

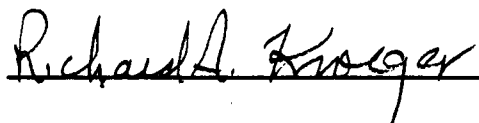
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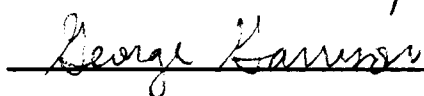
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AN IMPROVED TECHNIQUE FOR NORMALIZING
ENGINE-OUT MINIMUM-CONTROL
AIRSPEED DATA

A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville

Barry Gerard McCarthy

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To Mom and Dad--thanks for everything!

ABSTRACT

Due to the expenses associated with flight testing today's aircraft, there is present an ever-increasing demand for improved, more efficient test methods. In light of these demands, this study proposes a new, more direct method for determining the minimum-control airspeed for aircraft operating under asymmetric power.

This new technique for normalizing minimum-control airspeed data is applied to a light, twin-engine, propeller-driven aircraft with the critical engine inoperative. In order to determine accuracy and efficiency, this method is compared with the extrapolation technique presently in use for determining standardized values.

Results show the technique proposed in this study to be equally precise and more efficient when compared to present-day methods. Unnecessary flight hours and time spent reducing data is substantially reduced while still maintaining today's standards for accuracy.

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NOMENCLATURE

ϕ	Bank angle
β	Sideslip angle
δ_a	Aileron deflection
δ_r	Rudder deflection
σ	Sidewash angle
σ_d	Density ratio (ρ/ρ_0)
Y_β	Sideforce due to sideslip
Y_{δ_a}	Sideforce due to aileron deflection
Y_{δ_r}	Sideforce due to rudder deflection
l_β	Rolling moment due to sideslip
l_{δ_a}	Rolling moment due to aileron deflection
l_{δ_r}	Rolling moment due to rudder deflection
N_β	Yawing moment due to sideslip
N_{δ_a}	Yawing moment due to aileron deflection
N_{δ_r}	Yawing moment due to rudder deflection
W	Weight
l_T	Rolling moment due to asymmetric thrust
F_N	Net thrust
l_c	Lateral distance from engine's center line to aircraft's center line
a_v	Lift curve slope of vertical tail ($dC_L/d\alpha$)
η_v	Vertical tail efficiency factor (q_v/q_∞)
q	Dynamic pressure
S_a	Area of aileron section
S_v	Area of vertical tail

S	Reference area
l_v	Horizontal distance between vertical tail's center of pressure and airplane's center of gravity
D_w	Windmill drag
L	Lift
V_{MCA}	Minimum-control airspeed
TER	Trailing edge right
RWD	Right wing down
MP	Manifold pressure
OAT	Outside air temperature
E	Equilibrium

CHAPTER I

INTRODUCTION

The most critical situation pilots of multi-engine aircraft face is an engine failure. The asymmetric thrust configuration brings two problems to pilots: 1) regaining control immediately following the failure of one or more engines, and 2) maintaining control in steady flight with one or more engines inoperative. Solutions to these problems are functions of pilot performance and aircraft capabilities [1].

The general aviation community has been led to believe that an aircraft operating on asymmetric power can be safely and successfully flown if the airspeed is maintained at or above the minimum control airspeed, V_{MCA} . This is a dangerous misconception due to the effects of altitude, weight, center-of-gravity location, bank angle and sideslip angle on suggested minimum airspeeds. In order for an aircraft to be effectively flown during an engine failure, a pilot must know precisely the minimum-control airspeed applicable to the operating conditions and how the prior-mentioned parameters affect that airspeed [2].

The lateral-directional control problem associated with asymmetric power is dependent on the aircraft's ability to counteract the induced yawing and/or rolling moments with pilot-control inputs and aircraft stability [3]. This control problem is critical for aircraft operating with the highest available thrust--sea-level takeoff. More extensive studies, such as that conducted by Bervin [2], indicate that the severity of the problem increases as weight decreases and as center of gravity moves aft. This configuration defines the maximum operational value

for V_{MCA} and establishes the directional-control requirement for multi-engine aircraft.

Since physical and safety limitations prohibit testing where control is critical, flight-test organizations rely on extrapolating their data to sea level to determine the minimum-control airspeed at minimum weight, V_{MCA_S} [4]. While yielding good results, this method is costly in terms of flight hours and time required to reduce data. These associated high costs demonstrate the need for a more efficient method for determining the weight-normalized value for minimum-control airspeeds.

This thesis examines the feasibility of using a more direct technique to determine normalized values from minimum-control airspeed data. This proposed technique, while taking into account the effects of weight and bank angle, will allow for the determination of V_{MCA_S} from a minimal amount of flight data. In order to determine accuracy and efficiency, this investigation shall compare this method with the previously-mentioned extrapolation technique.

CHAPTER II

BACKGROUND

For a multi-engine aircraft to maintain level unaccelerated flight with one or more engines inoperative, all forces and moments acting on the aircraft must be in balance. Assuming adequate power is available for level flight, the longitudinal and vertical forces and moments will balance leaving only those for the lateral-directional case to be considered.

Assuming no acceleration, the lateral-directional equations of motion describing an aircraft operating on asymmetric thrust can be stated as [2]

$$Y_{\beta} \beta + Y_{\delta_a} \delta_a + Y_{\delta_r} \delta_r + W \sin \phi = 0, \quad (1)$$

$$l_{\beta} \beta + l_{\delta_a} \delta_a + l_{\delta_r} \delta_r + l_T = 0, \quad (2)$$

$$N_{\beta} \beta + N_{\delta_a} \delta_a + N_{\delta_r} \delta_r - F_N l_c = 0. \quad (3)$$

Lateral Forces

Sideforce Due to Sideslip, Y_{β}

Although the wing and wing-body interference may modify Y_{β} , the primary contribution is from the vertical tail. When an airplane is subjected to a sideslip angle β , the vertical tail experiences a change in angle of attack, α_T . This change of α_T , caused by the sidewise inflow of air, generates a force rotating the aircraft in a direction

to reduce β . The magnitude of this sideforce can be expressed in non-dimensional form as [3]

$$C_{Y_{\beta_v}} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} . \quad (4)$$

Sideforce Due to Lateral-Control Deflection, Y_{δ_a}

Deflection of lateral-control devices located close inboard on the wings disturbs the airflow around the fuselage enough to generate a sideforce. In addition, a dihedral angle causes the lift vectors resulting from deflected ailerons to generate a lateral component. Both lateral forces are generally very small and often neglected in many studies.

Sideforce Due to Rudder Deflection, Y_{δ_r}

Deflecting the rudder-control surface causes the vertical tail-rudder combination to act as a cambered airfoil. This airfoil generates a sideforce whose magnitude may be expressed in nondimensional form as

$$C_{Y_{\delta_r}} = a_v \tau_v \eta_v \frac{S_v}{S_w} . \quad (5)$$

Side Force Due to Bank Angle, $W \sin \phi$

During a bank, an aircraft experiences a gravitational component of sideforce. The magnitude of this sideforce is dependent on the magnitudes of bank angle and weight, and is expressed nondimensionally by replacing the weight term with the lift coefficient. Since this

component of sideforce acts through the aircraft's center of gravity, $C_L \sin \phi$ generates no rolling or yawing moments [3,5,6].

Lateral Moments

Rolling Moment Due to Sideslip, l_β

Commonly referred to as "dihedral effect," the rolling moment induced by sideslip is vital to the handling qualities of an aircraft. l_β is a measurement of an aircraft's tendency to roll in a direction opposite to an imposed sideslip. A stable dihedral effect results in l_β being slightly negative since a right sideslip ($\beta > 0$) would induce a left roll ($l < 0$). Geometric dihedral, wing sweep, placement of the wing on the fuselage, and the vertical tail are major contributors toward l_β . The resulting rolling moment generated by the vertical tail can be approximated nondimensionally using

$$C_{l_\beta} = -a_v \left(1 - \frac{d\sigma}{d\beta}\right) \eta_v \frac{S_v}{S_w} \frac{Z_v}{b}. \quad (6)$$

Interested readers can find more detailed methods in References [7] and [8].

Rolling Moment Due to Lateral-Control Deflection, l_{δ_a}

Lateral control is achieved by deflecting the ailerons to induce a rolling moment by altering the spanwise-lift distribution. Hence, l_{δ_a} or lateral-control power is a measurement of the rolling moment created by a given aileron deflection. As evident from its nondimensional form,

$$C_{\ell_{\delta_a}} = a_a \left(\frac{S_a}{S_w} \right) \frac{\bar{y}}{b}, \quad (7)$$

lateral-control power is a function of the aileron airfoil section, the area of the aileron in relation to the area of the wing, and the location of the wing's center of pressure. While this relationship is a good approximation, $C_{\ell_{\delta_a}}$ can be determined more accurately using strip integration or from determinations of the spanwise lift distribution [8,9].

Rolling Moment Due to Rudder Deflection, ℓ_{δ_r}

This cross-coupling effect is caused by the center of pressure of the fin-rudder combination being located above the longitudinal axis; a rudder deflection produces a sideforce which generates a rolling moment. The magnitude of this moment may be expressed nondimensionally as

$$C_{\ell_{\delta_r}} = C_{Y_{\delta_r}} \frac{z_v}{b}. \quad (8)$$

Rolling Moment Due to Asymmetric Thrust, ℓ_T

This rolling effect results from the live-engine wing being partially immersed in high energy slipstream air, while the dead-engine wing suffers a degradation in lift due to a slight loss in dynamic pressure. While the wings remain at the same angle of attack, there is more lift on the live-engine wing which induces a rolling moment, ℓ_T . This moment increases with angle of attack and power setting and is the largest term in the roll equation [3,4,10].

Directional Moments

Yawing Moment Due to Sideslip, N_β

Referred to as "weathercock stability," N_β is the most predominant term in Equation (3) and represents the directional stability of the aircraft. Since contributions from the wing, fuselage, nacelles, and propeller normal forces are all destabilizing, the vertical tail has to be designed to offset these influences and provide some level of directional stability. For positive directional stability, the design of the vertical tail requires a positive sideslip (relative wind right) to induce a positive yawing moment (nose right). The contribution provided by the vertical tail is a function of its ability to generate the necessary sideforce. This contribution can be expressed in nondimensional form as

$$C_{N_{\beta_v}} = a_v \left(1 - \frac{d\sigma}{d\beta} \right) \eta_v \frac{S_v}{S_w} \frac{l_v}{b_w} . \quad (9)$$

Total directional stability can be approximated by summing the contributions from the wings, fuselage, nacelles, propeller normal forces and the vertical tail [8].

Yawing Moment Due to Lateral-Control Deflection, N_{δ_a}

Upon deflection, lateral-control surfaces normally generate directional moments. On most aircraft the generated yawing moments oppose the initial aileron inputs such that a left roll results in a right yaw. From the definition for induced drag, this adverse yaw is due to increased drag resulting from increased lift on the wing with the

aileron trailing edge down. For spoiler systems, the resulting yaw is proverse and is attributed to the increase in profile drag [4].

Yawing Moment Due to Rudder Deflection, N_{δ_r} :

The ability for an aircraft to generate yawing moments is a function of the sideforce capable of being produced by the vertical tail's control surface. The magnitude of this yawing moment is a measure of rudder effectiveness or "rudder control power" and is expressed non-dimensionally as [3,10]

$$C_{N_{\delta_r}} = - a_v \tau_v \eta_v \frac{S_v}{S_w} \frac{l_v}{b_w} . \quad (10)$$

Yawing Moment Due to Asymmetric Thrust, $F_{N_c} l_c$

Flight on asymmetric power is characterized by a yawing moment generated by the failed engine. In coefficient form, the magnitude of this moment can be expressed as

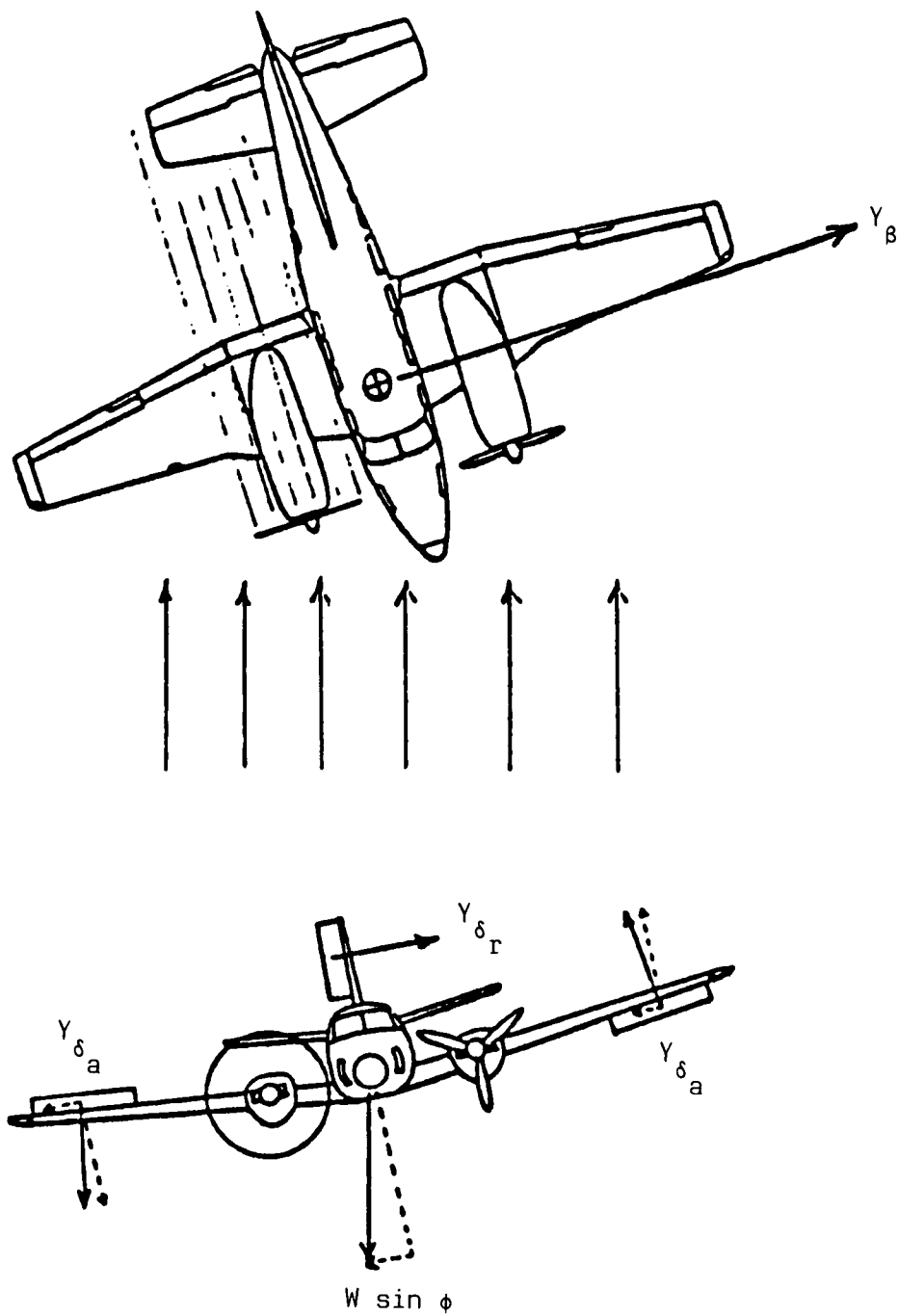
$$C_{N_a} = \frac{F_{N_c} l_c}{qSb} \quad (11)$$

where

$$F_N = F_2 + D_w . \quad (12)$$

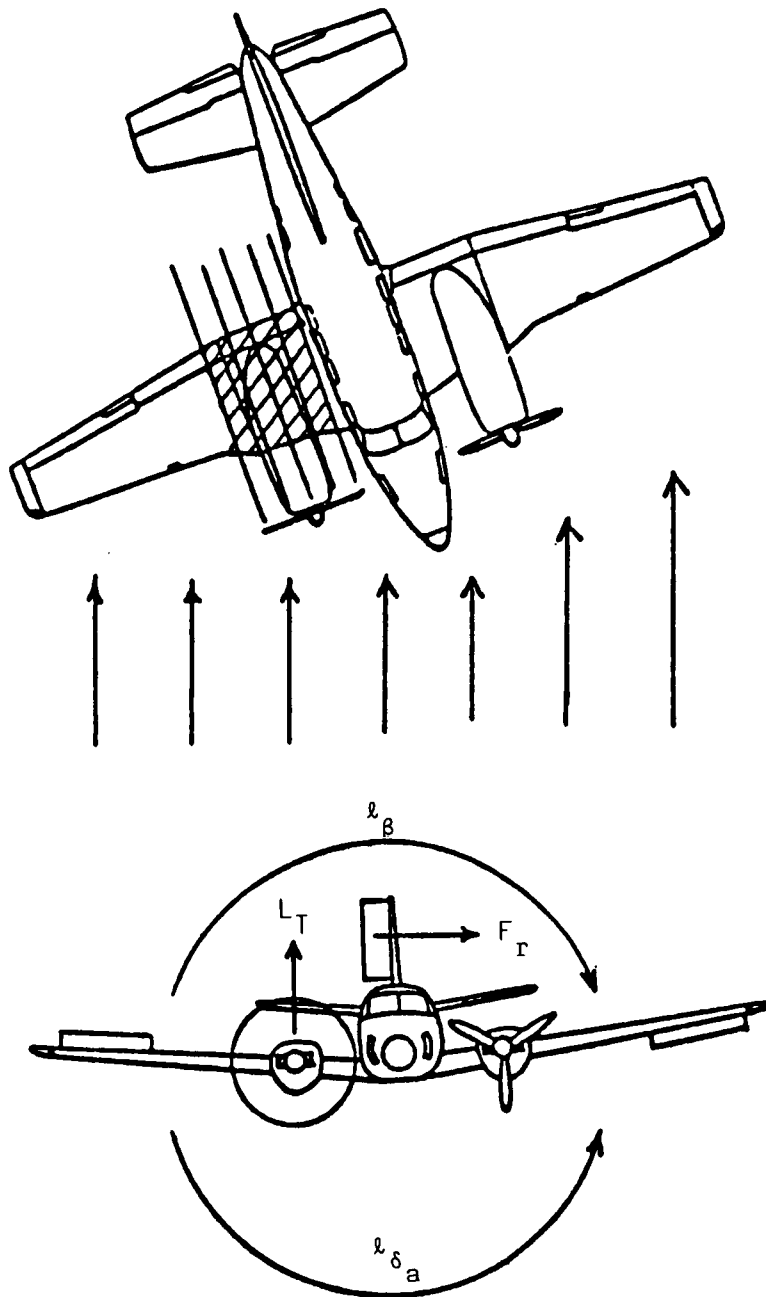
Note that the asymmetric power yawing-moment coefficient increases with an increase in operative engine thrust, the distance of operative engine from aircraft's center of gravity, and a decrease in airspeed [2,4].

