STOL HIGH-LIFT DESIGN STUDY

Volume I. State-of-the-Art Review of
STOL Aerodynamic Technology

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FOREWORD

This report was prepared for the Air Force Flight Dynamics Laboratory by The Boeing Company. The study was conducted under USAF Contract F33615-70-C-1277, "STOL High-Lift Design Study", with Captain Garland S. Oates as Project Engineer for the Air Force.

The study was performed from January through December, 1970 by Mr. Colin Widdison of the Vertol Division and Mr. Fred May of the Military Airplane Systems Division.

This technical report has been reviewed and is approved.

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ABSTRACT

The state-of-the-art of STOL aerodynamic technology for selected lift/propulsion concepts has been surveyed to identify the available test data and prediction methods in the literature. The report consists of two volumes.

In Volume I important technology areas and information necessary for the evaluation of STOL aircraft aerodynamics are listed; the aerodynamic test data and prediction methodology relevant to the deflected slipstream and externally blown flap concepts are assessed with emphasis on the latter; an empirical method for the prediction of the longitudinal aerodynamic characteristics of externally blown flap configurations is presented; and high-lift technology for five lift/propulsion concepts is assessed in application to a medium-sized STOL transport.

Volume II consists of a Bibliography that resulted from a literature search for aerodynamic information related to seven lift/propulsion concepts suitable for STOL aircraft. The Bibliography contains references to about 900 reports classified by concept and by technology area.

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SUMMARY

The state of the art of STOL aerodynamics has been evaluated by assessing the validity and applicability of available analytical and empirical prediction methods.

The important technological areas in STOL aircraft aerodynamic design are outlined and a list is presented of the information needed to permit STOL aerodynamic design, analysis, and evaluation.

An extensive literature search was made of aerodynamic prediction methods and data applicable to seven distinct STOL concepts:

Externally Blow Flaps
Deflected Slipstream
Jet Flap
Mechanical High Lift Devices (including boundary layer control)
Fan-in-Wing
Tilt Wing
Direct Jet Lift

The results of the literature search are summarized in a comprehensive bibliography of nearly 900 references. The bibliography comprises Volume II of this report.

Primary emphasis was placed on researching prediction methods applicable to externally blown flap and deflected slipstream configurations. The literature search revealed a considerable amount of potentially useful test data and a number of prediction methods related to the deflected slipstream configuration. In contrast, it was discovered that relatively little test data exists for the externally blown flap configuration and that there was a complete lack of analytical methods applicable to that concept. As a result, the program activity was redirected at mid-term to include the development of a method for the prediction of the aerodynamic characteristics of externally blown flap configurations.

The methods available for the deflected slipstream concept were reviewed and analyzed and one method was evaluated against test data.

The gaps and deficiencies in the methods and data related to both of the above concepts have been defined and an outline has been given for the research programs that are needed to correct the voids.

A comparison of five lift/propulsion concepts is made in terms of their applicability to a near term Medium STOL Transport airplane. The concepts studied were Externally Blown Flaps, Internally Blown Flaps, Mechanical High Lift Devices, Augmentor Wings and Direct Jet Lift.

1. INTRODUCTION

This survey of the state-of-the-art of STOL aerodynamics was initiated by the necessity of providing information concerning the technical base for the design of STOL aircraft. In particular, the aim was to assess the current state of the art of aerodynamics as applicable to near-term STOL transport aircraft.

The approach taken in the survey was first to limit the survey to include only those STOL concepts that were in a reasonably advanced state of development and could therefore be considered as likely candidates for the near-term STOL transport role. Seven concepts were chosen and are listed in the summary. Next, the important technology areas of STOL aerodynamics and the information required in evaluating the aerodynamic characteristics of a STOL aircraft design were defined and used as guidelines in the conduct of a literature search. The literature search resulted in a bibliography of about 900 references, most having abstracts. The results of the literature search are classified into technology areas and STOL concepts and are presented in Volume II of this report.

From the outset of the study the concepts of major interest were the externally blown flap and deflected slipstream types. The literature search revealed several prediction methods that were applicable to calculation of the characteristics of deflected slipstream configurations, but none applicable to the externally blown flap concept. Consequently, it was decided that the major emphasis of the remainder of the study should be placed upon the development of a suitable prediction method for such configurations. Such a method, based upon jet flap theory has been developed.

The analysis of methods and data related to the deflected slipstream concept consists of a critical survey of a number of prediction methods and correlation of a small amount of flight and wind tunnel test data with one of the methods.

The work related to the externally blown flap and the deflected slipstream concepts is presented in two separate parts of the report. Each self-contained portion (Sections 4 and 5 of the report) includes an assessment of the gaps and deficiencies in methods and data, recommendations for corrective programs and a list of related references.

2. FACTORS IMPORTANT TO THE DESIGN AND OPERATION OF STOL AIRCRAFT

2.1 INTRODUCTION

Short takeoff or landing performance requirements impose special problems that are unique to STOL aircraft. This section includes a review of the aerodynamic factors that influence the design and operation of the airplane. These factors include the basic aircraft requirements for obtaining STOL performance, criteria and margins for safe operation, characteristics of the airplane in ground effect, and testing requirements and techniques.

2.2 REQUIREMENTS FOR OBTAINING STOL PERFORMANCE

In the context of this study the term STOL includes only those types of airplanes that achieve short takeoff and landing performance without unduly sacrificing cruise performance. In this sense, successful STOL transport aircraft are characterized by good cruise speed capability and comfortable ride qualities.

To obtain short takeoff performance the airplane must either takeoff at a very low speed or employ a sufficiently high thrust level to accelerate to the takeoff speed in the required distance. In fact, both measures are taken, takeoff speed being lower and thrust/weight ratio being higher than for a conventional aircraft. Figure 1 illustrates a typical relationship between takeoff distance and a parameter combining takeoff speed and thrust/weight ratio for a given configuration.

The attainment of short air distance during landing results from a steep approach path and ground run is minimized by touching down at the lowest possible speed and obtaining the maximum deceleration.

Figure 2 indicates typical approach speeds required to achieve short field landing performance using two different sets of ground rules for calculating the distance. Figure 3 shows the results of a parametric study of landing performance in which the major factors that affect landing distance were systematically varied. Starting from a base point representing an airplane designed for conventional landing, a number of things can be done to reduce its landing distance. However, large changes in parameters such as reverse thrust, ground roll friction and approach angle, have relatively little effect, while reduction in stall speed is very powerful. The conclusion is that in order to achieve STOL landing performance, the airplane must be designed to fly slowly.

Figure 4 shows the combinations of wing loading and usable lift coefficient required to achieve the low speed operation required for short takeoff and landing. It is seen that either low wing loadings — of the order of 65 psf or less — or powered lift systems are required to achieve landing distances of 2000 feet.

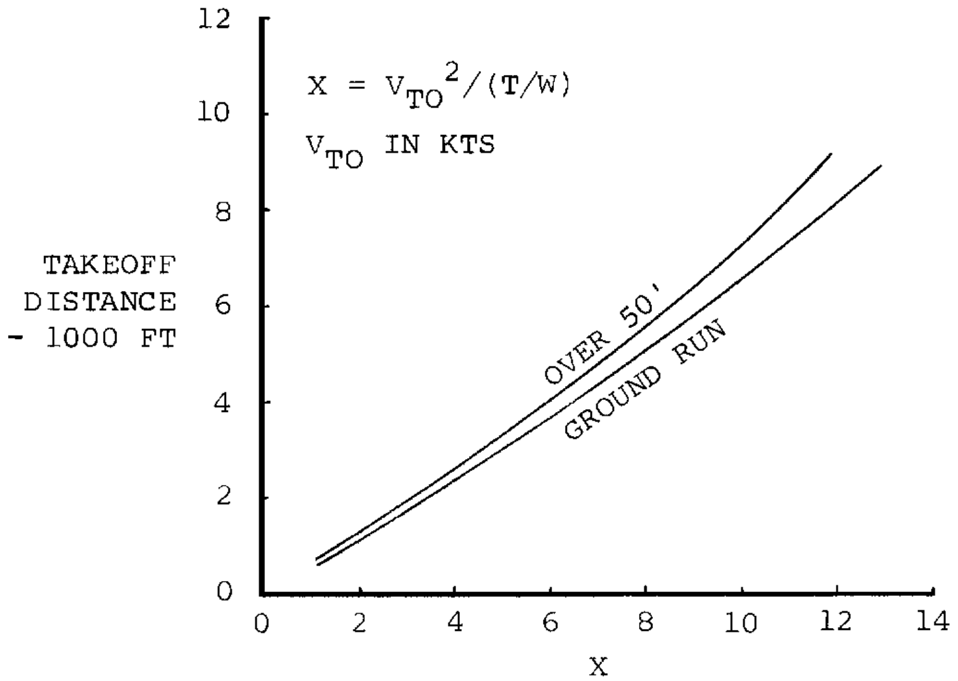


FIGURE 1. VARIATION OF TAKEOFF DISTANCE WITH TAKEOFF SPEED AND THRUST/WEIGHT RATIO

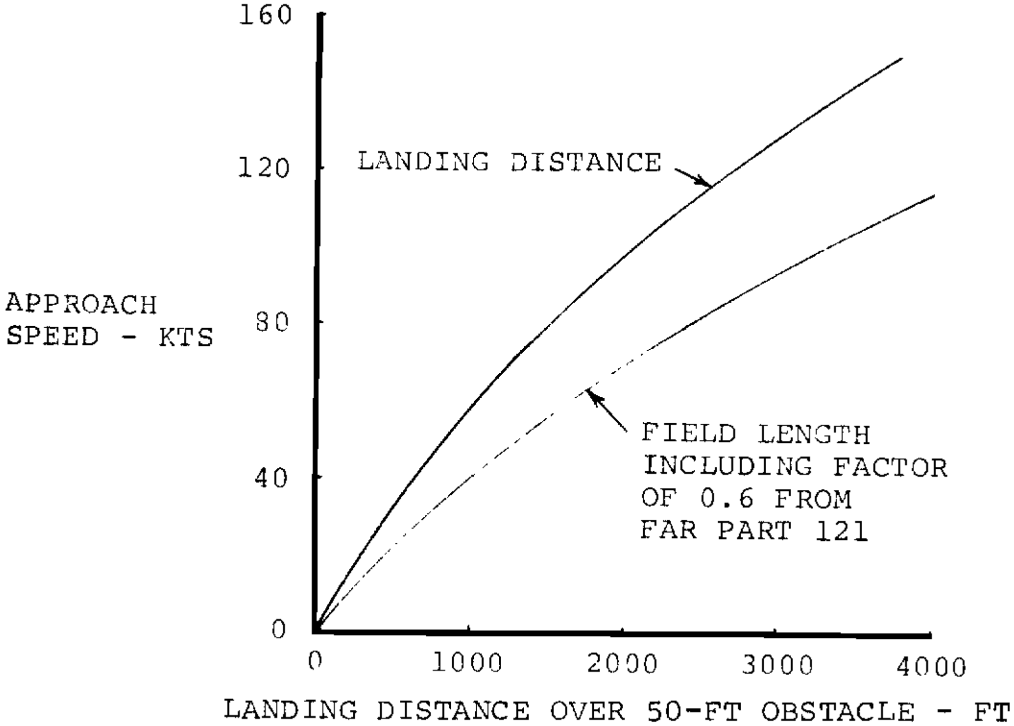
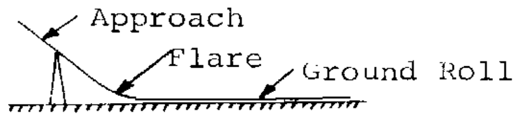


FIGURE 2. VARIATION OF LANDING DISTANCE AND FIELD LENGTH WITH APPROACH SPEED



*Base Point Wing Loading = 80 PSF Thrust Loading = 0.3
Aspect Ratio = 6 $C_{L_{max}}$ = 2.0

Approach Angle =-3° Approach Velocity = 120 Knots
Rate of Sink at Impact = 0 Ground Roll Friction Coefficient =0.225
Reverse Thrust = 30% Velocity at Which Engines are Idled=40 Knots
Sea Level Standard Day

