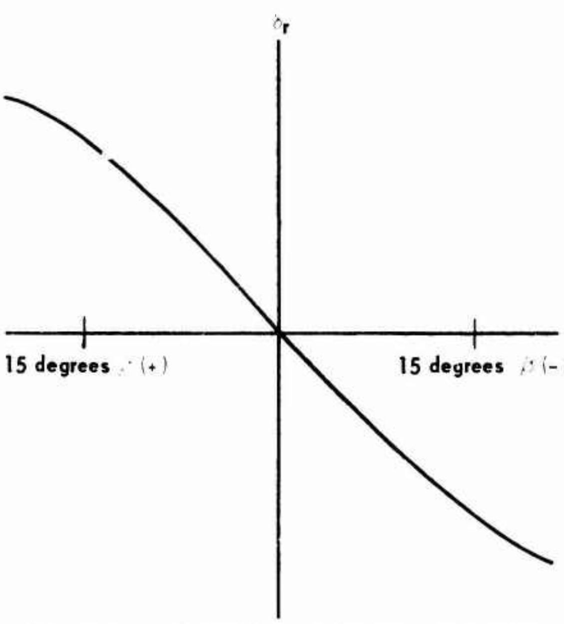


FIGURE 7.4 RUDDER DEFLECTION vs SIDESLIP

MIL-F-8785, paragraph 3.3.6.1 requires that right rudder pedal deflection ($+\delta_r$) accompany left sideslips ($-\beta$). Further, for angles of sideslip between ± 15 degrees, a plot of $\partial\delta_r/\partial\beta$ should be essentially linear. For larger sideslip angles, an increase in β must require an increase in δ_r. In other words, the slope of $\partial\delta_r/\partial\beta$ cannot go to zero.

It should be remembered that drastic changes occur in the transonic and supersonic speed regions. In the transonic region where the flight controls are most effective, a small δ_r may give a large β and thus $\partial\delta_r/\partial\beta$ may appear less stable. However, as speed increases, control surface effectiveness decreases, and $\partial\delta_r/\partial\beta$ will increase in slope. This apparent change in C_{n_β} is due solely to a change in control surface effectiveness and can give an entirely erroneous indication of the magnitude of the static directional stability if not taken into account.

It is now necessary to investigate the control free stability of an aircraft. As discussed in the theory of static directional stability, a plot of rudder force required versus sideslip, $\partial F_r/\partial\beta$, is an indication of the rudder-free static directional stability of an aircraft. It was shown that a plot of $\partial F_r/\partial\beta$ must have a negative slope for positive rudder-free static directional stability. MIL-F-8785 paragraph 3.3.6.1 requires that a plot of $\partial F_r/\partial\beta$ be essentially linear between ± 10 degrees of β from the trim condition. However, at greater angles of sideslip, the rudder forces may lighten but may never go to zero, or overbalance. These requirements are depicted in figure 7.5.

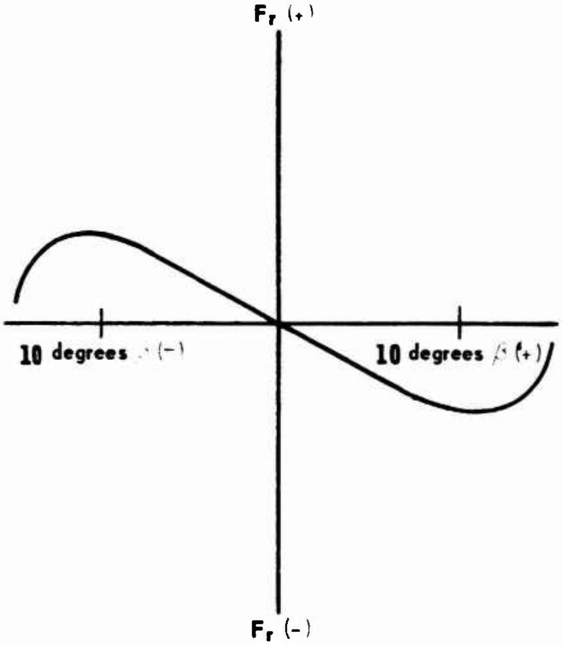


FIGURE 7.5 CONTROL FREE SIDESLIP DATA

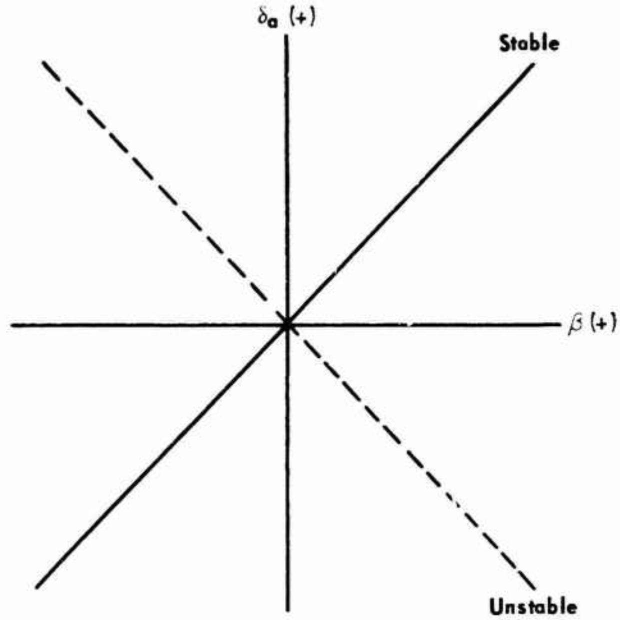


FIGURE 7.6 CONTROL FIXED SIDESLIP DATA

The control force information in figure 7.5 is acceptable in accordance with MIL-F-8785 as long as the algebraic sign of F_r/β is negative. It can be seen that at very large sideslip angles, the slope $\partial F_r/\partial\beta$ may be positive. This is acceptable as long as the rudder force required does not go to zero.

Static lateral characteristics are also investigated during the sideslip test. It was shown in the theory of static lateral stability that $\partial\delta_a/\partial\beta$ may be taken as an indication of the control-fixed dihedral effect of an aircraft, $C_{\ell_{\beta}(\text{Fixed})}$. For stable dihedral effect, it was shown that a plot of $\partial\delta_a/\partial\beta$ must have a positive slope. MIL-F-8785 specifies that right aileron control deflection shall accompany right sideslips and left aileron control shall accompany left sideslips. A plot of $\partial\delta_a/\partial\beta$ for stable dihedral effect is presented in figure 7.6.

MIL-F-8785, paragraph 3.3.6.3b, limits the amount of stable dihedral effect an aircraft will exhibit by specifying that no more than 75 percent of full aileron cockpit control deflection will be required in any of the mandatory sideslip tests required by paragraph 3.3.6. MIL-F-8785 paragraph 3.3.6.3b limits the amount of dihedral effect an aircraft may have. It states that no more than [illegible] pounds of aileron stick force or 20 pounds of aileron wheel force is allowed for sideslips which may be experienced in operational employment.

The theoretical discussion of control free dihedral effect revealed that $\partial F_a/\partial\beta$ will give an indication of $C_{\ell_{\beta}(\text{Free})}$, and that for stable dihedral effect $\partial F_a/\partial\beta$ is positive. Refer to figure 7.7. MIL-F-8785 paragraph 3.3.6.3 states that the left aileron force should be required for left sideslips and that a plot of $\partial F_a/\partial\beta$ should be essentially linear for all of the mandatory sideslips tested.

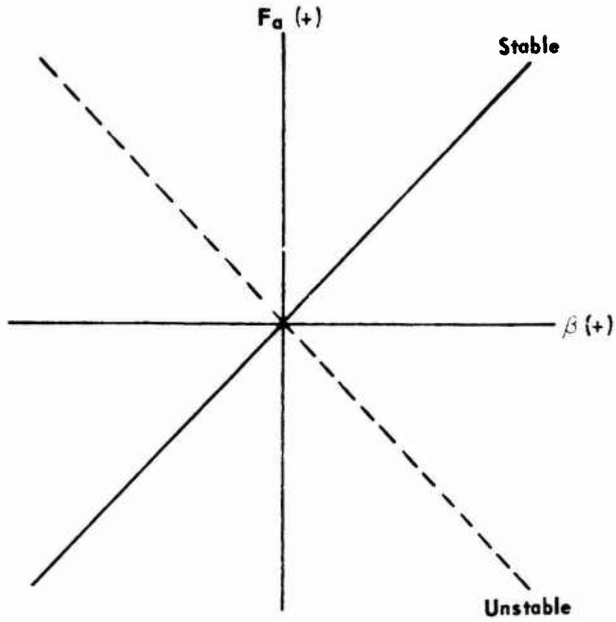


FIGURE 7.7 CONTROL FREE SIDESLIP DATA

MIL-F-8785, paragraph 3.3.6.3a does permit an aircraft to exhibit negative dihedral effect in wave-off conditions as long as no more than 50 percent of available roll control or 10 pounds of aileron control force is required in the negative dihedral direction.

A longitudinal trim change will most likely occur when the aircraft is sideslipped. MIL-F-8785 paragraph 3.2.3.7, places definite limits on the allowable magnitude of this trim change. It is prefered that an increasing pull force accompany an increase in sideslip angle and that the magnitude and direction of the trim change should be similar for both left and right sideslips. The specification also limits the magnitude of the control force accompanying the longitudinal trim change depending on the type of controller in the aircraft (stick or wheel). A plot of elevator force versus sideslip angle that complies with MIL-F-8785 is presented in figure 7.8.

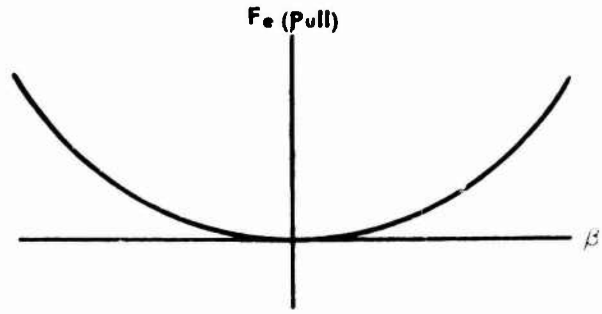


FIGURE 7.8 ELEVATOR FORCE VERSUS SIDESLIP ANGLE

Paragraph 3.3.6 of MIL-F-8785 outlines the sideslip tests that must be performed in an aircraft. The specification requires that sideslips be tested to full rudder pedal deflection, 250 pounds of rudder pedal force, or maximum aileron deflection, whichever occurs first. Often sideslips must be discontinued prior to reaching these limits due to controllability or structural problems.

The following is a complete list of MIL-F-8785 paragraphs that apply to sideslip tests:

3.2.3.7

3.3.6

3.3.6.1

3.3.6.2

3.3.6.3, 3.3.6.3a, 3.3.6.3b

7.3 LIMITATIONS

On student data missions at the Aerospace Research Pilot School it is not possible to conform to all aspects of MIL-F-8785 that concern sideslip tests. Therefore, certain general limitations must be applied to all student sideslip tests.

1. In the cruise configuration, sideslip tests will be conducted at three different altitudes. At each altitude, three different airspeeds will be investigated.

2. In the power approach configuration, sideslip tests will be conducted at 10,000 feet AGL, at the minimum speed for normal final approach.

3. Sideslip tests will be conducted within the limitations outlined in the appropriate aircraft Flight Manual.

4. Within these limitations, the student test program will be set up to investigate all of the requirements concerning sideslip tests that are outlined in the applicable paragraphs of MIL-F-8785.

5. To void excessive data reduction, all results will be plotted against V_i.

Sample data plots of sideslip test results are presented in figures 7.9 and 7.10.

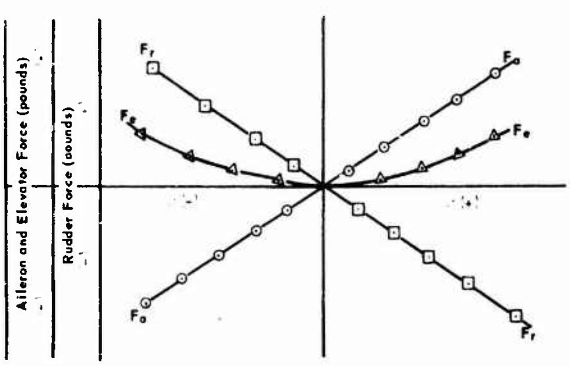


FIGURE 7.9 STEADY STRAIGHT SIDESLIP CHARACTERISTICS CONTROL FORCES VERSUS SIDESLIP

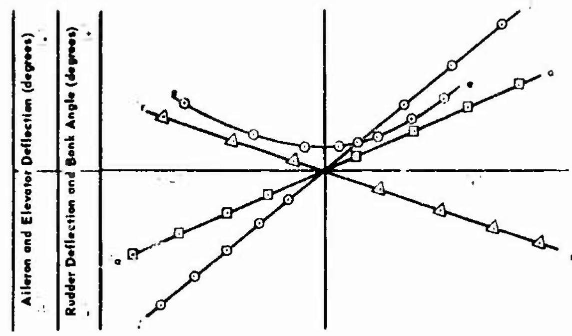


FIGURE 7.10 STEADY STRAIGHT SIDESLIP CHARACTERISTICS CONTROL DEFLECTIONS AND BANK ANGLE VERSUS SIDESLIP

7.4 AILERON ROLL FLIGHT TEST TECHNIQUE

Generally, the aileron roll flight test technique is used to determine the rolling performance of an aircraft and the yawing moments generated by rolling. Roll coupling is another important aircraft characteristic normally investigated by using the aileron roll flight test technique. The roll coupling aspect of the aileron roll test will not be investigated at the Aerospace Research Pilot School. However, the theoretical aspects of roll coupling will be covered in a later course.

It is necessary to understand the basic mechanics involved in the aileron roll flight test technique. The aircraft is first trimmed in the desired configuration at the desired altitude-airspeed combination. Then, the lateral control is abruptly placed to a particular control deflection (1/4, 1/2, 3/4, or full). Normally, the desired control deflection is obtained by using some mechanical restrictor such as a chain stop. With the lateral control at the desired deflection, the aircraft is rolled through a specified increment of bank. For control deflections less than maximum, the aircraft is normally rolled through 90 degrees of bank. Because of the higher roll rates obtained at full control deflection, it is usually desirable to roll the aircraft through 360 degrees of bank when using maximum lateral control deflection. To facilitate aircraft control when rolling through a bank angle change of 90 degrees, start the roll from a 45-degree bank angle. During the roll, a mechanical recorder, such as an oscillograph, is used to record the following information: aileron position, aileron force, bank angle, sideslip and roll rate. Aileron rolls are normally conducted in both directions to account for roll variations due to engine gyroscopic effects. Aileron rolls are performed with rudders free; with rudders fixed; and are coordinated with $\beta = 0$ throughout roll.

Caution should be exercised in testing a fighter type airplane in rolling maneuvers. The stability of the airplane in pitch and yaw is lower while rolling. The incremental angles of attack and sideslip that are attained in rolling can produce accelerations which are disturbing to the pilot and can also cause critical structural loading. The stability of an airplane in a rolling maneuver is a function of Mach number, roll rate, dynamic pressure, angle of attack, configuration, and control deflections during the maneuver.

The most important design requirement imposed upon ailerons or other lateral control devices is the ability to provide sufficient rolling moment at low speeds to counteract the effects of vertical asymmetric gusts tending to roll the airplane. This means, in effect, that the ailerons must provide a minimum specified roll rate, and a rolling acceleration such that the required rate of roll can be obtained within a specified time, even under loading conditions that result in the maximum rolling moment of inertia (e.g., full tip tanks). The steady roll rate and the minimum time required to reach a particular change in bank angle are the two parameters presently used to indicate rolling capability. Pilot opinion surveys reveal that time to roll a specified number of degrees provides the best overall measure of rolling performance.

The minimum rolling performance required of an aircraft is outlined in MIL-F-8785, table VI. This rolling performance is expressed as a function of time to reach a specified bank angle. Table VI is supplemented further by roll performance required of Class IV airplanes in various flight phases. The specific requirements for Class IV airplanes are spelled out in MIL-F-8785, paragraphs 3.3.4.1a, 1b, 1c, 1d. Paragraph 3.3.4.2 and table VII of MIL-F-8785 specify the maximum and minimum aileron control forces allowed in meeting the roll requirements of table VI and the supplemental requirements concerning Class IV aircraft. Paragraph 3.3.2.3 specifies the maximum rudder force permitted for coordinating the required rolls.

In addition to examining time required to bank a specified number of degrees and aileron forces, F_a, it is necessary to examine the maximum roll rate, P_{max}, to get a complete picture of the aircraft's rolling performance. Therefore, in any investigation of aircraft rolling performance, the maximum roll rate obtained at maximum lateral control displacement is normally plotted versus airspeed.

Paragraph 3.3.4.3 of MIL-F-8785 states that there should be no objectionable nonlinearities in roll response to small aileron control deflections or forces. To investigate this area, it is necessary to observe the roll response to aileron deflections less than maximum - such as 1/4 and 1/2 aileron deflections.

MIL-F-8785, paragraph 3.3.4.5 states that it should be possible to raise a wing by using the rudder pedal alone, and that right rudder pedal force should be required for right rolls. Further, it states that with the aileron cockpit control free, it should be possible to produce a roll rate of 3 degrees per second by use of rudder pedal forces of 50 pounds or less. Turn coordination requirements are spelled out in MIL-F-8785, paragraph 3.3.2.4 for steady turning maneuvers.

The other area of prime interest in the aileron roll flight test is the amount of sideslip that is developed in a roll and the phasing of this sideslip with respect to the roll rate. Associated with this characteristic is the roll rate oscillation. These factors influence the pilot's ability to accomplish precise tracking tasks.

The following is a complete list of MIL-F-8785 paragraphs that apply to aileron roll tests:

3.3.2.3

3.3.2.4

3.3.4

3.3.4.1, 3.3.4.1a, 3.3.4.1b, 3.3.4.1c, 3.3.4.1c

3.3.4.3

3.3.4.4

3.3.4.5

3.3.5

3.3.5.1, 3.3.5.1a

3.3.5.2

7.5 SCHOOL TEST LIMITATIONS

The following limitations will apply to all student data missions at the Aerospace Research Pilot School:

1. In the cruise configuration, aileron roll tests will be conducted at three different altitudes. At each altitude, three different airspeeds will be investigated.

2. In the power approach configuration, aileron roll tests will be conducted at final approach speed at 10,000 feet AGL.

3. Aileron roll tests will be conducted within the limitations outlined in the appropriate aircraft Flight Manual.

4. Within these limitations, the student test program will be set up to investigate all of the requirements concerning aileron roll tests that are outlined in the applicable paragraphs of MIL-F-8785(ASG).

5. To avoid excessive data reduction, all results will be plotted against V_i. Sample plots of aileron roll data are presented in figure 7.11.

7.6 DEMONSTRATION MISSION

To unify all that has been said concerning the sideslip and aileron roll flight test techniques, a complete description of an example demonstration mission in the T-33 will be presented.

1. After engine start, and with aileron boost on, visually position the ailerons to approximately 1/4, 1/2 and 3/4 deflections. Mark each position on the instrument panel with masking tape. Prior to leaving the ramp, a ground shot will be taken to ascertain proper functioning of the force and control deflector indicator. The ground shot will consist of a con-

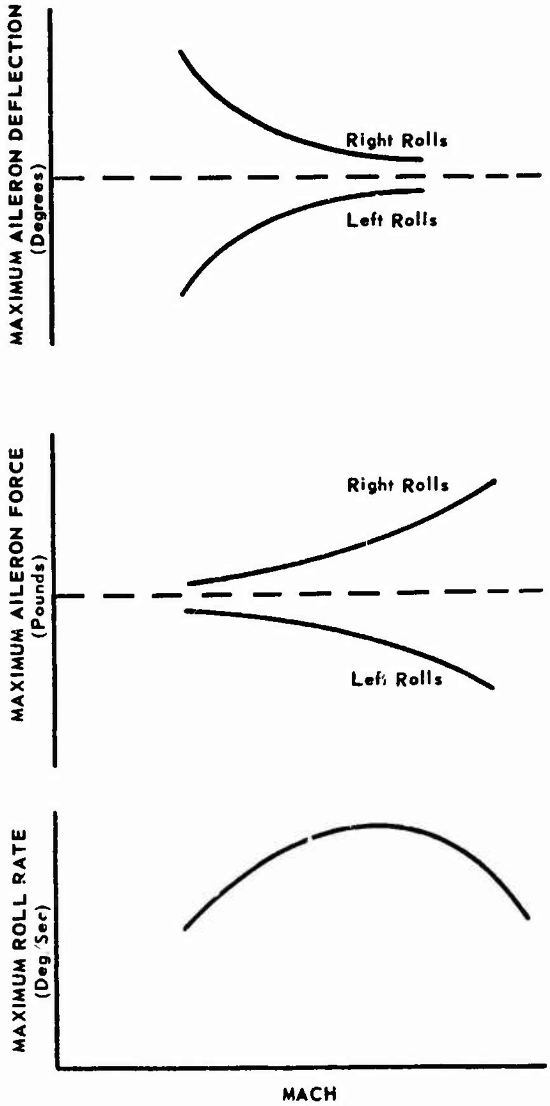


FIGURE 7.11 MAXIMUM AILERON ROLL TEST RESULTS ONE ALTITUDE

tinuous recording of all control deflections and forces from neutral to full positive, to full negative, and then back to neutral. This will help eliminate confusion in reading the film and the oscillograph paper. While taxiing, the pilot should practice making 1/4, 1/2, 3/4, and full aileron deflections.