Figures 1-3 summarize the forces and moments involved during a flight on asymmetric power with the port engine inoperative. Approximations for these forces and moments have been previously discussed using



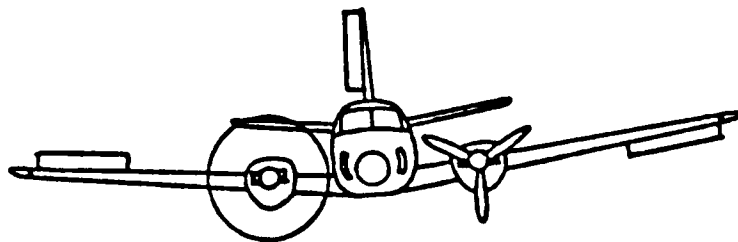
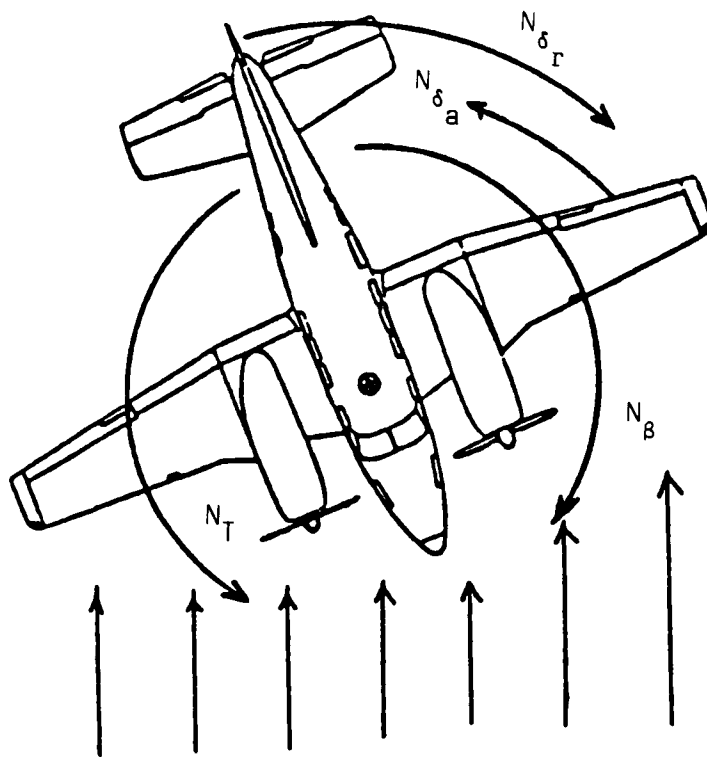
$$Y_\beta + Y_{\delta_a} \delta_a + Y_{\delta_r} \delta_r + W \sin \phi = 0$$

Figure 1. Lateral Forces Involved During Flight on Asymmetric Power with Zero Sideslip and Five Degrees Angle of Bank.



$$\ell_\beta \beta + \ell_{\delta_a} \delta_a + \ell_{\delta_r} \delta_r + \ell_T = 0$$

Figure 2. Lateral Moments Involved During Flight on Asymmetric Power.



$$N_\beta + N_{\delta_a} \delta_a + N_{\delta_r} \delta_r - N_T = 0$$

Figure 3. Directional Moments Involved During Flight on Asymmetric Power.

conventional nondimensional form. These coefficients are defined as products of derivatives and changes in appropriate variables. For example,

$$\frac{l}{qSb} = C_l$$

and

$$\frac{\partial C_l}{\partial \beta} \beta = C_{l_\beta} \beta.$$

Defining each variable in this manner yields the following set of lateral-directional equations:

$$C_{Y_\beta} \beta + C_{Y_{\delta_a}} \delta_a + C_{Y_{\delta_r}} \delta_r + C_L \sin \phi = 0 \quad (13)$$

$$C_{l_\beta} \beta + C_{l_{\delta_a}} \delta_a + C_{l_{\delta_r}} \delta_r + C_{l_T} = 0 \quad (14)$$

$$C_{N_\beta} \beta + C_{N_{\delta_a}} \delta_a + C_{N_{\delta_r}} \delta_r - C_{N_a} = 0. \quad (15)$$

Directional-Control Problem

In most flight conditions it is desired to maintain zero sideslip. If an aircraft possesses positive weathercock stability and is truly symmetrical, then it will tend to fly in this condition. However, yawing moments may act on an airplane as a result of slipstream rotation, a nonsymmetrical flow field associated with turning flight, or, in the case of this study, an asymmetrical thrust configuration [1].

An aircraft's ability to operate under asymmetric power is a function of the aircraft's lateral-directional control capabilities. (Longitudinal control is usually not critical and is maintained by variations in pitch angle, angle of attack, and elevator deflection [11].) Hence, the flying qualities during an engine failure are dependent on the aircraft's ability to maintain equilibrium in response to increasing asymmetric moments. This equilibrium is established by balancing the forces and moments in Equations (13) through (15) with various combinations of bank and sideslip angles, i.e., aileron and rudder deflections.

If a pilot chooses to maintain equilibrium flight using zero sideslip, an expression for the necessary rudder requirement may be obtained from Equation (15) and written as

$$\delta_{r_{E_\beta=0}} = \frac{(F_2 + D_w) \ell_c}{qSb C_{N_{\delta_r}}} . \quad (16)$$

For simplification, $C_{Y_{\delta_a}}$, $C_{N_{\delta_a}}$, and $C_{\ell_{\delta_r}}$ are assumed to be zero.

With zero sideslip there is no sideforce; however, some bank angle must be utilized to counteract the lateral force generated by the rudder input. From Equation (13), this bank angle can be expressed as

$$\phi_{E_\beta=0} = - \frac{C_{Y_{\delta_r}} \delta_r}{C_L} = - \frac{C_{Y_{\delta_r}}}{C_L} \left(\frac{F_N \ell_c}{qSb C_{N_{\delta_r}}} \right) ,$$

or

$$\phi_{E_\beta=0} = - \left(\frac{C_{Y_{\delta_r}}}{C_{N_{\delta_r}}} \right) \left(\frac{F_N}{W} \right) \left(\frac{\ell_c}{b} \right) . \quad (17)$$

Several important relationships may be obtained from Equations (16) and (17). For equilibrium using zero sideslip, the rudder requirement increases with asymmetric thrust, decreasing airspeed, and lateral engine placement from CG. It is inversely proportional to rudder control power. Considering a negative asymmetric moment (port engine operative) and $\beta = 0$, for trim, a negative rudder deflection (trailing edge right) and a positive bank angle (right wing down) will be required. This bank-angle requirement is independent of velocity and, in most cases, is fairly small, $\phi \leq 5^\circ$ [3].

If conditions exist forcing a pilot to maintain zero bank angle, the rudder requirement necessary for equilibrium, steady heading flight, as derived from a simultaneous solution of the sideforce and yawing moment equations, can be expressed as

$$\delta_{rE} \phi = 0 = \frac{\frac{F_{N_c} \ell_c}{qSb}}{C_{N_{\delta_r}} \left[1 - \frac{C_{N_\beta} C_{Y_{\delta_r}}}{C_{Y_\beta} C_{N_{\delta_r}}} \right]}$$

or

$$\delta_{rE} \phi = 0 = \frac{\delta_{rE\beta=0}}{\left[1 - \frac{C_{N_\beta} C_{Y_{\delta_r}}}{C_{Y_\beta} C_{N_{\delta_r}}} \right]} \quad (18)$$

Typically, the denominator in Equation (18) is significantly smaller than one, thereby substantially increasing the rudder requirement over the zero sideslip case. This increase in δ_r is necessary to balance

the sideforce due to sideslip (C_{Y_β}) and results in a higher minimum-control airspeed as illustrated in Figure 4 [12].

Under a zero bank angle configuration, there will be a sideslip angle required to balance the sideforce equation. From Equation (13) this angle is expressed

$$\beta_{E_{\phi=0}} = - \frac{C_{Y_{\delta_r}}}{C_{Y_\beta}} \cdot \delta_r.$$

Hence, to balance the same asymmetric condition, this configuration requires an increase in rudder deflection over the previous case and a negative sideslip angle. Such increases only serve to decrease the performance during an already dangerous situation [11].

If necessary, flight under asymmetric power can be maintained without deflecting the rudder. For equilibrium flight, the sideslip and bank-angle requirements necessary to balance the asymmetric moments may be determined from Equations (15) and (13) and expressed as

$$\beta_{E_{\delta_r=0}} = \frac{F_{N_c} \ell_c}{qSb C_{N_\beta}} \quad (19)$$

$$\phi_{E_{\delta_r=0}} = - \frac{C_{Y_\beta}}{C_L} \beta. \quad (20)$$

For a negative asymmetric moment the resulting sideslip is positive and relatively large, particularly at low airspeed, high-operative engine thrust, and low directional stability. The bank angle is into the operative engine and also quite large, approximately 15 degrees.

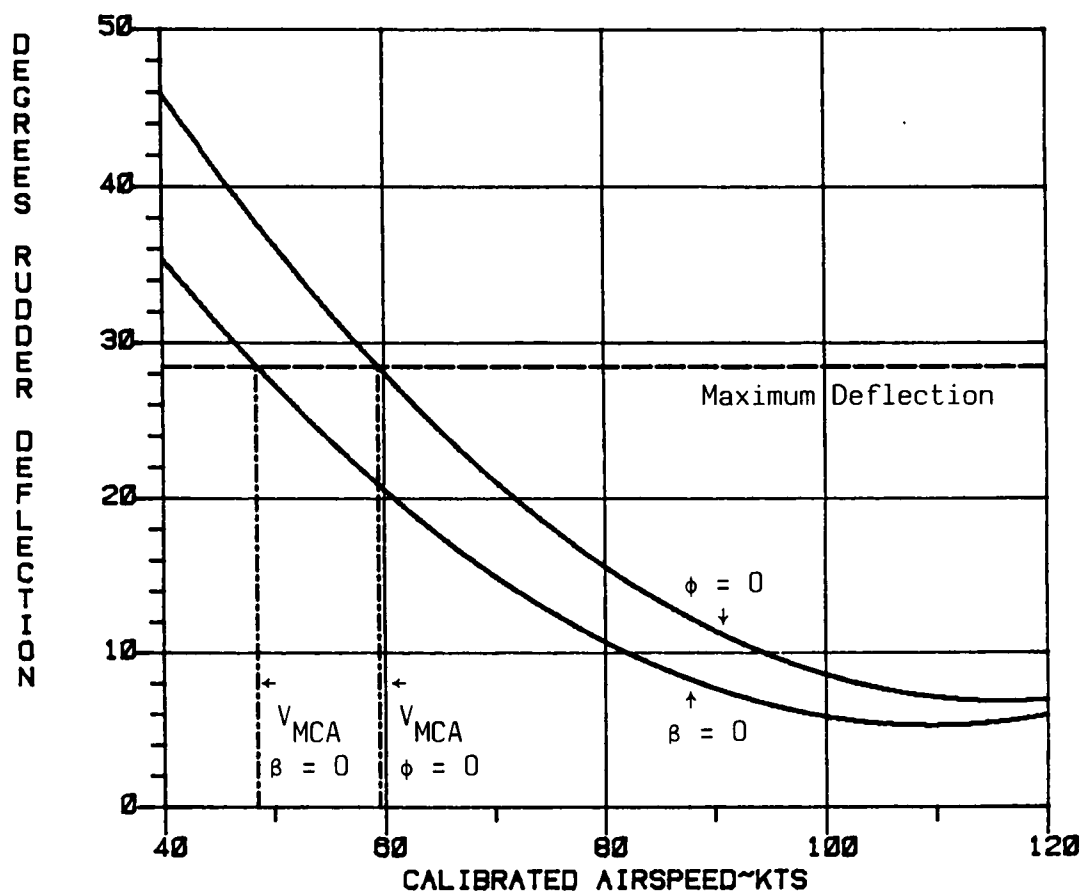


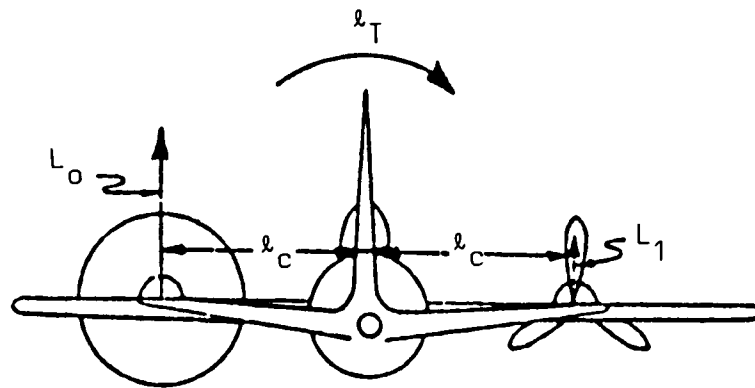
Figure 4. V_{MCA} as Defined by Maximum-Control Surface Deflection for $\beta = 0$ and $\phi = 0$ Flight Configurations.

The large requirement for bank angle is necessary to balance the side-force due to sideslip and to counteract the rolling moment due to asymmetric thrust and that induced by the large sideslip angle. The large β and ϕ configuration results in more drag than the previous two cases, and it is potentially dangerous since large sideslip may cause the vertical tail to stall. Also, 15 degrees of bank angle will generate a large side acceleration due to gravity making the aircraft even more difficult to fly [3,9,12].

From the previous analysis, holding zero sideslip while maintaining some bank angle yields the most optimum performance. Since FAR 23.149 allows a bank angle up to five degrees during testing under asymmetric power, the flight configuration chosen for this study holds $\beta = 0$ and $\phi = 5^\circ$.

Lateral-Control Problem

Generally, lateral controllability under asymmetric power is not as critical as the directional control requirements. However, for propeller-driven airplanes, the differences in slipstream over the wings may generate large rolling moments. The higher dynamic pressure over the live-engine wing will generate unbalanced lift vectors inducing a rolling moment toward the inoperative engine (Figure 5). If the wings are almost completely immersed in propeller slipstream, lateral-control requirements may determine minimum-control airspeeds. The combination of sideslip from the operative engine and positive dihedral effect complicates lateral control further. To counteract the rolling moment



$$l_T = L_0 l_c - L_1 l_c$$

$$= l_c (L_0 - L_1)$$

$$C_{l_T} = \frac{l_T}{qSb}$$

$$C_{l_T} = \frac{C_L l_c}{b} \frac{(L_0 - L_1)}{w}$$

Figure 5. Rolling Moment Induced by Asymmetric Power.

due to asymmetric thrust (ℓ_T) and sideslip, the required aileron deflection can be determined from the rolling moment equation and is equivalent to [3,4]

$$\delta_{aE} = - \frac{1}{C_{\ell_{\delta a}}} \left(\frac{C_{L\ell c}}{bW} (L_0 - L_1) + C_{\ell_{\beta}} \beta \right).$$

Performance Problem

The reduction in power available due to an engine failure (Figure 6), coupled with increased drag, greatly reduces range capability, service ceiling and climb performance. This degradation in excess power allows little mismanagement of attitude or configuration during takeoff and in the initial climb phase on asymmetric power. The severity of the performance problem increases with altitude due to the reduction in power available caused by decreased atmospheric pressure. Operating in this reduced envelope greatly depends on pilot experience and skill. Decision speeds and distances plus any options available to the pilot after failure of one or more engines are functions of this reduced level of performance [7].

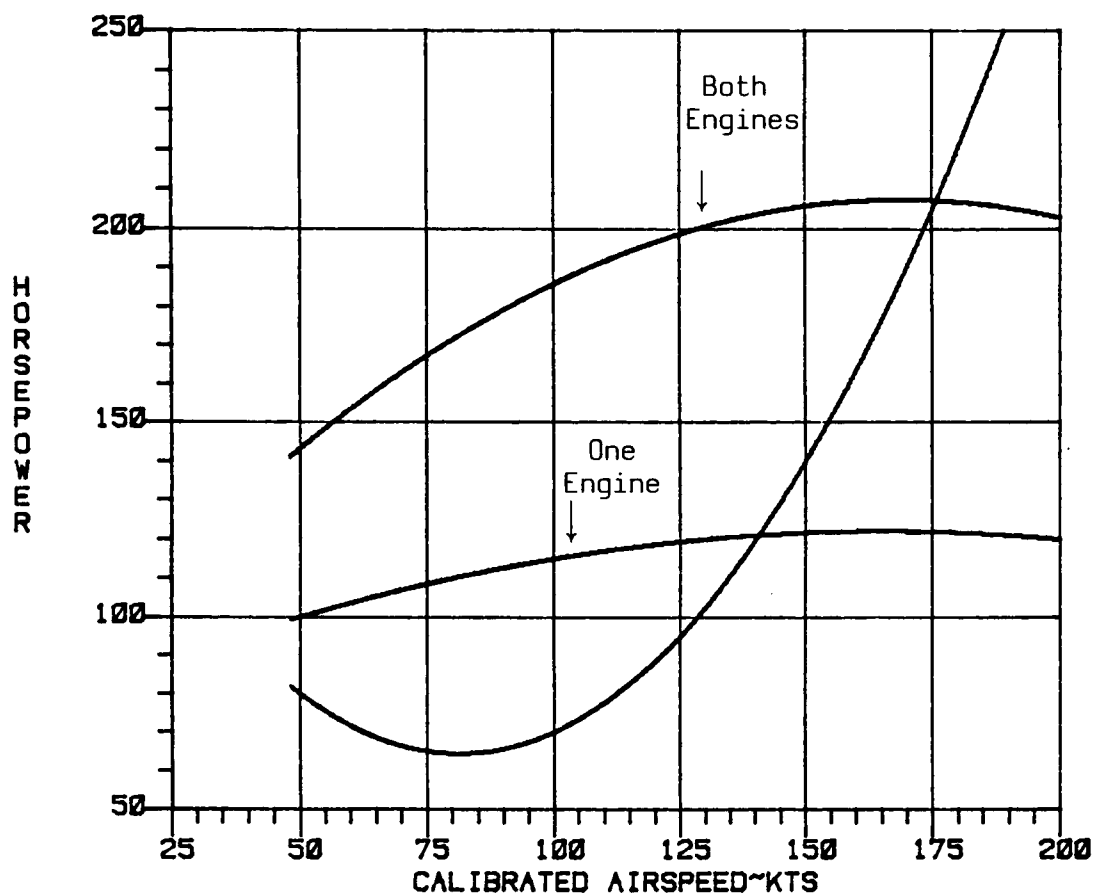


Figure 6. Generalized Performance Curves Illustrating Loss in Power Available Due to an Engine Failure on Twin-Engine Aircraft.

CHAPTER III

TEST PROCEDURES AND TECHNIQUES

Flight on asymmetric power is characterized by the control problem experienced as a result of the yawing moments generated by the engine-out configuration. From the previous section, the magnitude of the yawing moment can be stated nondimensionally [4],

$$C_{N_a} = \frac{(F_2 + D_W) l_c}{qSb} .$$

In straight, level, unaccelerated flight,

$$C_{N_a} = \frac{(F_2 + D_W) l_c C_L}{Wb} . \quad (21)$$

The lowest speed at which aerodynamic control can be maintained over an aircraft with the critical engine inoperative is defined as the minimum-control airspeed, V_{MCA} . (On multi-engine propeller-driven aircraft, the left outboard is critical due to propeller fin effect [13].) This airspeed is determined by available lateral- or directional-control deflections and/or the control forces involved necessary to counteract the rolling or yawing moments [1,9]. Regardless of the limiting factor(s), from Equation (21) the critical asymmetric condition which establishes the maximum operational value for V_{MCA} is low altitude, high thrust setting, and low weight. Since flight testing at this condition is not recommended due to safety limitations, an extrapolation technique is presently used to obtain the desired information.

Normal testing procedures consist of establishing stabilized flight with all engines operating at maximum take-off power then shutting down the critical engine while maintaining steady, straight flight without reducing the thrust on the live engine or operating propeller or trim controls. Stabilize at predetermined airspeeds (3-5 kt. increments) and record sideslip angle, control surface deflections and, if necessary, control forces. The minimum-control airspeed can be determined for the test condition by maintaining equilibrium flight until maximum control (directional control is usually critical) deflection or force is reached, or by extrapolating the test data to maximum surface deflections (Figures 7-9) [1].

Due to increased engine thrust at lower altitudes, V_{MCA} increases as altitude decreases. Holding weight and CG constant while repeating the above procedure at various altitudes provides an extrapolation to sea level V_{MCA} for the test configuration (Figure 10).

Since the critical condition is defined at sea-level minimum weight, the above procedures must be repeated for different weights allowing for an extrapolation to the standardized value for V_{MCA} (Figure 11). A complete summary of the test requirements and data-reduction procedures is given in Figures 12 and 13.

Because this method involves failing the critical engine prior to approaching the minimum conditions, the final result defines a static V_{MCA} . A dynamic engine failure would require V_{MCA} to increase due to several factors including pilot reaction time and skill. Hence, the dynamic V_{MCA} equals the sum of the static value and an increment of airspeed necessary to handle the dynamic case as determined through qualitative evaluation [4].