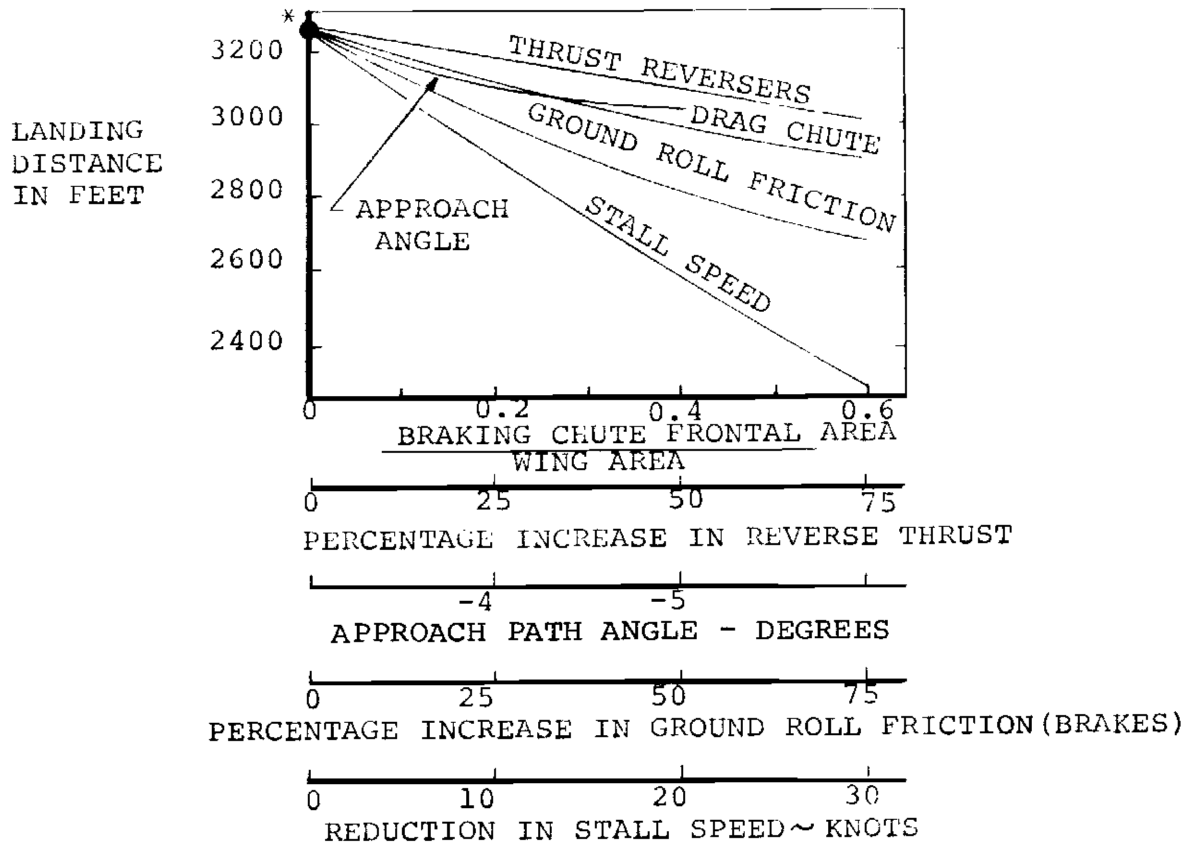


FIGURE 3. POSSIBLE IMPROVEMENTS IN STOL LANDING

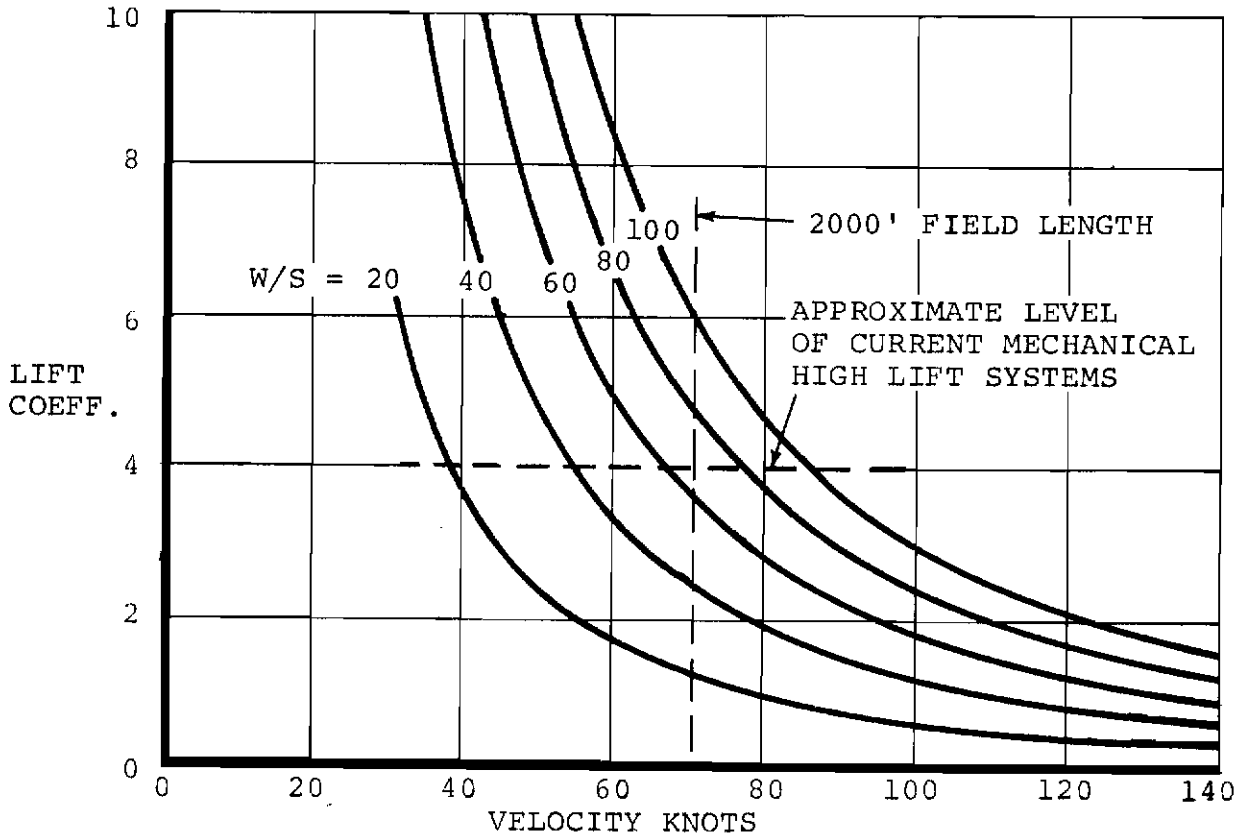


FIGURE 4. LIFT COEFFICIENT FOR LEVEL FLIGHT

Airplanes relying on conventional aerodynamic control surfaces have a limit on the minimum flight speed because of the low dynamic pressure achieved. Thus, to attain speeds sufficiently low to perform STOL maneuvers, new techniques are required for aircraft and flight path control. Figure 5 illustrates the minimum usable speed for a variety of STOL aircraft expressed as a function of the wing loading.

It is seen that to achieve the low speeds required for STOL performance while using wing loadings in the 60 psf to 100 psf range, new control techniques utilizing propulsive augmentation of the aerodynamic controls are required.

The effect of wing loading, aspect ratio and wing sweep on the vertical acceleration an aircraft is subjected to when it flies into a vertical gust of one foot per second is presented in Figure 6 for an aircraft flying at 300 knots at sea level. It shows that low wing loading leads to high gust sensitivity and is the most influential parameter of those shown. The Δg due to a given gust varies as $1/(W/S)$ for any given flight speed.

It is therefore seen that to meet the basic STOL objectives of short field operation — which requires low speed capability — and comfortable ride qualities, one solution is to use high wing loading to minimize gust sensitivity and a powered lift system to achieve low speeds.

Another solution would be to use a lower wing loading and employ a gust alleviation system to ensure necessary ride qualities. This solution reduces the thrust requirement for powered lift STOL systems and provides a natural means for achieving acceptable noise levels.

Figure 7 shows the reduction in g per fps of vertical gust velocity that can be obtained by the use of a LAMS-(load alleviation and mode stabilization) controller.

STOL aircraft designed for military application and intended for operation from rough semi-prepared sites have additional design requirements. A high degree of agility and maneuverability on the ground is required if the aircraft is to be placed in revetments or other concealed areas and for movement in confined spaces. Recirculation of flows due to use of reverse thrust on semi-prepared strips can lead to significant foreign object damage to engines.

2.3 CRITERIA AND MARGINS

2.3.1 Performance Criteria

It is necessary, for safety of operation, to ensure that the aircraft maintains a sufficient margin away from any dangerous condition which could occur due to possible deviations from the normal operating condition. These deviations include atmospheric gusts, critical engine failures, and wave-off or other required