2. Climb to 20,000 feet and trim the aircraft in the cruise configuration at 250 KIAS using the back side trim technique. Obtain a photo panel shot with the aircraft trimmed in this condition.

3. If a yaw string is available and it is not centered during the trim shot, align it with the longitudinal axis of the aircraft and obtain a photo panel shot. If a sideslip indicator is available for inflight use, this zero sideslip shot can be used to calibrate it. The photo panel shot with zero sideslip should be examined on the ground to ascertain that the zero sideslip indication, as obtained from the calibration book, is correct.

4. During the sideslip test, altitude will be lost if trim power and 250 KIAS are maintained. Therefore, this test will be conducted at 20,000 ±1,000 feet. After the trim shot at 20,000 feet has been obtained, note trim power and climb to 21,000 feet.

5. The instructor pilot will demonstrate the stabilized sideslip flight test technique. Starting from a zero sideslip condition, establish a steady straight sideslip of approximately two degrees while holding trim airspeed. This may best be done by applying a small amount of rudder and then coming in with just enough bank to hold a constant heading. A point on the outside horizon will provide the pilot with the most accurate means of holding a constant heading. The needle of the turn and slip indicator may also be an aid in holding constant heading, zero yawing velocity flight.

Constant trim velocity should be maintained as the sideslip is increased in approximately 2 degree increments until reaching the maximum sideslip obtainable (if no other restriction applies), or until a dangerous flight condition is anticipated. A photopanel shot will be taken at each stabilized sideslip condition. Sideslips in the T-33 will be discontinued at rudder buffet, or at plus or minus 14 degrees of sideslip in the cruise configuration to prevent inadvertent tumbling and possible structural damage. The T-33 is restricted from full rudder deflection sideslips. If no sideslip gauge is available in the cockpit, the following guide may be used: At approximately 12 degrees of sideslip, the standard airspeed indicator will jump approximately 5 knots. The pilot may continue the sideslip investigation approximately two degrees past this point if no other adverse indications are noted. If an airspeed calibration is available for the boom airspeed system at various angles of sideslip, this correction should be used while attempting to hold a constant airspeed. If such a calibration is not available, an indication of the magnitude of the position error due to sideslip can be obtained by placing the aircraft in a sideslip and rapidly coming back to zero sideslip while noting the magnitude of the airspeed change. This correction can be ignored if this change is only one or two knots. It should be immediately apparent whether back or forward stick is necessary to hold a constant airspeed as the sideslip is increased, and thus the correct control movement should be anticipated as the sideslip is increased. Throughout the test, lead all inputs with the rudder. This technique, coupled with slow, deliberate control inputs, will help keep Dutch roll at a minimum. If a Dutch roll should develop, stop it with the rudder; aileron inputs will only reinforce the motion. After the maximum sideslip point is reached, the aircraft should be smoothly returned to trim and then similar sideslip points should be made in the opposite direction. Smoothness in this test, as in all tests, is imperative in order to get good stabilized points quickly. Anticipation of correct control movement is a great aid in establishing good test points quickly. A record should be kept of beginning and final photo correlation numbers as well as a list of any unreliable points.

6. The pilot will practice the stabilized sideslip flight test technique. Maintain altitude at 20,000 $\pm$1,000 feet.

7. The instructor pilot will demonstrate the slowly varying sideslip method. This is an alternate method of obtaining sideslip information. Starting from zero sideslip, continuously increase the sideslip angle at not more than one degree per second while maintaining heading and velocity. A continuous photo record is taken out to the maximum sideslip angle. This method is considerably more difficult to fly properly than the stabilized sideslip method.

8. The pilot will practice the slowly varying sideslip method. Maintain 20,000 ±1,000 feet.

9. At the completion of the sideslip practice, the pilot will return the aircraft to 21,000 feet and 250 KIAS. The instructor pilot will demonstrate the aileron roll flight test technique. Using trim power and holding 250 KIAS, roll the aircraft into a 45 degree bank. Stabilize the aircraft in this condition, holding the rudder pedals fixed to hold trim sideslip. The recording trigger is depressed prior to starting a roll to permit some leader on the oscillograph paper. The stick is rapidly moved to 1/4 aileron deflection, and the aircraft is rolled to 45 degrees of bank in the opposite direction. This control input should be a "step input." The pilot may avoid stick "bobble" by using both hands on the stick. The recording trigger is held depressed throughout the roll. The airspeed should be held as close as possible to the aim V_i throughout the roll. When the roll is complete, rapidly re-establish a 45-degree bank at 250 KIAS. This will prevent needless altitude loss and airspeed excursion. This procedure is repeated in the opposite direction. When the aircraft has been rolled in both directions at 1/4 aileron deflection, repeat the procedure at 1/2 aileron deflection. The full deflection aileron roll is started from wings level flight at 250 KIAS. Apply full aileron deflection in the desired direction of roll. In order to determine exactly how much aileron force it required to hold full aileron deflection, it will be necessary to slowly relax the control force prior to completing 360° of roll. Once on the ground, the oscillograph trace will allow the pilot to match a control force against the point where the aileron deflection first starts to decrease. The full deflection aileron roll will be repeated in the opposite direction. For the sake of demonstration, the instructor pilot may roll in only one direction for each control deflection tested. The instructor will then demonstrate a step aileron input that will accomplish a 45°-45° roll in approximately 6 seconds.

10. The pilot will practice the aileron roll test with rudders fixed. When complete, he will repeat the full deflection roll with rudders free. He will then practice the step aileron input needed to get a 45°-45° roll in approximately 6 seconds.

11. The pilot will descend to 10,000 feet pressure altitude in an area that will allow at least 5,000 feet terrain clearance. The aircraft will be placed in the power approach configuration, i.e., gear down, full flaps, speed brakes up. Obtain a trim shot at 10,000 feet and 120 knots plus fuel using the backside trim technique. Obtain a photopanel shot with the aircraft trimmed in this condition. Also, obtain a photopanel shot in a zero sideslip condition.

12. This test will be conducted at 10,000 ±1,000 feet. After the 10,000 foot trim shot has been obtained, note trim

power and climb to 11,000 feet.

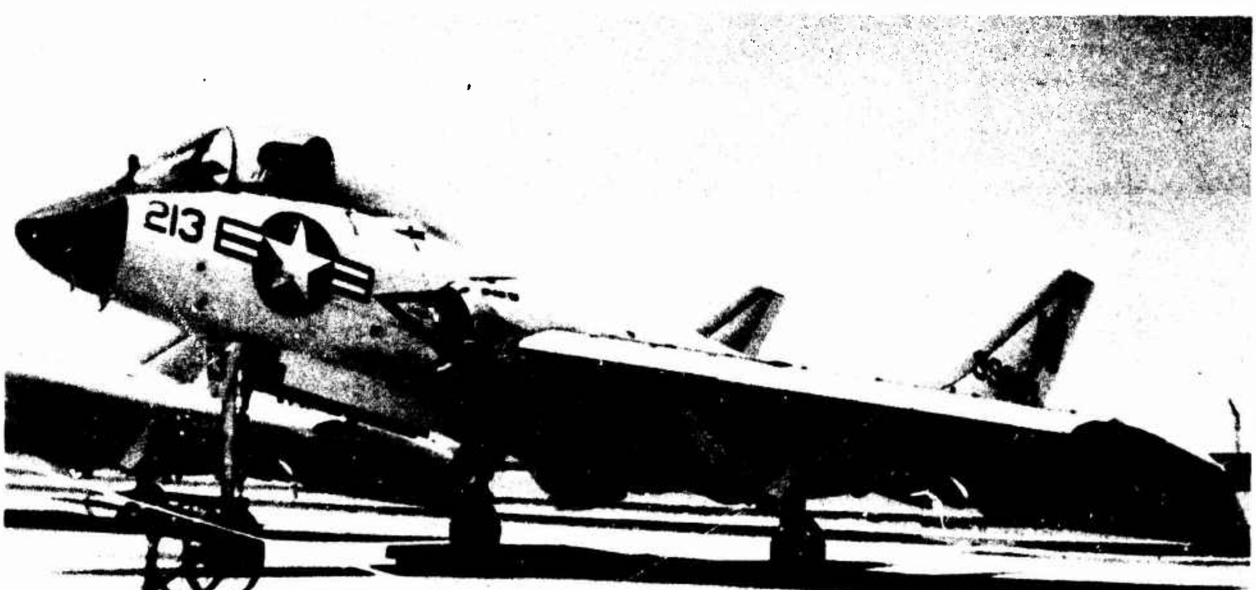
13. The instructor pilot will demonstrate the stabilized sideslip flight test technique in the power approach configuration. Because of the low "q", rudder forces will be very light and care should be exercised to avoid overcontrolling. Sideslips in the T-33 will be discontinued at rudder buffet or at plus or minus 10 degrees of sideslip in the power approach configuration to prevent inadvertent tumbling. Care should be exercised in returning from maximum sideslip to the trim condition. Climb when necessary in order to remain within the allowable altitude band ($\pm$1,000 feet).

14. The pilot will practice the stabilized sideslip flight test technique in the power approach configuration.

15. The instructor pilot will demonstrate the aileron roll flight test technique in the power approach configuration. During this low "q" condition, considerable sideslip will develop. Recover from the roll using aileron only. Overcontrolling or putting in incorrect rudder inputs can create a hazardous situation. Therefore, 360-degree rolls will not be accomplished in the power approach configuration.

16. The pilot will practice the stabilized sideslip flight test technique in the power approach configuration with the rudders fixed. When this is completed, he will repeat the 1/2 aileron deflection point with rudders free. Recovery will be made with rudders free.

17. Landing will be made from a simulated flameout pattern set up by the instructor pilot.



CHAPTER 8

ENGINE-OUT OPERATION

● 8.1 INTRODUCTION

The problems associated with an engine failure in a multi-engine aircraft may be classified into two types; control problems and performance problems. The severity of one may greatly overshadow the other in certain aircraft; but in general, the pilot is confronted with a generous portion of both.

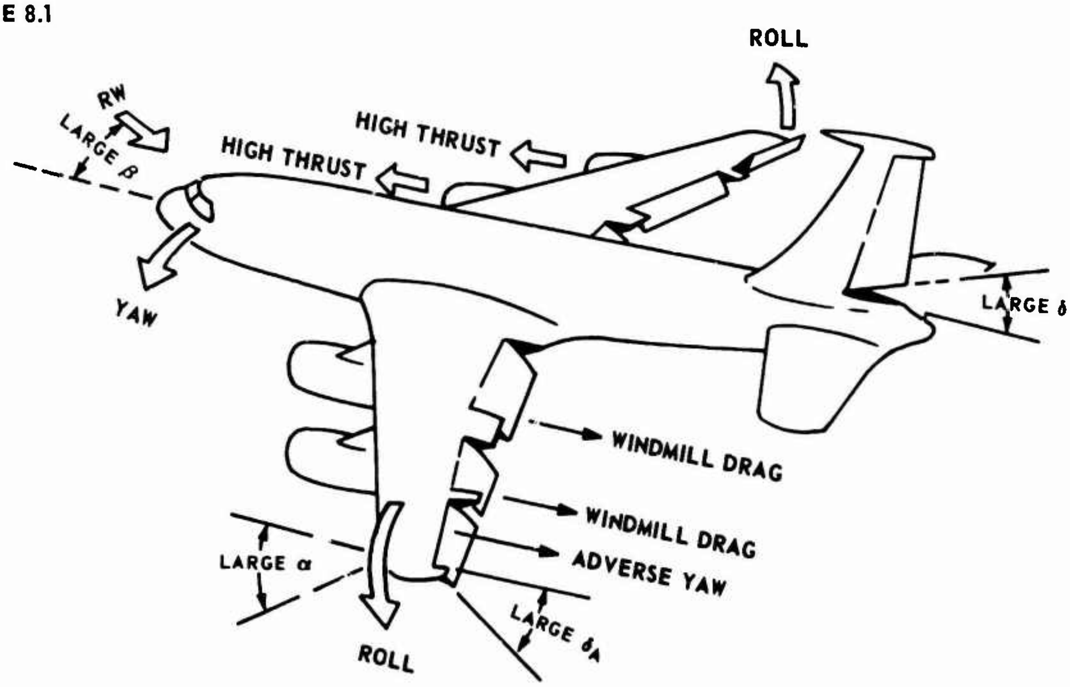
● 8.2 THE CONTROL PROBLEM

The control problem may be simply stated - the pilot must be able to achieve and maintain straight unaccelerated flight following the loss of an engine. Thus the engine-out control problem can be divided into cases; the non-steady state dynamic case and the steady state equilibrium case. The dynamic case begins when an engine fails and terminates when the equilibrium case has been achieved.

When a pilot intentionally shuts down an engine in an aircraft with adequate control authority to maintain equilibrium, the dynamic case is usually not severe and the transients encountered are mild. If, however, an engine fails suddenly on takeoff, or the pilot makes a sudden application of go-around power to asymmetric engines, a potentially divergent rolling and yawing motion can ensue.

These hazardous dynamic situations are caused by a rapid sequence of events, as illustrated

FIGURE 8.1



in the following hypothetical sequence:

1. The aircraft is in a critical flight phase such as takeoff or go-around when a large yawing moment due to asymmetric thrust appears very suddenly. The aircraft yaws rapidly through a large angle.

2. The pilot allows a large sideslip angle to develop because of the high yaw rate and the surprise factor. A rolling moment into the bad engines is generated by the dihedral effect. This rolling moment is augmented by wing blanking on swept-wing configurations and by asymmetric slipstream effects on propeller aircraft.

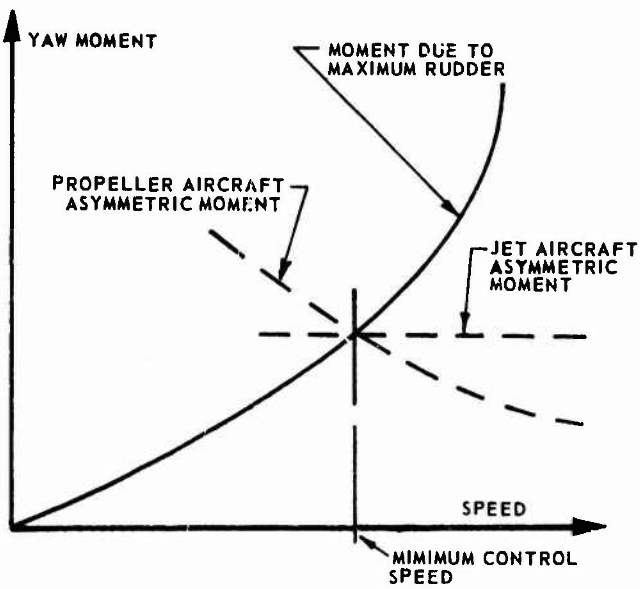
3. As the angular momentum builds up in roll and yaw, larger compensating moments, over and above the steady state requirements, are required to arrest the motion. Large control deflections are required because of the reduced control effectiveness at slow speed, and adverse yaw adds to the forcing moment. If full control is insufficient to achieve equilibrium, a power reduction on the good engines will be required.

4. But a power reduction aggravates an already critical performance problem. Speed is difficult to maintain because of decreased thrust and increased drag. If the down-going wing, which is at a high angle of attack because of the slow speed and the rolling velocity, is allowed to reach stall, the dynamic case may terminate without ever reaching equilibrium.

The severity of such responses is difficult to predict by theoretical analysis, and flight test of critical situations is required to establish safe flight boundaries. Slow speed restrictions due to decreased control effectiveness are most common, although others may exist in the supersonic range due to reduced stability. The dynamic case boundaries are usually (although not necessarily) more restrictive than those due to the equilibrium case.

For every given set of asymmetric conditions there is a speed below which aerodynamic control is insufficient to maintain the equilibrium case. This is called the minimum control speed.

FIGURE 8.2



Obviously, this minimum speed will vary with the prevailing conditions. Not so obvious, however, is the fact that for a given condition the equilibrium case can be maintained with different combinations of bank angle, sideslip angle, and rudder deflection and that the minimum speed will vary according to the combination used.

Engine-out definitions and terminology are not standard throughout the aviation industry and in any discussion it must be clearly understood what the conditions are, and that everyone is talking about the same thing. Several more or less standard definitions are discussed below.

Minimum Control Speed:

It is possible that there will be no minimum control speed for a multi-engine aircraft because it can be controlled up to aerodynamic stall. This is the desired situation. MIL-F-8785 (para 3.3.9.2) specifies that straight flight must be possible during takeoff at any speed above minimum takeoff speed and further specifies the control forces and deflections that may be used to accomplish this. This establishes the minimum control speed required for every possible gross weight condition. It might therefore seem that a minimum control speed is only of academic interest, but there may be instances where a multi-engine aircraft could meet the specifications at takeoff but be operated at a speed in some operational or approach flight phase which would be lower than minimum takeoff speed and hence the asymmetric thrust minimum control speed must still be determined by flight test.

Ground Minimum Control Speed:

Control of asymmetrically powered multi-engine aircraft on the ground also presents a problem that must be considered. If a pilot loses the most critical engine during takeoff roll, he must decide whether to continue the takeoff or abort. MIL-F-8785 (para 3.3.9.1) specifies that the pilot must be able to maintain a path on the runway that does not deviate more than 30 feet from the original path if he decides to continue the takeoff and is above the refusal speed (based on the shortest runway from which the airplane is designed to operate). If the pilot decides to abort, the directional control requirements are still specified but the pilot is allowed to use additional controls such as nosewheel steering and differential braking. Flight (ground) testing is required to show compliance with the specification so a ground minimum control speed can be determined. This is defined as the lowest speed at which directional control can be maintained on the ground when the most critical engine fails during takeoff roll.

8.3 THE PERFORMANCE PROBLEM

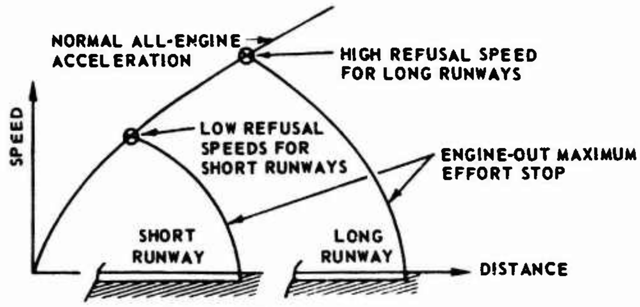
Reduced climb performance, service ceiling and range capability accompany an engine failure as a natural result of decreased thrust and increased drag. The effect of engine failure on takeoff performance, however, is a complex subject requiring additional definitions and operational techniques.

Takeoff Performance:

The basic requirement is simple; at every instant throughout the takeoff roll the pilot must have an acceptable course of action available to him in the event of engine failure. During the first part of the roll, this action will be to abort the takeoff and stop. Beyond a certain point the action will be to continue the takeoff with the engine failed. The dividing point between these courses of action is a function of aircraft performance.

Consider an aircraft at a particular configuration and gross weight starting its takeoff roll on a given day. For a given runway length there is a maximum speed to which it can accelerate on all engines, lose the critical engine and then just complete a maximum effort stop at the far end of the runway. This speed, called the refusal speed, is relatively high for long runways and relatively low for short ones.

FIGURE 8.3

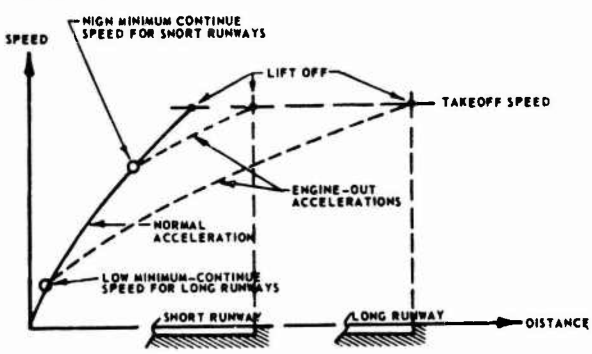


REFUSAL SPEED

REFUSAL SPEED is thus defined as the <u>maximum</u> speed to which the aircraft can make a normal takeoff acceleration, lose the critical engine at that speed and then stop on the remaining runway. Stopping technique and devices to be used must be specified.