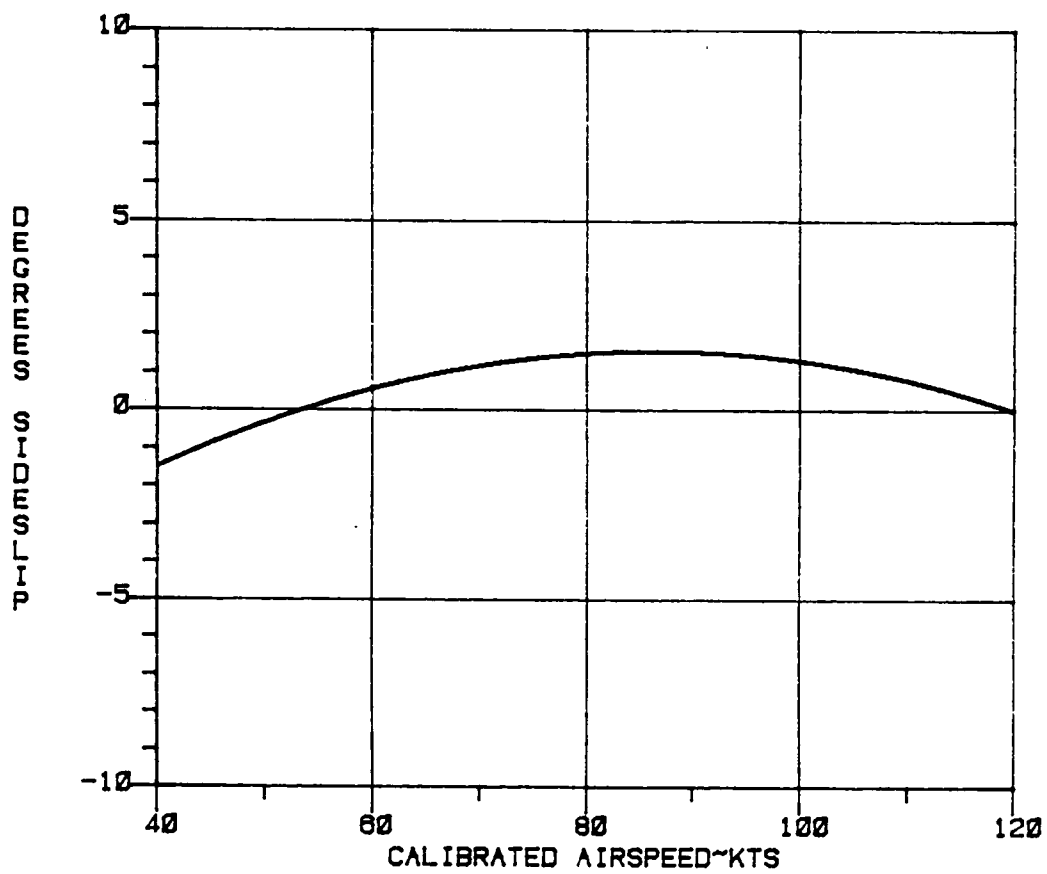


Figure 7. Sideslip Versus Calibrated Airspeed for Determining V_{MCA} for One Altitude.

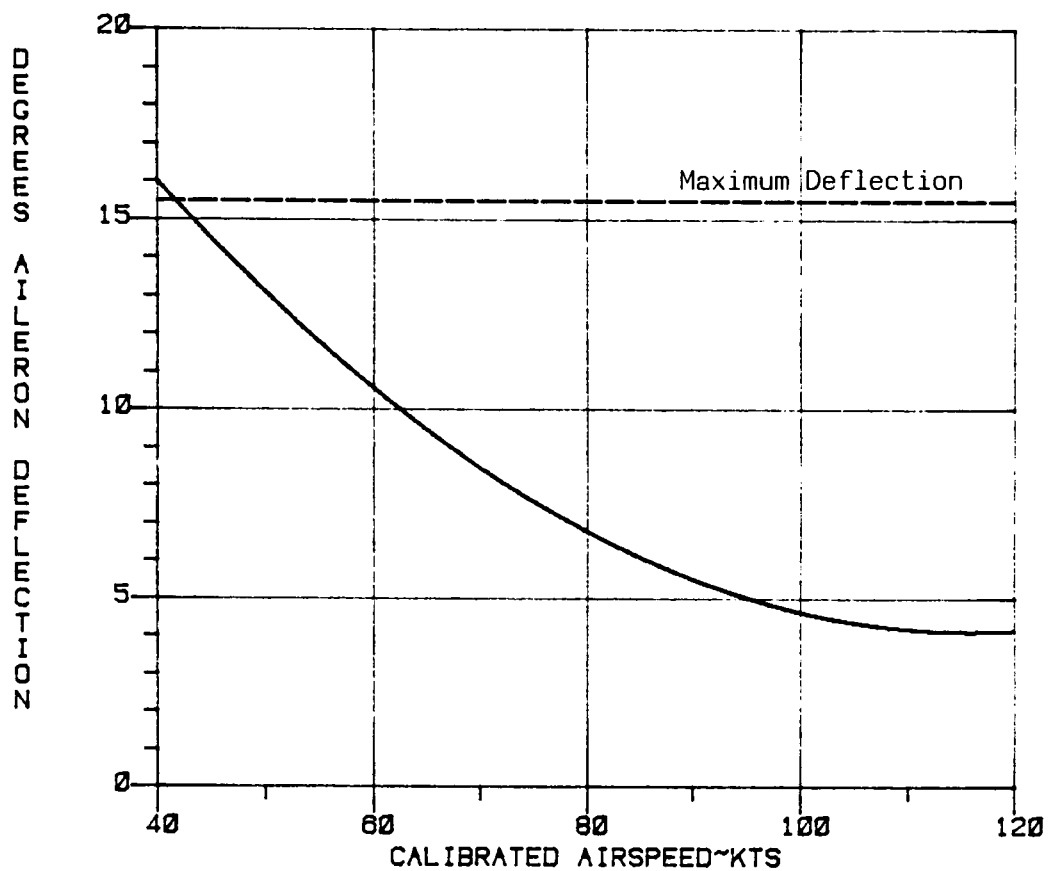


Figure 8. Aileron Deflection Versus Calibrated Airspeed for Determining V_{MCA} for One Altitude.

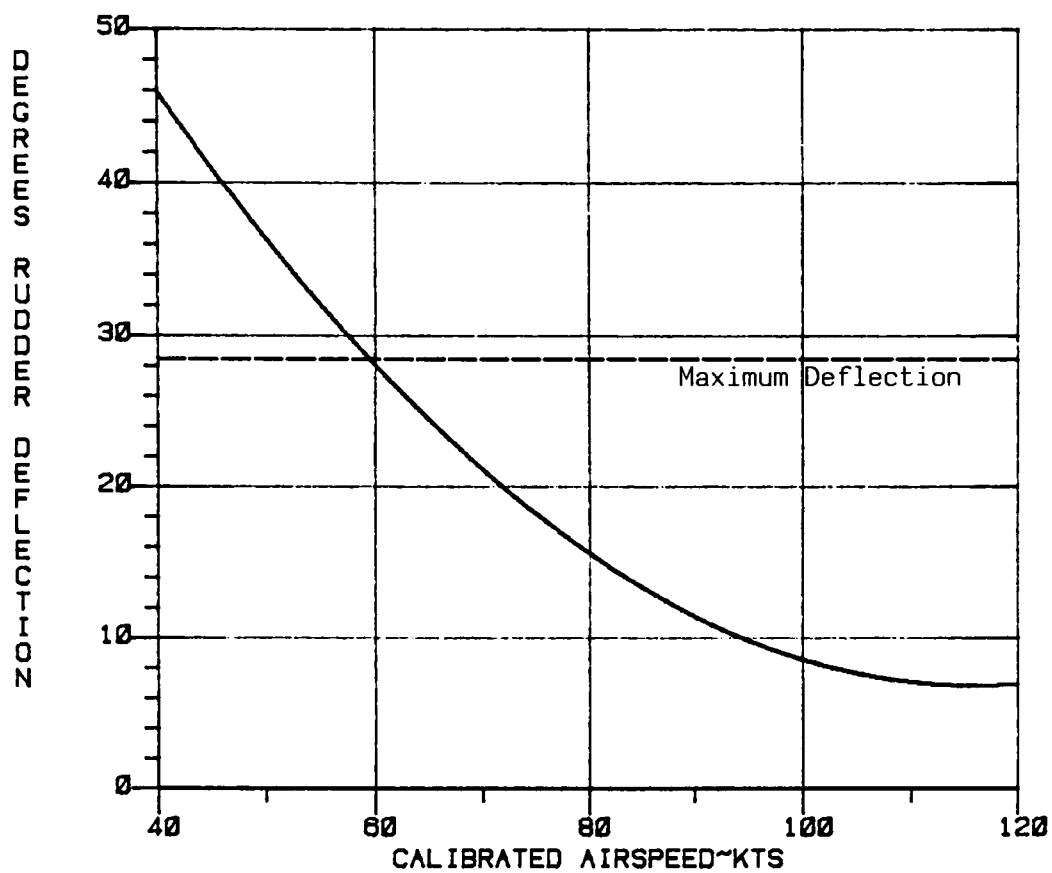


Figure 9. Rudder Deflection Versus Calibrated Airspeed for Determining V_{MCA} for One Altitude.

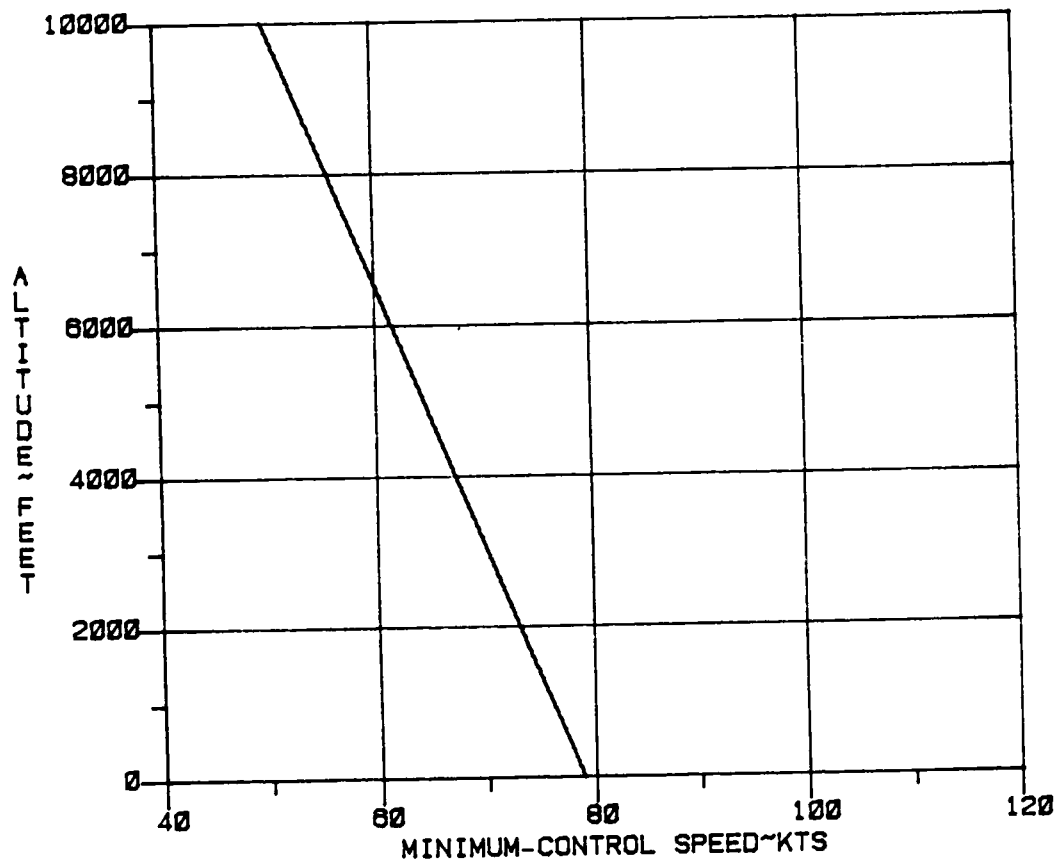


Figure 10. Altitude Versus Minimum-Control Speed for One Weight Extrapolated to Determine Sea-Level V_{MCA} .

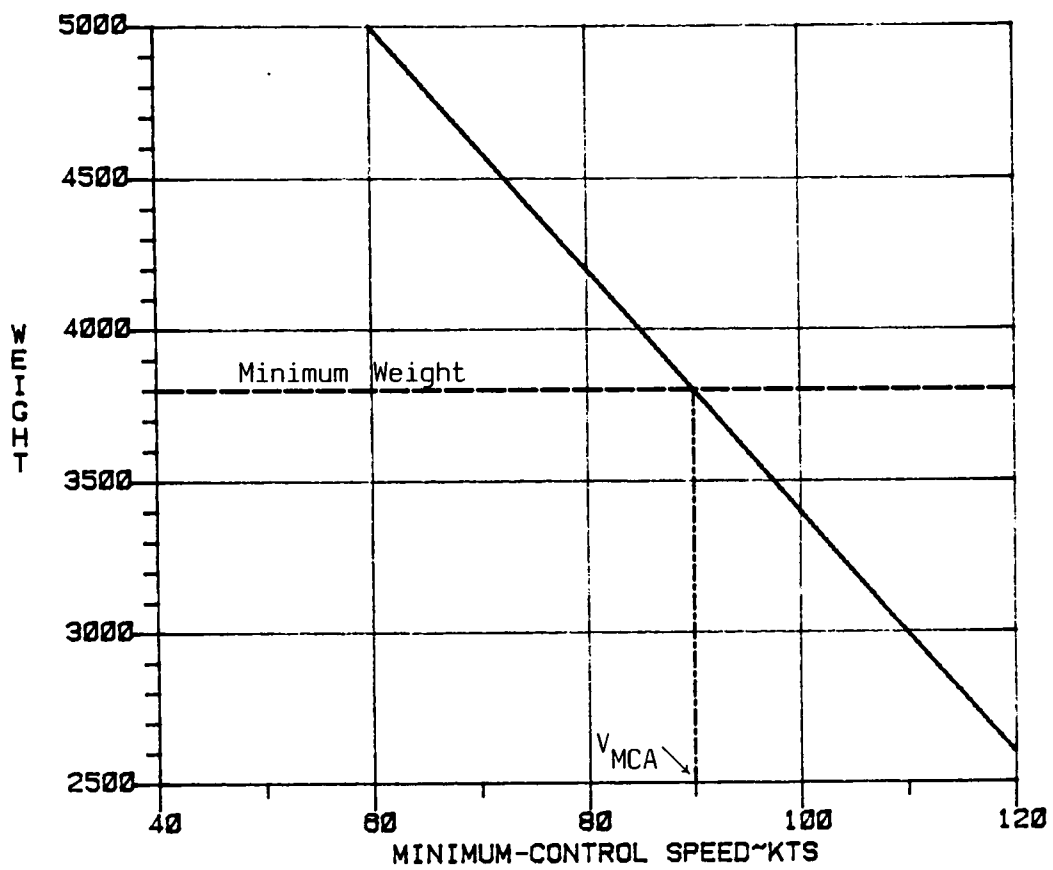


Figure 11. Weight Versus Minimum-Control Speed for Defining Standardized Value for V_{MCA} .

Stabilize aircraft at
maximum take-off power

Shut down critical engine
while maintaining zero degrees
sideslip and five degrees
angle of bank

Reduce airspeed in small
increments and record: β , δ_a ,
 δ_r , V_i , RPM, MP_i , OAT, and ϕ

Repeat for at least three
different indicated pressure
altitudes

Repeat for at least
three different weights

Figure 12. Flight Test Requirements for Engine-Out, Minimum-Control Speed Determination.

Correct indicated data for
instrument and position errors

Plot for each altitude: β vs V_C ,
 δ_a vs V_C , δ_r vs V_C , and
extrapolate to maximum-control
surface deflections (V_{MCA_t})

Plot extrapolated test values for
 V_{MCA} versus calibrated airspeed
to obtain V_{MCA} at sea level

Plot test weight versus
calibrated airspeed and extrapolate
to minimum-weight V_{MCA}

Figure 13. Data Reduction Procedure for Engine-Out, Minimum-Control Speed Determination.

CHAPTER IV

THEORY

The scope of this investigation is to prove the reliability of an apparently more efficient method for normalizing minimum-control airspeed data. This technique was originally derived by Les Bervin, an Aerospace Engineer and Flight Test Pilot for the Federal Aviation Administration. According to Bervin, by defining an airframe's ability to produce yawing moments in response to any applied lateral forces and equating the airframe's maximum capability with those forces produced by the power plants, the standardized conditions can be determined.

A plot defining an airframe's ability to provide directional restoring moments can be generated through an evaluation of its lateral-directional stability and control derivatives (Equations (13) through (15)). These derivatives may be estimated using the conventional methods discussed earlier, determined in flight or obtained through wind-tunnel testing; the later generally being the best since one parameter at a time can be varied while holding the others constant [2].

In 1971, NASA conducted a comprehensive evaluation of the aerodynamic characteristics of an executive-type jet by placing a full-scale Learjet 24D in the Ames 40- by 80-foot wind tunnel. The following values for the lateral-directional derivatives are taken from Reference [14] assuming 10 degrees angle-of-attack and center-of-gravity location at 31.5 percent mean aerodynamic chord.

$$\begin{array}{lll}
C_{Y_{\beta}} = -0.0115 & C_{Y_{\delta_a}} = 0.0003 & C_{Y_{\delta_r}} = -0.0024 \\
C_{L_{\beta}} = 0.0025 & C_{L_{\delta_a}} = 0.0031 & C_{L_{\delta_r}} = -0.0003 \\
C_{N_{\beta}} = 0.0026 & C_{N_{\delta_a}} = -0.0003 & C_{N_{\delta_r}} = 0.0013
\end{array}$$

Substituting these values into Equations (13) through (15), and eliminating one of the five unknowns by assuming maximum rudder deflection ($\delta_{r_{MAX}} = 30^\circ$) yields the following set of equations:

$$-0.0115 \beta + 0.0003 \delta_a = 0.072 - C_L \sin \phi, \quad (22)$$

$$-0.0025 \beta + 0.0031 \delta_a = 0.009, \quad (23)$$

and

$$0.0026 \beta - 0.0003 \delta_a - C_{N_a} = -0.039. \quad (24)$$

A plot of the asymmetric directional-moment coefficient as a function of $C_L \sin \phi$ can now be generated for the airframe by assigning values for the lateral-lift component and solving the three equations simultaneously for C_{N_a} . This line defines the airframe's ability to provide restoring moments in response to any applied lateral forces (Figure 14).

Similar plots defining the asymmetric moment produced by the engines can be obtained from engine-performance data. General Electric's model specifications for the CJ610-6 engine are given in Table 1 and utilized in plotting performance capabilities on a sea-level standard day, and at 5,000 and 10,000 feet on a standard day (Figure 15).

$C_L \sin \phi$	β	δ_a	C_{N_a}
0.00	-6.32°	-2.18°	0.0232
0.02	-4.54°	-0.76°	0.0274
0.04	-2.77°	+0.67°	0.0316
0.06	-0.99°	+2.10°	0.0358
0.08	+0.79°	+3.53°	0.0400
0.10	+2.56°	+4.96°	0.0442
0.12	+4.34°	+6.38°	0.0484
0.14	+6.12°	+7.81°	0.0526

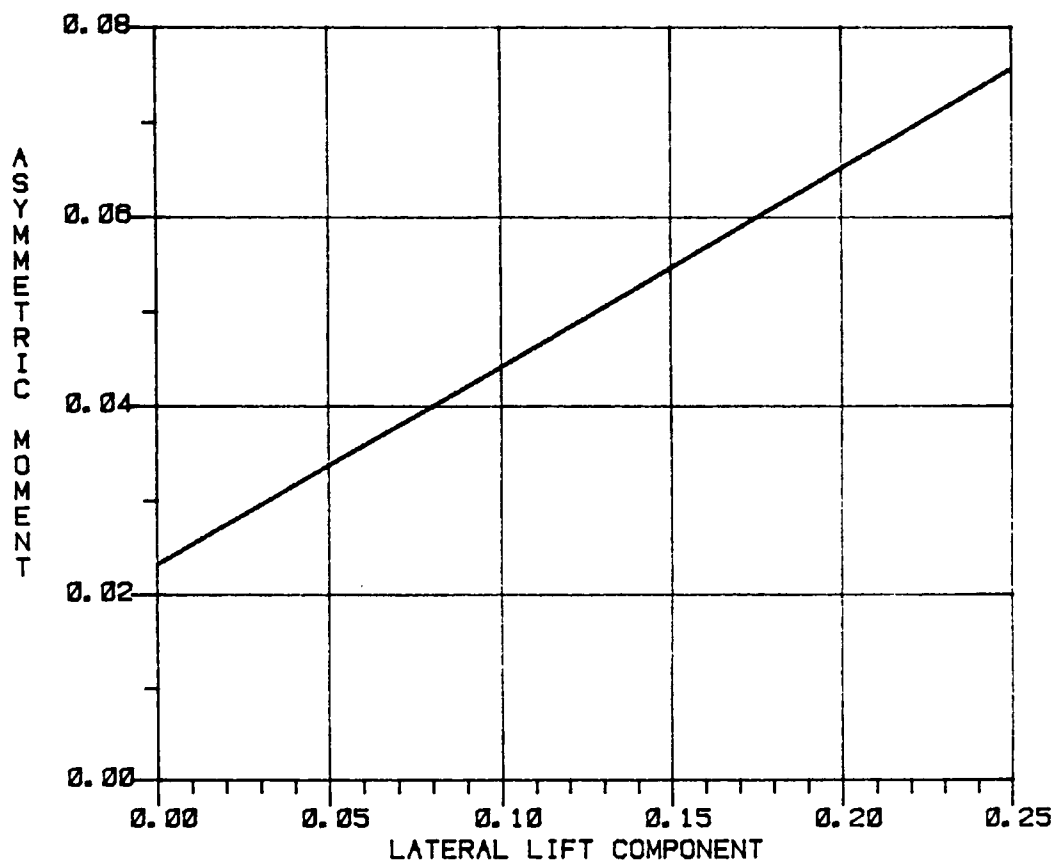


Figure 14. Asymmetric Moment (C_{N_a}) Versus Lateral Lift Component ($C_L \sin \phi$) Defining the Airframe's Restoring Capabilities as Derived from Equations (22-24).