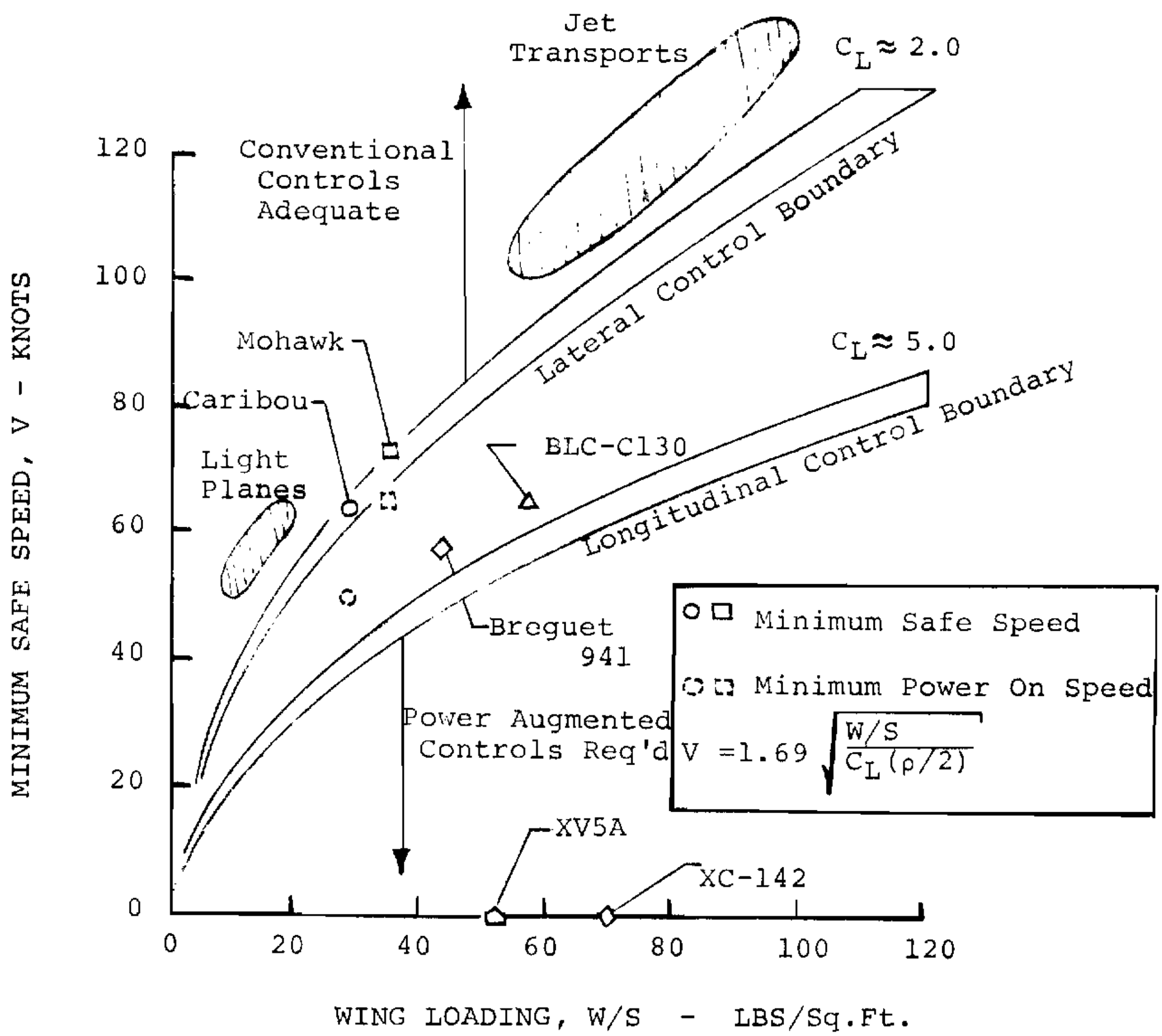


FIGURE 5. MINIMUM SAFE SPEED

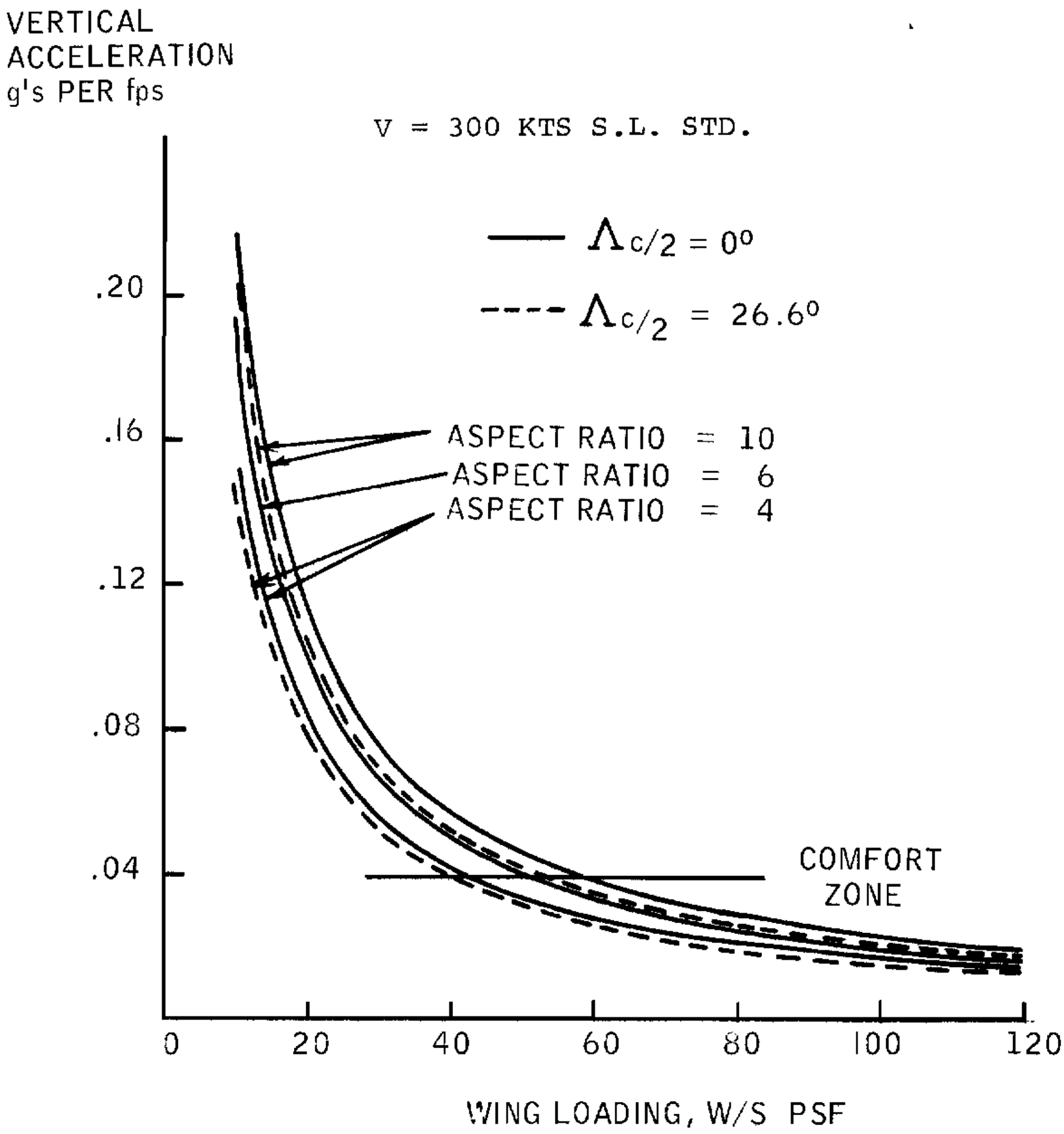


FIGURE 6. GUST SENSITIVITY OF RIGID-BODY MOTION

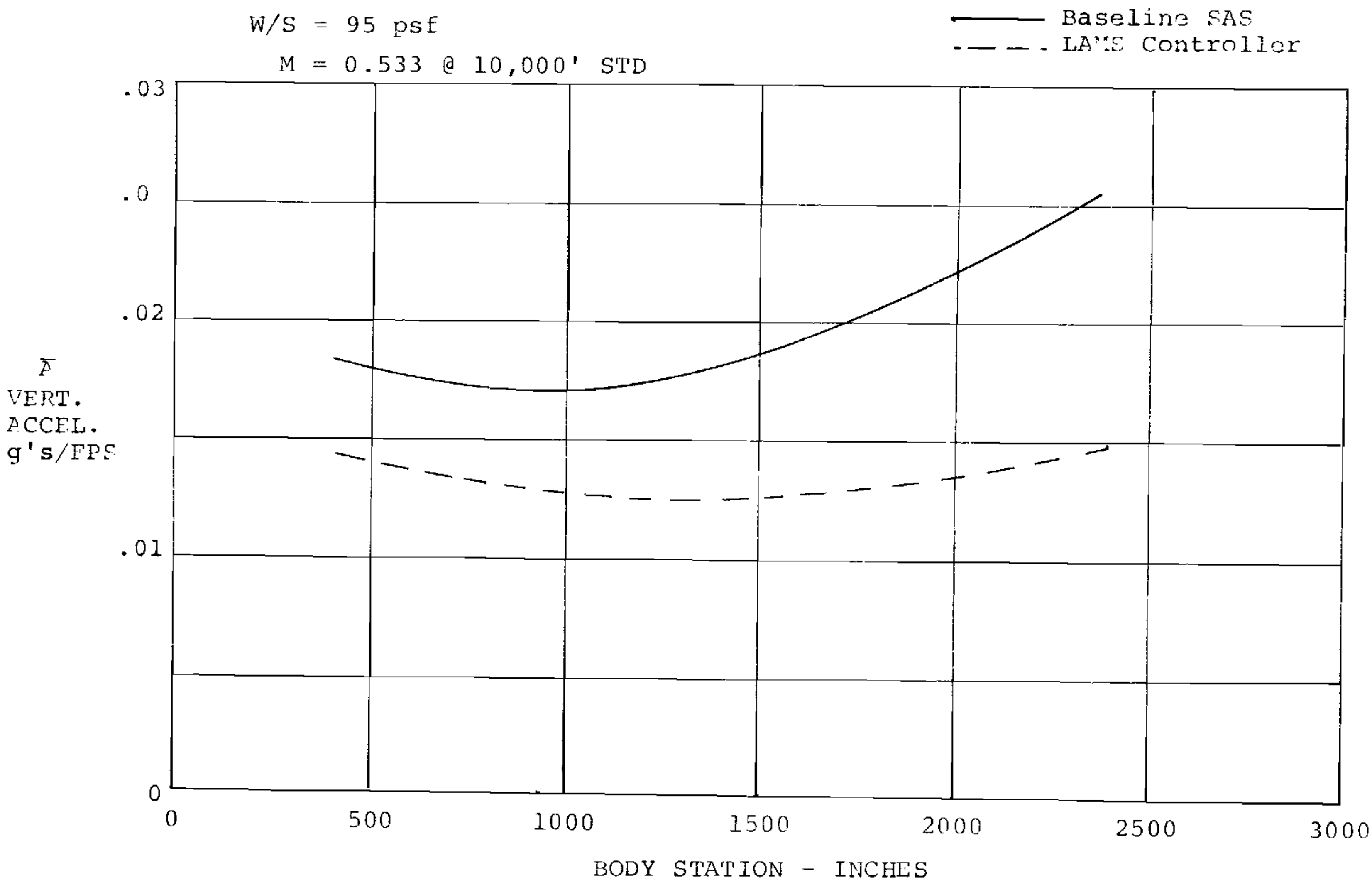


FIGURE 7. VERTICAL GUST ALLEVIATION

flight path corrections. Gusts effectively change the flight speed (longitudinal gusts) or angle of attack (vertical gusts), imposing the requirement for speed and angle of attack margins. Flight path adjustments include transient altitude changes and turns. These impose a requirement for a margin on normal load factor available. For conventional aircraft, normal load factor, angle of attack, and speed are directly related. Therefore, these required margins are traditionally ensured by application of a specified margin on flight speed above stall speed.

Since the lift of a STOL airplane can depend strongly on the power setting as well as the flight speed and attitude, it is necessary to develop STOL criteria that differ from those applied to conventional aircraft. Angle of attack, speed, load factor, and power margins must be independently applied for the STOL aircraft.

Figure 8 shows the influence of STOL margins on the flight envelope of an aircraft. While the aircraft represented is a four engine turboprop, the principles it illustrates are valid for any STOL airplane. Presented in the top portion are steady state rates of climb or sink for various power settings as a function of speed. Also shown are lines of constant flight path angle. On the bottom portion of the figure the maximum load factor attainable versus speed is shown for 1, 2, 3 and 4 engine operation.

The criteria for this aircraft were that it be able to pull 1.2g with all engines operating, 1.1g with one engine out and have a 3-degree climb-out angle with one engine out. For this particular aircraft configuration and weight it is seen that the 1.2g requirement with all engines operating is least critical, with a minimum speed of about 23 knots, the 1.1g requirement with one engine out is more critical and that the 3-degree climb-out angle is the most critical requirement as it establishes the highest speed of about 40 knots in order to be satisfied. A required margin on flight speed is not shown but could have been included in a similar manner.

Compared with conventional aircraft flight experience, STOL flight experience is relatively sparse. STOL experience in the approach maneuver is summarized in Table I from Reference 1. The table illustrates maximum descent angles, and minimum flight speeds for eight STOL airplanes that have been flown by NASA pilots. Also included are the approach speeds used, ratio of approach speed to minimum speed and the margin of approach speed over the minimum. The table also indicates the factors that determine the minimum speed. It should be noted that the minimum speed is not necessarily fixed by $C_{L_{max}}$.

Factors such as those above are interpreted into ground rules which strongly affect the stated or advertised STOL performance as it is calculated for any given vehicle. These ground rules form the basis for standardizing the comparison between competing concepts and include such factors as obstacle height, rate of descent at touchdown, load factor in flare, if a flare is used, time delays between

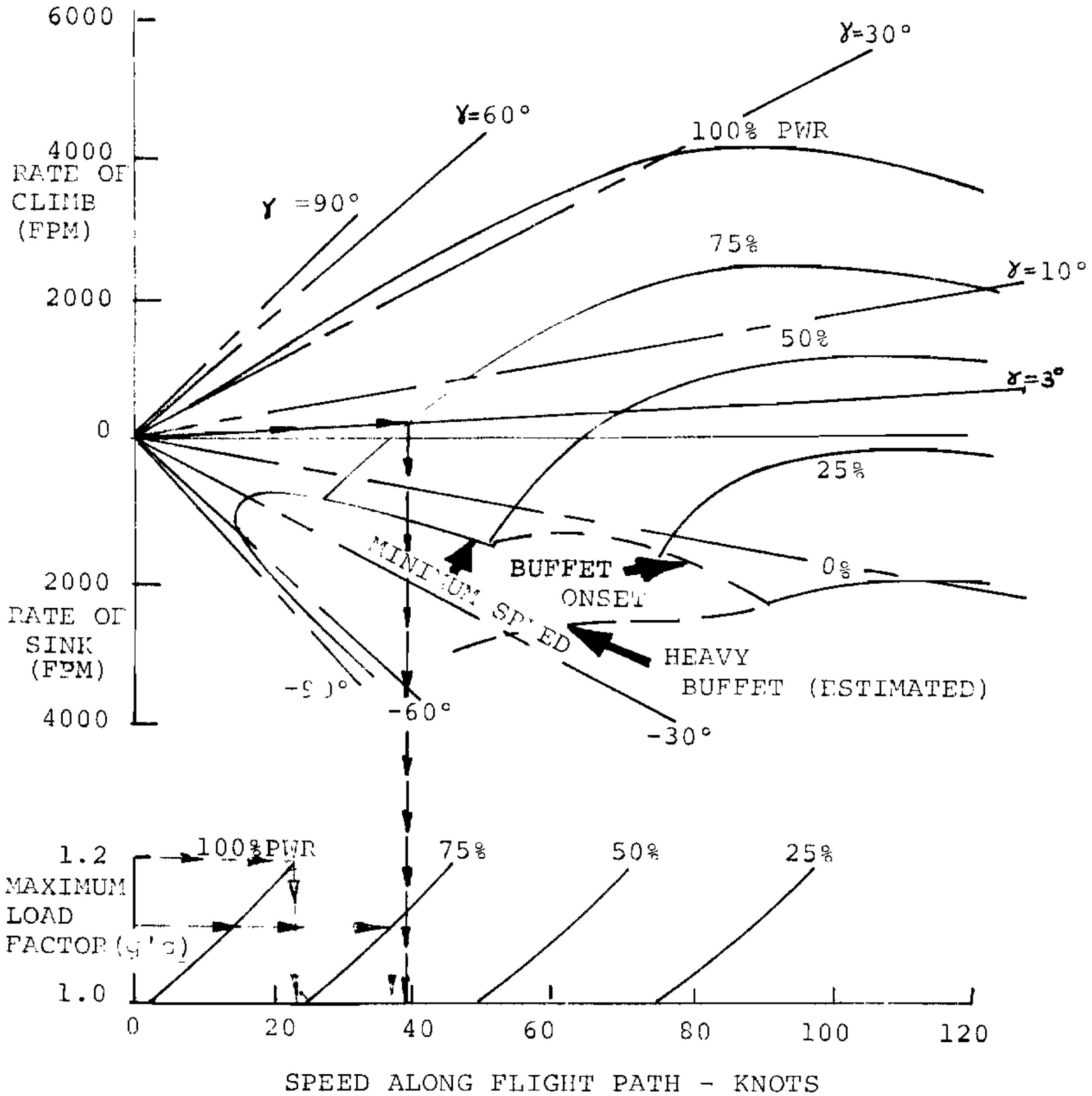


FIGURE 8. STOL MARGINS

touchdown and activation of brakes, spoilers, etc., deceleration on the ground and relationship between field length and actual takeoff and landing distances.

TABLE I

SUMMARY OF STOL FLIGHT EXPERIENCE

Aircraft	γ, deg	V_{min} knots	Limited by	$\frac{V_a}{V_{min}}$	ΔV_a KTS
YC-134A	-1 -4 -9	64 68 77-1/2	Stall	1.17 1.15 1.24	11 10 19
NC-130B	-1 -2 -4-1/2 -8	55 56-1/2 61-1/2 68	Stall	1.15 1.15 1.14 1.20	8 10 9 13
VZ-3RY	-16	36	Lateral Control	1.11	4
UF-XS	-3-1/2 -5		Stall		
CV-48	-6 -7-1/2	47 51	Stall	1.17 1.18	8 9
BR-941	-4 -5 -6 -7-1/2	49 50 52 53	V_{min}	1.16 1.15 1.14 1.13	8 8 7 7
C-8A	-7	60	Stall	1.17	10
367-80	-3	75	Stall	1.2	15

Figures 9 through 12 serve to illustrate the sensitivity of takeoff and landing performance to ground rules applied to the execution of these maneuvers.

One of the most important relationships is that between the flare maneuver and an acceptable sink speed at touchdown. It is possible that for STOL aircraft a flare should not be used but instead that the allowable sink speed be increased and the landing gear designed, at a cost in weight, to absorb the higher landing loads. The reason that this philosophy may apply is because the distance covered in the flare maneuver is so dependent on individual pilot technique. If a flare is to be used special instrumentation may be required to enable the pilot to start the flare at very precise heights and speeds and to pull the correct g during the flare in order to have consistent and repeatable landing distances.