Now consider the same aircraft making the same takeoff under identical conditions. For a given runway length there is a minimum speed to which it can accelerate on all engines, lose the critical engine at that speed and then continue the takeoff with the engine failed, getting airborne just at the far end of the runway. This speed (a "minimum-continue" speed) varies with runway length in a manner opposite that of refusal speed, i.e., it is relatively low for long runways.

FIGURE 8.4



The gap between the minimum-continue speed and the Refusal Speed reflects the size of the safety margin provided by a given runway for the particular conditions.

FIGURE 8.5

SAFE TAKEOFF

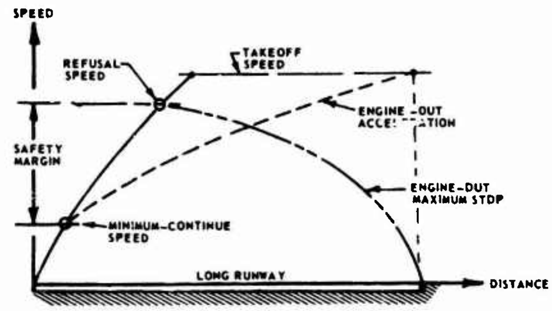


FIGURE 8.6

UNSAFE TAKEOFF

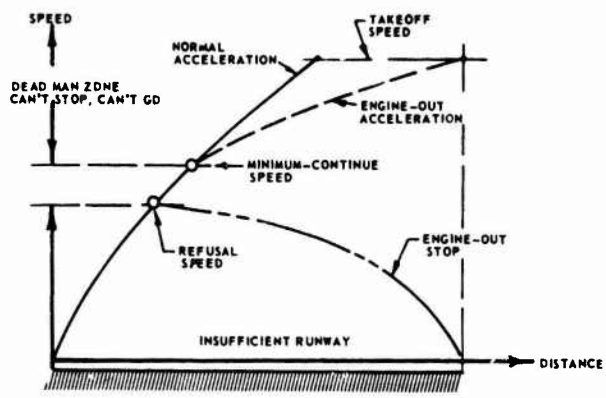
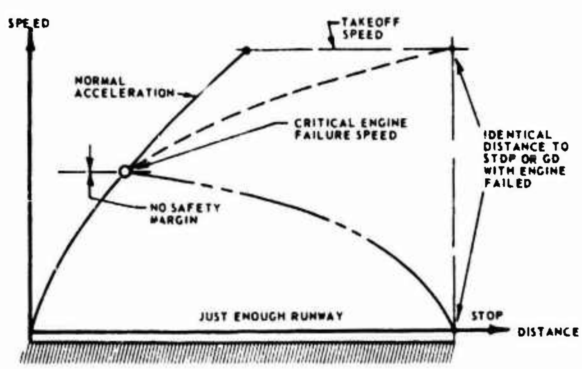


FIGURE 8.7

CRITICAL FIELD LENGTH



Critical Engine Failure Speed:

If the critical engine fails at this speed, the distance required to complete the takeoff is identical to the distance required to stop. The total runway required to accelerate to this speed, lose an engine, and then stop or go is the Critical Field Length.

Initial Climb Performance:

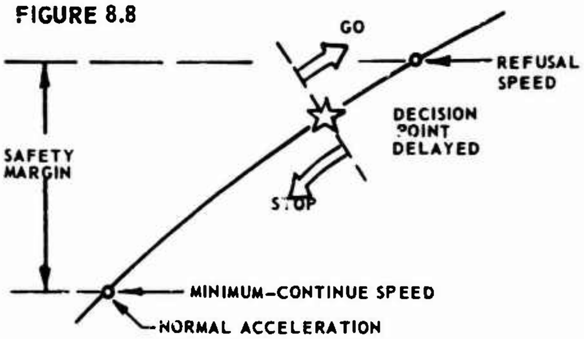
The period between lift-off and attainment of best engine-out climb speed can be very critical. Military multi-engine aircraft are routinely loaded to gross weights that provide as low as 50 feet-per-minute rate of climb with an engine inoperative. Obviously, this level of performance allows little margin for mis-management of attitude or configuration. Flap retraction may have to be accomplished incrementally on a very tight speed schedule to keep sufficient lift for a positive climb gradient without excessive drag. Unexpected characteristics may be encountered in this phase. For example, the additional drag due to doors opening might make it desirable to delay gear retraction until late in the clean-up phase or in another instance, the time available to obtain the clean configuration might be limited by the supply of water injection fluid if dry thrust is insufficient to maintain the climb. Careful flight test exploration of this phase is an obvious requirement.

Decision Speed/Distance:

All the performance discussions above are concerned with what the aircraft will actually do. It still remains for the pilot or operational authority to decide at what particular speed or distance the course of action will change from stop to go in the event of engine failure. This defines the decision point.

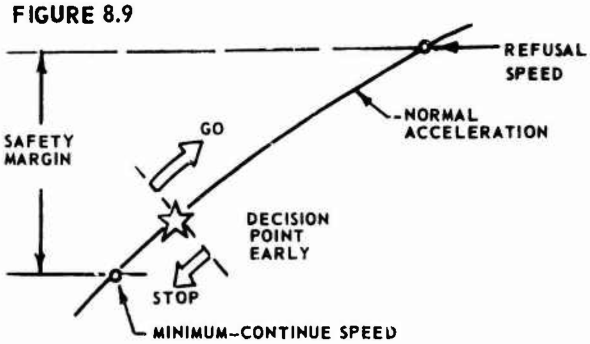
If the initial climb performance is going to be critical on the takeoff in question, the decision point may be near the higher speed end of the safety margin.

FIGURE 8.8



The B-47 illustrates the opposite case. This aircraft has a very poor record for successful aborts and is operated with the decision point relatively early (near the low speed end) of the safety margin.

FIGURE 8.9



Other cases may be decided by the nature of the overrun or the terrain beyond the runway, i.e., is it better to go off the far end almost stopped or almost flying?

8.4 ENGINE-OUT FLIGHT TESTING

Military aircraft are usually designed with relatively low safety margins in order to attain the desired performance - the marginal engine-out climb capability previously mentioned is an example. In fact, during war emergency operation the gross weight may be so high that engine-out operation is not possible at all. Flight tests of these critical phases, on or near the ground, require a high level of crew skill and proficiency; each point must be carefully planned and flown.

Such tests are a normal part of the Category I and II testing of a new aircraft. They also play a vital part in side-by-side evaluations of assault or VSTOL transports where the ability to carry a useful load in and out of a given landing area is frequently limited by engine-out performance. Individual evaluations to determine if an aircraft meets the contractor's guarantees may also hinge on this area of operation.

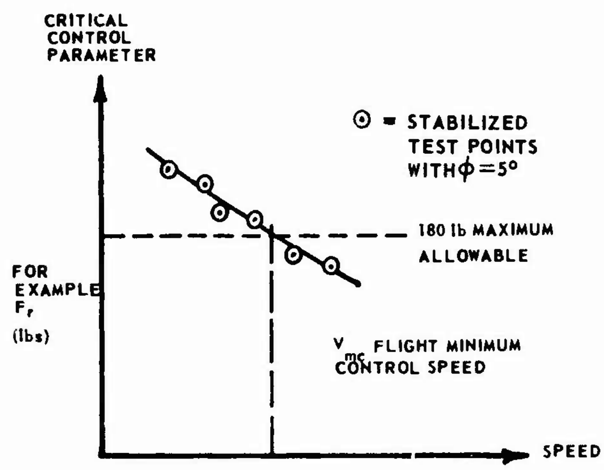
Flight Minimum Control Speed:

It will be shown later in paragraph 8.5 that an aircraft with an engine inoperative can be stabilized in straight (unaccelerated) flight in various combinations of bank angle, sideslip angle, and rudder deflection. For determination of minimum control speed, the maximum bank angle of 5 degrees allowed by MIL-F-8785 will be held constant, and the other two parameters adjusted as necessary to achieve straight flight.

The critical engine is always an outboard engine. For reciprocating aircraft with clockwise rotating propellers (looking forward), the left outboard is critical due to torque. Assuming there is no angular motion of the aircraft to provide gyroscopic couples from rotating engines, left or right is usually not critical on a jet powered aircraft (the distinction may be important for dynamic points, however).

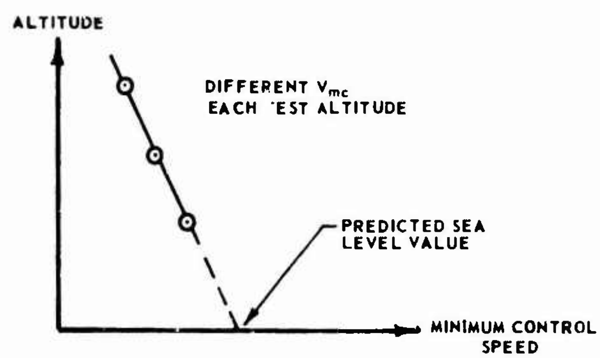
With the aircraft in the specified configuration, and with the critical engine failed, a series of stabilized points are recorded at decreasing speeds. A plot of the critical control parameter (this will most frequently be rudder force or deflection) versus airspeed is made to determine the minimum control speed.

FIGURE 8.10



The minimum control speed usually increases at lower altitude due to increased engine thrust; the test must be accomplished at more than one altitude, including one as low as is safely possible, to provide an extrapolation to sea level.

FIGURE 8.11



Care must be exercised to obtain points that are unaccelerated and well stabilized. The outside visual attitude is primary for maintaining airspeed, bank angle, and zero yaw rate, using the cockpit instruments for cross check. The ball, which in unaccelerated flight will always be at the bottom of the race, is the primary reference used to eliminate accelerations that result from unbalanced forces in the y direction. These lateral translations are difficult to discern visually.

Dynamic Engine Failure:

The military specifications (MIL-F-8785 para 3.3.9.3) require that a pilot be able to avoid dangerous conditions that might result from the sudden loss of an engine during flight. The method to test compliance with this specification is to stabilize with symmetrical power and dynamically fail the most critical engine. After observing a realistic time delay for pilot realization and diagnosis, the pilot arrests the aircraft motion and achieves the equilibrium engine-out condition. Since it obviously requires more control to arrest the motion than to maintain equilibrium, this dynamic situation must be considered in determining the minimum control speed.

Minimum control speed should not be set by any factor other than insufficient control. If the aircraft stalls before reaching the minimum control speed, a statement that "at this gross weight, the aircraft is controllable down to the stall" is preferable to calling the stall speed the "minimum control speed."

Ground Minimum Control Speed:

The ground minimum control speed will differ from the flight value because of:

1. The inability to use sideslip and the restriction on the use of bank angle.
2. Cross wind components.
3. The additional yaw moments produced by the landing gear, which in turn vary with: the landing gear configuration; the amount of steering used; the vertical loads on each gear; and runway condition.

There are two basic test methods, one involving acceleration and the other deceleration. If the aircraft will decelerate with the asymmetric power condition set up (symmetrical pairs of non-critical engines may also be retarded) the "back-in" method may be used. The test is started at a ground speed in excess of the expected minimum and the power condition is set. As the speed decreases, more aerodynamic control deflection is required; the speed where directional control cannot be maintained is the minimum control speed.

Some high performance aircraft accelerate in the test condition and the acceleration method is required. The asymmetric yawing moment is gradually increased (by throttle manipulation) as increasing speed provides more control. The speed where sufficient control is available to hold the full asymmetric power condition is the minimum control speed. This method requires considerable skill and coordination to obtain good results - the aircraft is essentially at minimum control speed throughout the acceleration.

Both of the methods above determine equilibrium control speeds. When sufficient experience has been obtained, sudden engine cuts are performed to determine if dynamic effects are more restrictive.

In-Flight Performance:

Normal performance flight test methods may be used to determine the climb, range, and endurance at altitude with engines inoperative.

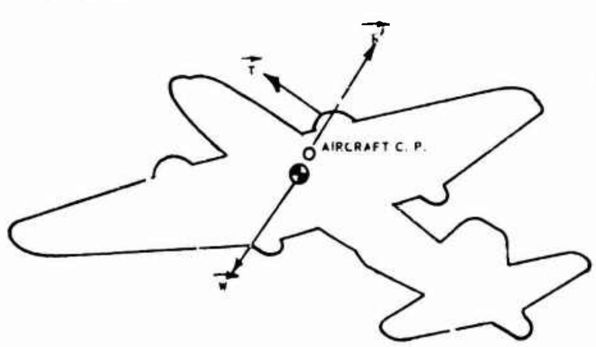
Landing Performance:

Restricted reversing capability and possible higher approach speeds required to maintain minimum safe speeds will affect landing performance. Normal flight test methods are valid, but caution must be exercised in go-around situations.

● 8.5 EFFECT OF BANK ANGLE ON THE EQUILIBRIUM CASE

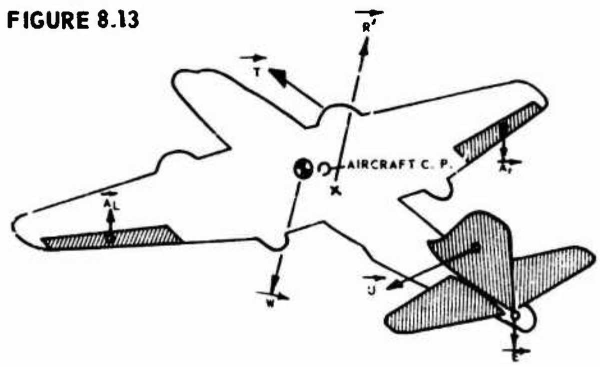
If torque and gyroscopic effects due to rotating engines or propellers are neglected, all the forces acting on an aircraft in flight with an engine inoperative are shown in figure 8.12.

FIGURE 8.12



The vector $\vec{R}$ is the total aerodynamic reaction acting at the aircraft center of pressure. This vector may be thought of as the sum of all the smaller reactions acting on the separate parts of aircraft. For the present discussion it is convenient to handle separately the smaller reactions that act on the ailerons ($\vec{A_r}$ and $\vec{A_L}$), the vertical fin and rudder ($\vec{U}$) and the horizontal stabilizer and elevator ($\vec{E}$), as shown in figure 8.13.

FIGURE 8.13

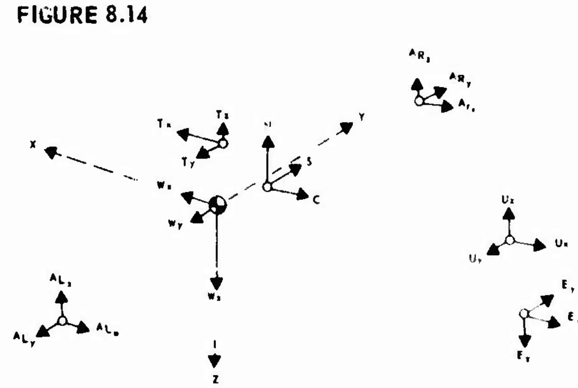


$\vec{R}$ is now the remaining portion of the total aerodynamic reaction, such that:

$$\vec{R} = \vec{R} + \vec{A_R} + \vec{A_L} + \vec{E} + \vec{U}$$

$\vec{R}$ has a point of action that is near, but not necessarily at, the aircraft c.p. Assuming mass to be symmetrically distributed, the weight vector acts through the cg at the origin of the body axis system. Because of the possibility of sideslip, none of the aerodynamic force vectors necessarily pass through a body axis, i.e., they may all produce moments in three directions. When all forces are resolved into components parallel to the body axes, the representation in figure 8.14 is obtained.

FIGURE 8.14



If the restriction of equilibrium (unaccelerated) flight is now imposed, six equations result:

Longitudinal	Lateral-Directional
(1) $F_x = 0$	(4) $F_y = 0$
(2) $F_z = 0$	(5) $L = 0$
(3) $M = 0$	(6) $N = 0$

The longitudinal equations are not critical in the achievement of equilibrium; they are balanced by the usual technique of stabilized points, i.e., variation of pitch angle (θ), angle of attack (α), and elevator deflection (δ_e).

If the forces of figure 8.14 are all moved to the cg and the moments lost by the move are replaced, the six equations can be expanded as shown below.

$$(1)\quad \underbrace{W_x}_{\text{equation is balanced with } \theta} + \underbrace{E_x + U_x + A_{R_x} + A_{L_x}}_{\text{control drag}} + \underbrace{C + T_x}_{\text{other drag}} = 0$$

$$(2)\quad \underbrace{N}_{\text{equation is balanced with } \alpha} + E_z + \underbrace{U_z + A_{R_z} + A_{L_z}}_{\text{very small}} + W_z + T_z = 0$$

$$(3)\quad \underbrace{M/_E}_{\text{equation is balanced with } \delta_e} + M/_T + M/_N + M/_C + \underbrace{M/_U + M/_{A_{R,L}}}_{\text{very small}} = 0$$

The lateral-directional equations are of most interest in achieving equilibrium. The roll equation (5) is usually not critical, although in some cases lack of aileron authority may be the limiting factor.

$$(5)\quad \underbrace{L/_{A_{R,L}}}_{\text{equation is balanced with } \delta_a} + L/_T + L/_U + \underbrace{L/_N + L/_S + L/_E}_{\text{very small}} = 0$$

It now remains to be shown that the side force equation (4) and the yaw equation (6) can be simultaneously balanced using an infinite number of combinations of bank angle (ϕ), sideslip (β), and rudder deflection (δ_r).

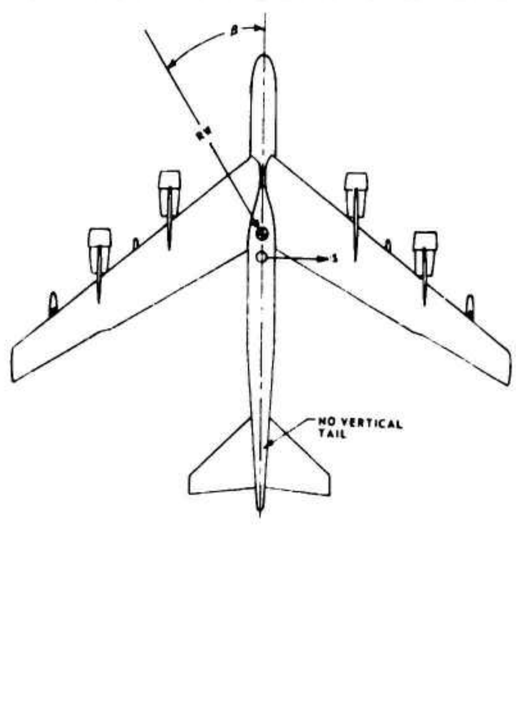
$$(4)\quad \underbrace{W_y}_{W \sin\phi} + U_y + S + \underbrace{E_y + A_{R_y} + A_{L_y}}_{\text{small values combined into } S} = 0$$

$$\boxed{\text{Side Force Equation: } W \sin\phi + U_y + S = 0}$$

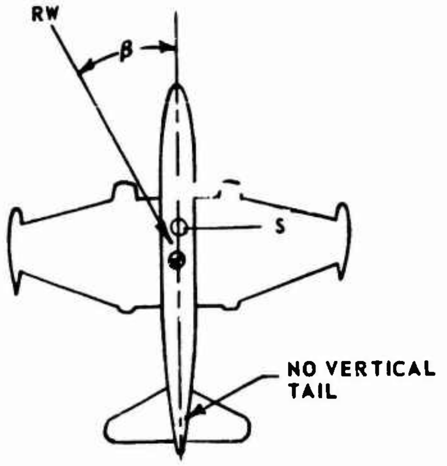
$$(6)\quad \underbrace{N/_T + \underbrace{N/_C + N/_E}_{\text{small}} + \underbrace{N/_{A_{R,L}}}_{\text{adverse yaw}}}_{\text{lumped together as } N_{Forcing}} + N/_U + N/_S = 0$$

$$\boxed{\text{Yaw Equation: } N_{Forcing} + N/_U + N/_S = 0}$$

The point of action of S is related to the directional stability with the vertical tail removed. Certain aircraft, such as B-52 with its long, slab-sided aft fuselage and swept wings, might have some directional stability without the vertical tail installed, in which case S would operate aft of the cg.



Aircraft more generally will be directionally unstable in this condition, and S will operate ahead of the cg.



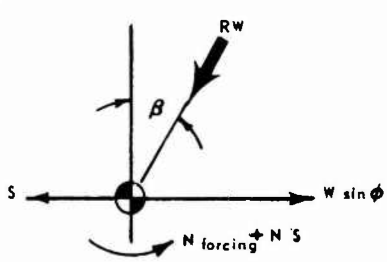
In either case S will have a short arm compared to that of U_Y and the sign of $N/_S$ (which in the discussion below will be considered unstable) or the effect of the other simplifying assumptions will not alter the diagrams below.