Table 1. General Electric's Model Specifications for CJ610-6 Engine Performance (takeoff thrust).

V_e	F_N	C_{N_T}	$C_L \sin \phi$
<u>Sea Level</u>			
0	2950	∞	∞
80	2820	0.0598	0.156
100	2780	0.0377	0.100
150	2710	0.0163	0.044
200	2690	0.0091	0.025
300	2650	0.0040	0.011
<u>5,000 Feet</u>			
80	2410	.0510	0.156
100	2390	.0324	0.100
130	2350	.0189	0.059
150	2350	.0142	0.044
200	2350	.0080	0.025
<u>10,000 Feet</u>			
80	2100	.0445	0.156
100	2100	.0285	0.100
130	2130	.0171	0.059
150	2150	.0130	0.044
200	2200	.0075	0.025

The intersection of the airframe curve and the sea-level standard engine curve defines the critical condition where the applied asymmetric moment is exactly counteracted by aircraft stability. Past this point, aerodynamic control is lost and regaining necessary airspeed is difficult, if not impossible (Figure 15). Hence, from this intersection the sea-level value for minimum-control airspeed can be determined.

Referring to Figures 15 and 16, and given a minimum weight of 9,000 lbs., V_{MCA_s} and F_{N_s} can be solved for as follows:

$$C_L \sin \phi_s = \frac{W \sin \phi_s}{qS}, \quad (25)$$

hence

$$V_{MCA_s} = \sqrt{\frac{295 W/S \sin \phi_s}{C_L \sin \phi_s}} = 86 \text{ kts} \quad (26)$$

and

$$C_{N_s} = \frac{F_{N_s} l_c}{qSb}, \quad (27)$$

which yields

$$F_{N_s} = 7.37 C_{N_s} V^2 = 2806 \text{ lbs.} \quad (28)$$

Therefore, sea-level minimum-control airspeeds can be determined by solving for the point of intersection between the airframe and standard engine curves. More generally, by defining the necessary curves

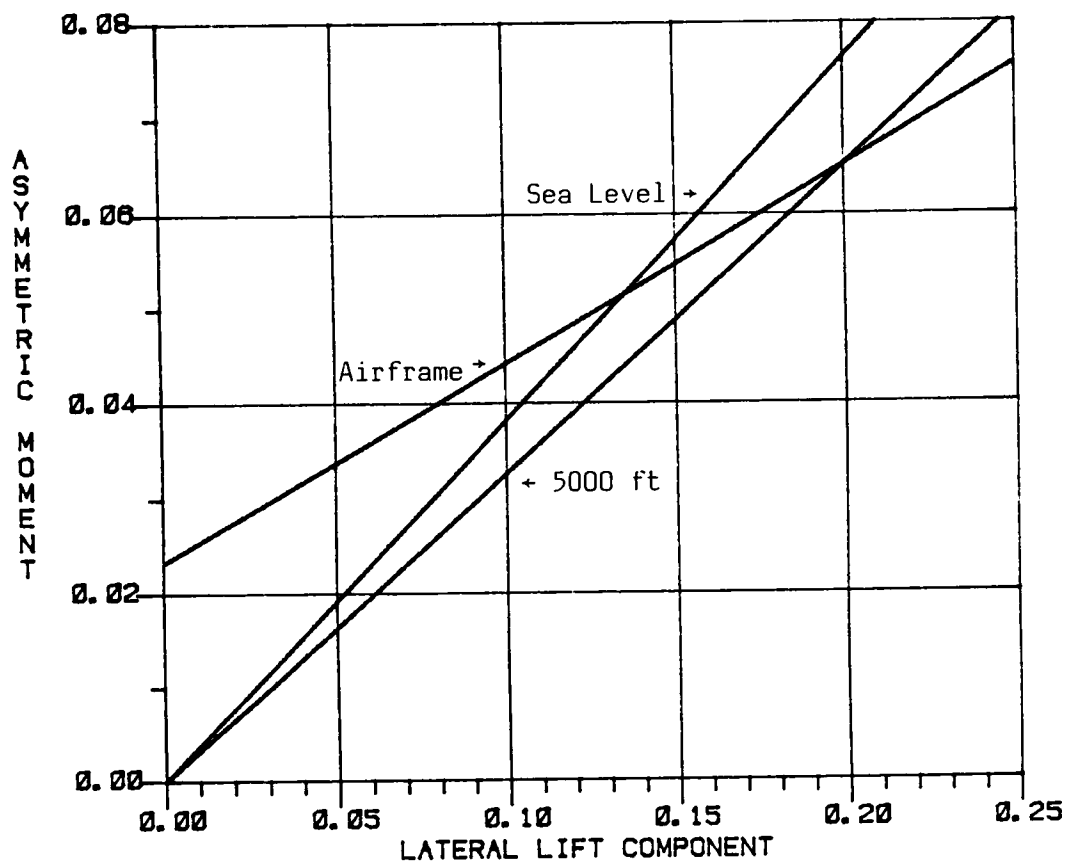


Figure 15. Asymmetric Moment (C_{N_a}) Versus Lateral Lift Component ($C_L \sin \phi$) for Airframe and CJ610-6 Engines.

	Wing	Horz. tail	Vert. tail
Aspect ratio	5.02	4.0	—
Taper ratio	0.507	0.469	0.493
Area, sq m	21.51 (23.77)	5.62 (6.00)	34.48 (37.08)
Sweep, deg (25% $\bar{c}$)	13	25	40
Airfoil section	64A109	64A008	64A010

All dimensions in meters (feet)

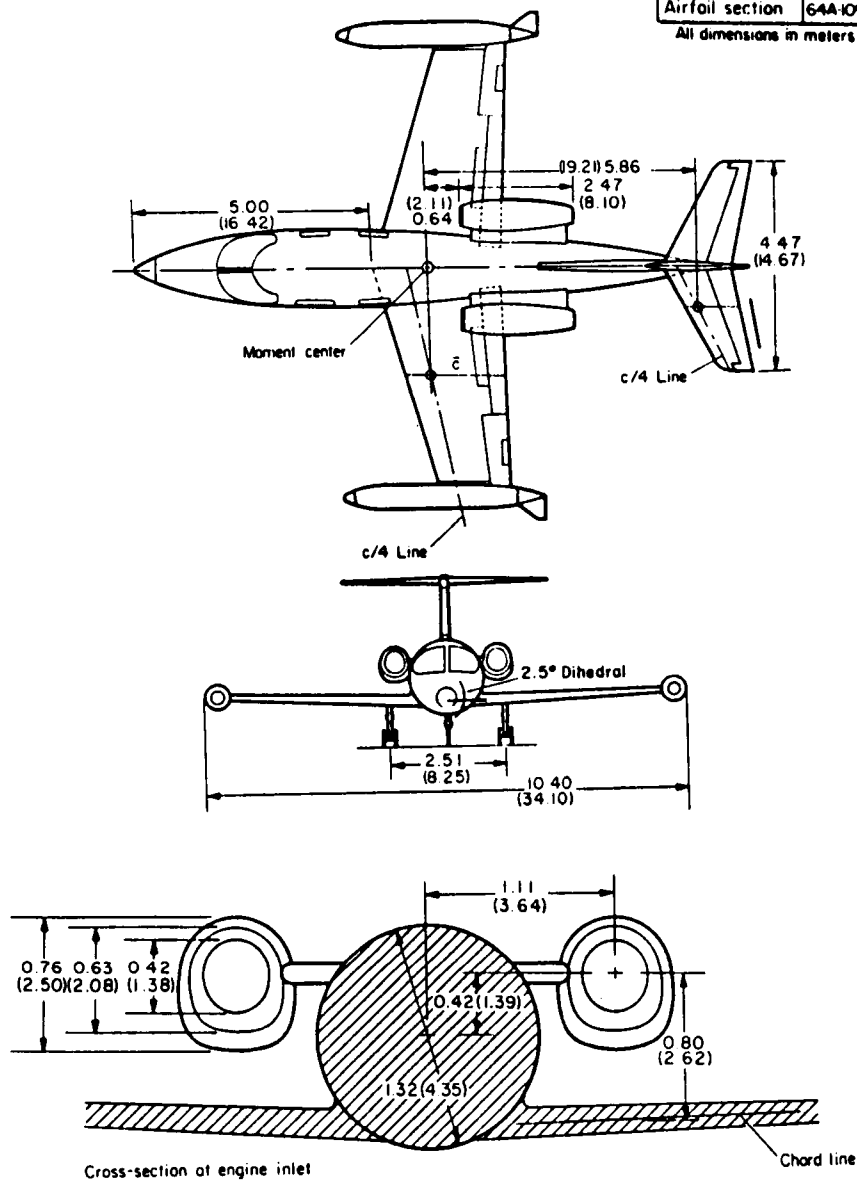


Figure 16. Necessary Dimensions for Learjet 24D as Taken from NASA TN-6573.

using flight-test data, V_{MCA_s} can be solved directly by equating the equations for the two curves.

From test data,

$$W_s = W_t = 9,000 \text{ lbs}$$

$$\phi_s = \phi_t = 5^\circ$$

$$F_{N_t} = 2,400 \text{ lbs}$$

$$F_{N_s} = 2,806 \text{ lbs}$$

$$V_{MCA_t} = 70.9 \text{ kts.}$$

Since minimum-control airspeed is defined at the point where the applied asymmetric moment is exactly balanced by the airframe, the test value for V_{MCA} can be utilized to define the airframe's capability. Doing so enables the following linear relationship to be formed:

$$C_{N_t} - C_{N_o} = \frac{dC_N}{dC_L \sin \phi} (C_L \sin \phi_t - C_L \sin \phi_o).$$

From Equations (25), (27), and Appendix A,

$$C_{N_o} = 0.0648 - 0.2 (0.1984) = 0.0251.$$

Hence, the airframe curve can be defined as

$$C_{N_a} = 0.0251 + 0.2 C_L \sin \phi.$$

If the standard engine curve is assumed to intersect the origin, defining this curve depends only on establishing one standard point. Since the yawing-moment coefficient and the lateral-force component are both functions of velocity, V_{MCA_t} can establish the proper coordinates,

$$C_L \sin \phi_s = \frac{295 W_s \sin \phi_s}{V_{MCA_t}^2 S} = 0.1984$$

$$C_{N_s} = \frac{295 F_{N_s} l_c}{V_{MCA}^2 S b} = 0.076.$$

Thus, defining the standard engine curve,

$$C_{N_s} = \left(\frac{C_N}{C_L \sin \phi} \right) C_L \sin \phi_s$$

$$C_{N_s} = 0.383 C_L \sin \phi_s.$$

With the two equations defined in this manner, the desired critical condition can now be solved. It follows that

$$C_{N_a} = C_{N_s}$$

yielding

$$0.0251 + 0.2 C_L \sin \phi = 0.383 C_L \sin \phi$$

and

$$C_L \sin \phi = 0.137.$$

Through Equation (26),

$$V_{MCA_s} = \sqrt{\frac{997.4}{C_L \sin \phi}} = 85.3 \text{ kts.}$$

To further simplify this procedure, referring to Figure 15, an expression for the weight-normalized value can be determined from test data by solving for the increment along the airframe curve from the test conditions to the standard conditions. From the point slope form defining the airframe line,

$$C_{N_t} - C_{N_s} = \frac{dC_N}{dC_L \sin \phi} (C_L \sin \phi_t - C_L \sin \phi_s)$$

and it can be shown that

$$C_L \sin \phi_s = \frac{C_{N_t} - 0.2 C_L \sin \phi_t}{\frac{C_{N_s}}{C_L \sin \phi_s} - 0.2}. \quad (29)$$

Solving for velocity in Equation (25) yields

$$V_{MCA_s} = \left[\frac{295 W_s \sin \phi_s}{S C_L \sin \phi_s} \right]^{1/2}. \quad (30)$$

Substituting Equation (29) into Equation (30) and reducing,

$$V_{MCA_s} = V_{MCA_t} \left[\frac{C_{N_s} - 0.2 C_L \sin \phi_s}{C_{N_t} - 0.2 C_L \sin \phi_t} \right]^{1/2}.$$

Substituting Equation (27) for the yawing-moment coefficient and since $C_L = \frac{W}{qs}$ in steady heading flight yields the desired relationship,

$$V_{MCA_s} = V_{MCA_t} \left(\frac{F_{N_s} \left(\frac{l_c}{b} \right) - 0.2 W_s \sin \phi_s}{F_{N_t} \left(\frac{l_c}{b} \right) - 0.2 W_t \sin \phi_t} \right)^{1/2} \quad (31)$$

Equation (31) defines the standardized minimum-control airspeed as the product of the test speed and the difference between the moment induced by the asymmetric thrust condition and the restoring moment provided by C_{N_β} . Applying the test data from the previous discussion demonstrates the above expression to be accurate to 1.2 percent.

$$V_{MCA_s} = 70.9 \left(\frac{2806 (0.107) - 0.2 (9000) \sin 5^\circ}{2400 (0.107) - 0.2 (9000) \sin 5^\circ} \right)^{1/2} = 85.0 \text{ kts.}$$

In the previous derivation, the airframe curve slope is generalized at 0.2 as justified in Appendix A. For this investigation,

$(dC_{N_a} / dC_L \sin \phi)_{\text{airframe}}$ is approximated as the ratio of C_{N_β} to C_{Y_β} .

The validity for this is given in Appendix B.

The difficulty in applying Equation (31) lies in determining its main parameter, net thrust. Test weight can be determined effectively by recording beginning and ending fuel quantities and noting the times of each data run. Hence, from a linear plot of fuel versus time, W_t can easily be determined.

For a propeller-driven aircraft with one engine inoperative, the net in-flight thrust can be determined from [4]

$$F_N = 550 \text{ BHP} \frac{\eta_P}{V_T} + D_W. \quad (32)$$

Once brake horsepower, propeller efficiency, and windmill drag are known, the problem becomes trivial.

For reciprocating engines, the most common methods for determining engine power are 1) the torque-meter method, 2) engine power charts, and 3) the fuel flow method [15]. Although the least accurate of the three, engine power charts are used for this study because of their ease of use and acceptable degree of accuracy. These charts are generated both from theory and engine testing and are supplied by the manufacturer.

Propeller-driven aircraft employ a propeller to convert mechanical energy produced by the engine to useful work or thrust. The effectiveness of this energy transfer is a function of the propeller's efficiency. This efficiency is defined as the ratio of power output to power input, and is expressed mathematically as [8]

$$\eta_P = \text{Power Out/Power In.}$$

Since

$$T = C_T \rho N^2 D^4$$

and

$$P = C_P \rho N^3 D^5,$$

then

$$\eta_P = \frac{C_T}{C_P} \frac{V}{ND}.$$

Using the above expression in conjunction with the correct propeller chart, as established through the propeller's activity factor, provides a direct method for computing propeller efficiency.

Very little research has been performed on negative thrust produced by propellers. Determining windmill drag for this investigation relies on charts and information found in References [15] and [17]. Hartman determines the negative thrust and torque characteristics of an aluminum alloy propeller for blade angles from 22 degrees to -23 degrees at $3/4$ radius and covers a $(\frac{V}{ND})^{-1}$ range from 0 to infinity [13]. The thoroughness of his report allows windmill drag to be effectively determined.

CHAPTER V

TEST PROGRAM

In order to determine if the method for computing V_{MCA_s} as described in Chapter IV is a viable alternative to the extrapolation technique discussed in Chapter III, several flights were conducted at The University of Tennessee Space Institute (UTSI) Flight Test Engineering Facility located at the Tullahoma Municipal Airport.

During September 1985, three flights were conducted using the department's Cessna 310 (ID No. N11UT) aircraft. Each flight was performed at a different weight, ballasted to allow minimal CG travel ($0.1\% \bar{C}$), and consisted of data runs at three different altitudes. Test data were corrected for instrument and position errors and extrapolated to obtain sea-level standard values and a normalized-weight value, similar to the procedure outlined in Figures 7 through 11. The same test data was then applied to Equation (31) with the hope of arriving at the same weight-normalized value.

The Cessna 310 is a twin-engine, low-wing monoplane with fully retractable gear. It is powered by two six-cylinder, horizontally-opposed, air-cooled, normally-aspirated Continental O-470M engines each rated at 240 horsepower at sea level. Both engines power two-bladed Hartzel HC 82XF-2/8433-4 constant-speed, full-feathering propellers. The necessary aircraft specifications for this investigation are given in Figure 17.

In each flight, the test minimum-control airspeed was determined by an extrapolation of the critical-control surface to maximum

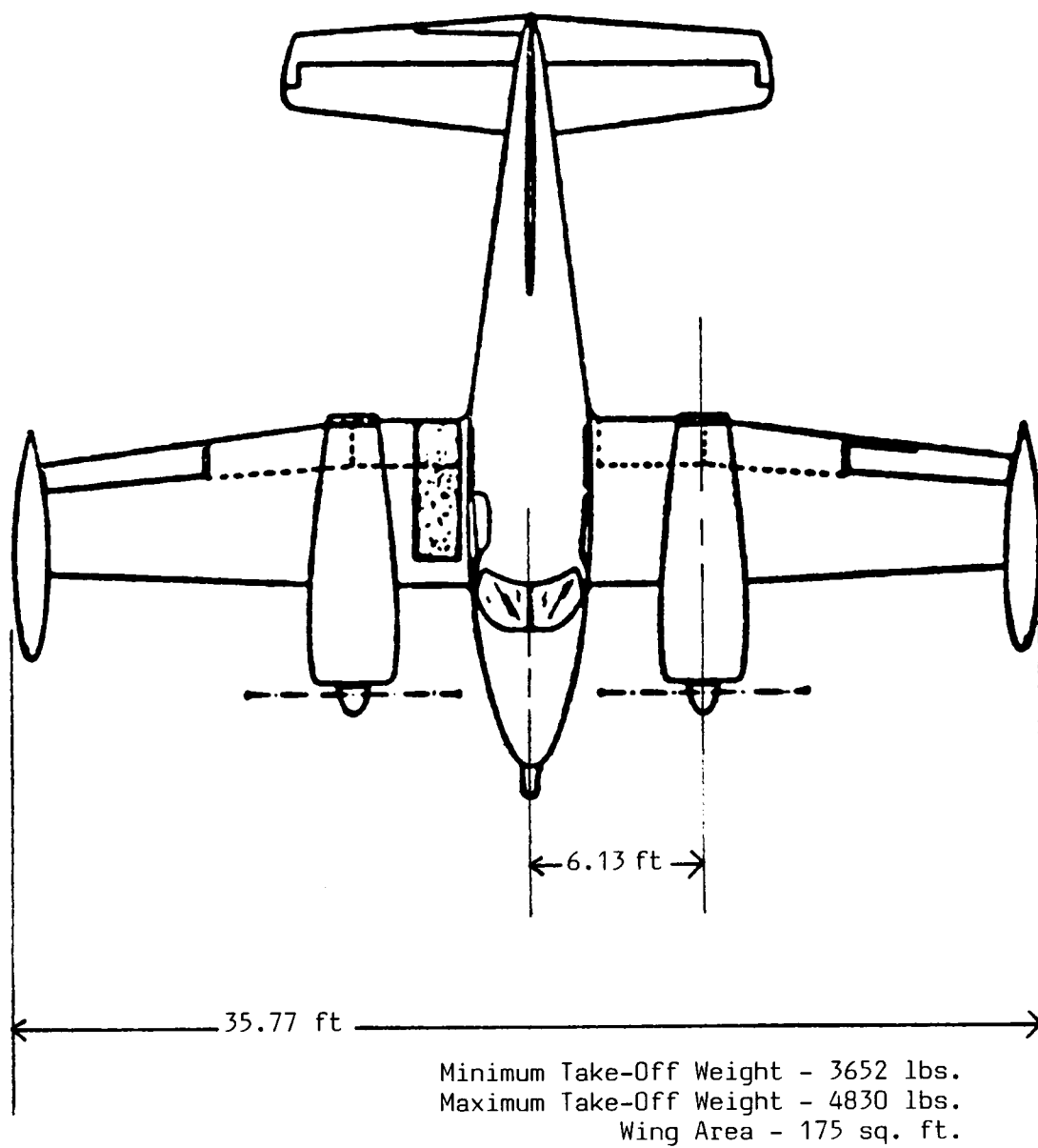


Figure 17. Specifications and Dimensions for Cessna 310 Aircraft.

deflection rather than flying to the point of minimum aerodynamic control as is the practice with military and industrial flight test programs. During the entire first flight, all major parameters (β , δ_a , and δ_r) were recorded and plotted against calibrated airspeed (Figures 18 through 26). As anticipated, directional control was established as the limiting factor eliminating plots of β versus V_c and δ_a versus V_c for the remaining flights. The results for Flights 2 and 3 are plotted in Figures 28 through 35. For increased accuracy, each plot was hand faired then generated using a weighted second-order curve fit.