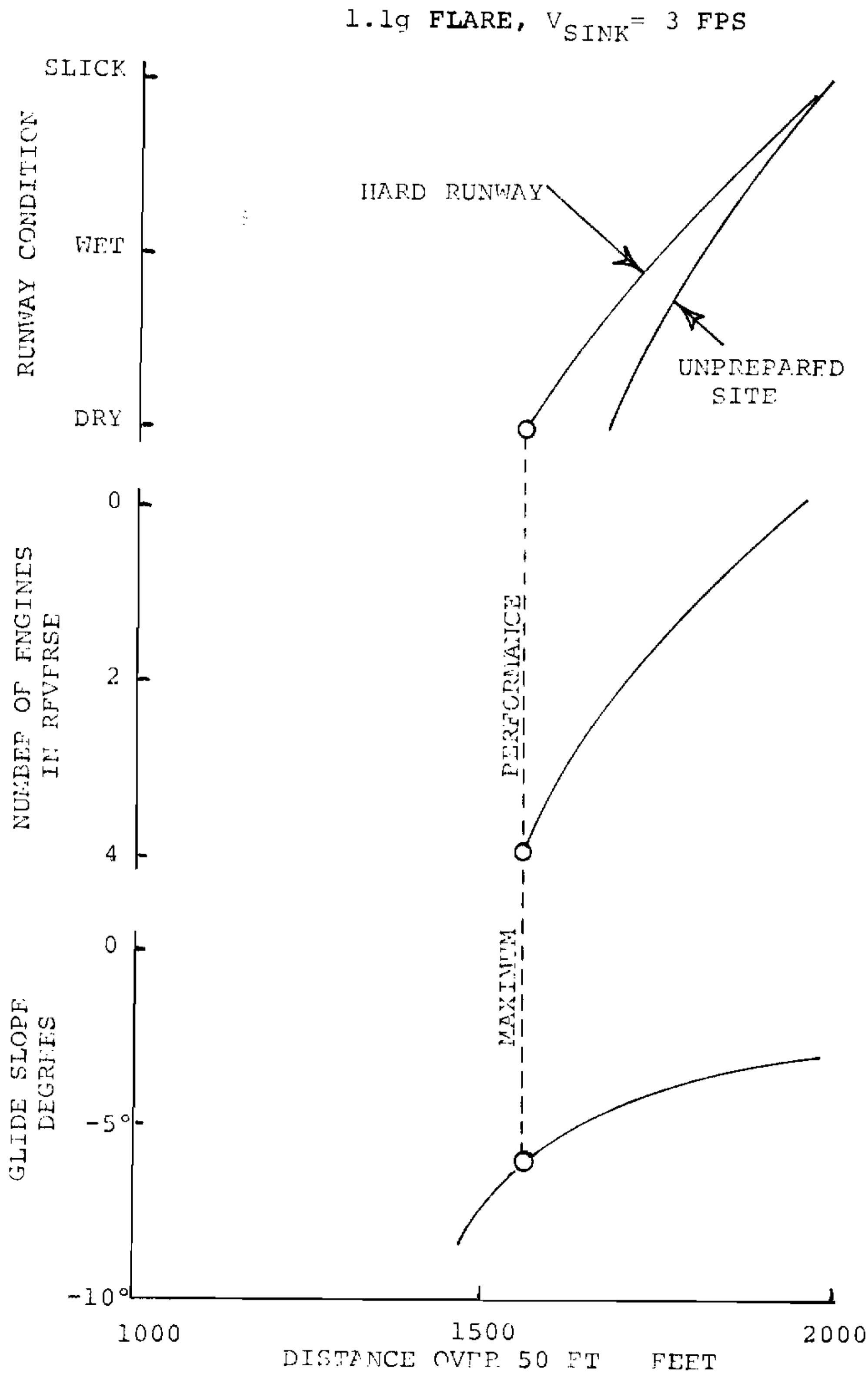


FIGURE 9. SENSITIVITY OF GROUND RULES

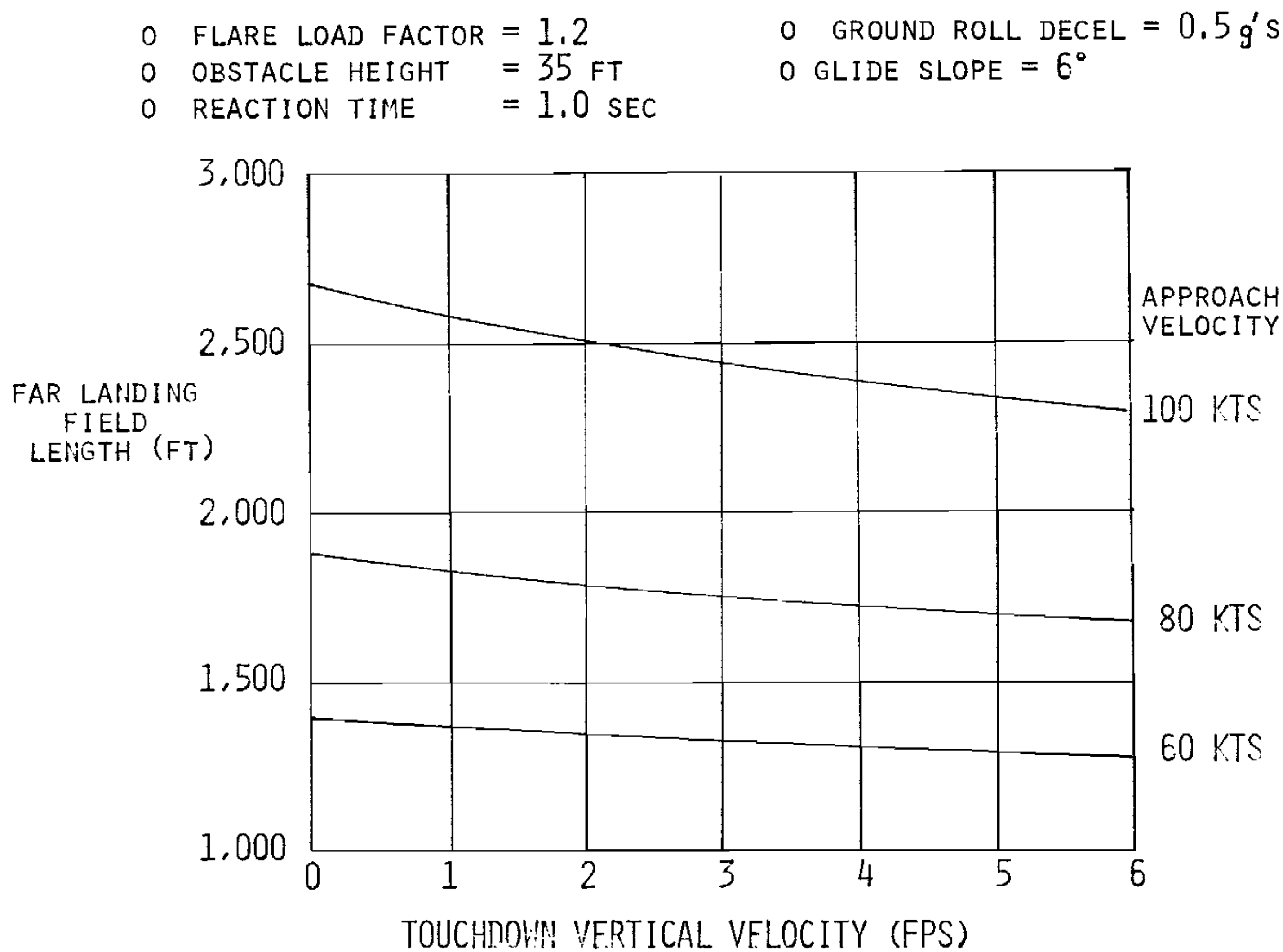


FIGURE 10. LANDING PARAMETRIC

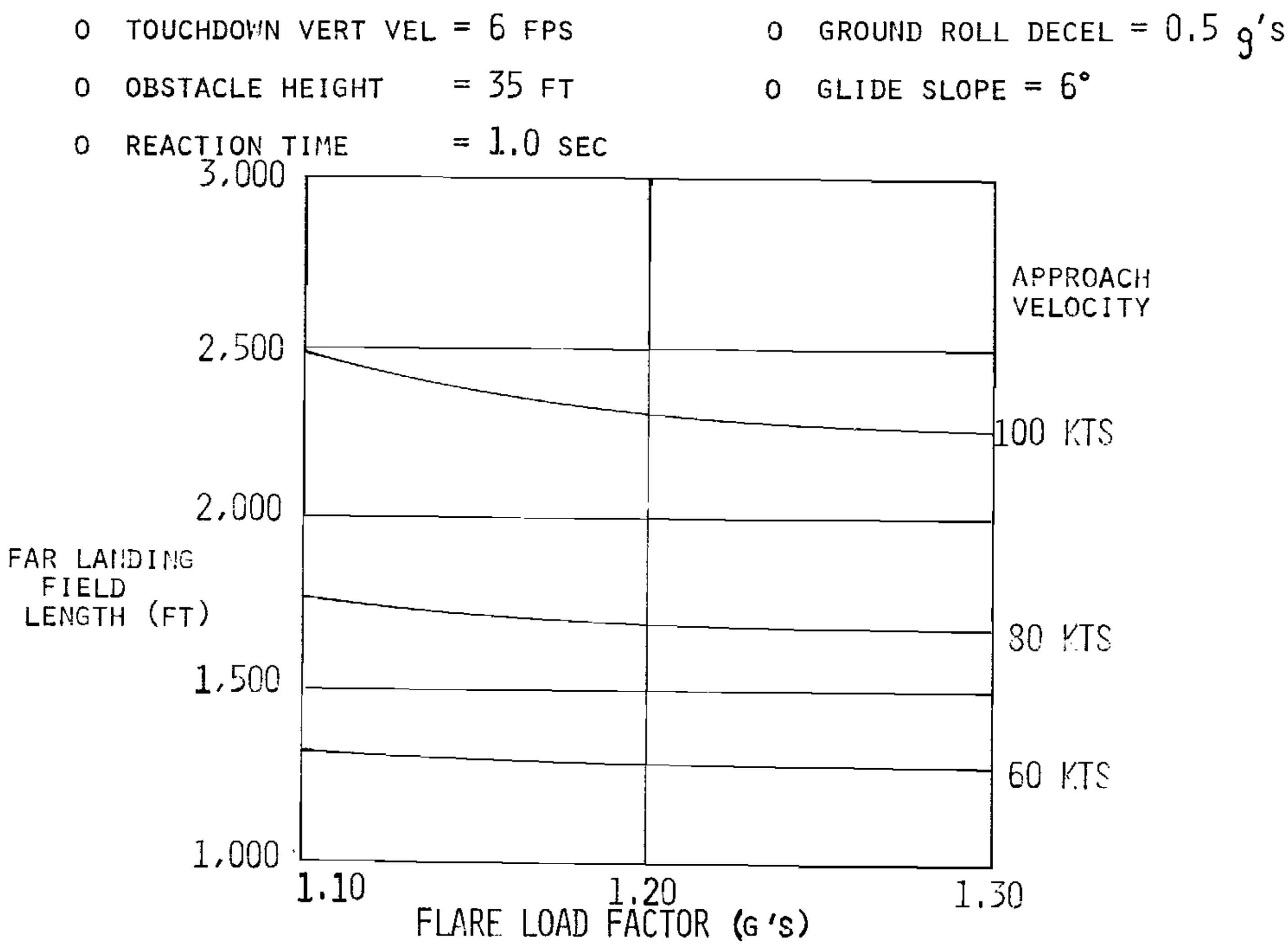


FIGURE 11. LANDING PARAMETRIC

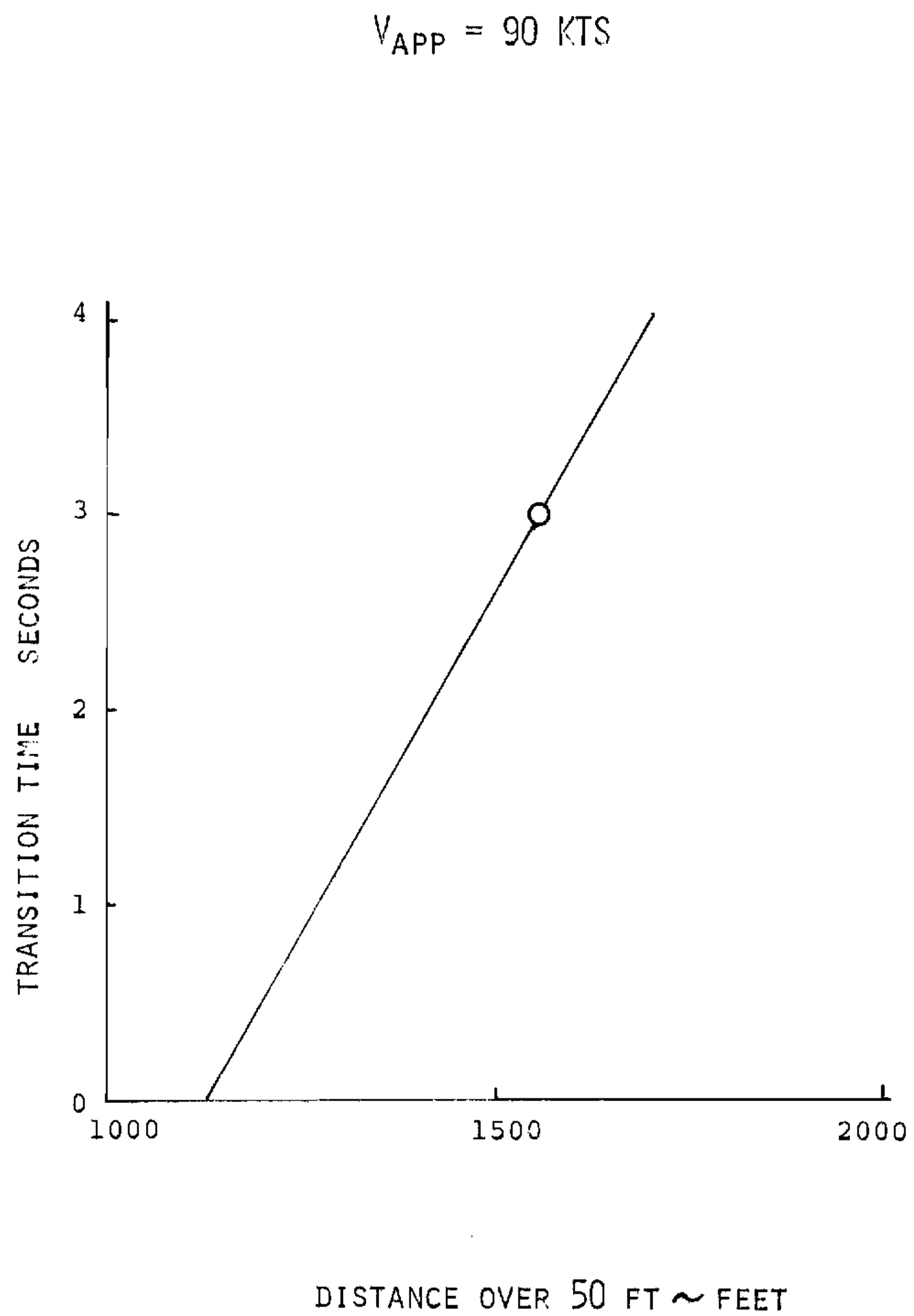


FIGURE 12. SENSITIVITY OF GROUND RULES TO TRANSITION TIME DELAY

Figure 13 shows the results of a study of the permissible takeoff speed, for a typical STOL airplane, with a variety of takeoff criteria considered. It indicates, for any given combination of wing loading and thrust-weight ratio, the minimum permissible takeoff speed and the criterion that dictated the minimum speed requirement. In this study the following criteria were considered:

1.2g normal load factor — all engines operating

$$\left.\begin{array}{l} \text{1.1g normal load factor} \\ V_{STALL} + 10 \text{ knots} \\ 1.2\ V_{STALL} \\ \gamma = 3^\circ \text{ steady climb} \end{array}\right\} \text{one engine inoperative}$$

For the range of wing loadings and thrust-to-weight ratios studied, the normal load factor requirements did not dictate the critical speed. As seen on the figure, three distinct regions appear, in each of which one of the remaining three criteria dictated the minimum speed. As would be expected, for low values of thrust-to-weight ratio, the climb angle criterion dictates speed, with the stall speed margins being critical at higher T/W.

2.3.2 Flying Qualities Criteria

Establishment of STOL flying qualities criteria is urgently needed for several reasons. Many of the criteria that have been developed for conventional aircraft are not applicable to the STOL regime flight characteristics and could limit STOL potential without ensuring safety. The criteria should provide safety at low speeds and reduce the pilot's work load during STOL maneuvers. It is important that the criteria should specify sufficient control power to allow precise and repeatable maneuvers at low speed.