Equilibrium with $\delta_r = 0$

β from good engine side

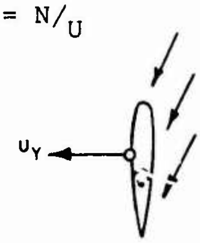
ϕ large

$\delta_r = 0$



$W \sin \phi = U_Y + S$

$N_{Forcing} + N/_S = N/_U$

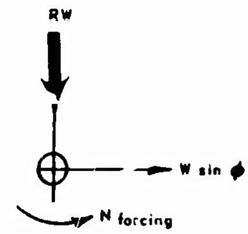


Equilibrium with $\beta = 0$

$\beta = 0$

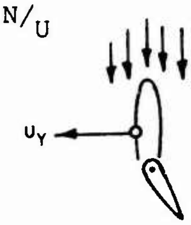
ϕ = reduced

δ_r = moderate



$W \sin \phi = U_Y$

$N_{Forcing} = N/_U$

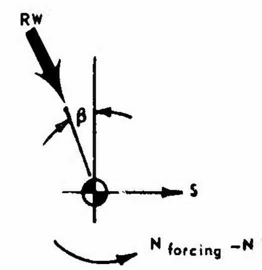


Equilibrium with $\phi = 0$

β from bad engine side

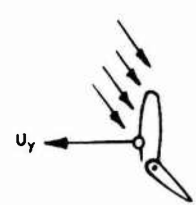
$\phi = 0$

δ_r = large



$U_Y = S$

$N_{Forcing} - N/_S = N/_U$

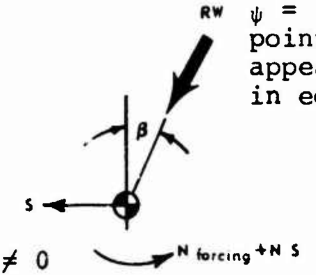


It may be shown by trial and error that the arrangements above are the only ones possible for the conditions specified provided the aircraft is truly following an unaccelerated flight path. For example, if the $\phi = 0$ condition is attempted with β from the good engine side, equilibrium cannot be obtained.

False $\phi = 0$ Point

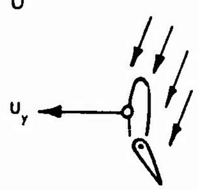
Aircraft is accelerating because of insufficient rudder

Because $\psi = 0$ the point may appear to be in equilibrium



$U_y + S \neq 0$

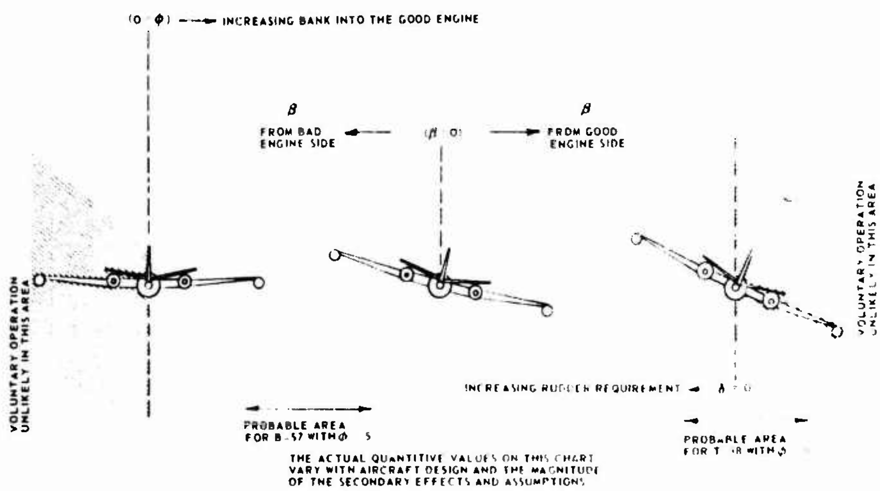
$N_{Forcing} + N/_S = N/_U$



Compared to the true $\phi = 0$ condition, more U_y is required to balance the yaw equation (since $N/_S$ is now added to $N_{Forcing}$) but it is obtained with less δ_r because of the favorable β at the tail. But the side force equation is not balanced ($U_y + S \neq 0$) and the aircraft is actually accelerating to the left. This condition, which can be readily attained in flight if insufficient rudder is used in the $\phi = 0$ condition, is difficult to see visually but can be recognized by a displacement of the ball to the right. If additional right rudder is applied until the ball returns to the bottom of the race, the sideslip will return to the bad engine side.

The relationship between ϕ, β, and δ_r revealed above is summarized in figure 8.15.

FIGURE 8.15



8.6 DEMONSTRATION MISSION

The student will fly an Engine-Out demonstration mission from the rear cockpit of the B-57.

Performance Evaluation - Single-Engine Climb:

1. Clean configuration trimmed for normal climb (300 kt).
2. Single-engine climb to assigned altitude with left engine at idle, right engine at 100%.
3. With time to climb to altitude starting at 5,000 feet and maintaining a constant heading, take stabilized camera readings at:

$\phi = 0°$ $\phi = 5°$	holding all forces constant
$\phi = 5°$	rudder forces only trimmed
$\phi = 5°$	rudder and elevator forces trimmed
$\phi = 5°$	all forces trimmed

Effect of Bank Angle on Yaw Control:

1. Clean configuration stabilized at assigned altitude at a speed slightly above minimum control speed.
2. Holding forces with left engine failed and right engine at 100%, record at trim airspeed:

False $\phi = 0°$ point

$\phi = 0°$

$\beta = 0°$

$F_r = 0$

ϕ into engine idle

Directional Control Test (MIL-F-8785 para 3.3.9.2):

1. The takeoff configuration is trim shot power, the same as required for the previous test.
2. Left engine idle, right engine 100%.
3. Record stabilized points at equal airspeed increments from trim speed to V_{min}. cont. first at $\phi = 0°$ and then repeat the test with $\phi = 5°$ and Sensitive Bank Angle Indicator Operating.

Asymmetric Power Test (MIL-F-8785 para 3.3.9.4):

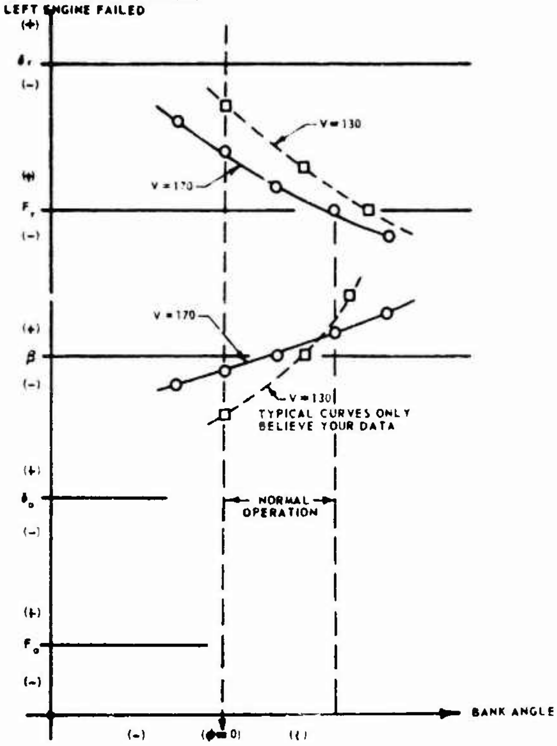
1. Configuration CR
2. Trim to 90 percent noseup
3. Set both engines at 96.5 percent power
4. Airspeed 160 kt
5. Use a sawtooth entry 1,000 feet below the assigned altitude. At the assigned altitude, an engine will be chopped to idle. Recover with no delay using the ailerons, feet on the floor.
6. Record data through recovery (start data 300 feet prior to reaching the assigned altitude).
7. Hand-record ψ_{max} and ϕ_{max} and qualitatively determine if MIL-F-8785 requirements are met.
8. Test. Repeat steps 1-7 using 150 KIAS (or 1.3 $V_{O_{min}}$ whichever is higher)

Trim Evaluation (MIL-F-8785 para 3.6.1.1):

1. Configuration - CR

FIGURE 8.16

EFFECT OF BANK ANGLE
LEFT ENGINE FAILED



2. Trim - as set for trim shot

3. Power: Left engine - As required

   Right engine - Idle

4. Airspeed - Flight Manual max - range single-engine cruise speed, 40,000 pounds gross weight, at assigned altitude

5. Altitude - as assigned

6. Test - with ϕ = 0°

7. Record - holding all forces
   - rudder trimmed; F_r = 0
   - aileron trimmed; F_a = 0
   - all forces trimmed

Dynamic Engine Failure (MIL-F-8785 para 3.3.9.3):

1. Configuration CR

2. Trim - as required

3. Power - 100 percent both engines

4. Airspeed - 140 KIAS

5. Start 1,000 feet below assigned altitude

6. Test - Climb through to the assigned altitude. The IP will chop an engine. Delay 3 seconds and then recover. Test each of the following recovery methods:
   - ailerons first, then rudder
   - rudder first, then ailerons
   - rudder and aileron simultaneously

7. Qualitatively record:
   - ϕ and β at 3 seconds after throttle chop
   - ϕ_{max} and β_{max} during recovery
   - Altitude lost in recovery

8. Photo record data throughout recovery

9. Repeat steps 1-8 @ 130 and 120 KIAS

Single-Engine Go Around Evaluation:

1. Configuration - PA

2. Trim - As required

3. Power - as required for descent at 135 KIAS,

rate of descent 500 feet per minute with both engines operating.

4. Altitude - 8,000 feet

5. Test - Qualitatively evaluate with an engine failed and ϕ - 5° maximum
   - Power required to maintain a 500 foot per minute descent with flaps up.
   - Power required to level off
   - Accelerate with gear up to 160 KIAS and go around.

6. Repeat Test 5 with the flaps down.

Single-Engine Traffic Pattern and Landing:

1. Qualitative analysis of single-engine traffic and landing qualities.

2. Test - Consult the Technical Order for proper single engine traffic and landing procedures (pay particular attention to WARNINGS CAUTIONS and NOTES).

3. Record qualitative comments.

Single-Engine Taxi Evaluation:

1. Record qualitative comments.

8.7 DATA

Data to Be Recorded:

The photopanel will be used for all stabilized points (parameters as prescribed in ARPS Special Instrumentation Requirements). Normally a continuous oscillograph record of dynamic points would be obtained. The student should record any applicable qualitative comments deemed necessary.

Data Presentation:

Plots similar to figures 8.10 and 8.11 will be presented in the report. Discussion of the minimum control speed and any other applicable findings should be included. A plot similar to figure 9.16 should also be prepared to illustrate the effect of bank angle.

Several methods of asymmetric data presentation are presently being developed. They are worthy of note. As aircraft size increases; gross weight and bank angle become important considerations in determining minimum control speeds. To normalize this consideration a plot of $C_L \sin \phi$ versus Thrust Moment is used. Temperature, pressure altitude, and dynamic pressure also effect the minimum control speed. To normalize these effects, (colder or hotter than standard day) a plot has been developed that depicts Thrust Moment versus Dynamic Pressure.

CHAPTER 9

DYNAMIC STABILITY

9.1 PURPOSE

The purpose of the dynamic stability flight test is to investigate an aircraft's primary modes of motion. This investigation will ascertain the acceptability of these modes - frequency and damping being the characteristics of primary importance.

9.2 AIRCRAFT MODES OF MOTION

The characteristic modes of motion of a modern aircraft are becoming of more interest as flight regions expand. An aircraft that has its mass primarily distributed along the fuselage and is designed for high speed flight could foster undesirable conditions during certain flight regions. The dynamic response of an aircraft to various pilot control inputs is important in evaluating its handling qualities. The aircraft may be statically stable yet its dynamic response could be such that a dangerous or impossible flight characteristic results. The aircraft must have dynamic qualities that will permit the design mission to be accomplished. One of the test pilot's prime responsibilities is to evaluate these handling qualities with respect to the expected mission.

An airplane usually has five major modes of free motion. (Phugoid, short period, rolling, Dutch roll and spiral.) This chapter will deal with two longitudinal modes first (phugoid and short period) then two lateral-directional modes (Dutch roll and sprial). The rolling mode is covered in Chapter VII.

There are several different forms that the modes of motion may take. Figure 9.1 shows four possibilities for aircraft free motion; a pure divergence, a pure convergence, a damped or an undamped oscillation. The aircraft, being a rather complicated dynamic system, will move in a manner that is a combination of several different modes at the same time. One of the problems of flight testing is to initiate the excitation input so that the various individual modes can be picked out and analyzed on an individual basis.

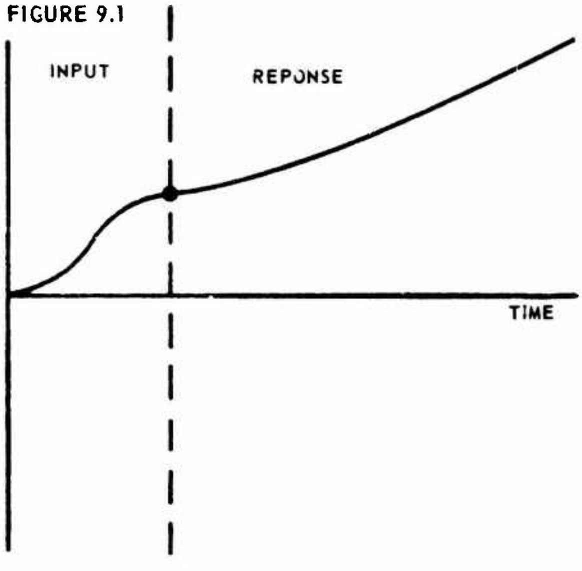
An airplane usually has two major longitudinal modes of free motion. One is the long period mode or "phugoid" which is essentially a variation in airspeed and pitch angle at nearly constant angle of attack. Its period is of the order of 20 seconds to 2 minutes. The other mode is of short period and is characterized by an oscillation of angle of attack and pitch angle at nearly constant airspeed. Its period is usually less than four seconds.

The phugoid mode is generally not considered an important flying quality because its period is usually of sufficient duration that the pilot has little difficulty in controlling it. However, under certain conditions it is possible for the damping to degenerate sufficiently so that the phugoid mode becomes important. The phugoid is characterized by airspeed, altitude, pitch angle, and rate variations while at essentially constant angle of attack.

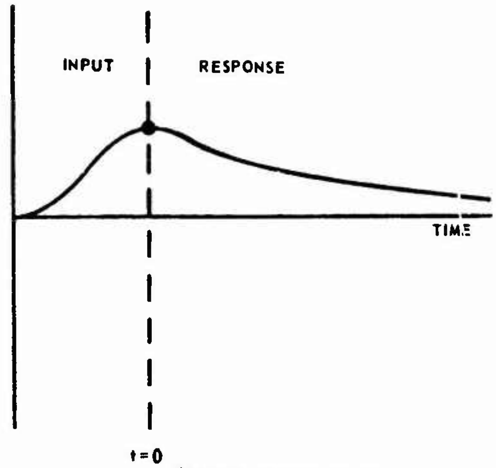
The short period mode is an important flying quality because its period can approach the limit of pilot reaction time and it is the mode which a pilot uses for longitudinal maneuvers in normal flying. The period and damping may be such that the pilot may induce an unstable oscillation if he attempts to damp the motion with control movements. Hence, heavy damping of this mode is desirable. In most airplanes the short period mode is sufficiently damped, but some airplanes must be fitted with artificial damping devices. These airplanes should be flight tested with dampers on and off. The short period is characterized by pitch angle, pitch rate, and angle of attack change while essentially at constant airspeed and altitude.

Damping is described in terms of damping ratio or number of cycles to damp to a specified fraction of initial amplitude. Although heavy damping of the short period mode is desired, investigations have shown that damping alone is insufficient for good flying qualities. In fact, very high damping may result in poor handling qualities. It is the combination of damping and frequency of the motion that is important.

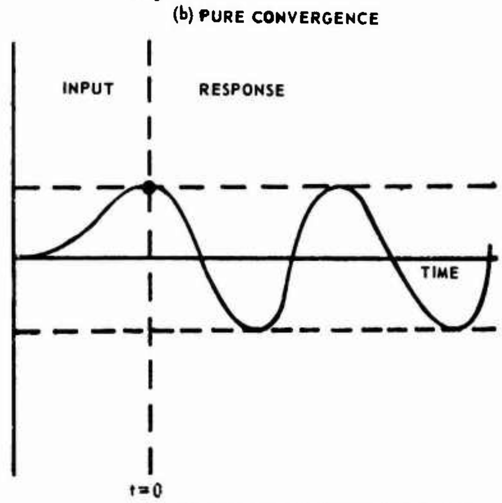
FIGURE 9.1



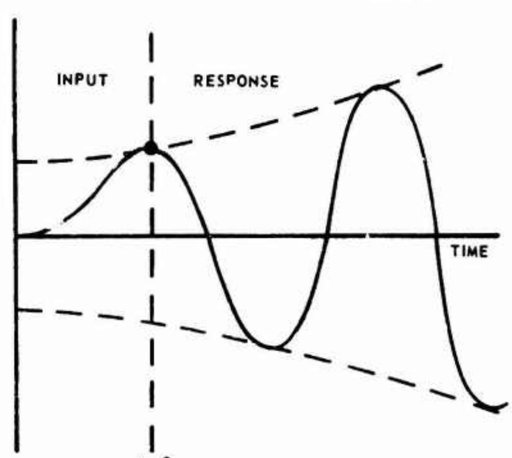
(a) PURE DIVERGENCE



(b) PURE CONVERGENCE



(c) OSCILLATION WITH ZERO DAMPING



(d) OSCILLATION WITH NEGATIVE DAMPING

The longitudinal modes should be flight tested for open-loop as well as closed-loop stability, since open-loop longitudinal modes can also be important. In open-loop motion, the elevator and control system is free to move (pilot does not hold the control) so that its motion is coupled with the longitudinal stick-fixed modes. The influence of the free elevator depends upon the magnitude, frequency and phase of elevator motion.

Tests for short period stability should be conducted from level flight at several altitudes and Mach numbers. Closed loop short period stability tests should be made also at various normal accelerations in maneuvering flight. This stability, when coupled to the pilot, is especially important to tracking and formation flying.

9.3 MILITARY SPECIFICATION REQUIREMENTS

MIL-F-8785 specifies that an aircraft's short period response, controls fixed or free, shall meet the requirements of frequency damping and acceleration sensitivity established in para. 3.2.2.1a, 3.2.2.1b, and figure 1. Residual oscillations shall not be greater than 0.05g at the pilot's station nor more than ± 3 mils of pitch excursion for category A Flight Phase tasks.

9.4 EXAMPLE TEST METHODS

The phugoid mode may be examined by stabilizing the airplane at the desired flight conditions and trimming the control forces to zero. Increase or decrease the airspeed by some small increment by the proper control pressure. For stick-fixed stability return the control to neutral and hold it fixed. For stick-free stability, return the control to neutral and then release it. After the control is released or returned, it may be necessary to maintain wings level by light lateral or slight directional pressure. Damping and frequency of phugoid motion may be changed appreciably by the presence of small bank angles (5 to 15 degrees). It may be very difficult to return the control to its trimmed position if the aircraft control system has a very large friction band. In such a case, the airspeed increment may be obtained by an increase or decrease in power and returning it to its trim setting or extending a drag device. In either case the aircraft configuration should be that of the trim condition at the time the data measurements are made.

To examine the short period mode, stabilize the airplane at the desired flight condition, (altitude, airspeed, normal acceleration). Trim the control forces to zero (for one g normal acceleration). Abruptly deflect the longitudinal control to obtain a change in normal acceleration of about one-half g. For stick-fixed stability, return the control to neutral and hold fixed. For stick free stability, release the control after it is returned to neutral (normally conducted only from one g flight). The aircraft response should be examined for positive and negative changes in normal acceleration. If the aircraft is equipped with artificial stabilization devices the test should be conducted with this device off as well as on. A note of caution: The abruptness and magnitude of the control input must be approached with due care! Use very small inputs until it is determined that the response is not violent. A suggested technique is to apply a longitudinal control doublet (a small positive displacement followed immediately by a negative displacement of the same magnitude followed by rapidly returning the control to the trimmed position). Start with small magnitudes and gradually work up to the desired excitation.

An input that is too sharp or too large could very easily excite the aircraft structural mode or produce a flutter that might seriously damage the airplane and/or injure the pilot.

Data Required:

For trim conditions, pressure altitude, airspeed, weight, cg position, and configuration should be recorded.

The test variables of concern are, airspeed, altitude, angle of attack, normal acceleration, pitch angle, pitch rate, control surface position, and control position.

Reduction and Presentation of Data:

Time histories of stick-fixed and stick-free oscillations should be presented. A complete analysis would present damping ratio and frequency as a function of flight condition. If the motion were non-oscillatory divergent, the instability could be represented by the time required to attain a certain parameter value from a trimmed condition.