Once the sea-level minimum-control airspeeds were determined (Figures 27, 31 and 35), a plot of weight versus V_{MCA} was generated to arrive at the normalized value for V_{MCA} (Figure 36).

Subsequent to the extrapolation method, the proposed, more efficient approach was applied. Test weight, standard and test thrust were determined as discussed in Chapter IV and implemented into Equation (31) to determine the weight-normalized value for V_{MCA} . To illustrate this procedure, consider Flight 1, Run 1:

$$W_t = 4,797 \text{ lbs.}$$

$$H_p = 6,900 \text{ ft.}$$

$$N_1 = 8.17 \text{ RPS}$$

$$N_2 = 41.67 \text{ RPS}$$

$$D = 6.67 \text{ ft.}$$

$$\sigma_d = 0.7804$$

$$\rho = 0.001856 \text{ slugs/ft}^3$$

$$V_{MCA_t} = 92.84 \text{ ft/sec} = 55.0 \text{ kts}$$

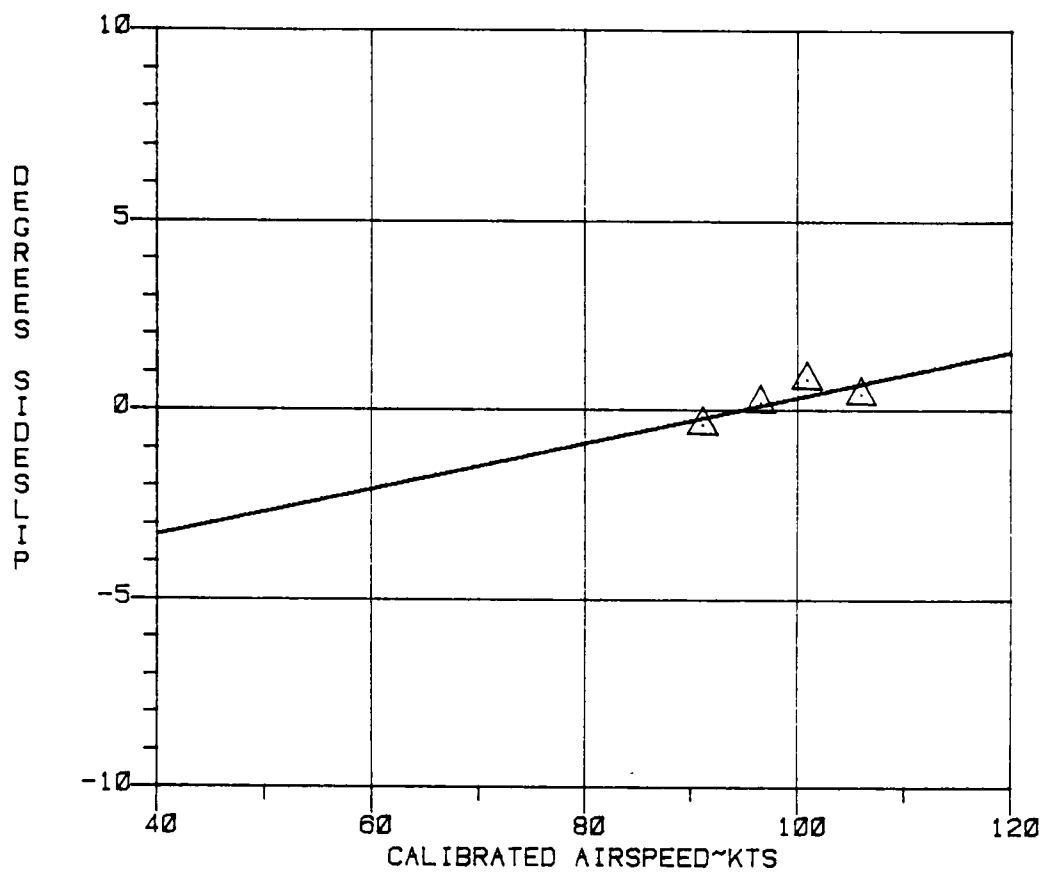


Figure 18. Sideslip Versus Calibrated Airspeed for Run 1, Flight 1 at 6900 ft. Pressure Altitude and a Test Weight of 4797 lbs.

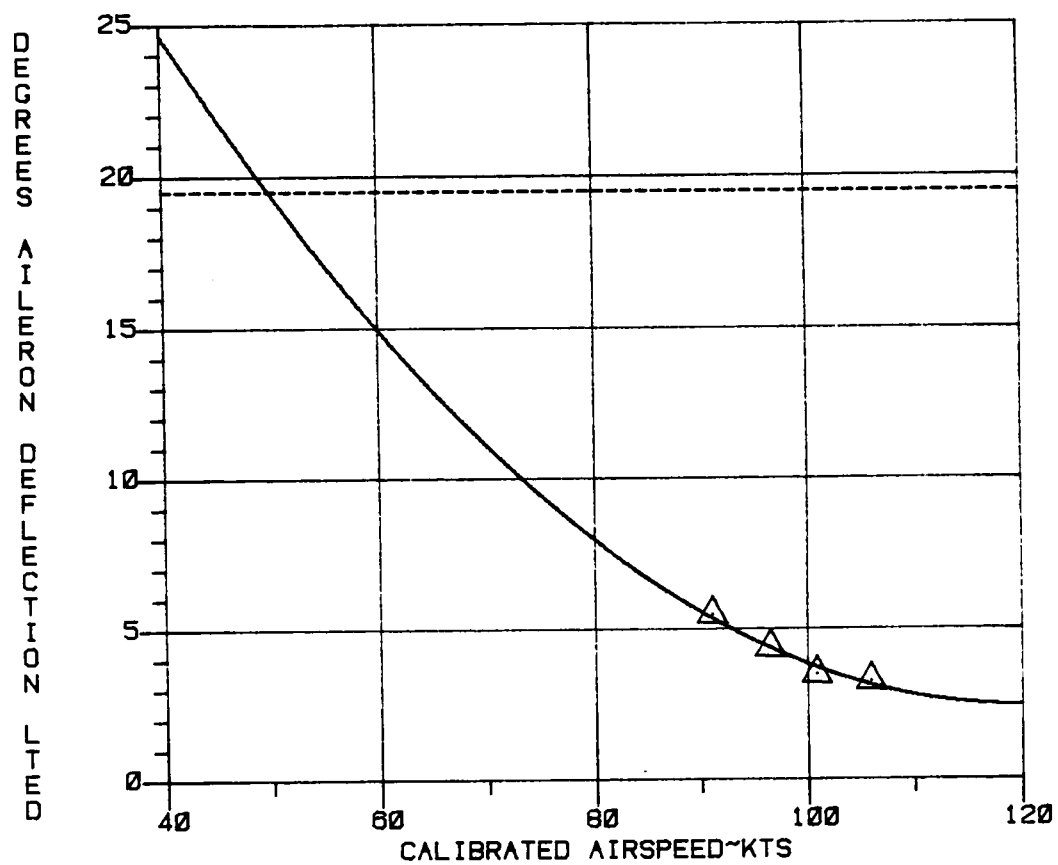


Figure 19. Aileron Deflection Versus Calibrated Airspeed for Run 1, Flight 1.

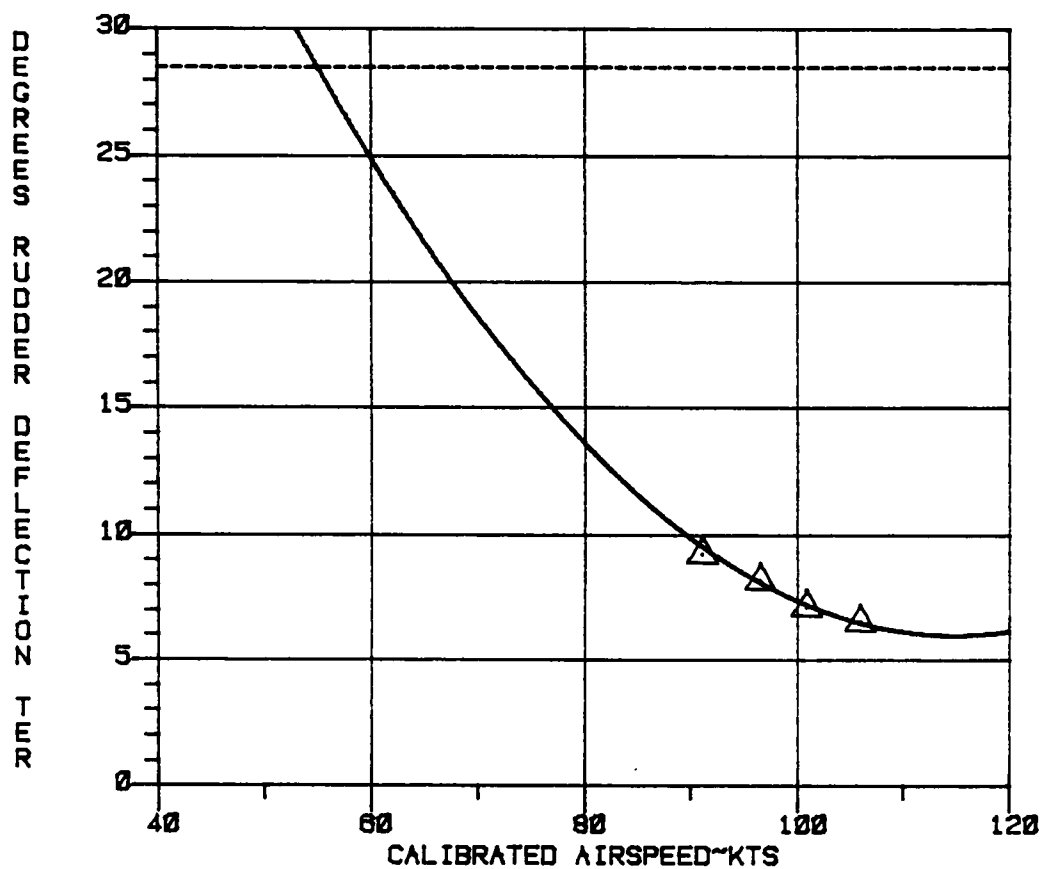


Figure 20. Rudder Deflection Versus Calibrated Airspeed for Run 1, Flight 1.

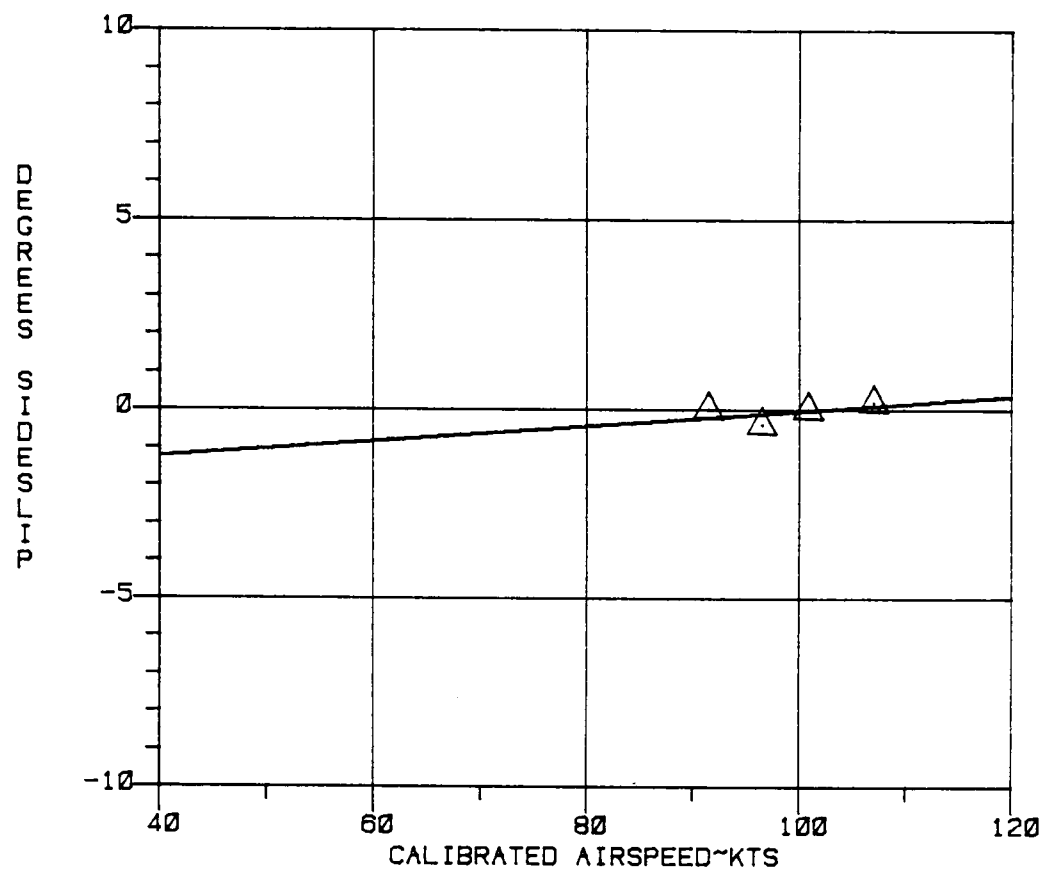


Figure 21. Sideslip Versus Calibrated Airspeed for Run 2, Flight 1 at 4800 ft. Pressure Altitude and a Test Weight of 4791 lbs.

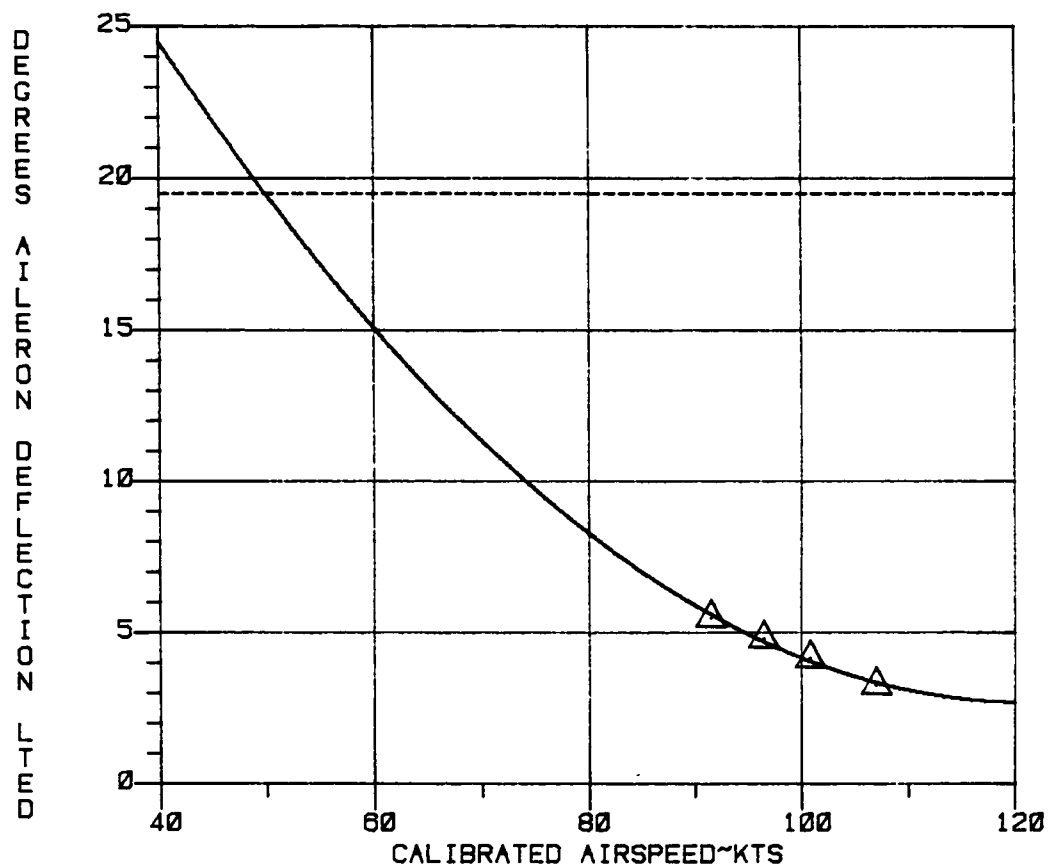


Figure 22. Aileron Deflection Versus Calibrated Airspeed for Run 2, Flight 1.

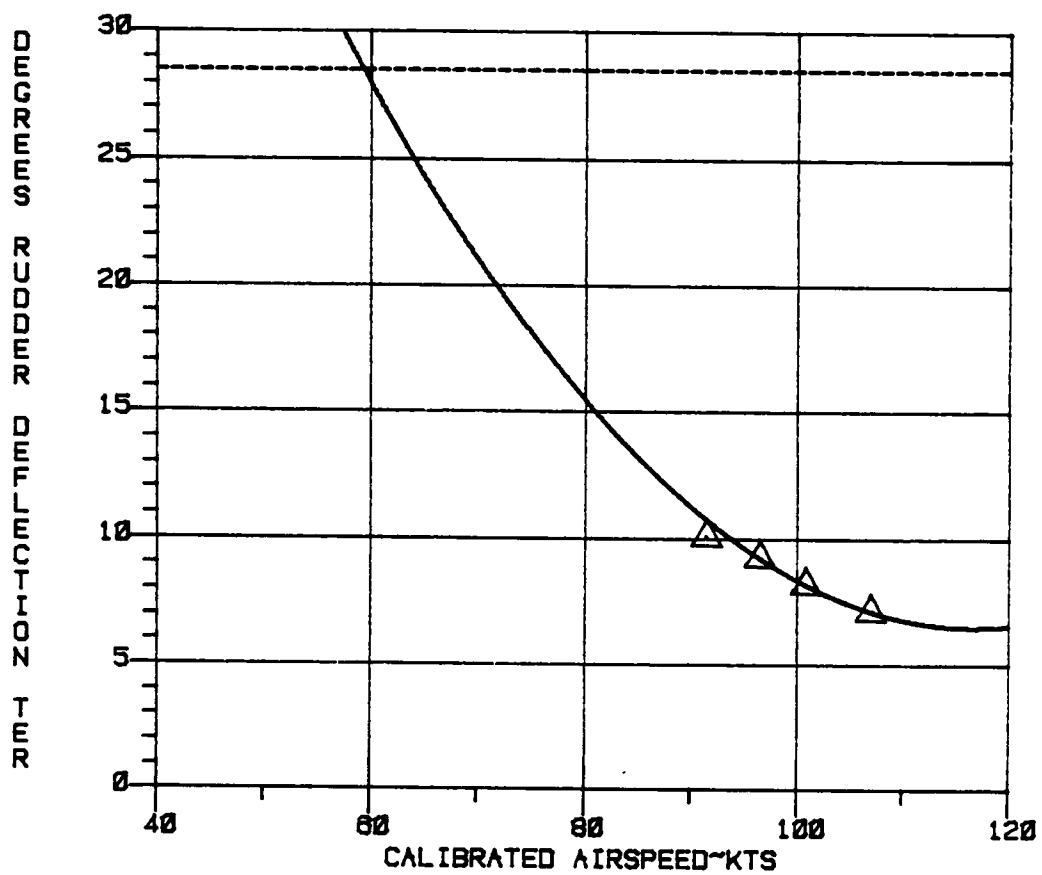


Figure 23. Rudder Deflection Versus Calibrated Airspeed for Run 2, Flight 1.