The main problems involving the selection of criteria are the small amount of flight experience and the variety of lift/propulsion concepts employed to achieve STOL performance. The criteria for conventional aircraft have been established over a long period from experience gained with many airplanes. Many conscientious and intelligent efforts to establish STOL criteria have been made by different private and government agencies. Due to the limited amount of flight experience, many criteria, by necessity, are based on personal opinion and as a result there are conflicting requirements.

Presently, no single flying qualities specification applicable specifically to STOL aircraft is available although several sets of suggested requirements have been compiled. Table II lists eight documents containing flying qualities specifications for conventional airplanes and helicopters and proposed recommendations for V/STOL or STOL airplanes.

In addition, Reference 2 represents a concerted attempt by the regulatory agencies and industry to produce an acceptable set of specifications. While the philosophy of a desired criteria appears well understood, quantification of criteria has been difficult and is incomplete at present. The most critical area of STOL aircraft at low speeds has been that of lateral control. This is particularly true of powered lift systems for the engine-out condition.

TABLE II

SAMPLING OF FLYING QUALITIES SPECIFICATIONS

- A proposed military specification for V/STOL flying qualities — Cornell Aeronautical Laboratory
- TND-5594; airworthiness considerations for STOL aircraft
- MIL-F-8785B; military specification flying qualities of piloted airplanes
- NASA TND-331; an examination of handling qualities criteria for V/STOL aircraft
- MIL-H-8501A; general requirements for helicopter flying and ground handling requirements
- MIL-H-8501B (proposed); general requirements for helicopter flying and ground handling requirements
- Agard report 408; recommendations for V/STOL handling qualities
- USAAML TR65-45; suggested requirements for V/STOL flying qualities

Table III shows a comparison of some important parameters which should be specified when flying qualities criteria are defined; this listing is taken from Reference 1.

The parameters that define control response are control power, force, linearity, cross coupling, and apparent damping. The parameters that specify stability and damping are also shown.

Since handling qualities are judged by pilots' opinions, the pertinent parameters were chosen so that they could be easily recognized, appreciated, and quantified, readily evaluated for compliance, and would include the effect of factors that influence the response or behavior of the aircraft.

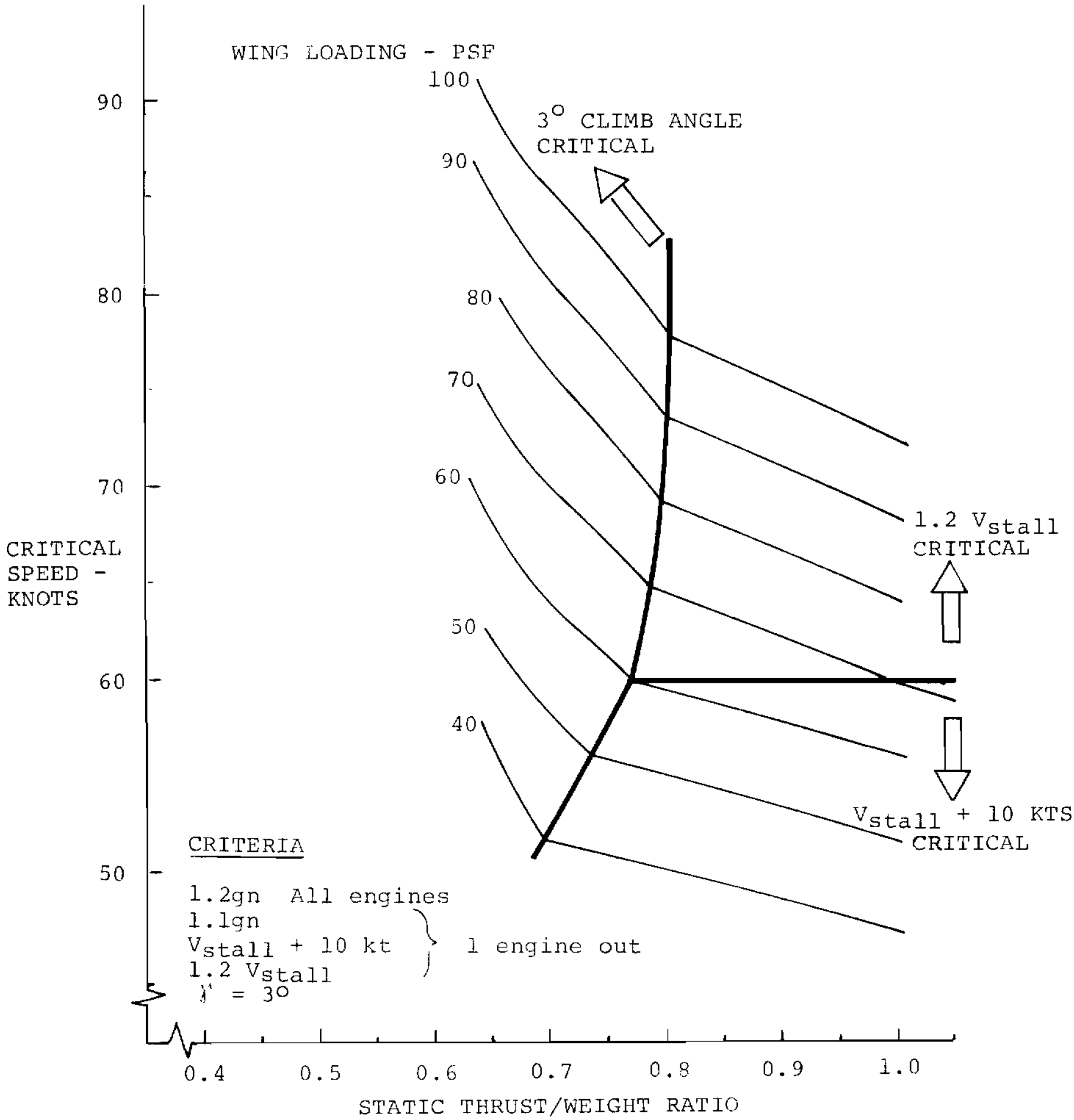


FIGURE 13. EFFECT OF CRITERIA ON MINIMUM TAKEOFF SPEED FOR A TYPICAL STOL AIRPLANE

TABLE III

COMPARISON OF FLYING QUALITIES PARAMETERS
(FROM REFERENCE 1)

CONTROL RESPONSE:

Item	Parameters to be Measured in Roll Axis	Yaw Axis	Pitch Axis
1. Control Power	Time to 30° Bank Angle	Steady-State Sideslip Angle	Time for 10° Attitude Change
	Roll Accel. within 1/2 SEC	Time for 15° change in heading	Pitch Accel. within 1 SEC
	Max. Control Deflection	Max. Pedal Deflection	Max. Control Deflection
2. Force	Max. Force to Achieve Item 1.	Force to Achieve Item 1.	Max. Force to Achieve Item 1.
3. Linearity	Roll Accel. per Unit Stick Defl.	Variation of Side-slip Angle with Pedal Deflection	Pitching Accel. per Unit Stick Displacement
4. Cross Coupling	$(\Delta\beta/\Delta\phi)MAX$	Effective Dihedral	
	$\Delta\theta$	Response About Longitudinal Axis	
	a_n/g		
5. Apparent Damping	Number of Control Reversals to Stabilize		Number of Control Reversals to Stabilize

STABILITY AND DAMPING:

	Lateral-Directional	Pitch-Axis
1. Stability	Directional: Period of Oscillation Spiral: Time to Double Amplitude Dihedral Effect: No Criteria, Insufficient Information	Insufficient information available
2. Damping	Directional: Time to Half Amplitude Lateral: Roll Time Constant	Level of $M\alpha$ should not be so low that short period motion is aperiodically divergent

It should be noted that this treatment is very preliminary and should be modified as new concepts of STOL aircraft with more advanced control and stabilization systems are developed and tested.

Table IV compares the control response and stability and damping criteria from Reference 1 and the proposed military specification for V/STOL flying qualities from Cornell Aeronautical Laboratory. This comparison is for the lateral axis. As can be noted, there are many areas where the proposed criteria are similar, and other areas where they are dissimilar. A comparison of the criteria about the directional and longitudinal axes would show other points of agreement and disagreement.

Figure 14 compares the maneuver control power criteria from several sources for the roll, yaw and pitch axes. These data were obtained from the sources which are referred to in the figure. Two facts should be noted. First, these data apply to an attitude or rate type control system. Other types such as translation control have no requirements defined to date. Second, some of the proposed requirements are independent of aircraft gross weight while others vary with weight.

Further work is necessary in this area to establish reasonable criteria for a wide variety of STOL lift/propulsion concepts. Additional research must be performed to define the gust, wind shear and crosswinds that are encountered in STOL operation. A systematic study should be made to relate control system operational characteristics to control power, sensitivity and damping especially in the IFR condition. Finally, more flight experience is required to define methods of operation in the terminal area and to define the guidance and displays needed.

2.4 GROUND EFFECT

Flying close to the ground causes modification of forces and moments due to distortion of the flow field around the aircraft.

Figure 15 illustrates the theoretical variation of lift in ground effect of a straight wing of aspect ratio 7. The prediction, based on a planar horseshoe vortex, serves to show that the magnitude of the ground effect increases with the amount of lift generated by the wing. At the high lift levels ($C_L \geq 4$) attained by STOL aircraft the planar horseshoe vortex model is an inadequate basis for predictions because of the large flow distortions involved and the complex interaction of the lifting and propulsion systems. Nevertheless, Figure 15 indicates that large lift variations in ground effect are possible for the STOL airplane. In addition to significant effects on forces and moments, the recirculation of flows created by the STOL airplane can cause hot gas reingestion, foreign object damage, and erratic dynamic motion of the aircraft.

TABLE IV

COMPARISON OF LOW SPEED FLYING QUALITIES CRITERIA
(REFERENCE 1 AND CAL V/STOL SPEC)
LATERAL-AXIS

CONTROL RESPONSE

Item	Parameter to be Measured	Level for Safe Operation (PR = 3.5) NASA TND-5594	CAL V/STOL Spec.
Control Power	Time to 30° Bank Angle	No More Than 2.4 SEC	No More Than 1.8 SEC
	Roll Acceleration within 1/2 SEC	More than 0.4 RAD/SEC^2	More than ≈ .36 RAD/SEC^2
	Maximum Control Deflection	No more than 60° wheel deflection or 5" of stick deflection	No more than 60° wheel deflection
Force	Maximum Force to Achieve Control Power	20 LB	3.3 LB → 15 LB.
Linearity	Roll Acceleration per Unit Stick Deflection	Should not increase	No objectionable non-linearities
Cross Coupling	$(\Delta\beta/\Delta\phi)_{MAX}$ $\Delta\theta$ a_n/g	0.3 Not noticeable Less than -0.1	Not excessive -- --
Apparent Roll Damping	Number of Control Reversals to Stabilize	No more than 2	--
LATERAL-DIRECTIONAL STABILITY AND DAMPING			
Lateral Damping	Roll Time Constant	Less Than 2 SEC	Less Than 1 SEC
Dihedral Effect	Lateral Stick Displacement	$\delta_{\ell_\beta} > 0$ $\delta_{\ell_\beta}\beta < .5\,\delta_{\ell MAX}$	$\delta_{\ell_\beta} > 0$ $\delta_{\ell_\beta}\beta_{MAX} < 0.75\,\delta_{LMAX}$
Spiral Stability	Time to double Amplitude	Not less than 20 SEC	Not less than 20 SEC
Dutch Roll	Frequency $\sim\omega_{nd}$ Damping Ratio $\sim\zeta_\ell$	$-\zeta_d\omega_{nd}$ Less than -.09 ζ_d greater than ≈ .10	$-\zeta_d\omega_{nd}$ Less than zero ζ_d greater than .08