Short period mode investigations have shown that frequency as well as damping is important in a consideration of flying qualities. This is so because at a given frequency, damping alters the phase angle of the closed-loop system (which consists of a pilot coupled to the airframe system). Phase angle of the total system governs the dynamic stability.

A. Phugoid:

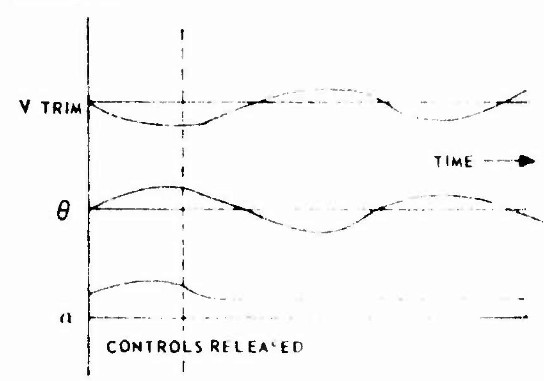
Stabilize the aircraft at the test altitude and the test airspeed. Smoothly increase the pitch angle until the airspeed reduce 10 to 15 knots below the trim airspeed. Very smoothly return the control column to the trim position and release all pressure. When the pitch angle reverses start timing. Record the maximum airspeed as the nose comes through level flight. The nose of the aircraft will continue to come up, reverse, and start down. Record the minimum velocity as the nose again passes through level flight. As the pitch angle reverses again mark the time. Continue the maneuver through 3 cycles.

Slight turbulence or imperfect lateral trim may result in wing roll during the pitch oscillations. If this should occur, then control the bank with smooth and light rudder pressures being careful not to excite aircraft Dutch roll.

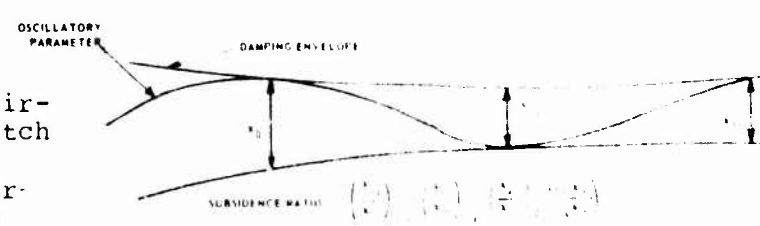
Data Reduction, Phugoid.

1. Plot a time history to include 3 cycles of the phugoid. Label airspeed, pitch rate and angle of attack.

FIGURE 9.2



2. Determine the frequency of the oscillation. Plot cycles versus time in a working plot.



3. Determine the phugoid damping ratio (ζ). Sketch the damping envelope on the oscillograph trace. Measure the width of the envelope at the peak values of the oscillation. Form the subsidence ratios (X_m/X_0). From figure 9.4 or 9.5 find the damping ratio for each subsidence ratio. Average these damping ratios. If the subsidence ratio is greater than 1.0 then use the inverse of that subsidence ratio. The damping ratio thus determined will be negative.

4. Determine the phugoid undamped nautral frequency (ω).

$$\omega_n = \frac{2\pi f}{\sqrt{1 - \zeta^2}}$$

$$f = \frac{\Delta \text{ cycles}}{\Delta \text{ time}}$$ from figure 9.3

FIGURE 9.3

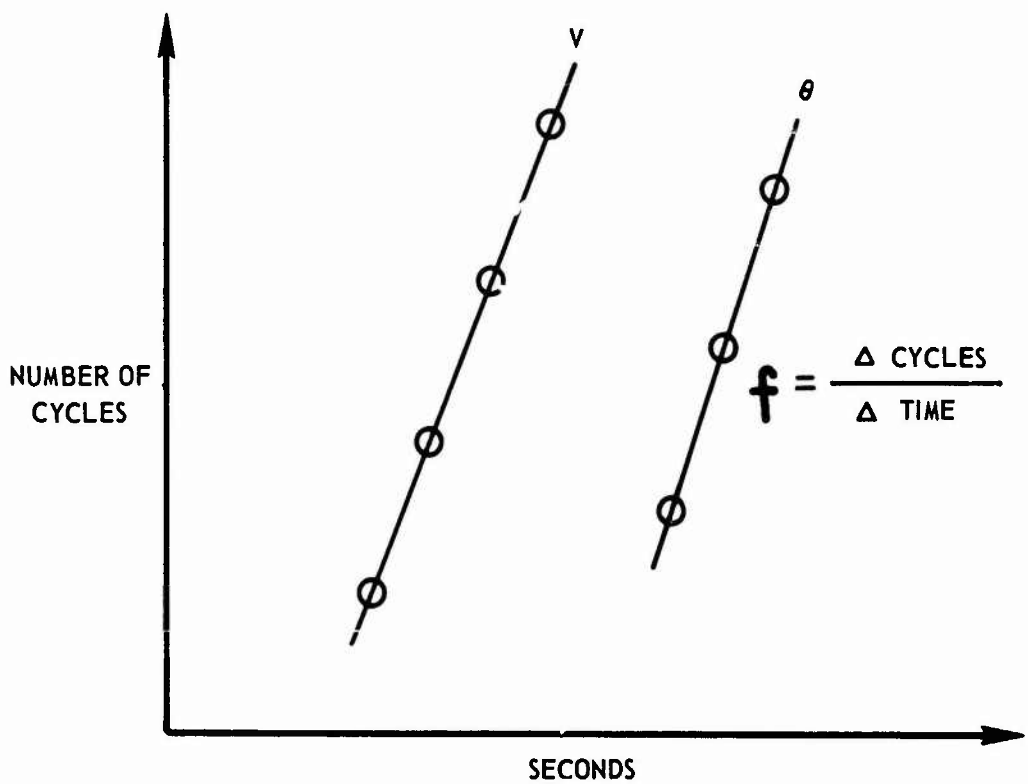


FIGURE 9.4

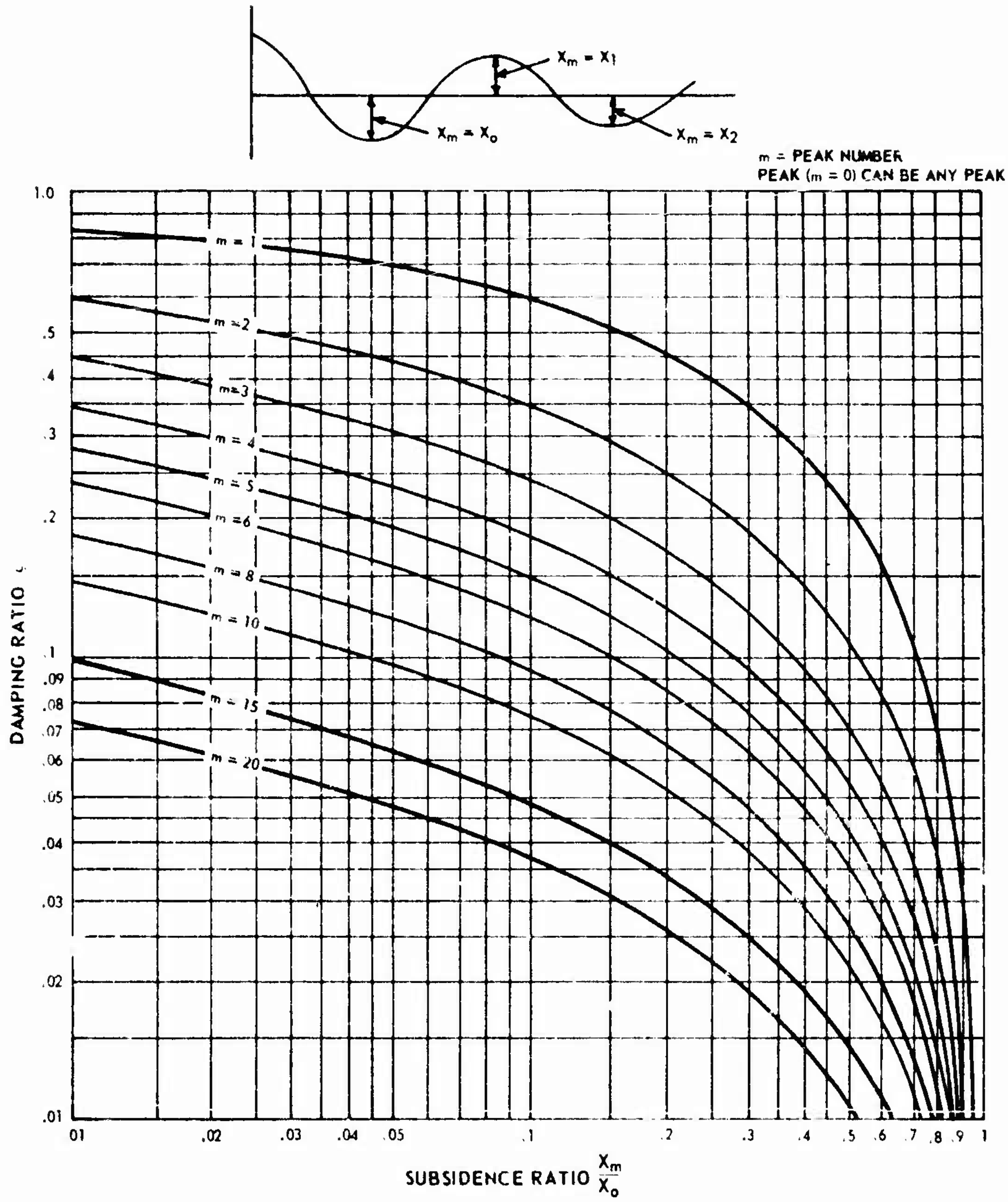
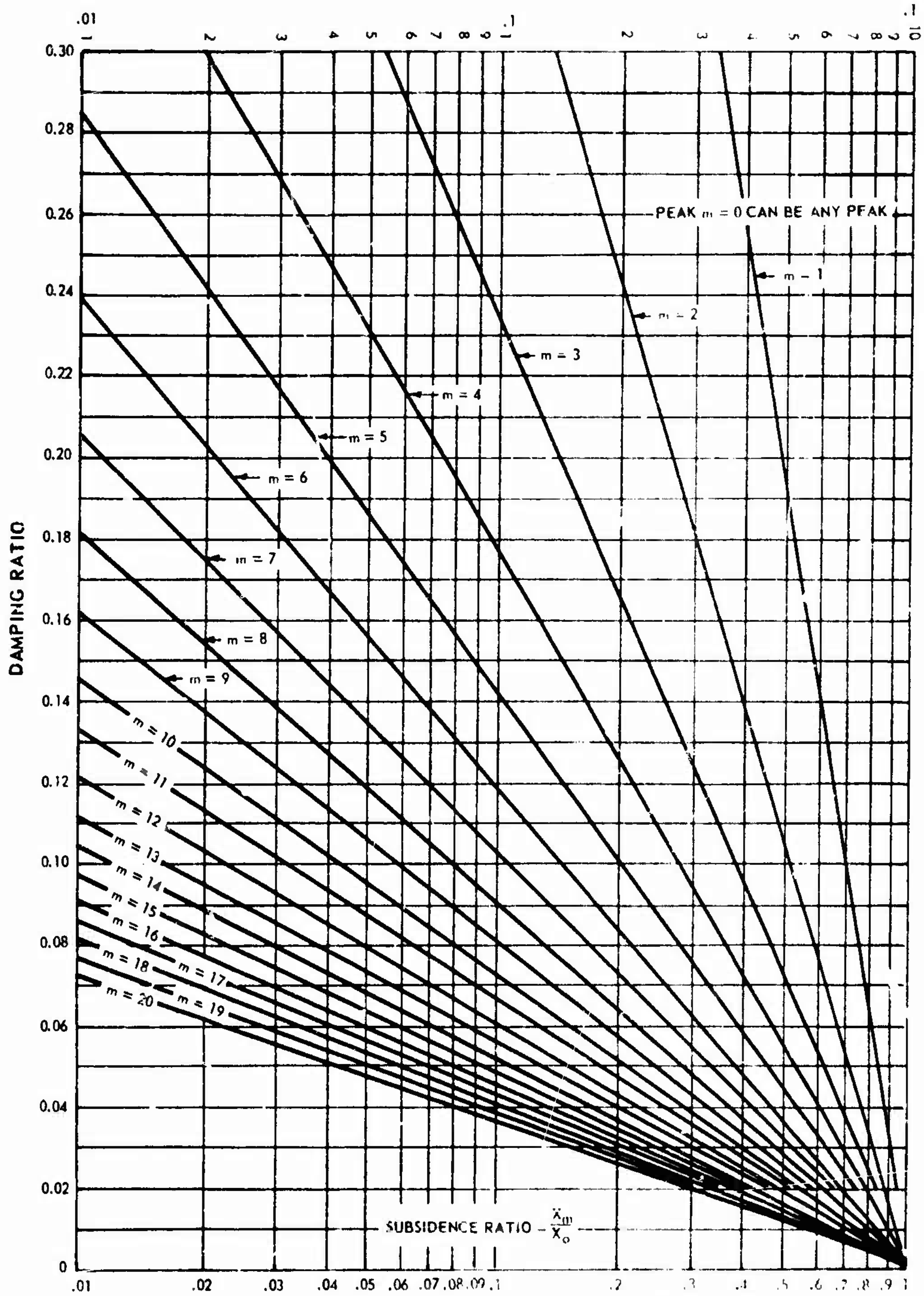

Xm = X1
Xm = Xo
Xm = X2
m = PEAK NUMBER
PEAK (m = 0) CAN BE ANY PEAK
1.0
.5
.4
.3
.2
.1
.09
.08
.07
.06
.05
.04
.03
.02
.01
m = 1
m = 2
m = 3
m = 4
m = 5
m = 6
m = 8
m = 10
m = 15
m = 20
DAMPING RATIO
.01
.02
.03
.04
.05
.1
.2
.3
.4
.5
.6
.7
.8
.9
1
SUBSIDENCE RATIO $\frac{X_m}{X_o}$

FIGURE 9.5



DAMPING RATIO AS A FUNCTION OF SUBSIDENCE RATIO

5. Plot phugoid frequency and damping ratio versus Mach number.

FIGURE 9.6

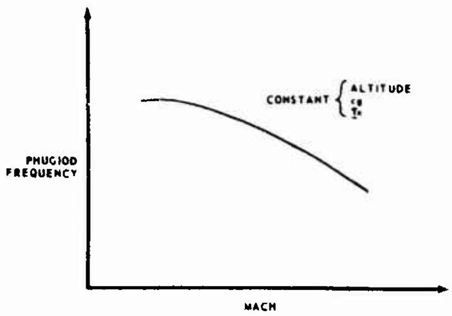
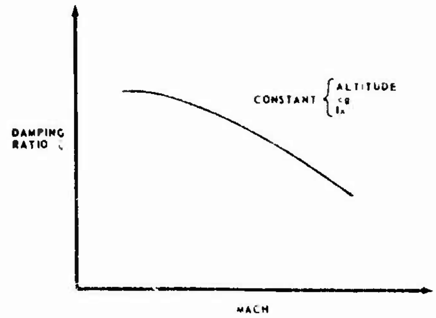
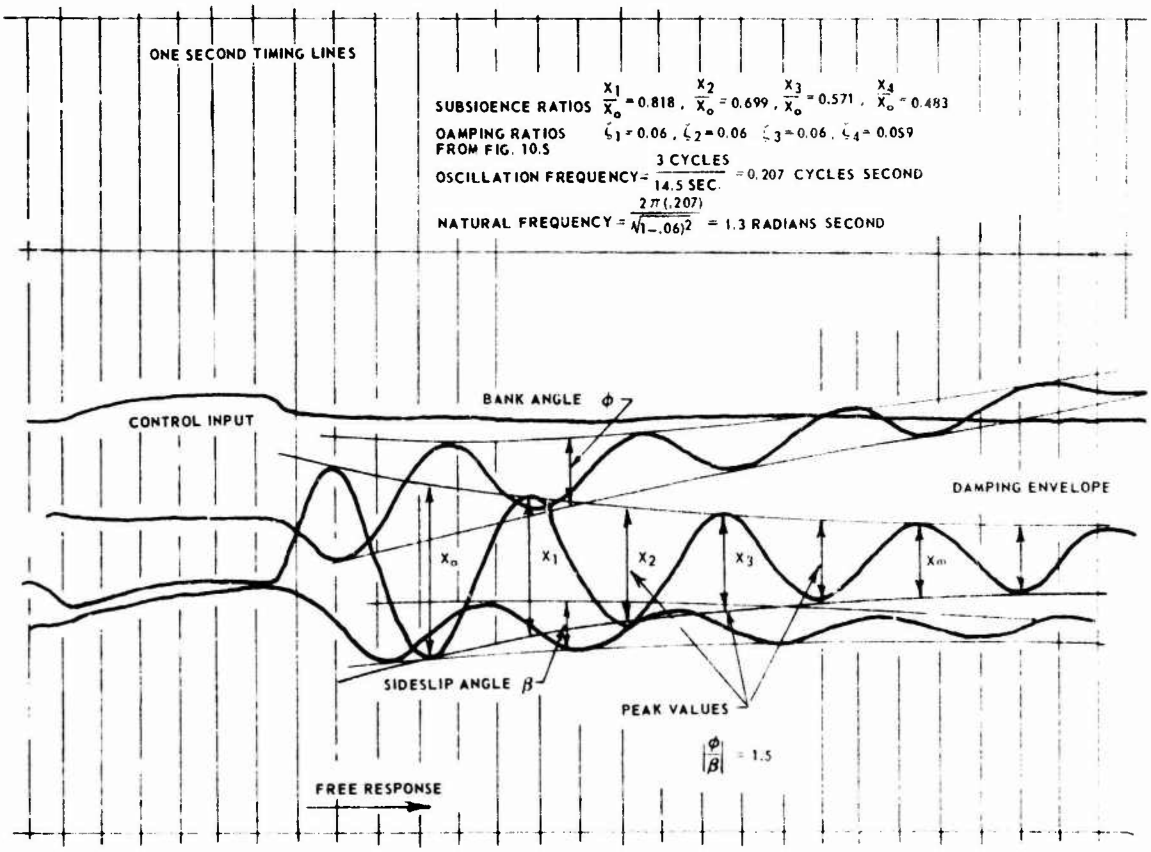


FIGURE 9.7



EXAMPLE NO. 1 DATA REDUCTION LIGHTLY DAMPEO OSCILLATION



OSCILLOGRAPH TRACE TIME →

6. In the test results include a short discussion on the effect the phugoid mode has on the aircraft handling qualities. This discussion should be presented with respect to an intended mission for the aircraft. Quantitatively compare the damping ratios with the requirements of MIL-F-8785.

B. Short Period:

Stick-Fixed.

Stabilize the aircraft at the test altitude on the test airspeed. Select oscillograph speed 4 and start recording. Smoothly but abruptly pull back on the control column; push it forward, and then rapidly return it to the trimmed position and hold it there. When the aircraft transient motion stops, stop recording data. Reverse the order of the input pulse and repeat. The airspeed and altitude should remain essentially constant during this maneuver. The data should be taken when the input pulse is approximately one-half g. This input pulse should be started small and gradually increased as the pilot's technique improves and if the aircraft response is satisfactory.

Stick-Free.

Stabilize the aircraft at the test altitude on the test airspeed. Select oscillograph speed 4 and start recording. Smoothly but abruptly pull back on the control column, push it forward, return it to approximately neutral and release. When the aircraft transient motion stops, stop recording data. Reverse the input order and repeat.

Data Reduction, Short Period.

1. Plot a time history of the aircraft response for closed-loop and open-loop. Label elevator deflection, control deflection, load factor, and angle of attack. Examine the phase angle between the stick movement and the actual elevator deflection for compliance with MIL-F-8785.

2. Determine the short period damping ratio (ζ). If the short period response is oscillatory and the damping ratio 0.5 or less proceed as outlined for the phugoid mode. If the damping ratio is between 0.5 and 2.0 then use figure 9.9. Select the point on the response curve at which the response is free. Divide the amplitude into the values 0.736, 0.406, and 0.199. Measure time values t_1, t_2 and t_3. Form the time ratios t_2/t_1, t_3/t_1, and $t_3 - t_2/t_2 - t_1$. Enter figure 9.9 at the Time Ratio side and find a damping ratio for each time ratio. For this damping ratio find a frequency time product for $(\omega_n t_1)$ $(\omega_n t_2)$ and $(\omega_n t_3)$. Average the damping ratios.