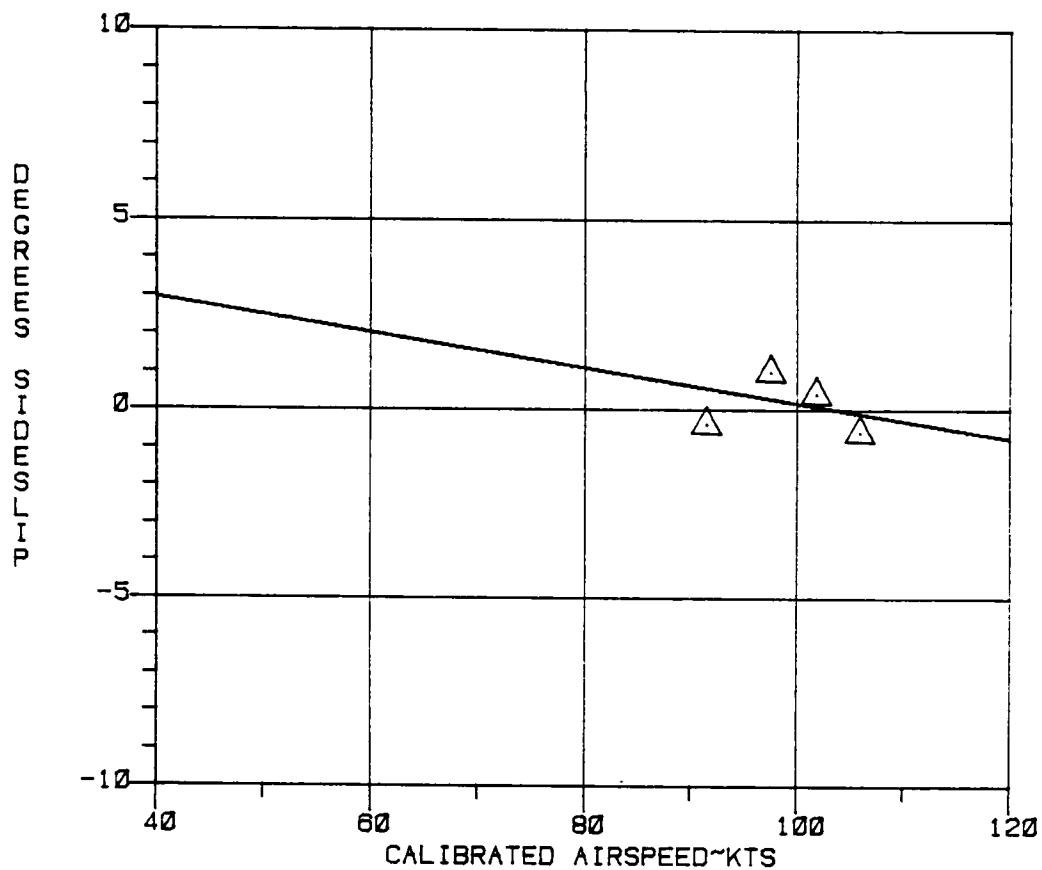


Figure 24. Sideslip Versus Calibrated Airspeed for Run 3, Flight 1 at 3250 ft. Pressure Altitude and a Test Weight of 4785 lbs.

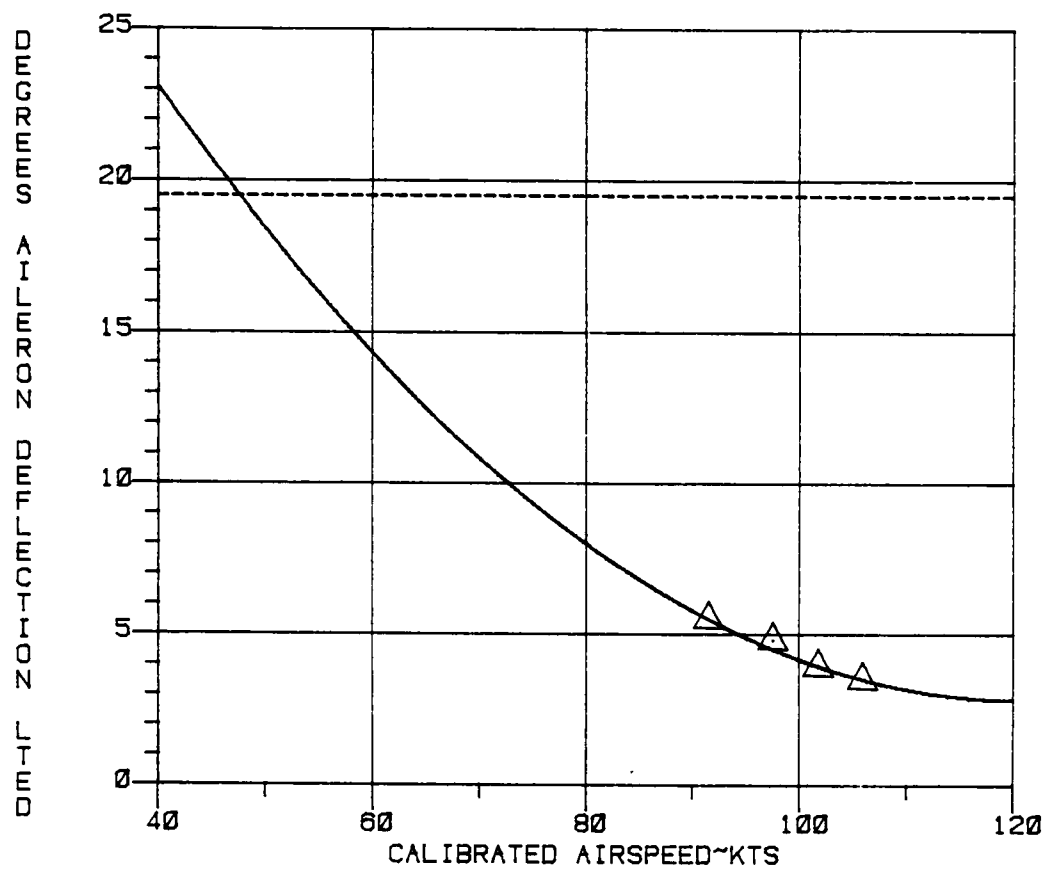


Figure 25. Aileron Deflection Versus Calibrated Airspeed for Run 3, Flight 1.

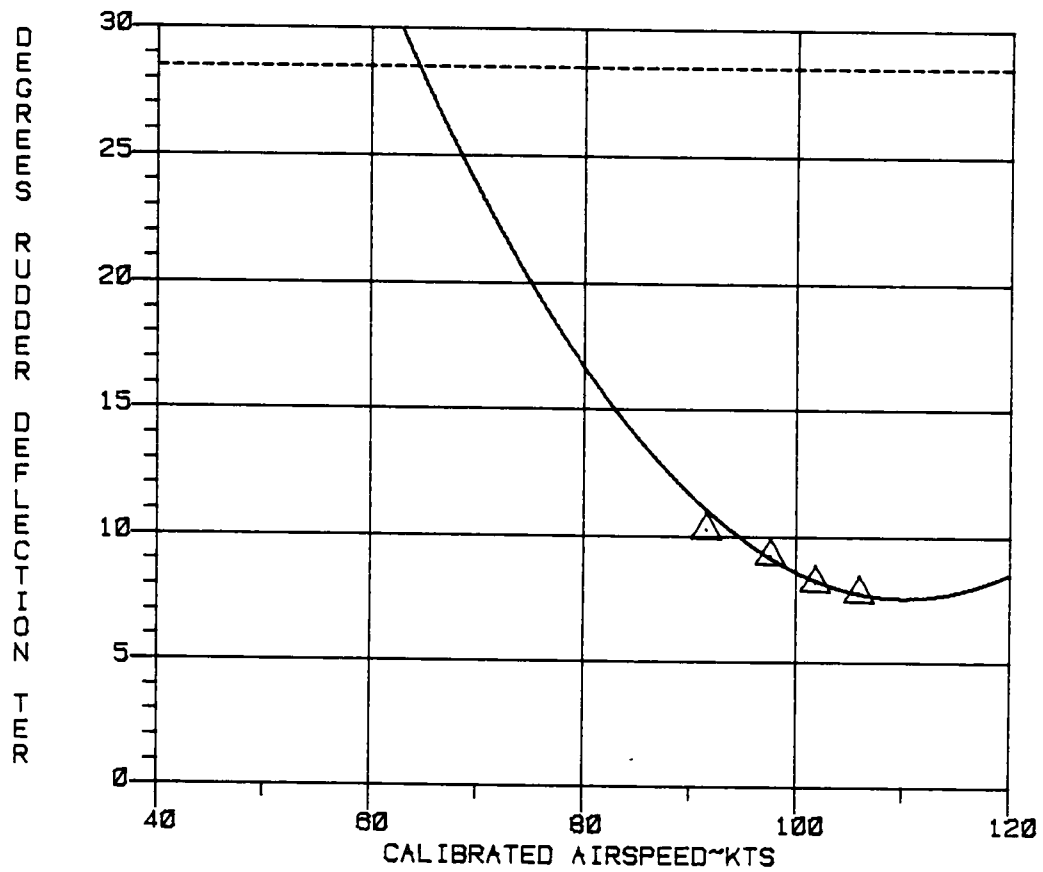


Figure 26. Rudder Deflection Versus Calibrated Airspeed for Run 3, Flight 1.

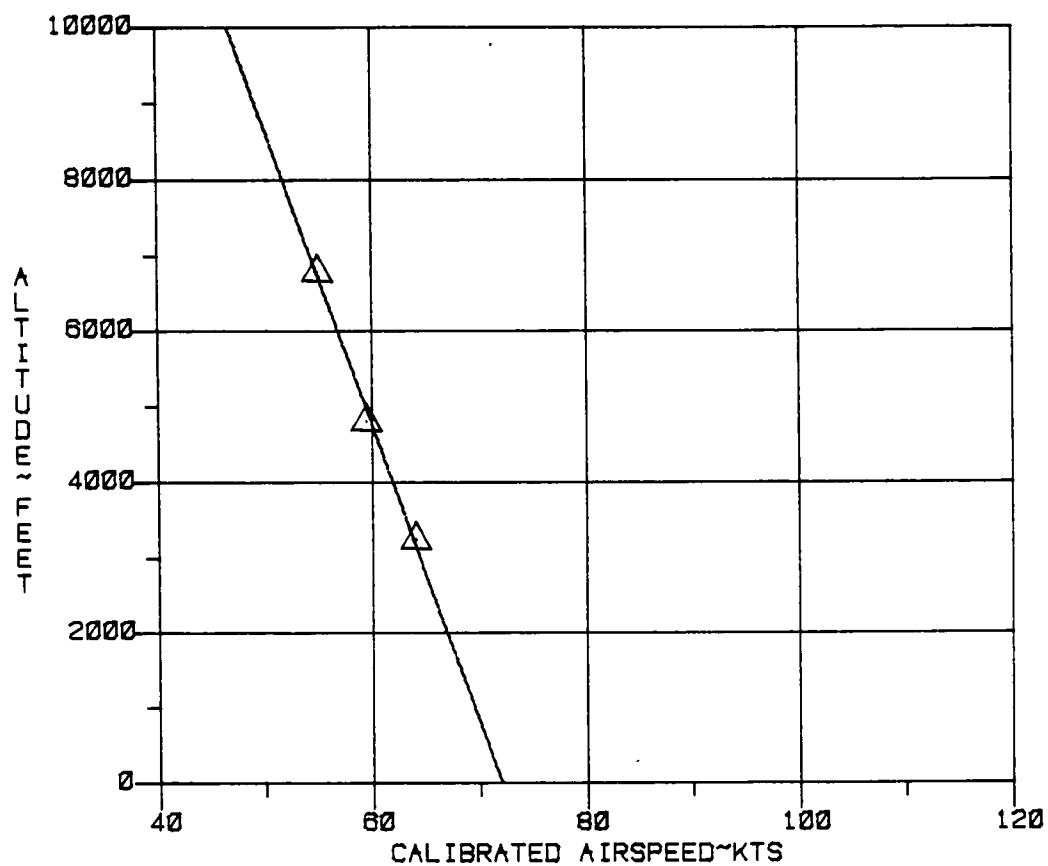


Figure 27. Altitude Versus Calibrated Airspeed for Flight 1.

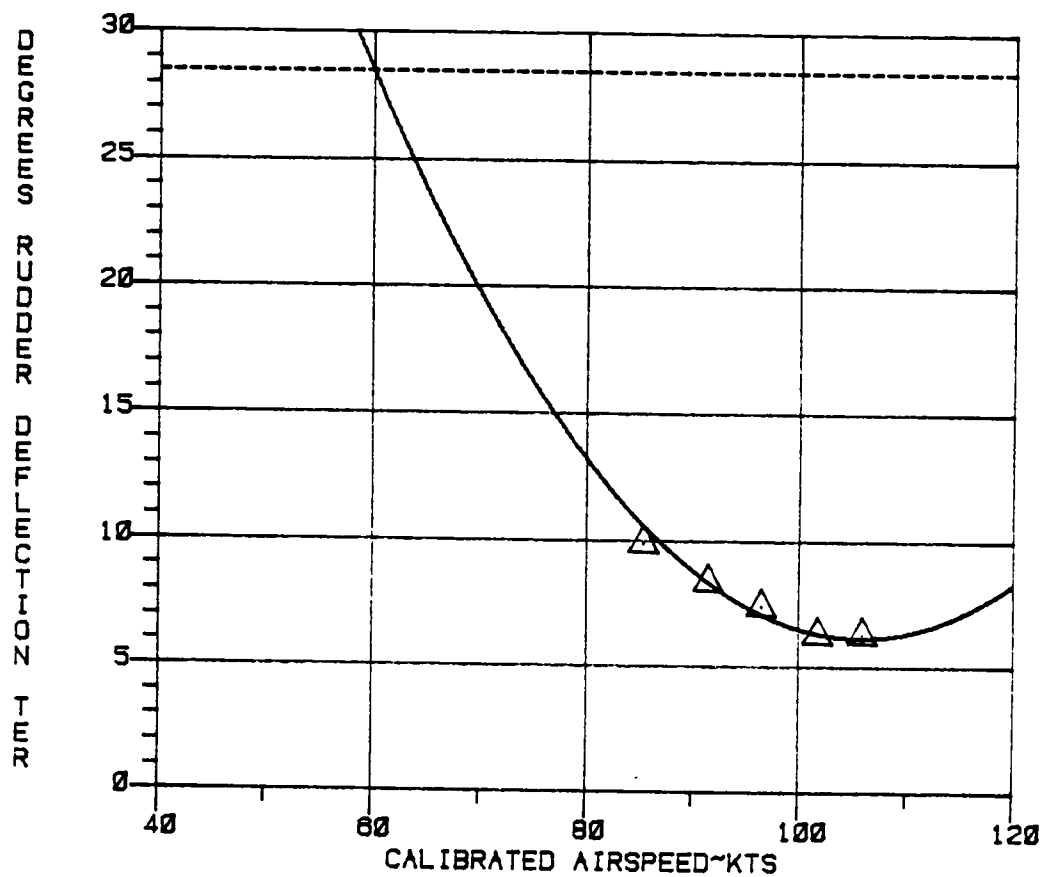


Figure 28. Rudder Deflection Versus Calibrated Airspeed for Run 1, Flight 2 at 7000 ft. Pressure Altitude and a Test Weight of 4465 lbs.

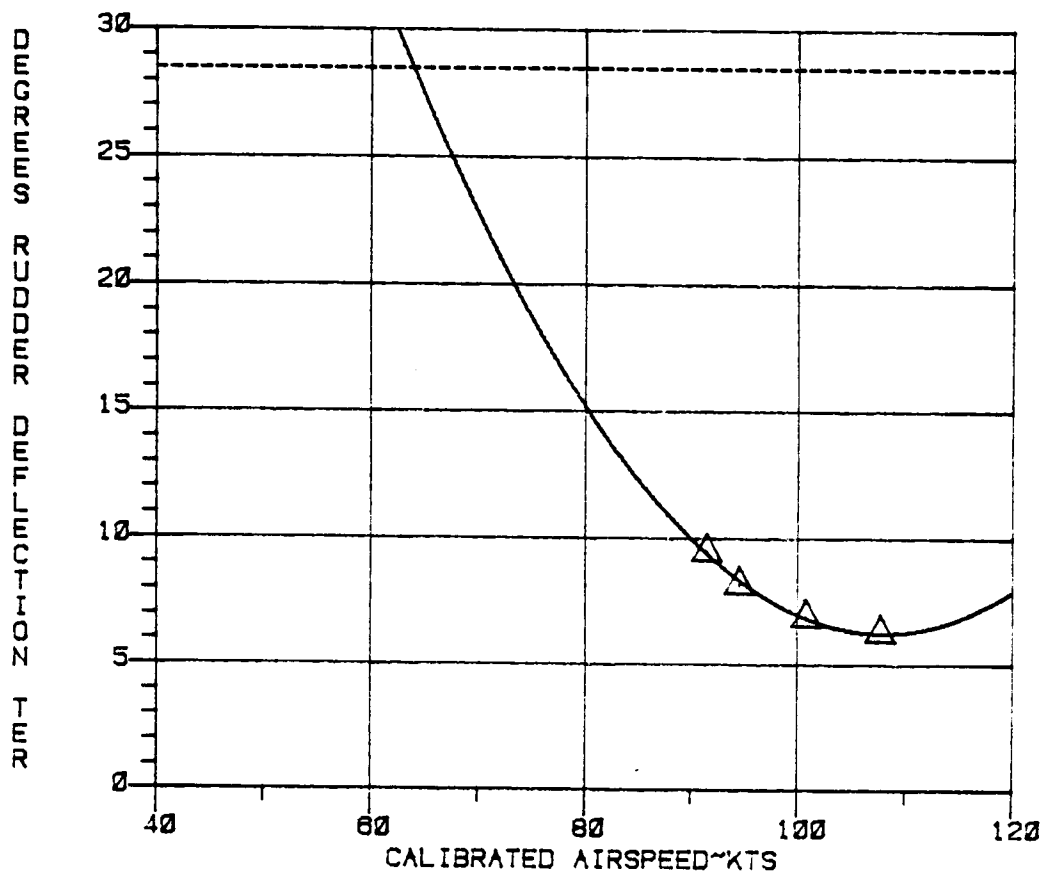


Figure 29. Rudder Deflection Versus Calibrated Airspeed for Run 2, Flight 2 at 5130 ft. Pressure Altitude and a Test Weight of 4459 lbs.