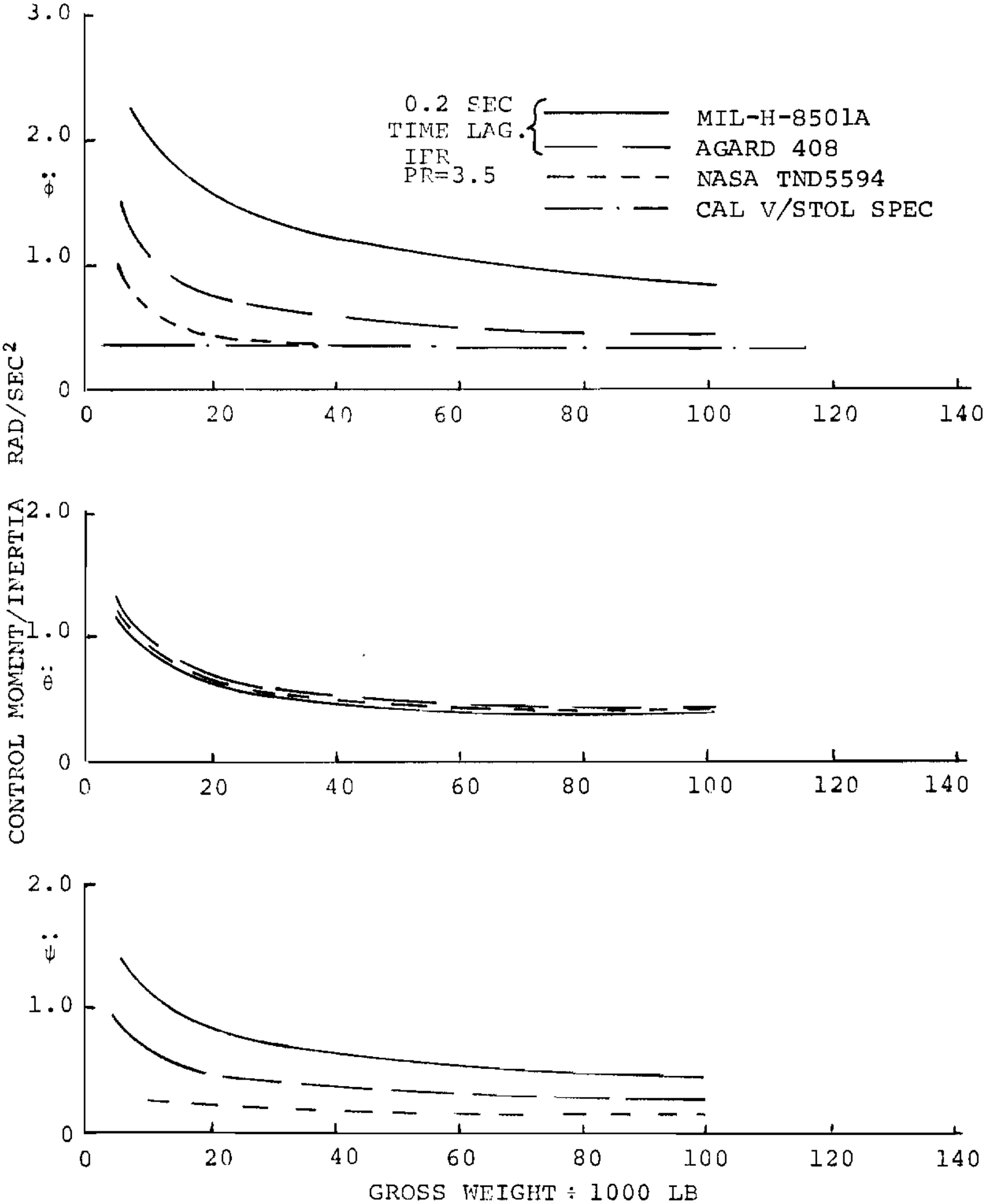


FIGURE 14. CONTROL POWER COMPARISON

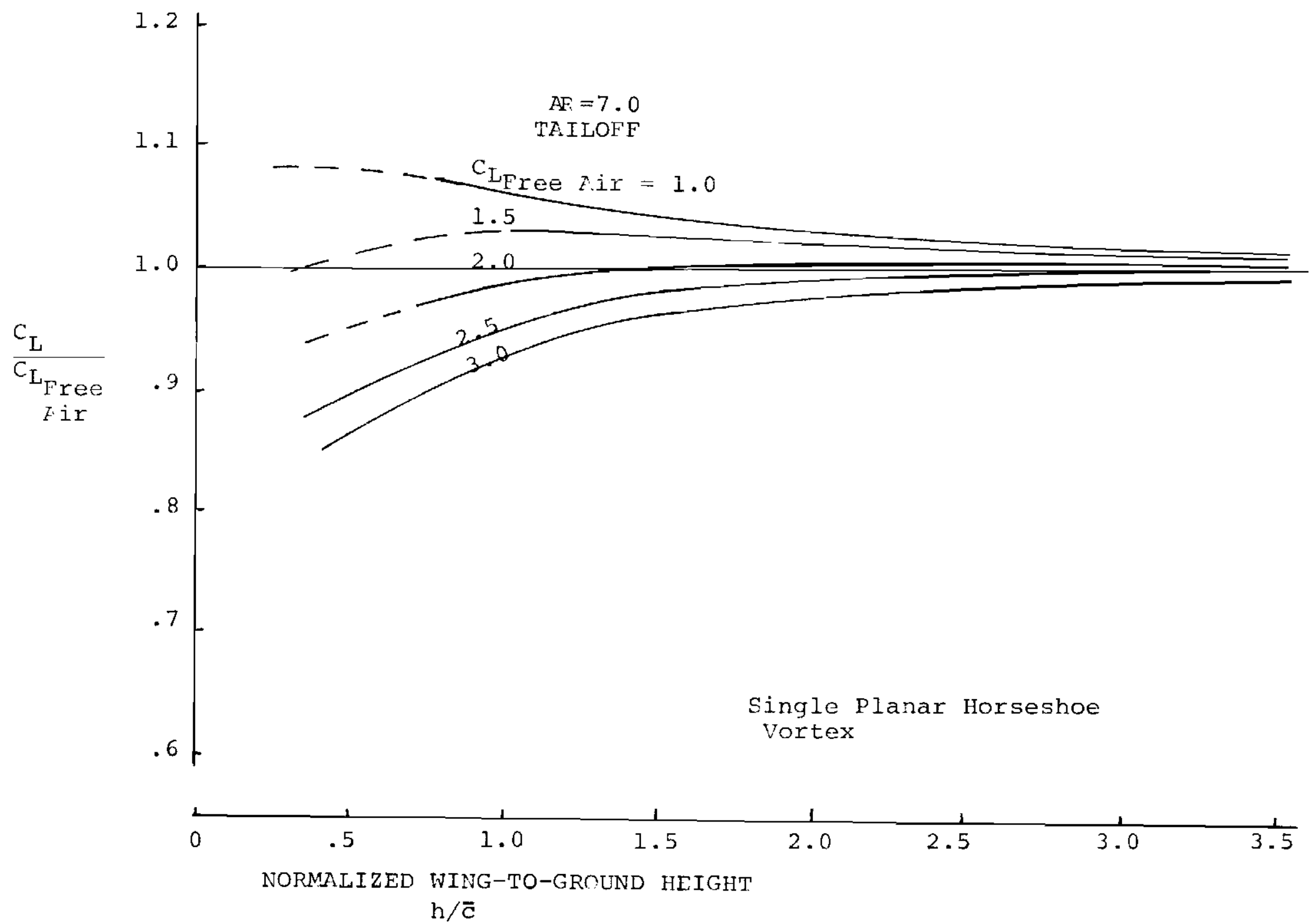


FIGURE 15. THEORETICAL GROUND EFFECT

Because of the variety of concepts employed for STOL aircraft, it is probable that each concept should be treated separately in calculation of the ground effects.

A typical measurement of the effect of configuration differences is shown in Figure 16. These data were obtained from wind tunnel tests of models of the Boeing 727 and 747 airplanes using a fixed ground board. It should be noted that the 747 test shows greater ground effect with tail on than the 727 test does, even though the C_L was smaller. In addition, the difference between tail-off and tail-on configurations is difficult to explain.

Recirculation of air has caused problems with some STOL airplanes, resulting in loss of directional control due to asymmetric drag changes occurring when the airplane was banked in ground effect. This roll-yaw coupling is illustrated in Figure 17 for the XC-142 configuration.

If the airplane is banked in ground effect the drag of the lower wing is reduced more than that of the upper thus producing adverse yaw. The tunnel tests indicate that at a wing incidence of 45°, a flap setting of 60° and a speed of about 25 knots, the adverse yawing moment will equal the yawing control moment available to the pilot at a bank angle of about 7.5 degrees.

Other ground effects that can occur include damage to and/or loss of power from the engines due to reingestion of hot exhaust gas, foreign object damage due to ground erosion by high velocity slipstreams and localized heating of the airframe by recirculation of exhaust gases.

Testing for ground effect in a wind tunnel can introduce the requirement for a moving ground board in order to avoid the strong interaction that would otherwise occur between the disturbed airflow about the airplane and the boundary layer on the tunnel floor. Figure 18, from Reference 3, presents the results of a study made to evaluate the criteria for determining the conditions under which the moving ground board is required. It is seen that if C_L is greater than about 4 (h/c) a moving ground board is required. This figure does not include concepts employing concentrated jets. Reference 4 indicates that moving belts are not required for those concepts.

2.5 STOL MODEL TESTING

2.5.1 Introduction

It has long been recognized that testing models of high lift configurations requires special modeling and testing techniques different from those employed for conventional aircraft models. The use of powered lift systems that obtain high lift from strong interactions between the lifting and propulsive systems dictates the use of powered models. Furthermore, the large flow deflections associated with high lift require that models tested in wind tunnels be small