FIGURE 9.8

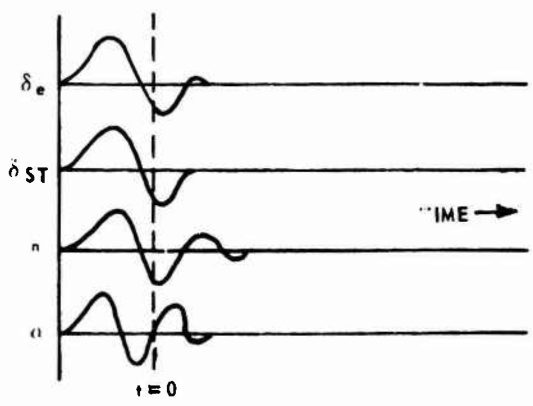
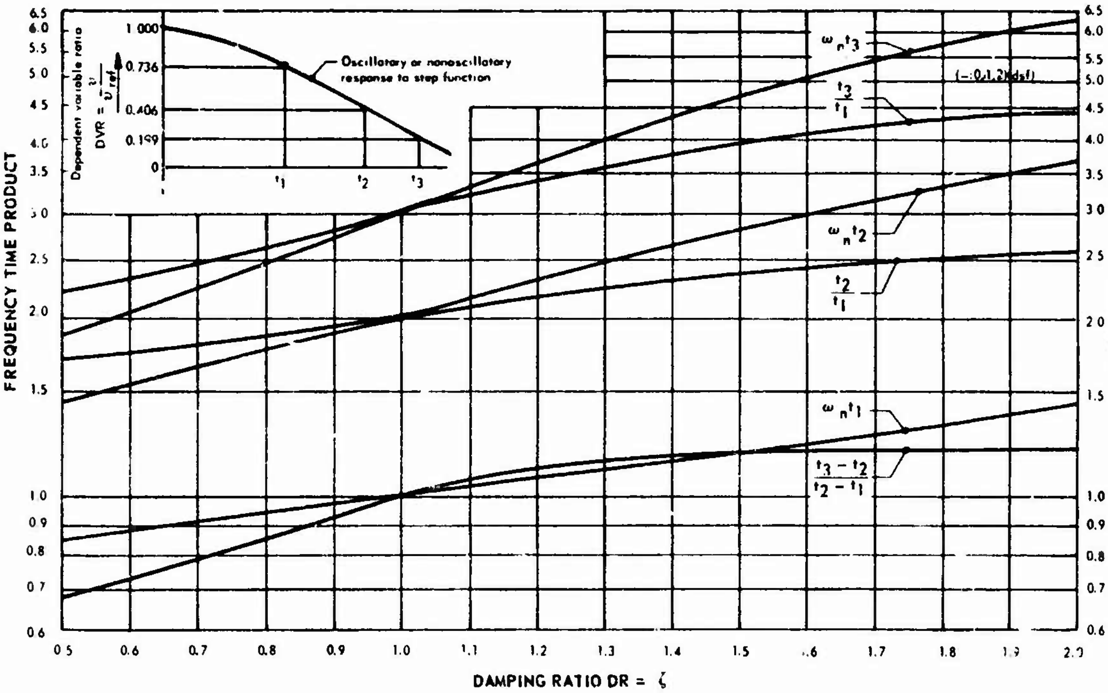


FIGURE 9.9



3. Determine the short period natural frequency (ω_n)

$$\omega_n = \frac{\omega_n t_3}{t_3}$$

$$\omega_n = \frac{\omega_n t_2}{t_2}$$

$$\omega_n = \frac{\omega_n t_1}{t_1}$$

Average the natural frequency.

4. Plot short period natural frequency and damping ratio versus Mach number.

FIGURE 9.10

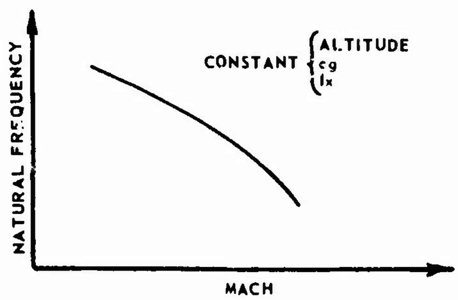
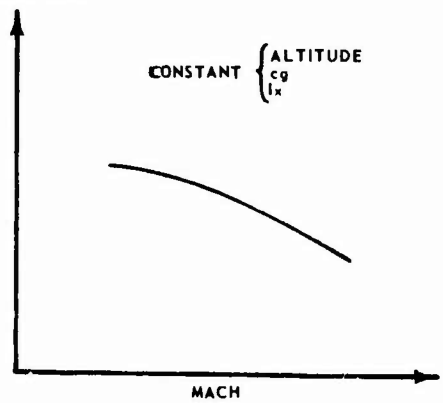


FIGURE 9.11



5. From the oscillograph trace, form the ratio, n_z/α. This is determined from the peak angle of attack which produced the peak "g." See where ω_n versus n_z/α is located in figure 1, MIL-F-8785.

6. In the test results include a short discussion on the effect of the short period characteristics have on the handling qualities of the aircraft.

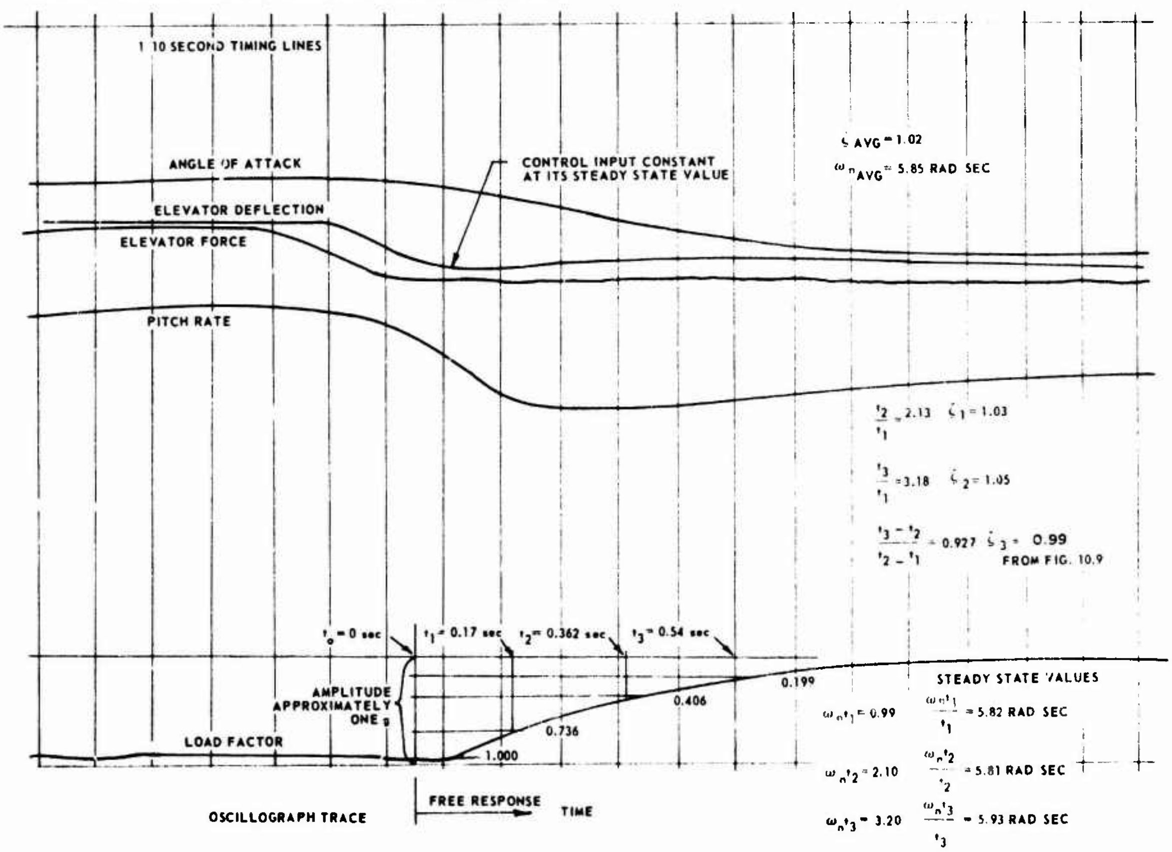
FIGURE 9.12

9.5 LATERAL-DIRECTIONAL DYNAMIC STABILITY

Dutch Roll:

The lateral-directional oscillations involve roll, yaw, and sideslip. The stability of the Dutch roll mode varies with airplane configuration, angle of attack, Mach number, and damper configuration. Dynamic stability of the lateral-directional modes is governed primarily by the static lateral and directional stabilities (C_{ℓ_β} and C_{n_β}), damping in roll and yaw (C_{ℓ_p} and C_{n_r}), and moments of inertia. The presence of a lightly damped oscillation adversely affects aiming accuracy during bombing runs, firing of guns and rockets and precise formation work such as in-flight refueling.

EXAMPLE NO. 2 DATA REDUCTION HEAVILY DAMPED RESPONSE



Stability of the oscillations is represented by the damping ratio; however, the frequency of an oscillation is also important in order to correlate the motion data with the pilot's opinion of handling qualities.

Military Specification Requirements:

Section 3.3 of MIL-F-8785 specifies in figure 2 the requirements for lateral directional handling qualities. It also states the residual oscillation that may be allowed for Category A Flight Phases.

Example Test Methods:

Release from Steady Sideslip.

Stabilize the airplane in level flight at test flight conditions and trim forces to zero. Establish a steady straight-path sideslip angle. Rapidly neutralize controls. Either hold controls for control-fixed or release controls for control free response. Start with small sideslip in case the aircraft diverges.

Rudder Pulse (Doublet).

Stabilize the airplane in level flight at test flight conditions and trim. Rapidly depress the rudder in each direction and neutralize. Hold at neutral for control-fixed or release rudder for control free response. For aircraft which require excessive rudder force in some flight conditions, the rudder pulse may be applied through the augmented directional flight control system.

Aileron Pulse.

Stabilize the airplane in level flight at test flight conditions and trim. Hold aircraft in a steady turn of 10 to 30 degrees of bank. Roll level at a maximum rate reducing the roll rate to zero at level flight. CAUTION . . . Such a test procedure must be monitored by an engineer who is thoroughly familiar with the inertial coupling of that aircraft and its effect upon structural loads and non-linear stability.

Data Required.

For trim condition, pressure altitude, airspeed, weight, cg position, and aircraft configuration should be recorded. The test variables of concern are: bank angle, sideslip angle, yaw rate, bank rate, control positions, and control surface positions.

Reduction and Presentation of Data.

Flight test data will be obtained as time histories. When determining the damping ratio the roll rate parameter usually presents the best trace.

Nonlinearities in the aircraft response may hinder the extraction of the necessary parameters. These can be induced by large input conditions. Small inputs balanced with instrument sensitivity give the best result.

moderately damped high frequency oscillation may be less satisfactory than a lightly damped low frequency oscillation. If the frequency is higher than pilot reaction time, the pilot cannot control the oscillation, and in some cases may reinforce the oscillation to an undesirable amplitude. Since it is the damping frequency combination which influences pilot opinion more than damping alone, some effort should be made to correlate this combination with pilot opinion of the lateral-directional oscillation.

At supersonic speeds, stability decreases with increased Mach number and altitude for constant g. An evaluation should proceed cautiously to avoid posible divergent responses that can result from nonlinear aerodynamics.

Control-Free Dutch Roll.

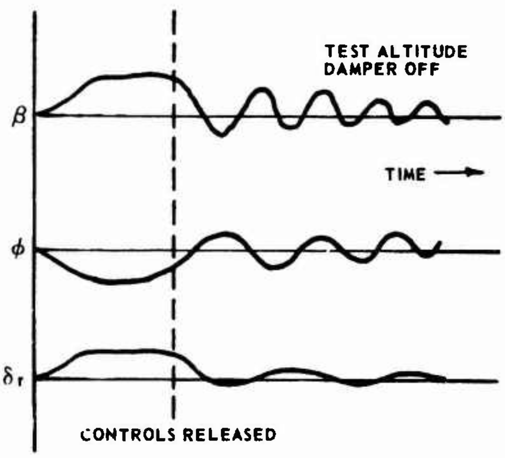
Stabilize the aircraft on test altitude and airspeed. The yaw damper and rudder power should be on. Select oscillograph speed 3 and start recording. Smoothly establish a steady straight sideslip using rudder and aileron. Release the controls. Start counting and timing oscillations when the aircraft nose reaches its extreme position from where it was released. Stop recording when the oscillation stops or after 5 to 8 cycles. Use caution and avoid any excessive sideslip angles.

Restabilize the aircraft with the yaw damper off. Select speed 3 on the oscillograph and start recording. Establish a steady straight sideslip and release controls. Start counting and timing cycles when the nose reaches its extreme position from the point of release. Stop recording after 5 to 8 cycles. Repeat the test with the rudder power off. Use caution in this configuration and avoid any excessive sideslip angles.

Data Reduction.

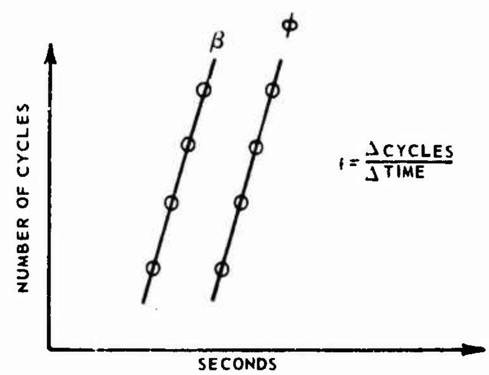
1. Sketch 5 cycles of the Dutch roll in each configuration, (damper and rudder power on, damper off, dampers and rudder power off). Label sideslip, bank angle and rudder deflection.

FIGURE 9.13



2. Determine the frequency of the oscillation. Plot cycles versus time on a working plot.

FIGURE 9.14



3. Determine the Dutch roll damping ratio (ζ) and natural frequency (ω_n) in the same manner as the phugoid mode.

4. Plot Dutch roll damping ratio and natural frequency versus Mach number for each configuration (damper on, damper off, rudder power off).

FIGURE 9.15

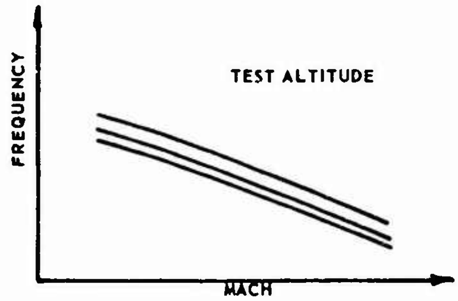
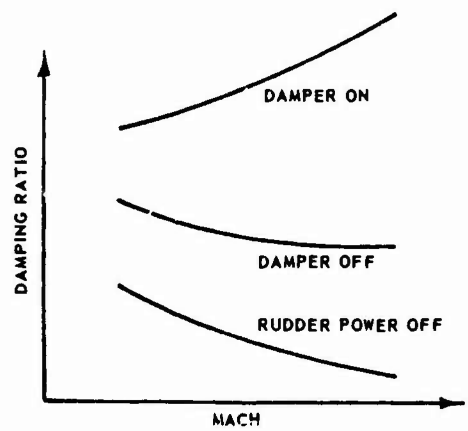
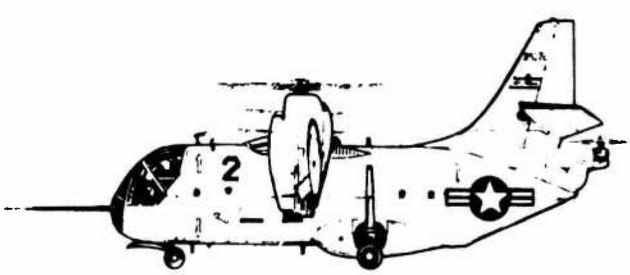


FIGURE 9.16



5. Compare the ω_n and ζ so obtained with figure 2 of MIL-F-8785 and determine compliance.



9.6 SPIRAL MODE

The spiral mode is, in general, relatively unimportant as a flying quality. However, a combination of spiral instability and lack of precise lateral trimmability may be bothersome to the pilot. This problem will be evaluated as a whole due to the difficulty in separating the effects.

The divergent motion is non-oscillatory, and is most noticeable in the bank and yaw responses. If an airplane is spirally divergent, it will, when disturbed and not checked, go into a tightening spiral dive. This divergence can be easily controlled by the pilot if the divergence is not too great.

Military Specification Requirements:

Spiral Stability is specified in MIL-F-8785 in table V. This table established limiting terms to double amplitude when the aircraft is put into a 20-degree bank and the controls freed.

Example Test Methods:

Trim the aircraft for hands-off flight, insuring that particular attention is given to lateral control and the ball being centered. Roll into a 20-degree bank in one direction, release the controls and measure the time it takes to reach 40 degrees of bank if the bank angle tends to diverge. Repeat the maneuver in a bank to the opposite side.

Data Required:

Aircraft configuration, weight, cg position, altitude and airspeed should be recorded. The test variables are bank angle, sideslip angle, control position, and control surface position.

Excitation of the spiral mode only is difficult because of its relatively large time constant. Any practical input using control surfaces would usually excite other modes as well. If a deficiency in lateral trim control exists, it is often difficult to determine what portion of the resultant motion following a disturbance is caused by the spiral mode. This flight test is used to determine if a combined problem of lateral trim and spiral stability exists. If test results show a definite divergence in hands-off flight, the problem exists.

Spiral divergence, on its own, is of little importance as a flying quality, which is well within the control capability of the pilot. The ability to hold lateral trim in hands-off flight for 10 to 20 seconds is important.

Spiral Mode:

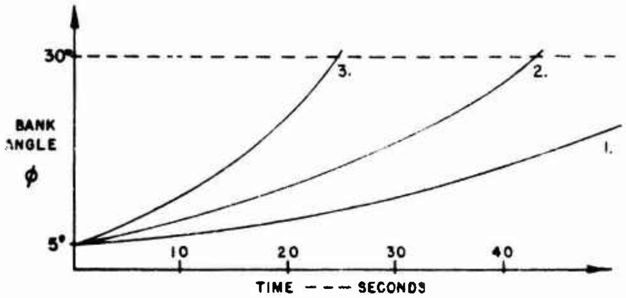
Stabilize the aircraft at test altitude and airspeed. Select oscillograph speed 2 and start recording. Establish a 20-degree bank and release controls. Time the motion to a 40-degree bank angle or 40 seconds elapse, whichever is shorter. Stop recording. Establish an opposite 40-degree bank and repeat. Repeat this procedure with the yaw damper off and then with the rudder power off.

Data Reduction:

1. Sketch a time history of the bank angle.

2. Average the time to double amplitude for right and left banks at each test condition. Compare with table V of MIL-F-8785.

FIGURE 9.17



3. Briefly discuss the spiral mode characteristics with respect to an intended mission.

9.7 DEMONSTRATION MISSION

The purpose of this mission is to demonstrate and practice the techniques used to investigate an aircraft's dynamic modes of motion. The aircraft used will be a B-57E.

Procedures:

1. Takeoff and climb to 20,000 feet. Practice rudder and stick inputs.

2. Obtain a trim point at 300 KIAS, 20,000 feet, in the CR configuration.

3. The IP will demonstrate the control inputs used to excite the short period mode. Vary frequency of input from structural frequency to less than the natural frequency. Inputs demonstrated will be singlets and doublets, stick-fixed and stick-free.

4. Practice inputs and observe aircraft motion. (Maintain 300 $\pm$10 knots and 20,000 feet $\pm$2,000 feet.)

5. The IP will demonstrate the methods of exciting the Dutch roll mode.

6. Practice inputs and observe aircraft motion. Estimate ϕ/β ratio, period of oscillation and number of overshoots.

7. Investigate spiral mode from a 20-degree bank angle. Note the effect that the out of trim condition has on results.

8. Obtain a trim point at 200 KIAS, 35,000 feet, in the CR configuration.

   NOTE: Maintain airspeed and altitude within ± 3 knots and ± 500 feet from trim points for remainder of mission.

9. Excite and observe the short period mode.

10. Investigate the Dutch roll motion with dampers off and with the rudder power on and off. Note ϕ/β ratio, overshoots, and period. <u>Use Caution</u> - <u>may be divergent</u>.