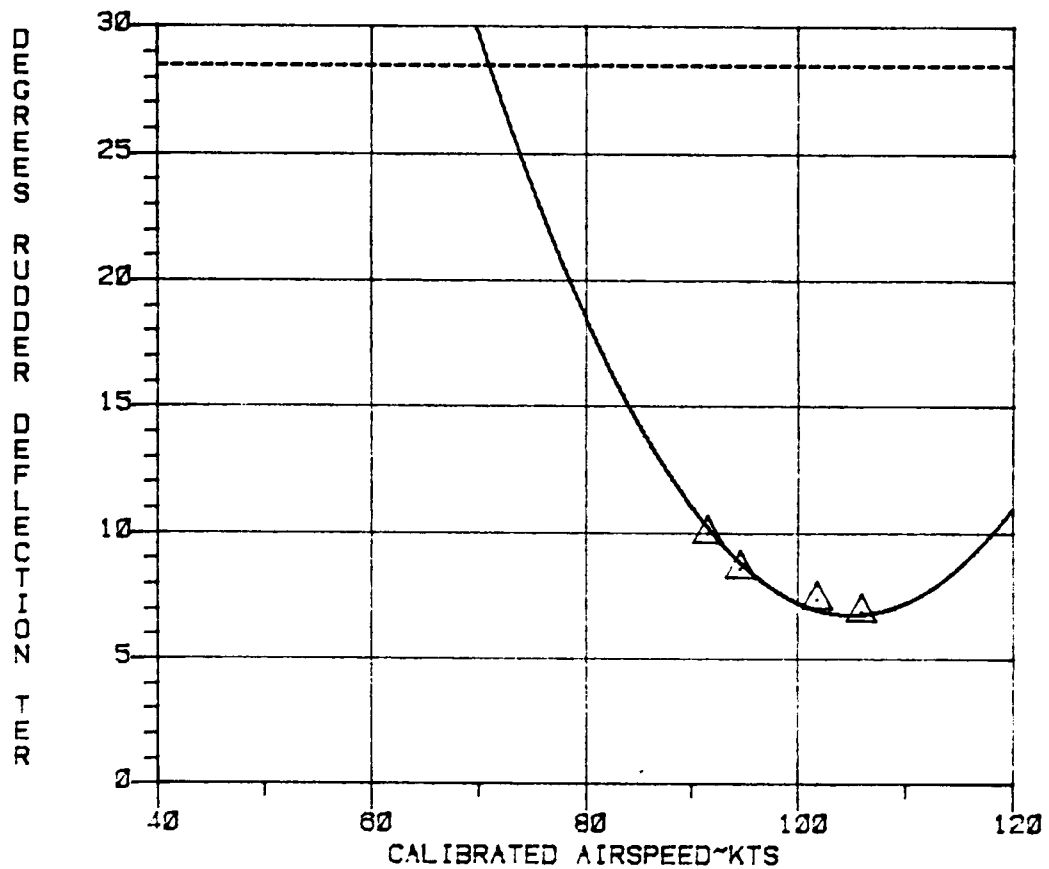


Figure 30. Rudder Deflection Versus Calibrated Airspeed for Run 3, Flight 2 at 3010 ft. Pressure Altitude and a Test Weight of 4453 lbs.

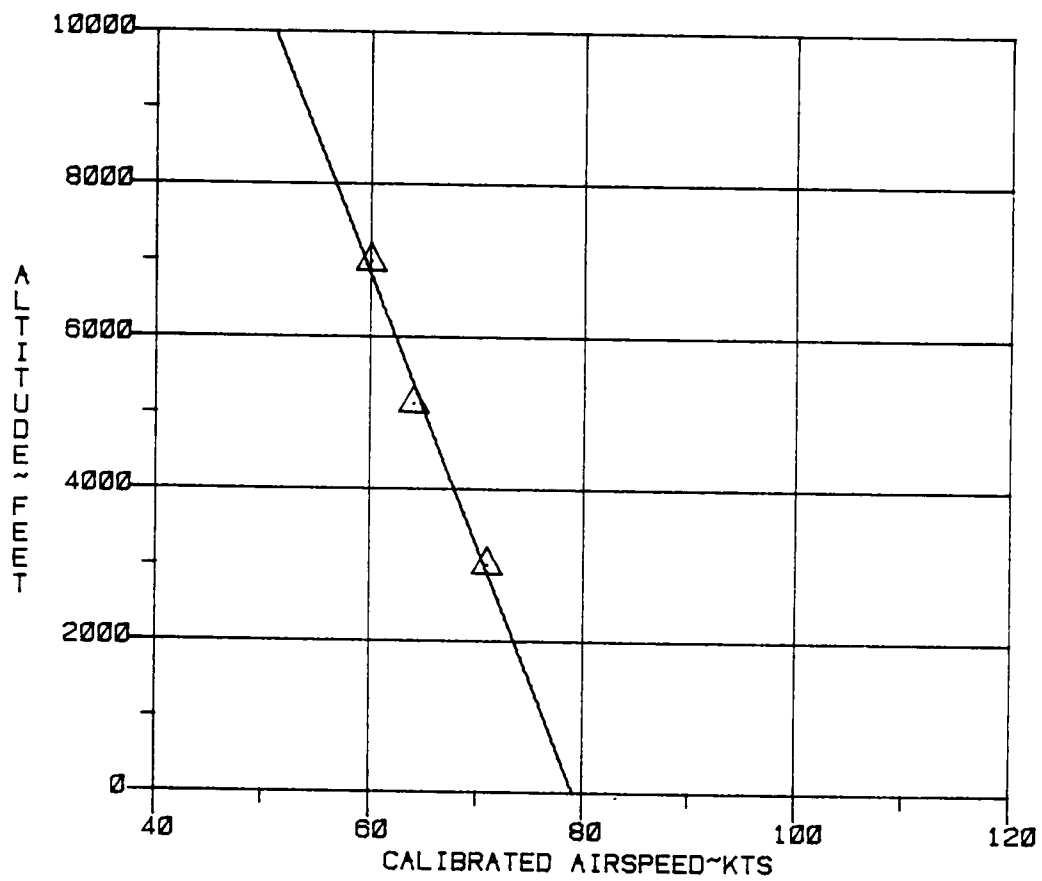


Figure 31. Altitude Versus Calibrated Airspeed for Flight 2.

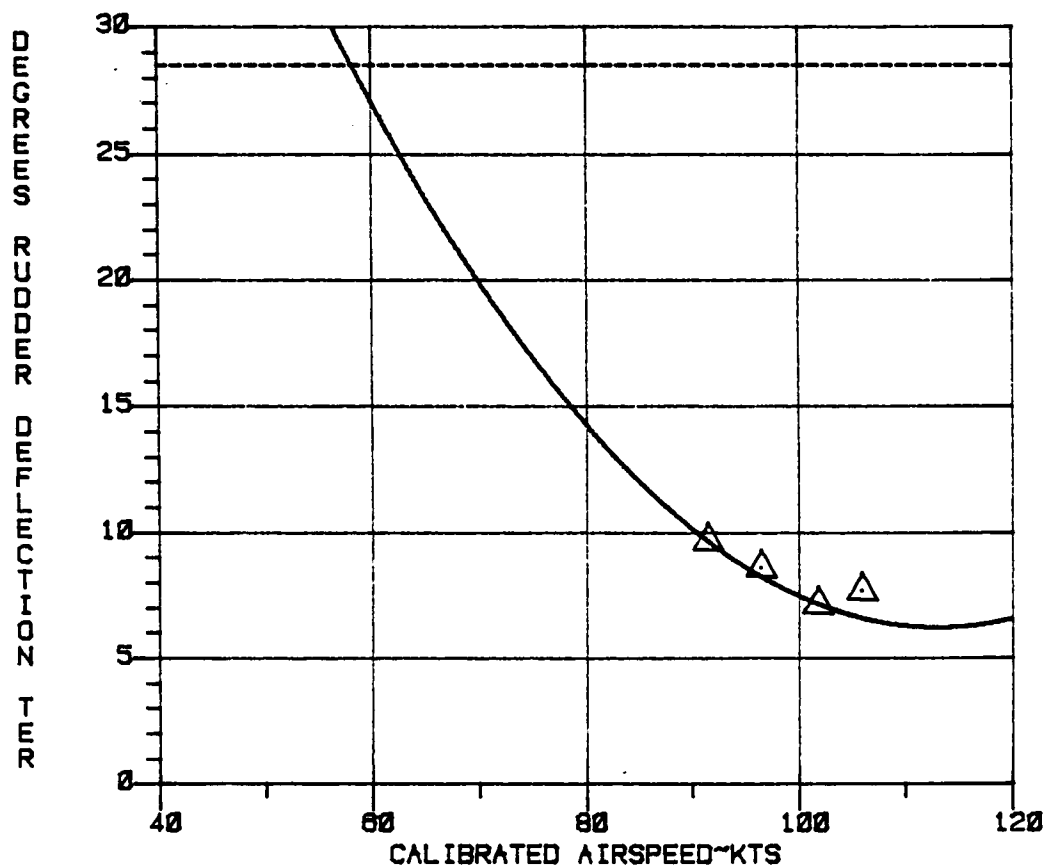


Figure 32. Rudder Deflection Versus Calibrated Airspeed for Run 1, Flight 3 at 6920 ft. Pressure Altitude and a Test Weight of 4607 lbs.

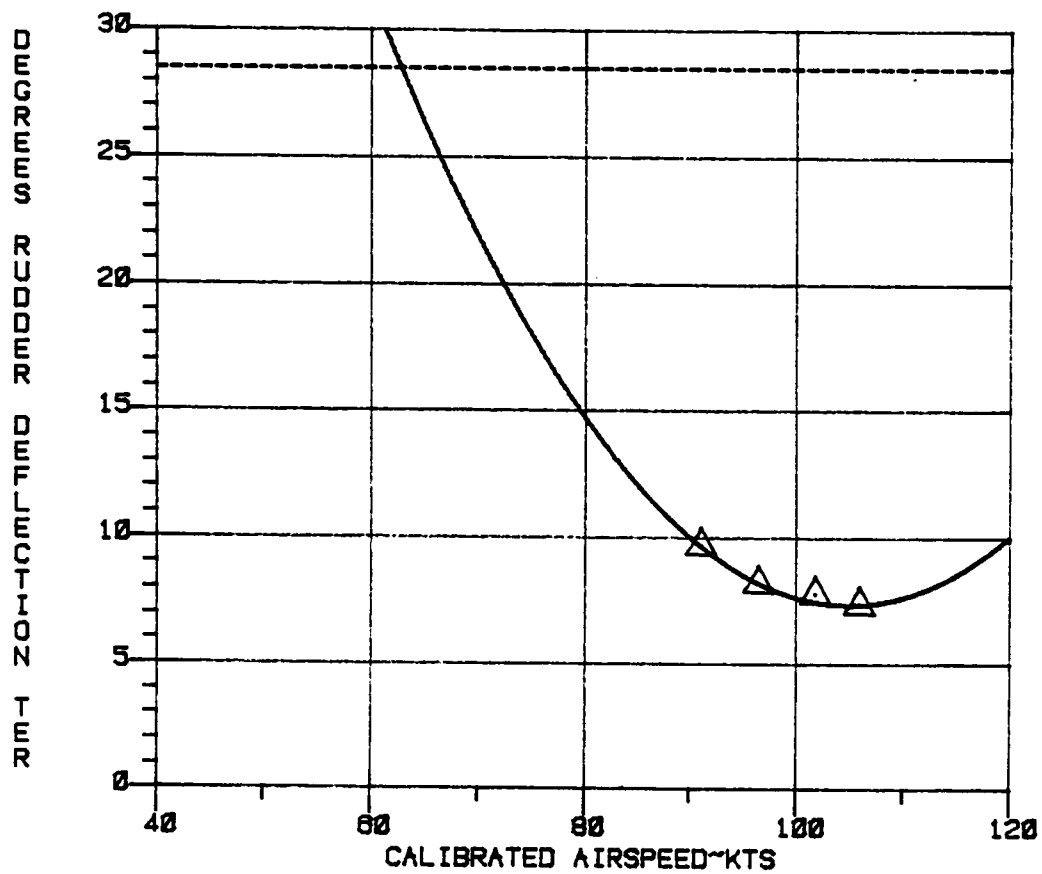


Figure 33. Rudder Deflection Versus Calibrated Airspeed for Run 2, Flight 3 at 5210 ft. Pressure Altitude and a Test Weight of 4610 lbs.

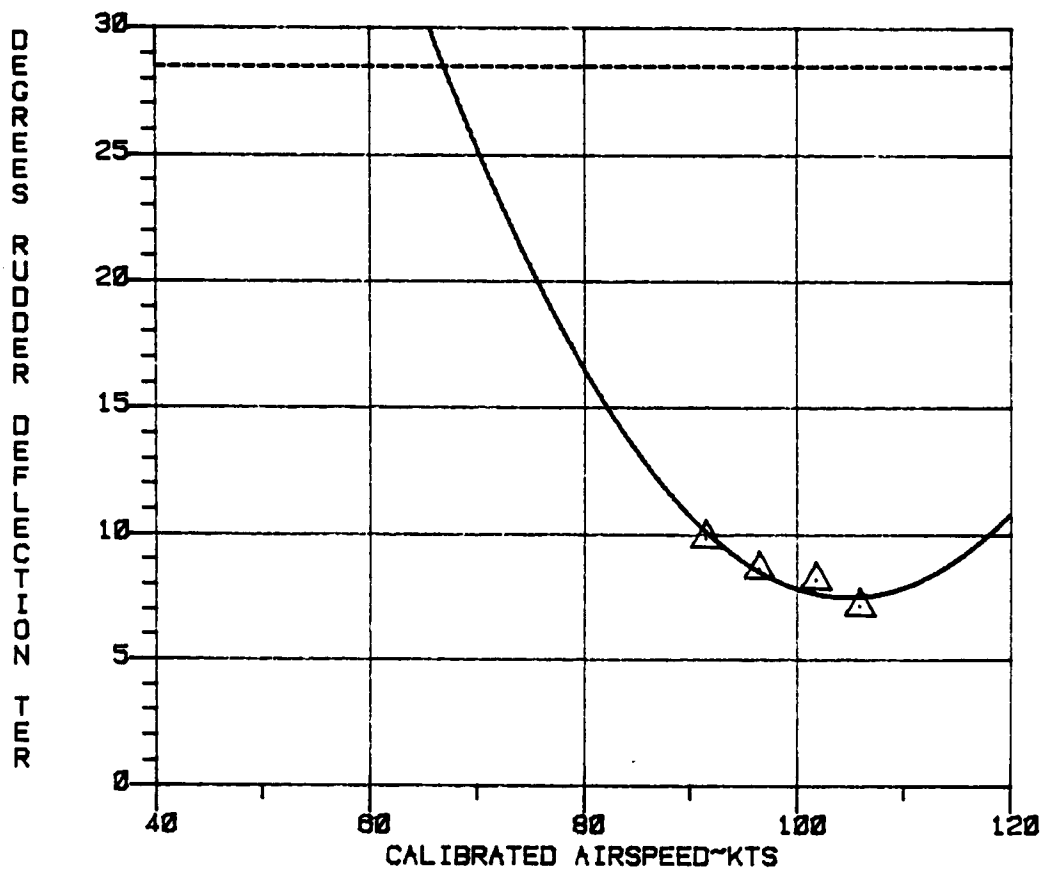


Figure 34. Rudder Deflection Versus Calibrated Airspeed for Run 3, Flight 3 at 3570 ft. Pressure Altitude and a Test Weight of 4595 lbs.

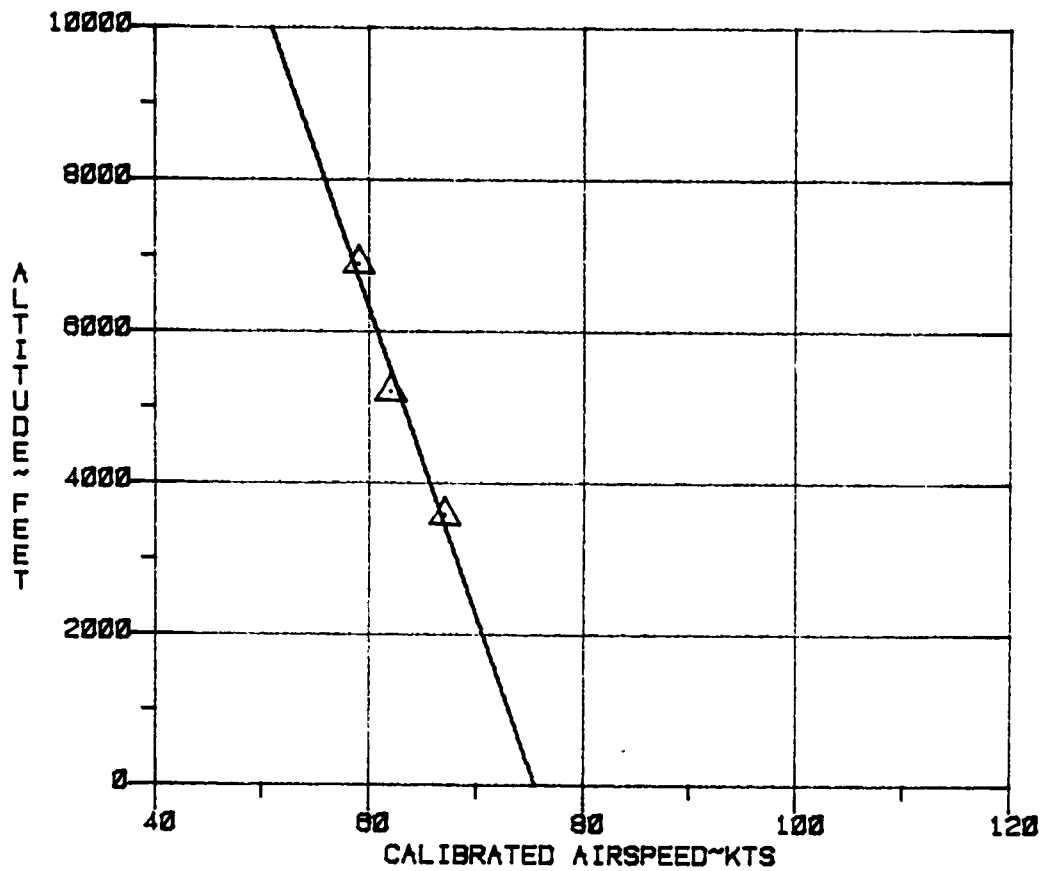


Figure 35. Altitude Versus Calibrated Airspeed for Flight 3.

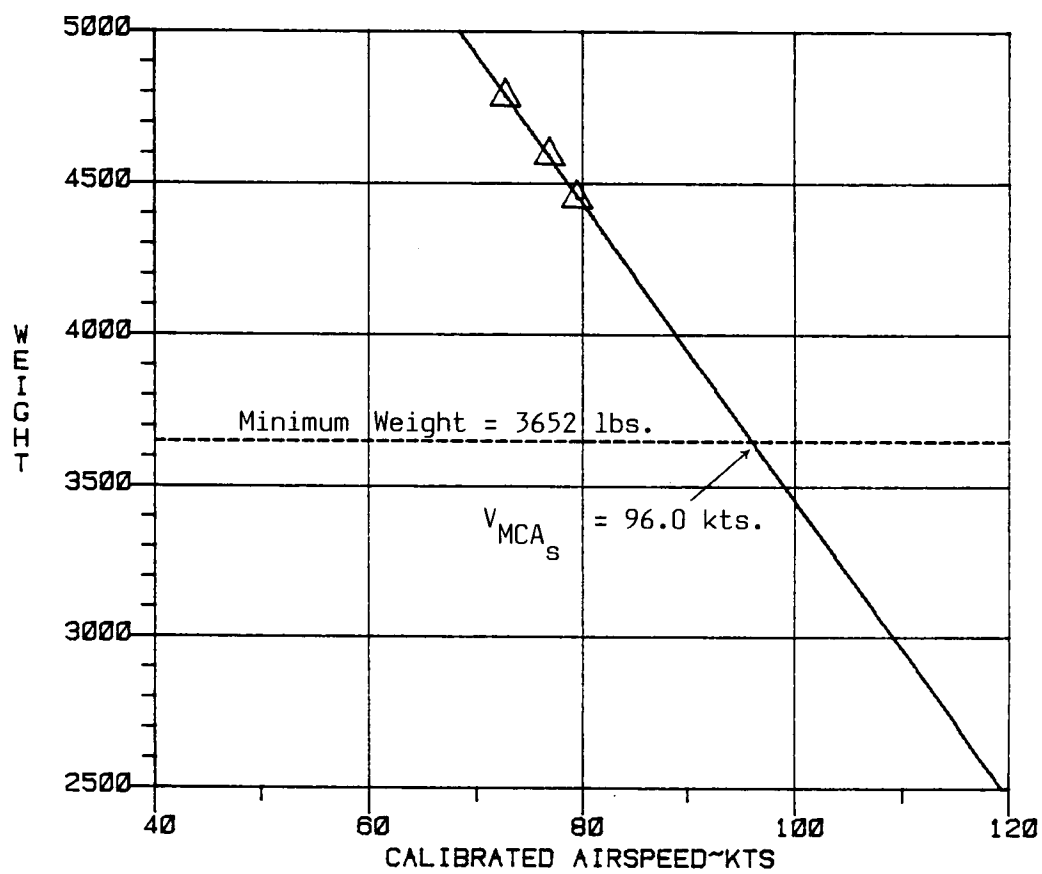


Figure 36. Weight Versus Calibrated Airspeed for Flights 1, 2, and 3.

$$MP = 22.0 \text{ in Hg}$$

$$BHP = 97,900 \text{ Ft lbs/sec}$$

$$W_s = 3652 \text{ lbs}$$

$$F_{N_s} = 680.11 \text{ lbs}$$

$$x_c = 6.125 \text{ ft}$$

$$b = 35.77 \text{ ft}$$

$$\phi_s = \phi_t = 5^\circ$$

$$dC_{N_a} / dC_L \sin \phi = 0.21.$$

Hence,

$$C_p = 0.055$$

$$J = 0.378$$

$$C_T = 0.091$$

$$\eta = 62.54\%$$

$$D_W = 29.18 \text{ lbs.}$$

Therefore,

$$F_{N_t} = 550 (178) \frac{0.6254}{105.09} + 29.18 = 611.79 \text{ lbs.}$$

The normalized value can now be determined. Using Equation (31),

$$V_{MCA_s} = 55 \left[\frac{680.11 (0.17) - 0.21 (3652) \sin 5^\circ}{611.79 (0.17) - 0.21 (4797) \sin 5^\circ} \right]^{1/2} = 95.4 \text{ kts.}$$

A summary of the results obtained from both methods is given in Tables 2 and 3.