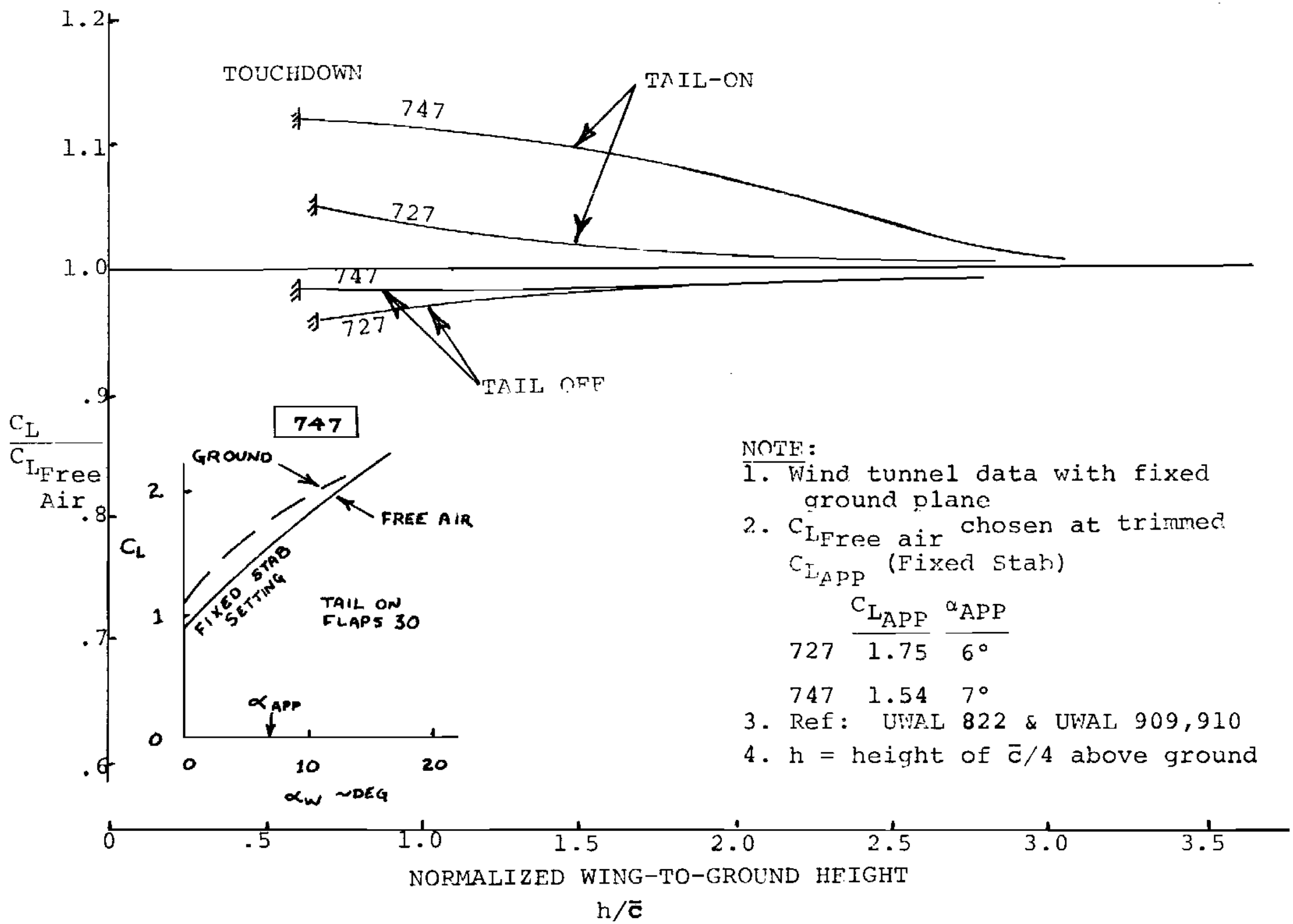


FIGURE 16. EFFECT OF CONFIGURATION ON GROUND EFFECT

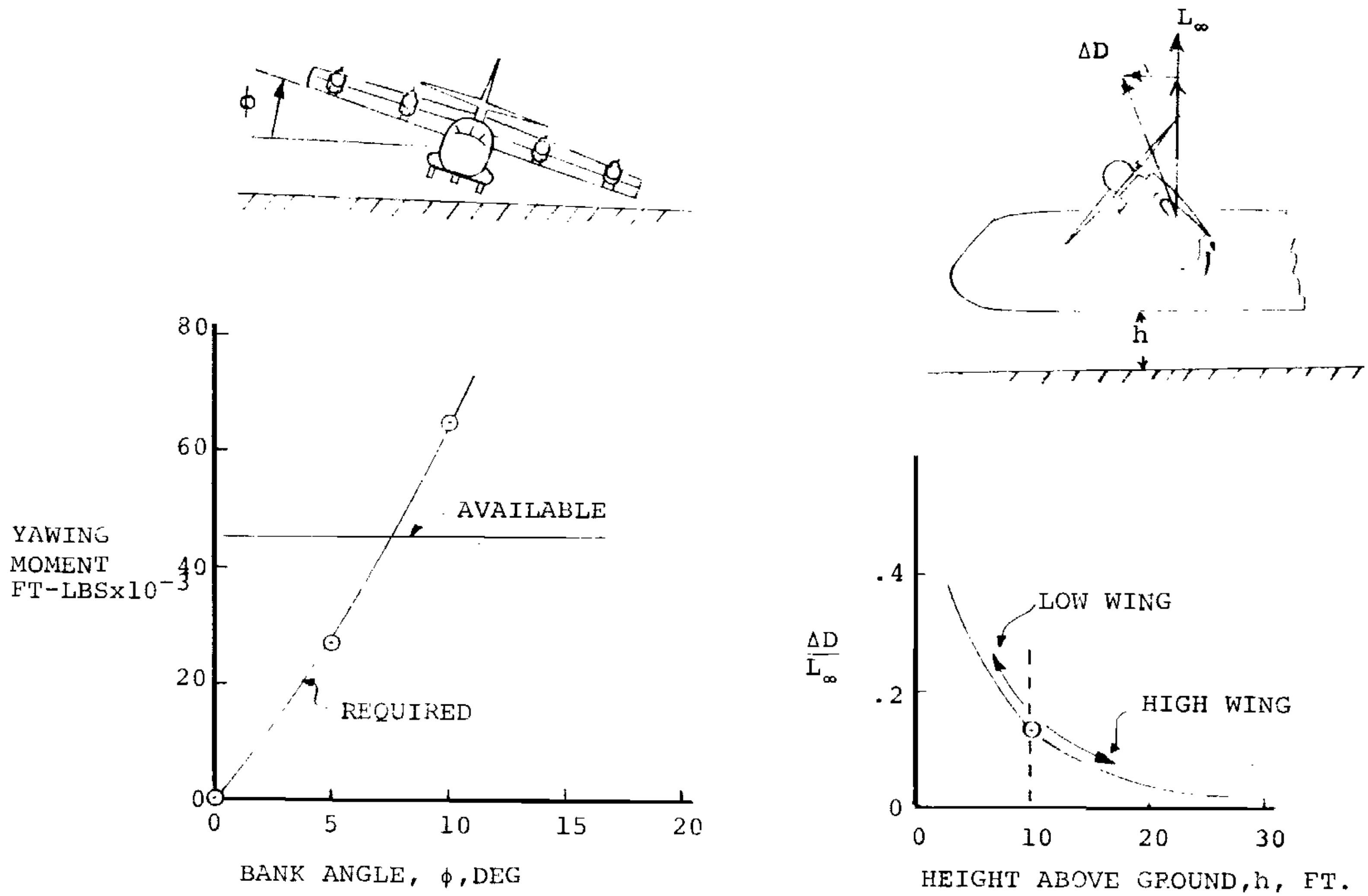


FIGURE 17. EFFECT OF BANK ANGLE ON YAWING MOMENT

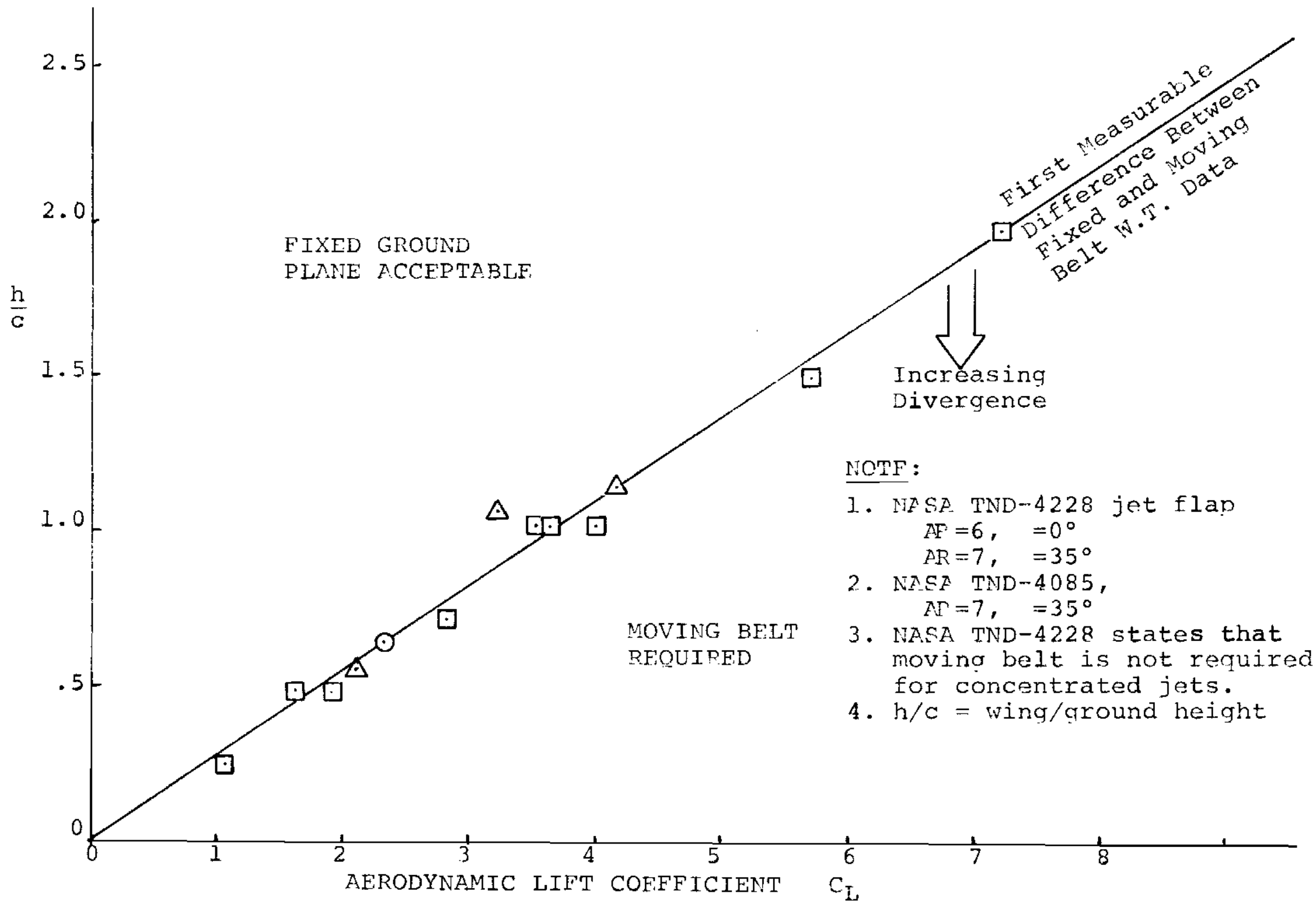


FIGURE 18. MOVING BELT TESTING REQUIREMENT FOR HIGH-LIFT WINGS

relative to the tunnel working section or that adequate predictions of wall corrections be available. These and other considerations have led to the construction of large wind tunnels and other suitable test facilities, the development of devices to simulate propulsion systems, new sophisticated instrumentation and spurred a continuation of the theoretical investigation of wind tunnel wall effect corrections.

2.5.2 Simulation of Propulsion Systems

Several different STOL high lift systems have been proposed using a wide variety of propulsive devices including turboshaft driven propellers, turbojets, turbofans and various systems employing gas generators to supply jet flaps, fans, etc. The requirement to simulate the operation of these systems with model-size motors can lead to significant complication of model testing. Motor size/model size matching is complicated by the requirement to provide high thrust coefficients without testing at extremely low tunnel speeds. Motor heating can cause interaction with balances, requiring careful attention to temperature compensation. Accurate means must be provided to determine propulsive forces. Routing of air supplies around balances requires that unusually detailed attention be paid to accurate, thorough calibration for data corrections. The large amount of powered testing performed on tilt wing and deflected slipstream configurations has led to the development of successful test techniques for propeller-driven concepts. The important considerations of such testing are described in more detail in Section 5.5.2.

2.5.3 Test Conditions

The high lift levels and large momentum deflections associated with STOL configurations at low speed can result in large wind tunnel wall effects, not predictable by the conventional wall correction methods, unless special attention is paid to model/tunnel matching. In addition, very small models are inappropriately scaled due to practical limits on model motor size, and the large models are required in order to obtain high thrust coefficients at acceptably high Reynolds numbers. This has led to the development of large V/STOL wind tunnels which have adjustable tunnel wall configurations — open, slotted, or closed — and with moving or fixed ground boards. Table V lists the major V/STOL wind tunnels currently in existence in North America.

Even in the larger tunnels a potential exists for significant flow distortion when the test conditions create large deflections of tunnel flow. Figures 19 through 22, taken from Reference 5, show the progressive buildup of flow distortion and interference fields with increasing downwash. A method of monitoring the tunnel flow conditions to determine the onset of large flow distortions has been recommended in Reference 5. The technique consists of observing the pressure distribution along the centerline of the tunnel floor. As shown in Figure 23, the pressure distribution is relatively smooth when the flow distortion consists only

TABLE V

CHARACTERISTICS OF V/STOL LOW SPEED WIND TUNNELS

Organization	Test Section			Installed Horsepower	Remarks
	Width	Height	Max. Velocity		
Boeing Vertol	20'	20'	270K	15,000	100K moving belt ground plane; 120 channel data system; slotted and open wall features.
General Dynamics (Convair)	8'	12'	260K	2,250	100FPS moving belt ground plane (ceiling mtd); model inverted for testing. Data log w/ computer fully operational.
Lockheed-Georgia	26' 23.5'	30' 16.25'	100K 200K	9,000	Single tunnel; 2 test sections tunnel features easy modification for varied test programs. Fully operational. On-line data log w/computer. Moving belt ground plane to be installed.

TABLE V - Concluded

Organization	Test Section			Installed Horsepower	Remarks
	Width	Height	Max. Velocity		
LTV-Dallas	21' 20' 10'	23' 15' 7'	35K 48K 200K	1,500	Single tunnel; 3 test sections; 100 FPS moving belt ground plane in 20x15 tunnel branch. Facility fully operational. Data system.
NASA-Ames	80'	40'	60K	12,000*	Capable of large to full-scale testing. Limited test applications.
NASA-Langley #1	22'	17'	250K	10,000*	Existing tunnel similar to Boeing-Vertol. Fully operational.
NASA-Langley #2	21.75'	14.5'	200K	8,000 (12,000)	New tunnel being completed. Moving ground belt w/suction. 12,000 HP for 30 minute duration.
NRC-Canada	30' 14'	30' 14'	120K 300K	10,000	Single tunnel; 2 test sections. Moving ground plane for future installation. Fully operational.