11. Obtain a trim point at M = 0.79 at 33,000 feet, in the CR configuration.

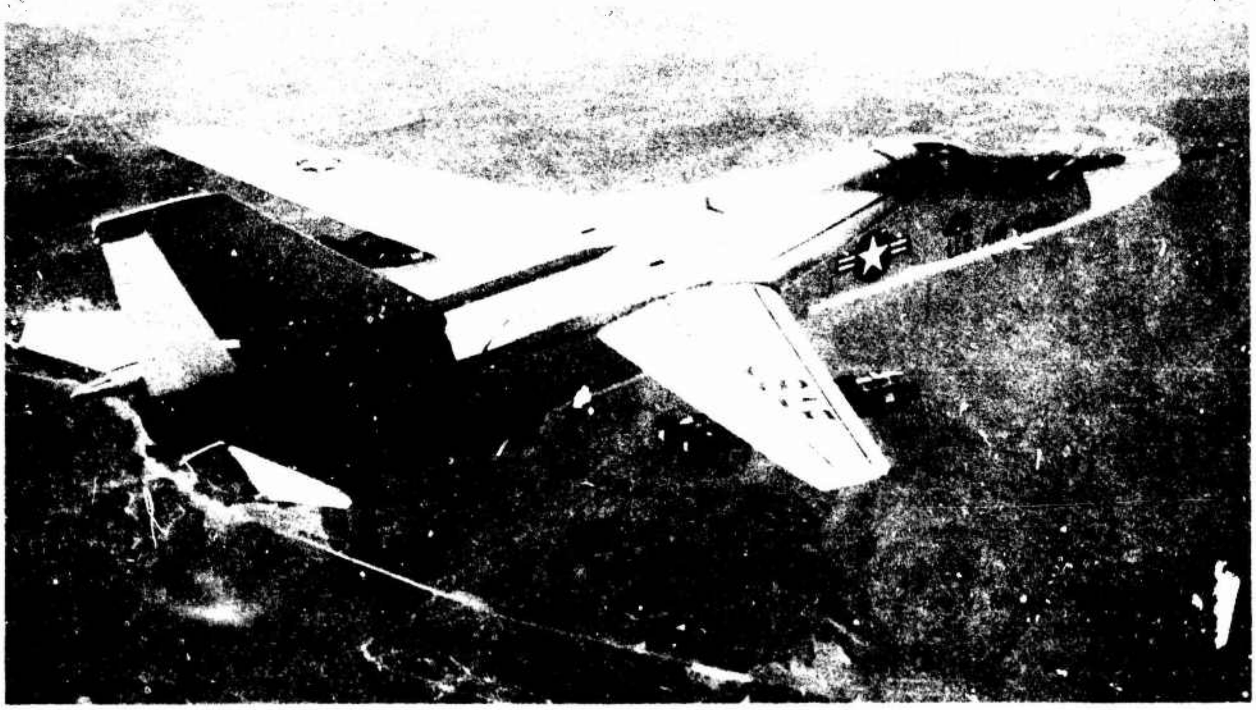
12. Excite the phugoid mode with a 3- to 4-knot Δ airspeed. Note the period and damping present.

13. Repeat No. 12 using a 10- to 12-knot Δ airspeed. Note the divergence due to M_u.

14. Investigate the short period mode.

15. Investigate the Dutch roll mode.

16. Land on the spot.



CHAPTER 10

QUALITATIVE FLIGHT TESTING

● 10.1 PURPOSE

The purpose of the qualitative flight test is to determine the maximum amount of information in the minimum amount of flying time in order to evaluate an aircraft with respect to its entire mission or some specific area of interest.

Qualitative flight testing has essentially the same purpose as quantitative flight testing, i.e., to determine how well the aircraft flies and how well it will perform its design mission. To accurately evaluate an aircraft from quantitative data requires analysis of large amounts of precisely measured data. The best a pilot can hope to do on a qualitative evaluation is to measure a limited amount of quantitative data. Thus, the test pilot's opinion on the acceptability of the aircraft is the important result and measured quantitative data when available is used primarily to support this opinion. Quantitative values of stick forces measured with a hand gage, for example, should be included in the report to support the pilot's opinion of acceptability. Estimations of values of stick force can be made if no reliable measurements are available or, qualifying terms such as "heavy", "medium", or "light" can be used to describe the forces. The point is that the difference in evaluating an aircraft qualitatively and quantitatively is a matter of degree. "Use what you've got." Pilot opinion supported by measured data is primary in qualitative testing while the reverse is true in quantitative testing. The general rule is to first decide how well the aircraft does its job and then use the quantitative data you can get to support your opinion.

● 10.2 PILOT OPINION

Naturally, all pilots will not have exactly the same opinion regarding the acceptability or unacceptability of a particular aircraft characteristic. No two people think exactly alike. However, the opinions of pilots with similar experience and background will usually not differ greatly, particularly with respect to the capability of an aircraft to perform a specific mission. In other respects, such as cockpit arrangements, the opinions may vary more markedly. For this reason, it is important for the qualitative test pilot to be as objective as possible in his evaluation. Guides which specify military requirements, such as MIL SPEC 203E and MIL-F-8785(ASG), should be used wherever possible to establish acceptability. However, it should be kept in mind that mere compliance with a set of requirements does not necessarily yield a satisfactory aircraft. The primary question is "will it do the job?", not "does it meet the specifications."

● 10.3 MISSION PREPARATION

A very limited amount of flight time is normally available for a qualitative evaluation. To acquire the information necessary to write an accurate and comprehensive report on an aircraft in this limited time requires a great deal of pre-flight study and planning.

The pre-flight preparation for a qualitative test is extremely important. It is almost impossible to put in too much time in planning for the flights. The amount of information acquired in the air will be directly proportional to the amount of preparation put in on the ground. A pilot who doesn't know what he is looking for is not likely to find it, and to know exactly what to look for in the evaluation requires considerable knowledge of the aircraft and its mission.

The precise mission of the aircraft is important in determining what specific investigations should be made in the evaluation. All fighters, for instance, do not have the same mission, and the characteristics of particular importance may not be the same. The roll characteristics of an air superiority fighter would be more important than for a long range strategic fighter, and the specific test plan should take this fact into account. Expected outstanding characteristics or weaknesses should also receive particular emphasis. Of course, the evaluation must be conducted within the cleared flight envelope of the aircraft, and the amount of flight time available may limit the number of altitudes, airspeeds and tests that can be investigated. However, concentration on the extremes of altitudes, airspeeds, etc., and the areas dictated by the primary mission will provide the best approach to the test planning.

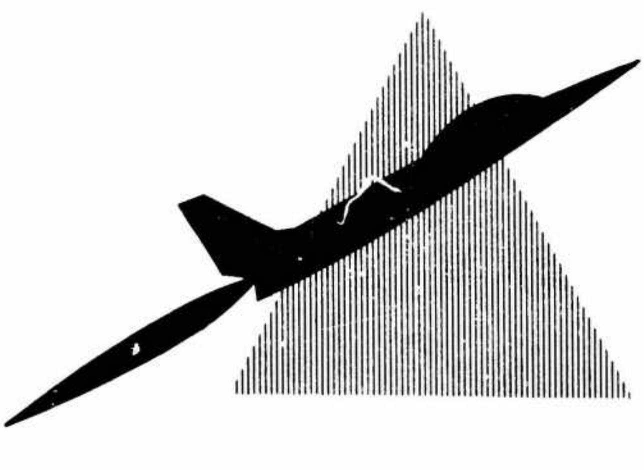
An outline of the test to be conducted and the various altitudes, airspeeds and configurations to be used will aid in organizing the flights and planning the flight data cards. The points included in the outline should be compatible with the time available for the evaluation but it is always wise to overplan the flight and include more than seems possible to accomplish in the allotted time. Leave yourself the option of skipping the less important parts of your plan if time or fuel run short. The sequence of tests should be such that as little time as possible is wasted. With proper planning a continuous flow from one investigation to the next is possible.

10.4 FLIGHT DATA CARDS

Before planning the flight data cards, as much as possible should be learned about the aircraft. Study the pilot's handbook if one is available, discuss the aircraft with the engineers, or with other pilots who have flown it, and get adequate cockpit time. The more the pilot knows about the aircraft and the more comfortable he is in it, the more thorough will be the evaluation. A pilot who doesn't know the aircraft procedures, both normal and emergency or who has to spend most of his time in the air looking for controls or switches will not be able to do much evaluating.

The flight data cards should be self explanatory and should include all the points it is desirable to investigate during the flight. They should be designed so that a minimum of writing is required in the air because time will not be available to write down more than a word or two about each point. Remember, however, to provide places in the flight plan to write down these necessary comments. Numerous forms for the data cards are possible but completeness and legibility are essential.

The following are some possible formats for planning check lists and flight data cards:



AIRCRAFT QUALITATIVE EVALUATION CHECK LIST

1. Location ______ Test Crew: Pilot ______
Date ______ Co-Pilot ______
Aircraft ______ Flight Mechanic ______
Runway Length ______ Observers ______
Weather ______
Runway Temperature ______
Press. Altitude ______
Surface Winds
5M ______ Climb Wind ______
10M ______ Freezing Level ______
15M ______
20M ______
25M ______

Weight and Balance: Operating Weight ______
Fuel Weight ______
MAC ______ % Gross Weight ______
T.O. Gross Weight ______
Est. Fuel Used ______
MAC ______ % Est. Landing Weight ______

Performance

T.O. Distance ______ Refusal Speed ______ T.O. Speed ______
Minimum Control Speed ______ Abort Landing Distance ______
Climb Schedule: 4M ______ 6M ______ 8M ______ 10M ______
12M ______ 14M ______ 16M ______ 18M ______
20M ______ 22M ______ 24M ______ 26M ______
28M ______ 30M ______
Cruise (Max Range) HI ______ Vi ______ Pwr ______

Operating Limitations

Gear Down ______ Flaps: 10 % ______ 50 % ______ 100 % ______
Landing Light ______ Cargo Doors ______ Para-defl ______
Dive Speeds: 30M ______ 25M ______ 20M ______ 15M ______
10M ______ 5M ______
Load Factor ______ Weight ______
Engine: Sync RPM ______ Slave RPM ______ Overspeed ______
TIT: T.O. ______ (MRP) Normal ______ Other ______
Airstart Vi ______ Torque ______ Max. ______ Cont. ______
Remarks: ______

Systems Operation:

DC Generators ______ Eng. AC Generators ______ Eng. ______
Booster Hyd ______ Eng. Utility Hyd ______ Eng. ______
Other:

Auxiliary Equipment Operation:

Auto-pilot ______
Other:

NOTES:

II. PRE-OPERATION EVALUATION

A. Support Equipment
   1. Power Unit
      Type
      Capacity
   2. Other

B. Cargo Compartment
   1. Entrance
   2. Egress
   3. Systems Accessibility
   4. Other

C. Flight Deck
   1. Crew Stations
      a. Pilot
         Seat Adjustment
         Clearance
         Vision
         Rudder
         Pedal Adjustment
         Restrictions
         Other

      b. Copilot

      c. Flight Mechanic

      d. Navigator

   2. Instrument Panel
      a. Flight Instruments
         Grouping
         Readability
         Adequacy

      b. Engine Instruments
         Grouping
         Readability
         Adequacy

      c. Warning Lights
         Placards
         Switches
         Controls

   3. Pedestal
      a. Engine Controls
         System Controls
         Switches
         Guards
         Placards
         Lights
         Feel Identification
         Accessibility
         Confusion Factor
         Arrangement

      b. Remarks

   4. Overhead Panel
      a. Engine Controls
         System Controls
         Switches
         Guards
         Lights
         Placards
         Accessibility
         Feel Identification
         Confusion Factor
         Arrangement

      b. Remarks

   5. Side Panels
      a. Switches
         CSs
         Lights

      b. Remarks

   6. Flight Controls
      a. Rudder
         Break-out Force
         Travel
         Adjustment
         Clearance
         Slop
         Friction

      b. Elevator
         Break-out Force
         Travel
         Slop
         Friction
         Clearance

      c. Control Wheel
         Aileron Break-out Force
         Travel
         Slop
         Friction
         Clearance
         Grip
         Switches

   7. General Comments

5. Vibration
   a. Noise
   b. Air vent deflectors
   c. Ventilation/heating
6. Control Required To Maintain Proper Taxi Speed
7. Remarks

D. Pre-Take-Off (line up at even 1,000 feet and check W/V)
1. Flight Control Check With Boost Operating
   a. b/o force
   b. rate
   c. deflection
   d. stop
   e. friction
2. Flaps Set__________Trim Set__________
3. Engine Power Check
   a. Acceleration
      Idle__________to__________(MRP)______Sec.
      Asymmetric__________
      Overshoot__________
   b. Stabilized conditions: OAT__________

Eng	%RPM	Torque	TIT	Throttle Pos
1				
2				
3				
4				

4. Brakes Hold At MIL PWR
5. Fuel reading__________lbs. W/V__________ ______kts.

E. Take-Off. (Use flight data on knee board)
1. Start Time Form BRAKE RELEASE TO START CLIMB__________
2. Brake Release Action
3. Directional Control. Rudder Effective__________kts.
4. Elevator Effective (nose wheel off)__________kts.
5. Aileron Control__________kts.
6. T.O. Distance__________ft. Lift-Off Speed__________kts.
   Time______sec.
7. Control Force__________Pitch__________Trim__________
8. Trim-Out - Raise Gear
   Time__________sec.
   Yaw__________
   Trim__________
9. Trim-Out - Raise Flaps
   Time__________sec.
   Trim__________
10. Acceleration to MINIMUM CONTROL SPEED
11. Acceleration to Climb Speed (1,000 ft)
12. Visibility and Pitch Angle__________
13. Remarks:

F. Climb (M N ________°, 90° to W/V).

1. Visibility ________
   Pitch Angle ________
2. Record: FUEL at START CLIMB ________

TIME	Hi	Vi	R/C	Ti	%RPM	TORQUE	TPT	Wf
	4M							
	6M							
	8M							
	10M							
	12M							
	14M							
	16M							
	18M							
	20M							
	22M							
	24M							
	26M							
	28M							
	30M							
	32M							

FUEL at LEVEL-OFF ________________

3. Check Cabin Pressurization:
   10M ________________
   15M ________________
   20M ________________
   25M ________________
   30M ________________ Note any fluctuations or surges.
4. Cabin Heat Adequacy
   a. Nesi glass
5. Remarks

G. Cruise

1. Vmax
   a. Hi ________
   b. Vi ________
   c. OAT ________
   d. Flt. Controls ________
   e. RPM ________
   f. Torque ________
   g. TIT ________
   h. Wf ________
   i. FUEL ________

2. Dynamics (Hi__________Vi__________) Note Control Position
   a. Phugoid
      1. Trim__________$V_{i_{in}}$__________V_{max}__________V_{min}__________
      2. Sec/cyc__________•__________Damping__________
   b. Porpoise Mode. input__________cycls__________ampl.__________
   c. Spiral stability
      1. RT Ø 10°__________°/__________sec.
      2. LFT " "__________°/__________sec.
      3. Remarks:
   d. Dutch Roll
      1. RT sideslip s/c__________Roll__________Yaw__________
         Damping__________(1)__________(2)__________(3)__________
      2. LFT sideslip s/c__________Roll__________Yaw__________
         Damping__________(1)__________(2)__________(3)__________
      3. (1) Norm (2) Damper Off (3) Rudder Power Off.
   e. Short Period
      1. Fixed (1.0g) Damping__________
      2. Fixed (-1.0g) " __________
      3. Free (1.0g) " __________
      4. Free (-1.0g) " __________
      5. Remarks:

3. Maximum Range Data
   a. Hi__________Vi__________OAT__________FUEL__________
   b. RPM__________Torque__________TPT__________Wf__________
   c. Remarks:

4. Systems Check: Hi__________Vi__________
   a. Engine shut-down, No.__________
      1. Time to feather__________Control force__________
      2. Procedure, etc:
   b. Engine restart
      1. Time to Normal power__________Surge__________Trim__________
      2. Procedure, etc:
   c. Anti-icing/de-icing system
      1. Full operation effect on engines__________
      2. Nesi glass
         Other
      3. Remarks:
   d. GTU/ATM operation__________
   e. Pressurization/heating__________
   f. Other:

5. Emergency Descent, Hi__________Vi__________(Initial)
   a. Time from cruise to start descent__________
   b. Procedure: G and F__________Clean__________Pressurization__________
   c. Time__________from CR to Hi__________at Vi__________
   d. Visibility__________Pitch__________Control__________
   e. Remarks:

6. Static Longitudinal Stability and Performance Hi ______

a. Acceleration check Trim at Max Range Vi ______

1. Decel to Vi ______ Control force *(Trim setting) ______
2. Speed/Pwr Vi ______ Rpm ______ Tq ______ TIT ______ OAT ______
   Speed/Pwr Vi ______ Rpm ______ Tq ______ TIT ______
3. Acceleration, (RESET TRIM), Time/10 kts (MRP) Initial Vi ______
   10 ______
   20 ______
   30 ______
   40 ______
   50 ______
   60 ______
   70 ______
   80 ______
   V/S ______ ft/min. Control forces/gradient
4. Remarks: FUEL ______

b. Trim Changes: Hi ______ Vi ______

1. Control boost off ______ on ______
2. Runaway Trim: Elev ______ Ail ______ Rud ______
   5 sec delay (build-up)

c. Turning Performance and Aileron Rolls. Cruise. (Build-up). FULL DEFLECT

1. 60° Ø, Time 360° ______ Vmax ______ Hi ______
2. 45° Lft - 45°Rt (FIX) Time for 90° ______
3. 45° Rt - 45° Lft (FIX) Time for 91° ______
4. 60° Ø, Time 360° ______ Vi ______ Hi ______
5. 45° Lft - 45°Rt (FIX) Time for 90° ______
6. 45° Rt - 45° Lft (FIX) Time for 90° ______
7. 60° Ø, Time 360° ______ Vi ______ Hi ______
8. 45° Lft - 45° Rt ______ (FIX) Time for 90° ______
9. 45° Rt - 45° Lft (FIX) Time for 90° ______
   POWER APPROACH
10. 45° Lft - 45° Rt (FIX) Time for 90° ______
11. 45° Rt - 45° Lft (FIX) Time for 90° ______

d. Spiral Stability PA Hi ______ Vi ______ Pwr ______

1. Rt Ø 10° ______ o/ ______ sec. (1 2 - 2).
2. Lft 10° ______ o/ ______ sec. (1 2 - 2).

e. Phugoid (Hi CL) ______

f. Sideslips, TRIM (L) Hi ______ Vi ______

1. Rt ______ °, Fr ______ Fa ______ Fs ______ dr ______ da ______ de ______
2. Lft ______ °, Fr ______ Fa ______ Fs ______ dr ______ da ______ de ______
   TRIM (CR) Hi ______ Vi ______
3. Rt ______ °, Fr ______ Fa ______ Fs ______ dr ______ da ______ de ______
4. Lft ______ °, Fr ______ Fa ______ Fs ______ dr ______ da ______ de ______
5. D.E. with rudder (Pick up wing) ______
6. Remarks: FUEL ______

7. Stalls, Gross Weight ______ Hi Trim ______

a. CR 1.0g TRIM Vi ______ Vw ______ Vs ______ Hi ______
b. CR 2.0g TRIM Vi ______ Vw ______ Vs ______ Hi ______
c. Remarks:
d. PA 1.0g TRIM Vi ______ Vw ______ Vs ______ Hi ______
e. PA 1.5g TRIM Vi ______ Vw ______ Vs ______ Hi ______

8. Asymmetric Power - Hi ____________
   a. Climb configuration (MRP, Climb Vi, Trimmed-out)
      NTC ________ Feather ________ No. 1 Eng. Rudder Free, 2 sec.
      Decel to 1.4 Vsl ________ kts. Ø and sideslip
      (Cond. permitting check 2 out on one side)
   b. T.O. Configuration at V_{max} Gear and T.O. Flaps (168 kts.)
      Fail 1 and 2 and decelerate holding Ø = ZERO.
      Vi_{min} ________. Check Ø = 5° and SIDESLIP = ZERO.
   c. AT Min control speed fail 3 and 4, Fr ________ Fa ________
      Fs ________ TRIM OUT HANDS OFF AT 1.2 Vsl ________
   d. Remarks:

9. Boost OFF Operation Hi ________ Vi ________ Pwr ________
   a. Asymmetric Control 1 and 2 idle, 3 and 4 MRP
   b. Response ________ Fr ________ Fa ________ Fs ________
   c. Remarks:

10. Descent
   a. CR Configuration Vi ________ V/S ________
      1. Visibility ________ Attitude ________
      2. Engine operation at idle ________
      3. Pressurization, systems, etc. ________
      4. Remarks:
   b. L Configuration Vi ________ V/S ________
      1. Visibility ________ Attitude ________
      2. Engine operation at idle ________
      3. Remarks:

11. Trim Changes Trim at Placard Speed, PLF
   a. Flaps to 50 % Vi ________ Hi ________ PLF/Trim
   b. Gear DOWN Vi ________ Hi ________ PLF/Trim
   c. Flaps to 100 % Vi ________ Hi ________ PLF/Trim
   d. Power to IDLE Vi ________ Hi ________ Trim
   e. Idle to HRP Vi ________ Att ________ Trim
   f. Gear UP Vi ________ V/S ________ Trim
   g. Flaps UP Vi ________ V/S ________ Trim

12. Asymmetric Power Go-around
   a. ________ Out, Pa Vi ________ Hi ________ Pwr ________
   b. Fr ________ Fa ________ Fe ________ Response and Control
   c. Remarks:

13. General Comments Prior to Completion of Flying.

H. Approach and Landing
   1. Pre-landing check: Operating Weight ________
      Alt Setting ________ Fuel Weight ________
      W/V ________ Landing GR WT ________
      Runway ________ Best Flare Speed ________
      (Pilot Pwr and Steer) Touchdown speed ________
      (Copilot Ailerons) V_{S1} ________
   2. Traffic pattern:
      a. Visibility ________ Control
      b. Power response ________
      c. Remarks:
   3. Landing:
      a. Flare ________ Response ________ Control ________
      b. Float ________ Characteristics in ground effect ________
      c. Touchdown ________ Nose-wheel off ________ Grd Idle ________
         Reverse ________ Brakes ________ Steering ________
      d. Directional control with ailerons ________
      e. Stopping distance ________
   4. Remarks:

I. Post-flight and Shut-down
   1. Normal procedures. Ease and time to accomplish ________
   2. Coordination ________
   3. Fuel ________
   4. Flight Time ________
   5. Squawks ________

J. Re-evaluate Cockpit and A/C in General

Fighter type aircraft - Two hour flight - Plan more than can be accomplished.