Table 2. Summary of Graphical Results for Flights 1, 2, and 3.

W_t (lbs)	H_p (ft)	V_{MCA_t} (kts)
<u>Flight 1</u>		
4797	6900	55.0
4791	4800	59.5
4785	3250	64.5
Extrapolated Sea-Level $V_{MCA} = 72.8$		
<u>Flight 2</u>		
4465	7000	60.0
4459	5130	64.0
4453	3010	71.0
Extrapolated Sea-Level $V_{MCA} = 79.5$		
<u>Flight 3</u>		
4607	6920	58.0
4601	5210	62.0
4595	3570	67.3
Extrapolated Sea-Level $V_{MCA} = 77.0$		
Extrapolated to Minimum Take-Off Weight = 96.0		

Table 3. Summary of Computational Results for Flights 1, 2, and 3.

Temp (°C)	W_t (lbs)	H_p (ft)	D_w (lbs)	η_p (%)	BHP	F_2 (lbs)	V_{MCA_t} (kts)	V_{MCA_s} (kts)	Error (%)
<u>Flight 1</u>									
12.8	4797	6000	29.18	62.54	178	582.61	55.0	95.4	0.63
15.6	4791	4800	33.08	62.16	191	594.58	59.5	95.3	0.73
17.8	4785	3250	37.60	65.00	197	606.65	64.5	96.0	0.00
<u>Flight 2</u>									
11.7	4465	7000	34.73	66.19	176	558.90	60.0	95.6	0.42
15.6	4459	5130	38.28	66.36	186	571.19	64.0	95.3	0.73
18.9	4453	3010	45.58	68.74	202	599.13	71.0	93.6	2.50
<u>Flight 3</u>									
12.6	4607	6920	31.44	64.68	177	568.13	58.0	96.5	0.52
15.6	4601	5210	34.77	65.25	186	578.88	62.0	96.6	0.63
18.5	4595	3570	40.93	67.07	197	595.61	67.3	95.7	0.31

Avg. V_{MCA_s} = 95.5 kts.

Subsequent to these calculations, it was noted that the sensitivity of the parameters in Equation (31) can lead to erroneous results. As illustrated in Figures 37 and 38, marginal calculations for propeller efficiency and windmill drag can effect net thrust calculations enough to fluctuate V_{MCA} by over 5 percent. Hence, accurate determination of in-flight net thrust is critical to the success of this method.

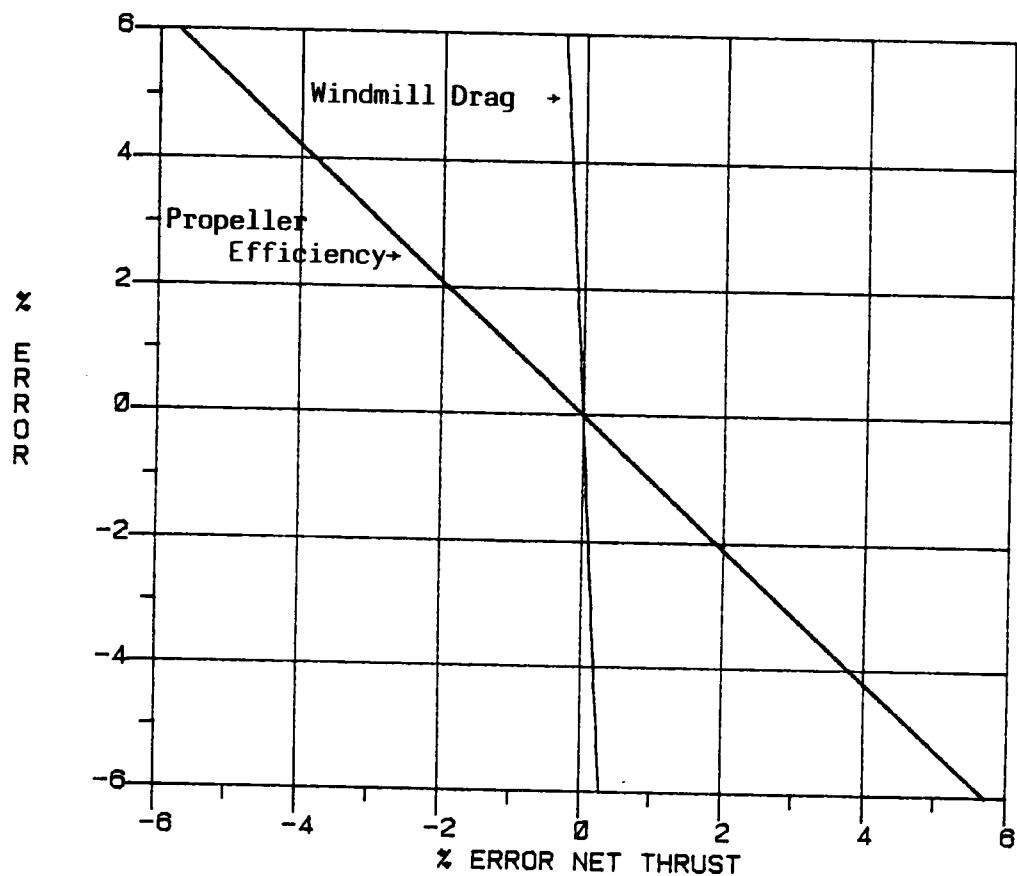


Figure 37. Errors Introduced into Net Thrust Calculations Due to Errors in Determining Propeller Efficiency and Windmill Drag.

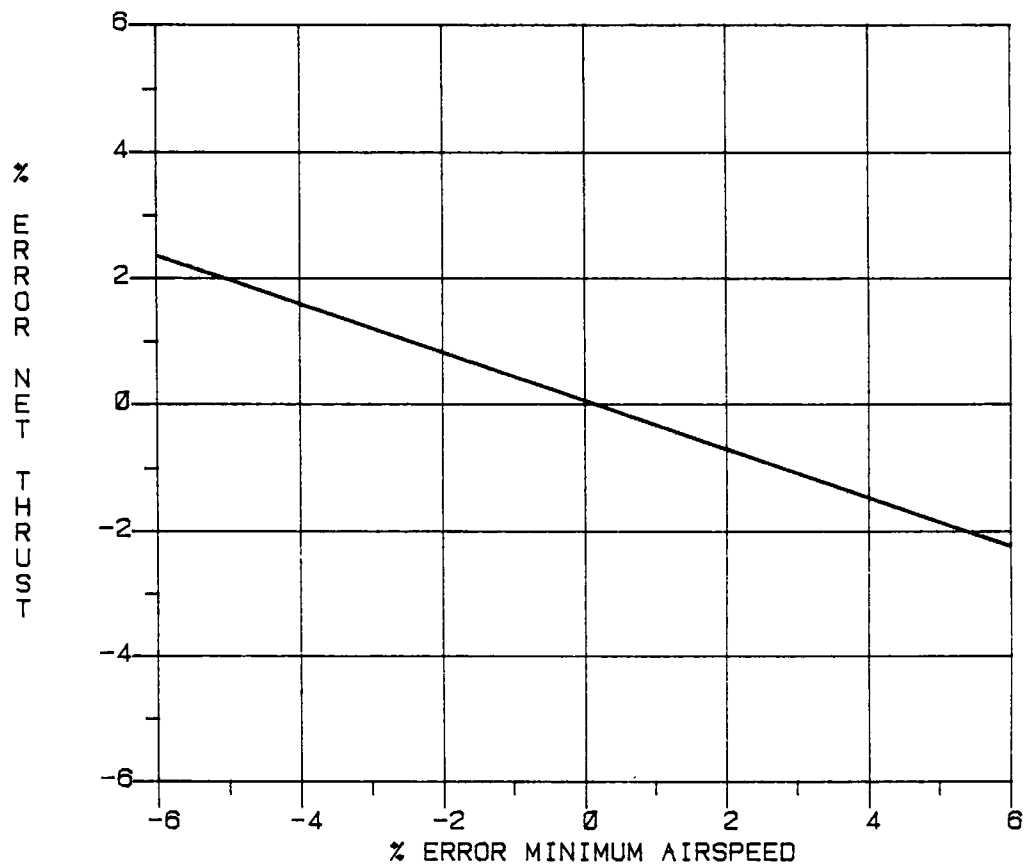


Figure 38. Error in Minimum-Control Airspeed Due to Errors in Determining Net Thrust.

CHAPTER VI

CONCLUSIONS AND RECOMMENDATIONS

This thesis investigates the possibility of using a more efficient method for flight testing aircraft operating under asymmetric power for the purpose of determining the maximum value for V_{MCA} . Summarized below are the results of this study.

Conclusions

The technique proposed in this study proves to be accurate to 0.52 percent on the average and demonstrated a time savings of 67 percent in-flight time and 84.7 percent in time spent reducing data when compared to the extrapolation method in use today.

Accurate determination of operative engine thrust, and in the case of propeller-driven aircraft, windmill drag and propeller efficiency are critical.

Recommendations

As is the case with any flight-testing program, early morning flights with smooth air are essential. Minor deviations from desired flight configurations can introduce errors due to simplifying assumptions.

Better methods need to be implemented for determining C_T and D_W . While propeller charts are adequate for this investigation, further studies need to be conducted on propeller efficiency and windmill drag to provide more accurate estimations and increase data confidence.

Due to the sensitivity of required data, on-board acquisition systems and data-monitoring systems should be mandatory for all test flights.

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APPENDICES

APPENDIX A

GENERALIZATION OF AIRFRAME SLOPE

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GENERALIZATION OF AIRFRAME SLOPE

In the general derivation of Equation (31), an approximation for $(dC_N/dC_L \sin \phi)_{\text{airframe}}$ was employed. This approximation is based on an average of C_N versus $C_L \sin \phi$ slopes for several different transport-class aircraft. This average is equal to 0.200 with a standard deviation of 0.009, as determined from Table A-1 [12].

Table A-1. A Sample of $(dC_N/dC_L \sin \phi)_{\text{airframe}}$ for Several Different Transport-Class Aircraft.

Aircraft	$dC_N/dC_L \sin \phi$
Learjet full-scale wind-tunnel tests	0.211
Learjet dedicated flight tests	0.202
747 TIR (D6-24250) stability and control derivatives used to correct V_{MCA} to higher bank angles	0.197
737-200 stability and control derivatives estimated from USAF DATCOM	0.209
737-200 C_N versus $C_L \sin \phi$ characteristics determined from engine simulator for JT8D-17AR engine approval	0.192
757-200 flight test data from V_{MCA} Certification Report D541 N003, Section C1.12.0006	0.190

APPENDIX B

APPROXIMATION OF AIRFRAME SLOPE

APPENDIX B

APPROXIMATION OF AIRFRAME SLOPE

The more accurate approximation for the slope of the airframe curve used in the final calculations of this investigation is determined by eliminating all of the secondary terms, as far as engine-out stabilization is concerned, in Equations (13) through (15). Then by working with only the roll axis, it is found that

$$C_{Y_{\beta}} + C_{Y_{\delta_r}} \delta_r = -C_L \sin \phi, \quad (B-1)$$

and

$$C_{N_{\beta}} + C_{N_{\delta_r}} \delta_r = C_{N_a}. \quad (B-2)$$

Solving for sideslip in Equation (B-2) and substituting into Equation (B-1) yields

$$\frac{C_{Y_{\beta}}}{C_{N_{\beta}}} \left(C_{N_a} - C_{N_{\delta_r}} \delta_r \right) + C_{Y_{\delta_r}} \delta_r = -C_L \sin \phi, \quad (B-3)$$

which can be put into a linear equation of the form

$$C_{N_a} = b + MC_L \sin \phi. \quad (B-4)$$

Hence,

$$C_{N_a} = C_{N_{\delta_r}} \delta_r - \frac{C_{N_{\beta}}}{C_{Y_{\delta_r}}} C_{Y_{\delta_r}} \delta_r - \frac{C_{N_{\beta}}}{C_{Y_{\beta}}} C_L \sin \phi \quad (B-5)$$

where

$$b = \text{intercept} = C_{N_{\delta_r}} \delta_r - \frac{C_{N_{\beta}}}{C_{Y_{\beta}}} C_{Y_{\delta_r}} \delta_r \quad (\text{B-6})$$

and

$$m = \text{slope} = - \frac{C_{N_{\beta}}}{C_{Y_{\beta}}}. \quad (\text{B-7})$$

Eliminating all secondary terms is justified through a trial-and-error determination of the slope of C_{N_a} versus $C_L \sin \phi$ by plugging in values for $C_L \sin \phi$ and solving the full set of equations for C_{N_a} . The slope of the calculated line is so close to $-C_{N_{\beta}}/C_{Y_{\beta}}$ that the difference is negligible [2].

From Reference [18] for the stability derivatives of the Cessna 310 aircraft,

$$C_{N_{\beta}} = 0.1444$$

and

$$C_{Y_{\beta}} = -0.6980.$$

Therefore, for the purpose of this investigation,

$$\left(\frac{dC_{N_a}}{dC_L \sin \phi} \right)_{\text{airframe}} = 0.21.$$

APPENDIX C
FLIGHT TEST DATA

FLIGHT TEST DATA SHEET

Date September 9, 1986 Model C-310 Ident. No. N 11UT Page 1 of 2
 Purpose: Determination of VMCA Flight No. 1
 Beginning Fuel 45 gal. L. 08 Ending Fuel 38 gal. L. 08 Time 9:45 a.m. to 10:23 a.m.
 Total Time 38 minutes
 Gross Wt. 4797 lbs Pilot Kimberlin
 C.G. 28.2% C Observer _____
 Config.: _____

Kts	V _i	LEFT		RIGHT		LEFT		RIGHT		°F		FEET		DEGREES				TIME
		RPM	RPM	RPM	RPM	MP	MP	MP	MP	DAT	DAT	Alt	Alt	δ _r	δ _a	β	β	
	104	2100	2500	2500	2500	12.0	12.0	22.0	22.0	54	54	7200	7200	-31	-15	2	2	10:08
	99	1900	2500	2500	2500	13.0	13.0	22.0	22.0	55	55	7020	7020	-34	-16	4	4	
	95	1800	2500	2500	2500	13.7	13.7	22.0	22.0	55	55	6720	6720	-39	-20	1	1	
	89	1600	2500	2500	2500	15.0	15.0	22.0	22.0	56	56	6600	6600	-44	-25	-2	-2	10:10
	105	2000	2500	2500	2500	12.5	12.5	23.5	23.5	58	58	5080	5080	-34	-15	1	1	10:12
	99	1800	2500	2500	2500	14.0	14.0	24.0	24.0	60	60	4760	4760	-39	-19	0	0	
	95	1700	2500	2500	2500	14.5	14.5	24.0	24.0	60	60	4600	4600	-44	-22	-2	-2	
	90	1550	2500	2500	2500	15.5	15.5	24.0	24.0	60	60	4520	4520	-48	-25	0	0	10:14

Comments:

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Page 2 of 2

Flight No. 1

Time 9:45 a.m. 10:23 a.m

Kimberlin

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F L I G H T T E S T D A T A S H E E T

Page 1 of 2

Flight No. 2

Date September 17, 1986 Model C-310 Ident. No. N 11UT

Purpose: Determination of VMCA

Time 9:12 a.m. to 9:52 a.m.

Beginning Fuel 19 gal. L. 08 Ending Fuel 10 gal. L. 08

Total Time 40 minutes

19 gal. R. 08 9 gal. R. 08 Gross Wt. 4465 lbs Pilot Kimberlin

C. G. 28.10% C Observer

Config.:

Kts	LEFT		RIGHT		LEFT		RIGHT		°F		FEET		DEGREES			
	V _i	RPM	RPM		RPM	MP	MP		OAT		Alt		δ _r	δ _a	β	TIME
	104	2050	2500		2500	7.0	21.5		51		7400		-30	-14	-5	9:31
	100	1950	2500		2500	7.0	22.0		51		7100		-30	-16	-1	
	95	1800	2500		2500	8.0	22.0		53		6900		-35	-20	-2	
	90	1600	2500		2500	8.5	22.0		53		6740		-40	-22	0	
	84	1500	2500		2500	9.0	22.0		53		6600		-47	-26	-3	9:34
	106	2000	2500		2500	7.0	23.5		57		5480		-30	-16	-3	9:35
	99	1850	2500		2500	7.0	23.5		60		5040		-33	-20	-1	
	93	1650	2500		2500	9.0	24.0		60		4900		-39	-24	+2	
	90	1550	2500		2500	9.0	24.0		60		4780		-45	-26	-3	9:38

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Page 2 of 2

Flight No. 2

Time 9:12 a.m. to 9:52 a.m.

Pilot Kimberlin

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Comments:

FLIGHT TEST DATA SHEET

Page 1 of 2

Date September 18, 1986 Model C-310 Ident. No. N 11UT

Flight No. 3

Purpose: Determination of VMCA

Beginning Fuel

Ending Fuel

Time 9:15 a.m. to 9:47 a.m.

Total Time 32 minutes

30 gal. L. 08 22 gal. L. 08 Gross Wt. 4607 lbs Pilot Kimberlin

30 gal. R. 08 22 gal. R. 08 C.G. 28.2% C Observer

Config.:

Kts	LEFT		RIGHT		LEFT		RIGHT		of		FEET		DEGREES				TIME
	V _i	RPM	RPM		MP	MP	MP	MP	OAT	Alt	Alt	Alt	φ	δ _r	δ _a	β	
	110	2200	2500		15.0	21.5	21.5	21.5	54	7440			5	-41	-17	-20	9:28
	104	2050	2500		15.5	22.0	22.0	22.0	51	6900			5	-37	-21	-2	
	100	1920	2500		16.0	22.0	22.0	22.0	55	6680			5	-34	-24	0	
	94	1750	2500		17.0	22.0	22.0	22.0	55	6560			6	-41		0	
	90	1600	2500		17.5	22.5	22.5	22.5	55	6400			10	-46	-31	0	9:31
	110	2100	2500		15.0	23.0	23.0	23.0	58	5420			8	-30	-19	0	9:33
	104	2000	2500		16.5	23.5	23.5	23.5	59	5320			9	-35	-23	0	
	100	1900	2500		17.0	23.5	23.5	23.5	60	5200			6	-37	-23	-1	
	95	1750	2500		17.5	23.5	23.5	23.5	61	5100			0	-39	-30	+1	
	89	1500	2500		19.0	23.5	23.5	23.5	62	5000			10	-46	-34	+1	9:35

Comments:

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Page 2 of 2Flight No. 3

Total Time 32 minutes

Pilot Kimberlin

Observer

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Comments:

VITA

Barry Gerard McCarthy, son of Lawrence and Irene McCarthy, was born July 28, 1962, in Chicago, Illinois. In 1967, his family moved to Nashville, Tennessee, where he later attended and graduated from Holy Rosary Academy and Father Ryan High School. In 1980, he entered Middle Tennessee State University and in May 1985 he graduated with his Bachelor of Science Degree. In the fall of that year he accepted a position as a full-time graduate research assistant at The University of Tennessee Space Institute. He was assigned to the Flight Research Group and in the Spring of 1987 was awarded the Master of Science Degree with a major in Aviation Systems. In May of the same year he married Kathy-Lee Smith and accepted a position with Douglas Aircraft Company in Long Beach, California.

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