EXTERNAL INSPECTION TOD START________

TOD FINISH________

Remarks:

COCKPIT EVALUATION

1. Ease of Entry ladder________________

   Steps________________
2. Location of Instruments and Controls
3. Adjustment of Seat and Controls
4. Comfort
5. Ease of Identification of:

   Switches

   Controls

   Emergency Devises

   Warning Lights
6. Egress - ground and Airborne

BEFORE STARTING CHECKS TOD________________

Remarks

Complexity:

STARTING ENGINES Fuel______ TOD______________

Complexity:

Ground Support:

Equipment______________

Personnei______________

BEFORE TAXI CHECKS TOD______________

Estimated Break-out Force

Longitudinal +______# - ________#

Lateral +______# - ________#

Directional +______# - ________#

Trim rate (Longitudinal) Aft________Sec

Fore________Sec

Flap Extension________sec Retraction___________sec

TAXIING Fuel________ TOD______________

RPM req to move________________

Visibiiity

Steering N.W.S.

Brakes

Visibility

Power required______rpm, fuel/flow__________pph

Runway temp_______°F. P.A. _____________ft.

TAKEOFF Fuel________# TOD______________

Do brakes hold in MIL PWR Yes No

Symmetry of brake release

Directional control

Rudder effective speed___________knots

Ease of rotoation

Lift-off speed______________knots

Estimated T/O distance__________feet

Gear up time_____sec Flaps up time______________sec

Trim changes Landing gear + - ________#

Flaps + - ________#

Are placards hard to exceed? Yes No

Visibility during T/O and Initial Climb

Adequacy of T/O trim setting:

Speed stability during acceleration:

CLIMB Fuel___ ____# TOD______________

Control during climb

Longitudinal

Directional

Lateral

Climb Schedule

Climb Schedule	5000 ft.	.89IMN	550
	10000 ft.	.89IMN	510
	15000 ft.	.90IMN	470
	20000 ft.	.905IMN	430
	25000 ft.	.910IMN	390
	30000 ft.	.915IMN	360
	35000 ft.	.92 IMN	320
	39000 ft.	.92 IMN	

LEVEL OFF FUEL________________# TOD___________

EASE

Attitude Change__________________________o

****

CRUISE 90 % RPM .86IMN (recommended cruise)

Start Fuel__________# TOD___________

Linear

Sideslip: Cl_β Hvy Med Lt Yes No

Cl_β Hvy Med Lt Yes No

Dutch Roll Period______________sec

Damping Hvy Med Lt

Cycles to Damp____________

CRUISE cont. 39,000 ft. .86IMN

PIO Tendency Yes No

Short Period Cycles to Damp________________

Period______________________sec

Do controls have dynamic tendency?

Yes No

Aileron Rolls: $t_{\phi 0}$

R L Adv. Yaw

1/2 deflection _____sec_____sec

Full deflect. _____sec_____sec

**********************DAMPERS OFF**********************

Linear?

Sideslip: C_{ℓ_β} Hvy Med Lt Yes No

C_{n_β} Hvy Med Lt Yes No

Dutch Roll: Period______________sec

Damping Hvy Med Lt

Cycles to damp___________

PIO Tendency Yes No

Short Period: Cycles to damp__________

Period_______________sec

********************************DAMPERS ON********************

Finish: Fuel________# TOD________________

Speed brake trim change Hvy Med Lt

Extend Push Pull

Retract Push Pull

MANEUVERING FLIGHT .90 IMN 39-35,000 ft.

Fuel________#

Initial buffet____________g

Heavy buffet____________g n_{max}__________g

Stick force Hvy Med Lt

Linear Yes No

****

ACCELERATION TO 1.20 IMN at 35,000 ft. (trim .90 IMN)

Start: Fuel________# TOD________________

NB Light L________sec R__________sec

NB Trim Change_______# Push Pull

Stick force gradient____________________

Transonic trim change__________________

Finish fuel____________# TOD________________

****

CRUISE 1.15 IMN 35,000 ft.

Start: Fuel____________# TOD________________

Linear?

Sideslips: C_{ℓ_β} Hvy Med Lt Yes No

C_{ℓ_β} Hvy Med Lt Yes No

Dutch Roll: Period____________sec

Damping Hvy Med Lt

Cycles to Damp____________

PIO Tendency Yes No

CRUISE cont 1.15 IMN 35,000 ft.

Short Period: Cycles to Damp_______-

Period__________sec

*****************************DAMPERS OFF************************

					Linear	
Sideslip	C_{ℓ_β}	Hvy	Med	Lt	Yes	No
	C_{n_β}	Hvy	Med	Lt	Yes	No

Dutch Roll:

Period__________sec

Damping Hvy Med Lt

Cycles to Damp__________

PIO Tendency Yes No

Short Period: Cycles to Damp__________

Period__________sec

***************************DAMPERS ON**************************

Aileron Rolls	t_{90} R	L	Adverse Yaw
1/2 deflection	_____sec	_____sec	_____
Full deflect	_____sec	_____sec	_____

Finish Fuel__________ TOD__________

****

SPEED BRAKE TRIM CHANGE 1.15-1-1.10 IMN

Hvy Med Lt

Extend Push Pull

Retract Push Pull

MANEUVERING FLIGHT 1.10 IMN 35-30,000 ft.

Fuel_______#

Initial buffet_______g Heavy buffet_______g

n_{max}_______g

Stick force Hvy Med Lt

Linear? Yes No

DECELERATION TO 210 knots 30,000 ft. (Long Stat)

Stick Force gradient____________________

****

CRUISE 210 knots 30,000 ft.

Start: Fuel__________# TOD________________

					Linear?	
Sideslips:	C_{ℓ_β}	Hvy	Med	Lt	Yes	No
	C_{n_β}	Hvy	Med	Lt	Yes	No

Dutch Rolls Period____________sec

Damping Hvy Med Lt

Cycles to Damp________________________

PIO Tendency Yes No

Short period: Cycles to Damp________________________

Period____________sec

############################DAMPERS OFF######################

					Linear?	
Sideslips:	C_{ℓ_β}	Hvy	Med	Lt	Yes	No
	C_{n_β}	Hvy	Med	Lt	Yes	No

CRUISE 210 knots at 30,000 ft.

Dutch Roll: Period____________sec

Damping Hvy Med Lt

Cycles to Damp________________________

PIO Tendency Yes No

Short Periods: Cycles to Damp________________________

Period______________________________

Finish: Fuel________# TOD______________

**************************DAMPERS ON***********************

AILERON ROLLS t_{90} Adverse Yaw

½ Deflection R_______sec L_______sec_______________

Full deflect R_______sec L_______sec_______________

****

MANEUVERING FLIGHT at 210 knots

Fuel__________#

Initial Buffet__________g Heavy Buffet_________g

"max" ___________g

Stick force gradient: Hvy Med Lt

****

STALLS Cruise Configuration 25,000 ft.

Fuel_________#

Cr V_W__________knots V_S__________knots

GLIDE V_W__________knots V_S__________knots

Remarks

POWER APPROACH CONFIGURATION

Gear extension__________sec

Flap extension__________sec

Asymmetric power at 155 knots

MIL RWR Rudder Force Hvy Med Lt

MAX TWR Rudder Force Hvy Med Lt

Trimability MIL MAX

STALLS: Fuel__________

V_W__________knots V_S__________knots

Remarks:

Trim at 160 knots

					Linear?	
Sideslip:	$C\ell_\beta$	Hvy	Med	Lt	Yes	No
	$C\ell_\beta$	Hvy	Med	Lt	Yes	No

Dutch Roll Period ________________ sec

Damping Hvy Med Lt

Cycles to Damp__________

PIO Tendency Yes No

Short Period Cycles to Damp_____ _____

Period________________sec

*************************DAMPERS OFF**********************

Dutch Roll Period_______________ sec

Damping Damping Hvy Med Lt

Cycles to Damp__________

PIO Tendency? Yes No

Short Period: Cycles to Damp__________

Period_______________sec

*************************DAMPERS ON*************************

AILERON ROLLS t_{90} Adverse Yaw

1/2 Deflection R_____sec L_____sec___________

Full Deflect R_____sec L_____sec___________

****

ACROBATICS

Loop

Immelman

Barrel Roll

INSTRUMENTS

Holding at 20,000 Ft.		250 knots	90-92 %
Penetration	S/B	270 knots	90 %
Initial	Clean	220 knots	94 %

Low Cone gear, 86 %, flaps, 155 knots

LANDING

Normal traffic pattern 60 % flaps

Single engine go-around closed pattern

Full stop Full flaps

Touchdown speed_______knots' marker_____________

****

TAXIING Fuel________# TOD________________

Engine acceleration Idle to mil__________sec

Turning radius___________feet

Re-evaluate cockpits

ENGINE SHUTDOWN

Check servicing for turn-around

Time_________

Oil___________qts

Hydraulic fluid_______qts

LOX____________liters

This data card is also for a fighter type aircraft - a one hour mission to evaluate the aircraft for a pilot training mission.

TOD____________ ________________beside A/C

START Procedure

F Flow____________RPM____________F Flow__________

Before Taxi Check

TOD__________________

TAXI

Power to roll________________Brakes S NS

Nosewheel steering Turn Rad.____________

NWS Off Brake turn________________

Canopy Operation

Visibility

TOD____________________________

LINE UP

Brakes Mil Pwr____________

Pump one brake

Engine Acc Time____________

RPM____________EGT______________FF__________

Throttle friction S NS

FUEL L__________ R______________

TOD__________________

TAKEOFF

Brake release

A/B light

NWS rel at Rudder Eff A/S__________

CONTROL FORCES L M H__________ lbs

NW LIFT OFF__________

T.O. ROLL__________ ft A/S__________

GEAR UP__________ sec. FLAPS UP__________ sec

Trim Changes__________

Noises

Press. Sys

Accelereleration Rotation

CLIMB

Schedule .9 to 35M

Control

Trim

Visibility

Dampers

35M Time__________ Fuel L__________ R__________

Throttle Mil Level Off

TOD__________________________________

SUPERSONIC

A/B Light time___________

TRIM CHANGES

STABILITY

DAMPERS	PULSE	CYCLE	TIME
ON	Elev		
	Rud		
OFF	Elev		
	Rud		

45' Roll

ONE ENGINE IDLE

Wind Up Turn to g Max.

A/S__________ "g"__________

Stick force gradient

Buffet

FUEL L__________ R____________

TOD______________________________________

TURNING PERFORMANCE 300 Kts ________________ sec

Zoom to Slow A/C

PWR STALL WARN __________ STALL ____________

230 Kts. Flight Roll

STABILITY

DAMPERS	PULSE	CYCLE	TIME
ON	Elev		
	Rud		
OFF	Elev		
	Rud		
Sideslip	6' Apx.		

CUT ONE ENGINE

EMERGENCY GEAR EXTENSION __________________ sec

AIRSTART

170 knots Flaps Down

Aileron Power

Cycle gear Flaps up TRIM

FUEL L ___________ R ________________________

TOD ____

DIVE 450 Kts 12M

CLOVERLEAF

BARREL ROLL

IMMELMAN

Level at 20 M inbound to VOR

200 Kts F FLOW____________

250 Kts F FLOW____________

300 Kts F FLOW____________

HIGH CONE

240 Kts. Gear Flaps Dive Brakes

1 g stall

200 Kts.

STABILITY Check

STALL RIGHT TURN 190 Kts

Clean up A/C 275 Kts. turn to ILS

350 Kts. Speed Brakes Decelerate

ISL Gear, Flaps, D/C 170 Kts

TOD____________________________________

SINGLE ENGINE GO-AROUND

SINGLE ENGINE TOUCH AND GO

RE-ENTER

PITCH OUT

NO FLAP LANDING

TRIM CHANGES

TAXI

AFTER LANDING CHECK

SHUTDOWN

10.5 GENERAL TECHNIQUES

The cockpit evaluation can normally be made while getting cockpit time prior to the first flight. MIL SPEC 203E specifies the standard cockpit arrangement for the various types of aircraft in considerable detail and should be used as a guide in making the cockpit evaluation. However, a summary of some of the points to note may prove helpful. These include: ease of entry, comfort, adjustment of seat and controls, location of basic flight instruments, size and legibility of instruments, accessibility of switches and controls, ease of identification of switches and controls, location and identification of emergency switches and controls, methods of escape (both on the ground and airborne), and general impression of cockpit layout.

Several points should be observed and recorded during the start and while preparing the aircraft for flight. These should be weighed against the aircraft's mission requirements. An all-weather interceptor, for example, should be capable of fast, uncomplicated starts to meet its alert and scramble requirements. Starts for other types may not be so critical; however, no starting procedure should be unnecessarily complex or confusing. Evaluation of the start should include: complexity of start, time to prepare for start, time to start, external power and ground support equipment required, ground personnel required, and time from start to taxi. The system checks and normal procedure requirements from start to taxi should also be evaluated.

An evaluation of the ground handling characteristics can be made while taxiing. How much power is required to start moving and to taxi at the desired speed? Is braking action required to prevent taxiing too fast? Is the visibility adequate? Is the directional control satisfactory? Is the braking action satisfactory? What is the turning radius of the aircraft? Does the aircraft require any auxiliary equipment such as removable wheels, escape ladders, etc? Is there any problem with clearing obstacles with any part of the aircraft?

The takeoff distance may be difficult to determine without assistance from outside personnel, but an estimate should be made using whatever aid is available such as runway distance markers. Use the recommended takeoff procedure, don't try to make a maximum performance takeoff. The normal ground roll will be of more interest than the minimum possible. Some of the other points to note in the takeoff include: ability of brakes to hold in military power, directional control during ground roll, rudder effective speed, nose lift-off speed, visibility after nose up and during initial acceleration and climb, force required to raise nose, any over-controlling tendencies, airborne speed, adequacy of recommended takeoff trim settings, time to retract gear and flaps, trim changes with retraction of gear and flaps, any tendency to exceed gear or flap speed limitations, effectiveness of trimming action during acceleration, and any distracting noises or vibrations.

The in-flight techniques differ very little from the techniques used in flying quantitative tests. However, it generally is not necessary to be as precise in holding airspeeds and altitudes. To do so would only waste time because differences caused by variations of a few hundred feet in altitude or a few knots in airspeed will not be qualitatively discernible so far as qualitative information is concerned. This is not an indorsement for being lax in flying the aircraft. Just don't waste time with preciseness that will not contribute to the evaluation of the aircraft. If speeds are critical, such as in the climb or in the pattern, then hold them as closely as possible. Otherwise, use good judgment in determining how close to an aim condition it is necessary to be and fly accordingly.

If the climb rate of the aircraft is relatively slow, it may be possible to get some stability information in the climb, i.e., stick pulses, sideslips, etc. Most present day fighter aircraft climb so rapidly that this may not be practical. If so, just record climb performance information (time, fuel, and indicated speed) at intervals of approximately 5,000 feet. Start the time at brake release. Intercept the climb schedule at a comfortable altitude and attempt to fly the recommended schedule precisely. Continue the climb only as far as is compatible with the objective of the flight. Unless climb performance is of primary importance, this will probably be to the altitude selected for the first series of investigations. General aircraft characteristics should be observed during the climb. How difficult is it to maintain the recommended climb schedule? Are the control responses smooth?; too fast?; too slow?; compatible? Is visibility adequate? Is there any buffet?; vibration or excessive noise? Are the ventilation and pressurization systems satisfactory? Are the normal procedures required complicated or excessively distracting? If dampers or other artificial stability devices are provided, check the applicable characteristics with them "ON" and "OFF".

The altitude selected for the first series of stability investigations may be at the tropopause since this is where the aircraft will probably have its best performance. However, if the designed operating altitude is considerably above this level it may be advisable to select an altitude at or near the aircraft's operating altitude. The stability maneuvers performed will be essentially the same at all the altitudes and airspeeds selected. These should be sufficiently spaced to assure descernible qualitative differences in the aircraft's characteristics.

The stability characteristics investigated should include: longitudinal and directional static stability, longitudinal and directional dynamic stability, aileron rolls, and maneuvering flight at several different airspeeds and altitudes. An investigation of the transonic trim changes also should be made. All the dynamic characteristics should be checked with the stability augmentation devices, if any, both "ON" and "OFF". With proper planning these investigations can be made in a minimum amount of time. The longitudinal static stability can be checked while accelerating to V_{max}, for instance. Once at V_{max}, the aircraft can be trimmed for approximately hands-off flight and the static directional stability checked by entering a steady sideslip out to maximum rudder deflection (if the aircraft is cleared to that limit). The periods of the dynamic modes can be timed using a stop watch or counting the seconds. Estimate the cycles to damp completely or to one-half amplitude as the case may be for all the modes.

Approach the aileron rolls cautiously. Make several partial deflection rolls before making any full deflection rolls. The time to reach 90 degrees of roll and the time to roll 360 degrees can be estimated using a stopwatch or again by counting the seconds. It is advisable to make rapid reversals of ailerons and other rolling maneuvers if these can be expected in operational use of the aircraft. The rolling characteristics should also be checked in accelerated flight as well as 1 g flight.

After completion of investigations at V_{max}, a windup turn to limit load factor can be made to check the maneuvering stability of the aircraft. Then zoom back to the original altitude and repeat these investigations at the second airspeed. The other altitudes and airspeeds can be checked in the same manner. Any differences resulting from the altitude or speed changes should be noted.

Stalls should be approached with caution if the aircraft is cleared for such a maneuver, and investigated in all configurations and types of entry. Determine the approximate stall warning margin, what defines the warning and the stall, and the aircraft characteristics in the stall and the recovery. If possible, determine the best method of breaking the stall and altitude loss in recovery from several points in the stall.

If possible, check the tactical mission capability of the aircraft. Simulated dive bombing runs or labs maneuvers could be made for a strategic fighter; for example. All the information obtainable will be helpful in writing an accurate and comprehensive report.

Fly the traffic pattern as recommended and, if fuel permits, make a go-around on the first pass. Note the power response, power required in the pattern, airspeed control and sink rate, trim changes with gear and flap extension, trimming action, buffet with gear extension, and general aircraft feel in the pattern. On the go-around, recheck the trim changes with gear and flap retraction and with drag device reaction. Don't forget to look at engine out characteristics if time and fuel permit. On the first landing in the aircraft it is probably not advisable to attempt to get the minimum landing roll. Make a normal touchdown and use normal braking action (use the drag chute if provided). Note the touchdown speed, the effects of any crosswind, directional control, nose lowering speed, etc. As with the takeoff, the normal landing roll is of more importance than the minimum possible.

While taxiing back to the parking area, review the flight, re-evaluate the cockpit, and attempt to determine whether the aircraft will perform its design mission and is safe and comfortable to fly. The opinion with everything fresh in mind is probably the most accurate possible. Continue this review of the flight immediately after leaving the aircraft. Put everything remembered about the flight and the impressions of the aircraft down on paper. Do this immediately and before talking to anyone about the airplane or the flight. Waiting or discussing points with other people may alter first hand impressions or cause important aspects of the flight to be forgotten.

10.6 DATA REDUCTION

10.6 The data reduction will consist of writing a comprehensive report on everything learned about the aircraft. A narrative form is normally used for qualitative reports. Comparisons with other aircraft can be used to assist in describing the aircraft. Care should be taken however, to insure that only aircraft familiar to most readers are used for comparison. Otherwise the comparison will mean nothing to them.

Keep in mind the purpose of the qualitative evaluation while writing the report. Mere figures are normally not enough to describe the stability of the aircraft, particularly on a qualitative evaluation since the data obtained are very limited. Analyze the aircraft characteristics in light of its ability to perform its design mission, give opinions of the aircraft's ability to do the job and support these opinions with the facts obtained on the evaluation flights. Comment on anything personnally disliked but be objective in condemning any shortcomings. Recommendations for specific changes in the aircraft are to be included in the report. The exact manner in which the aircraft should be fixed should not be specified or recommended. The test pilot's job is to evaluate the existing hardware and state what should be changed. It is then the manufacturer's responsibility to determine how to make the necessary changes.