*Estimated

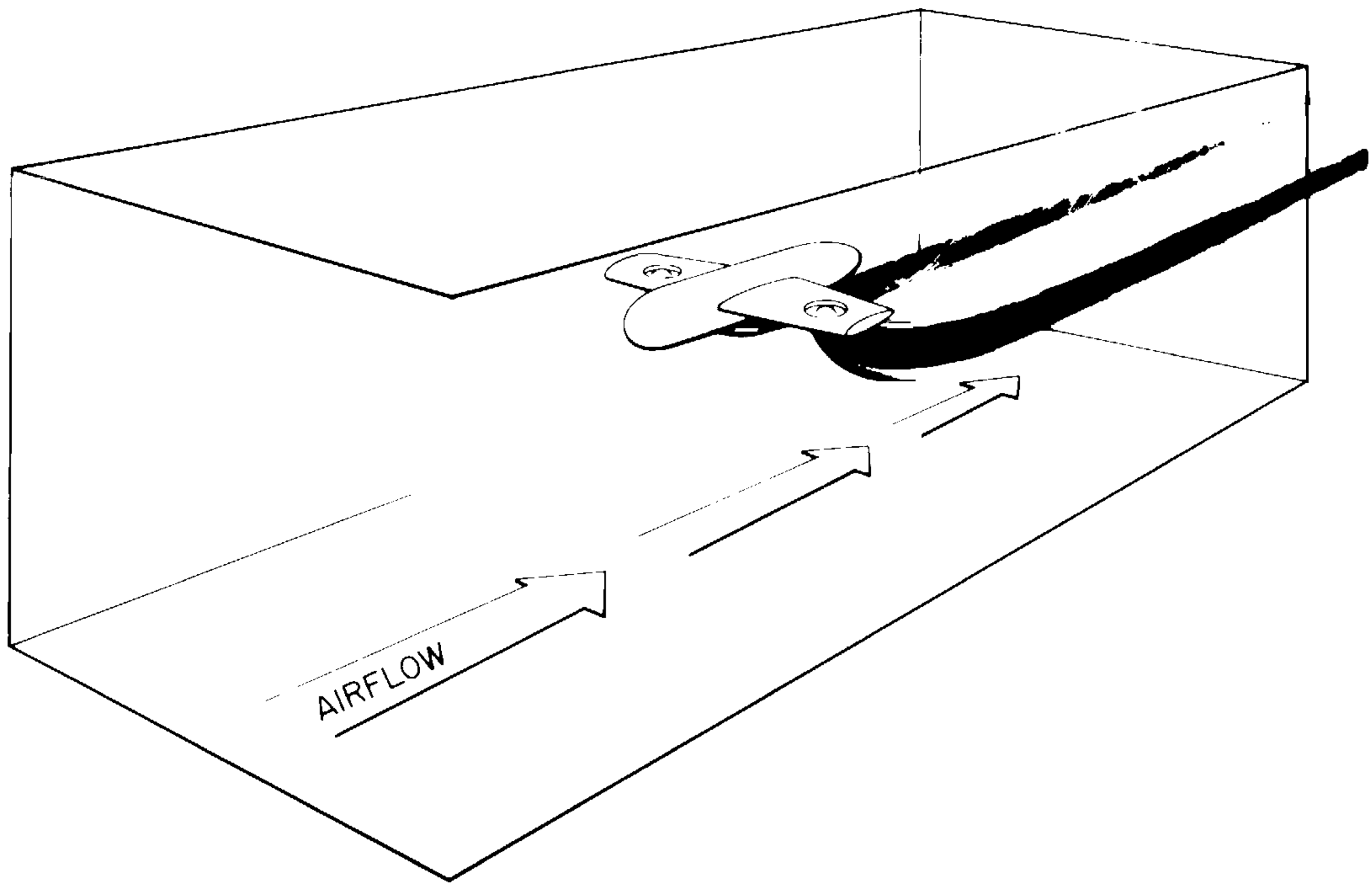


FIGURE 19. SCHEMATIC REPRESENTATION OF THE CONDITION FOR UNDISTURBED FLOW AT THE WIND TUNNEL WALLS

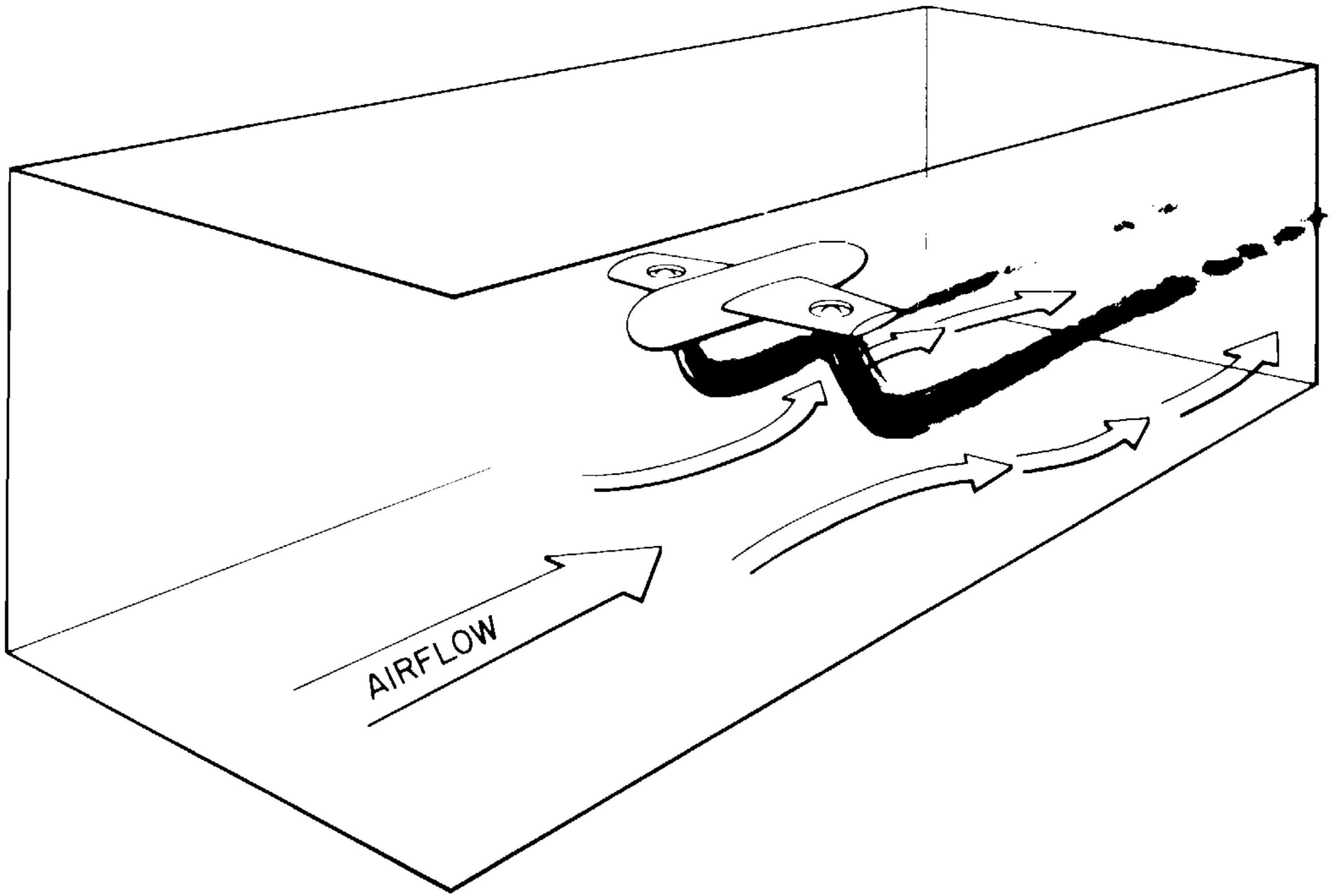


FIGURE 20. SCHEMATIC REPRESENTATION OF THE CONDITION FOR THE DISTURBANCE OF THE TUNNEL BOUNDARY LAYER

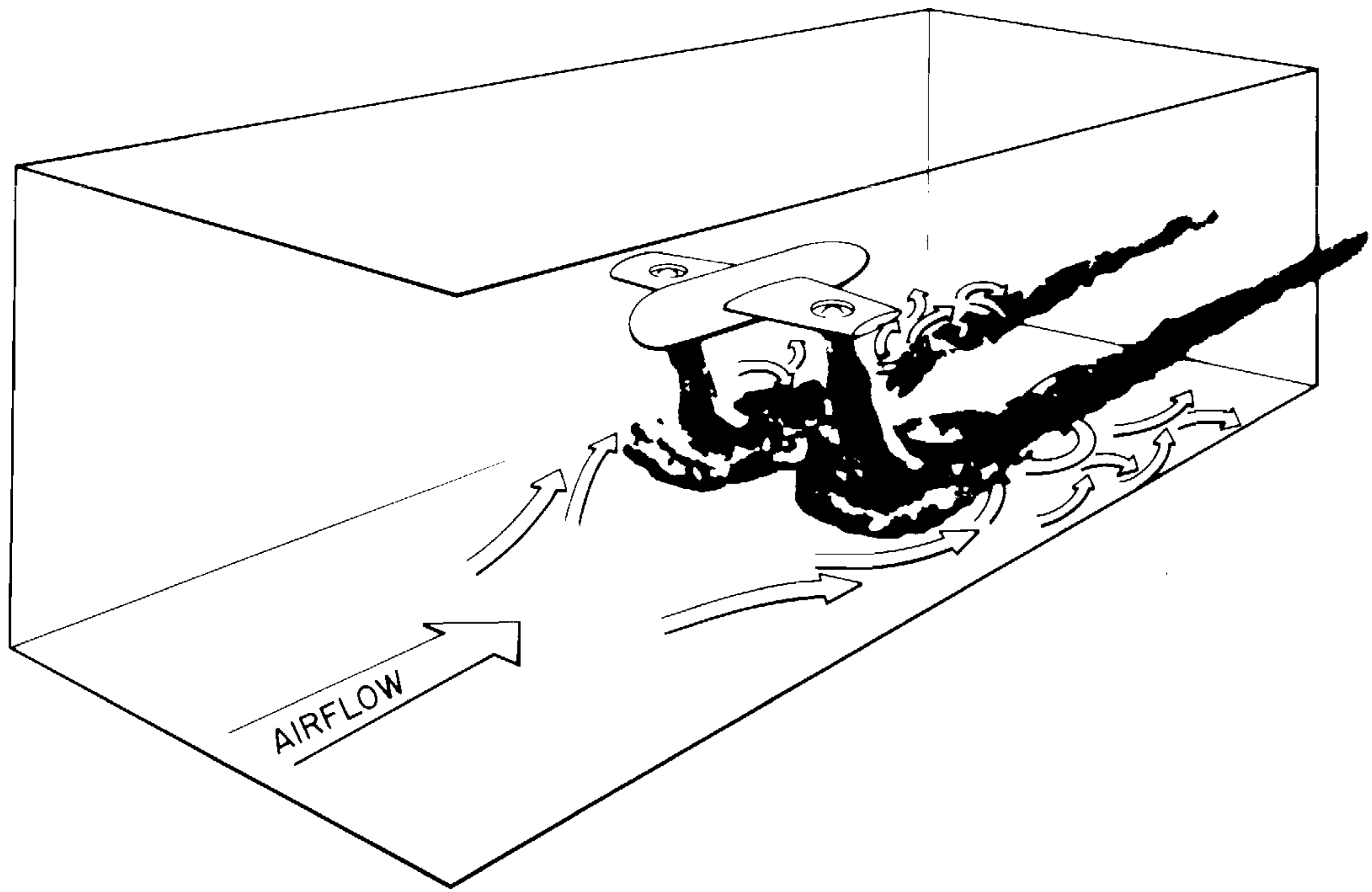


FIGURE 21. SCHEMATIC REPRESENTATION OF THE CONDITION FOR INCIPIENT STAGNATION

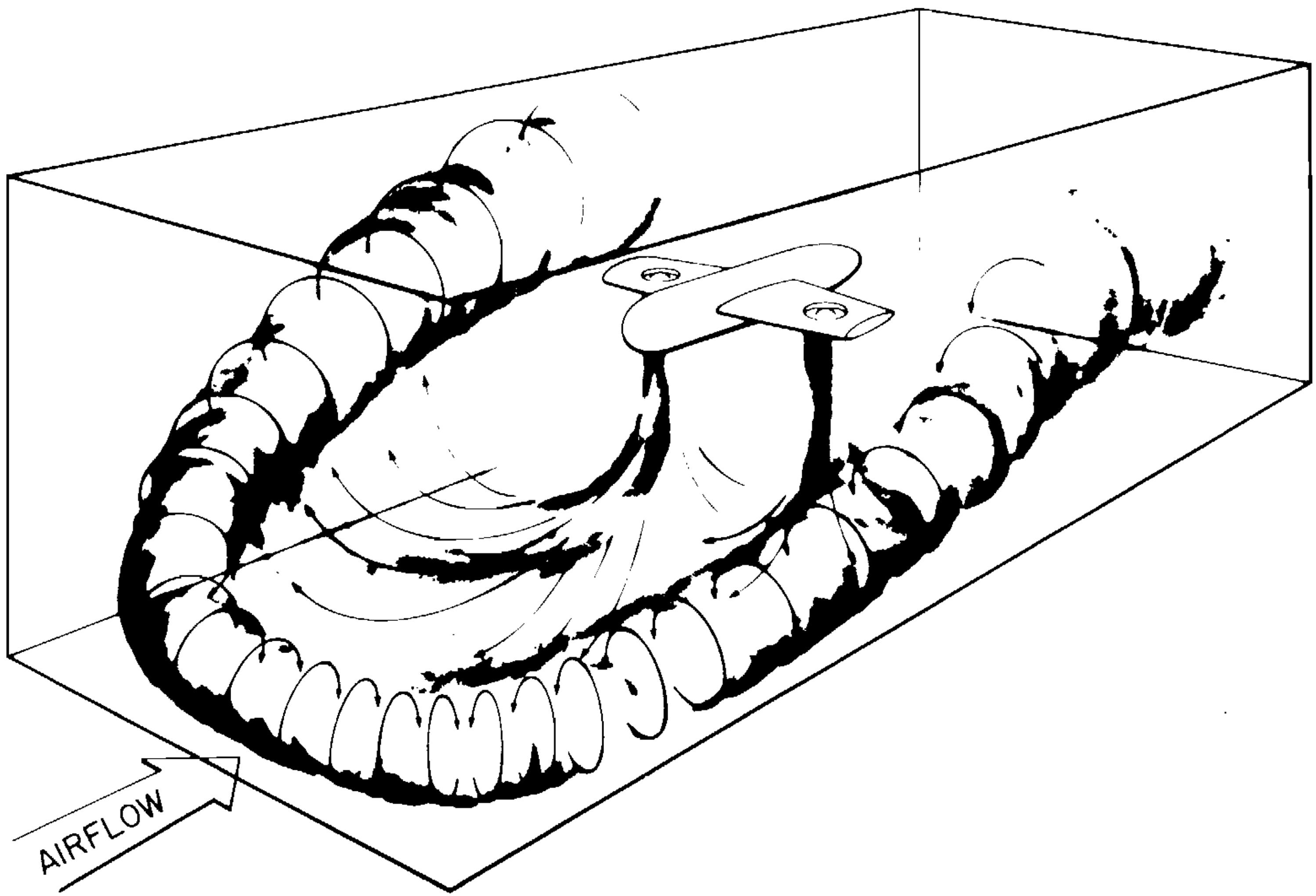


FIGURE 22. SCHEMATIC REPRESENTATION OF THE CONDITION FOR VORTEX FORMATION

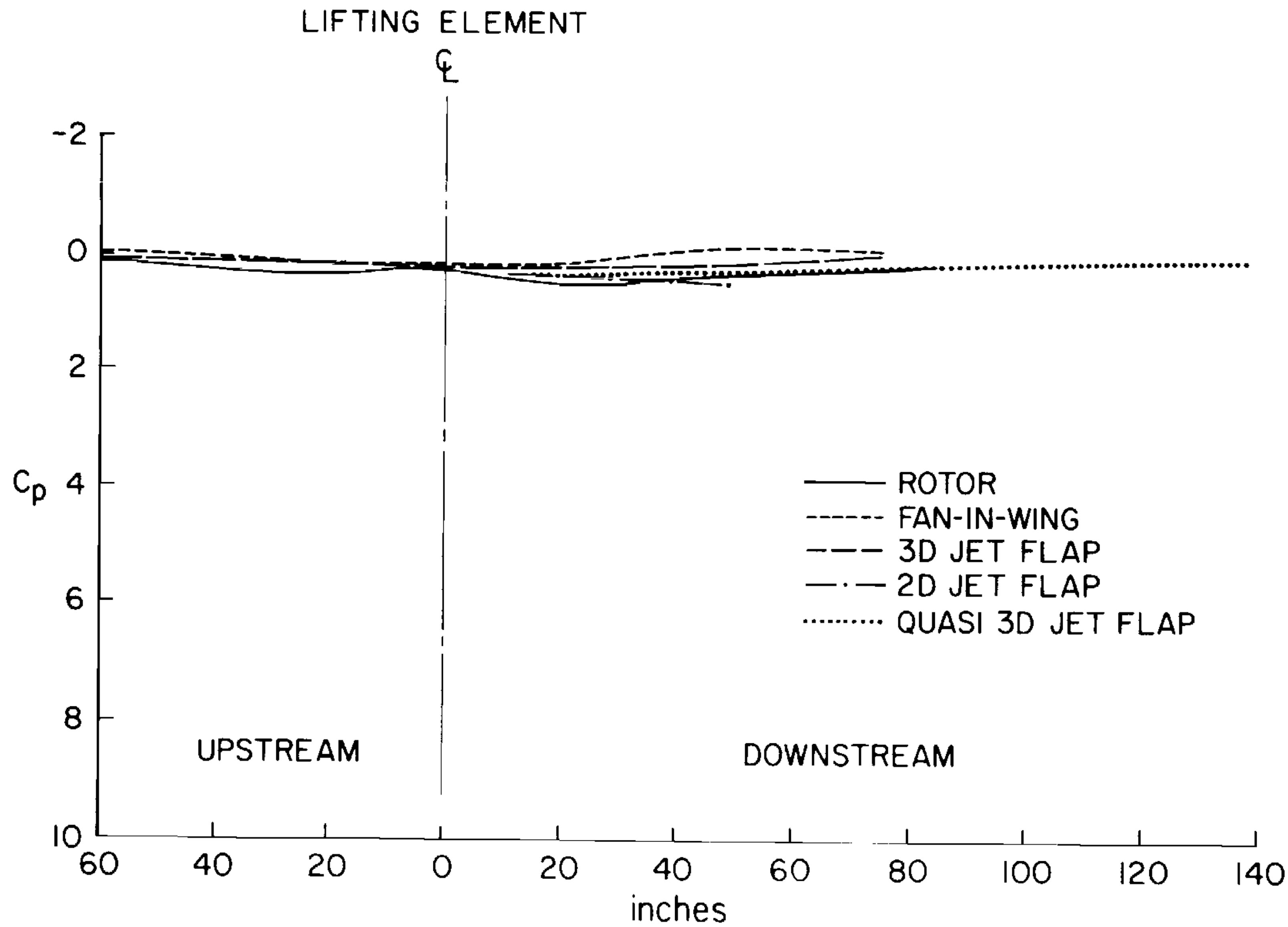


FIGURE 23. PRESSURE DISTRIBUTION SHOWING THE EFFECTS OF THE DISTURBANCE OF THE TUNNEL BOUNDARY LAYER

of minor disturbances of the tunnel boundary layer (such as Figure 20) but in Figure 24 becomes non-uniform at the point of incipient stagnation (Such as Figure 21). Reference 6 gives the results of a study to determine the largest model to tunnel size ratio permitted and indicates that the flow downwash angle due to the lift/propulsion system has a major effect on the acceptable model size.

Because of the complex flow interactions between the freestream and the propulsion system, great care must be taken to measure all important parameters and avoid unwarranted assumptions about parameters that might be expected to remain constant or behave in a predictable manner. Further, it is necessary to calibrate balances, propulsion units, etc. on an individual basis.

2.6 IMPORTANT TECHNOLOGY AREAS AND INFORMATION REQUIREMENTS

To serve as guidelines for the literature search described in Section 3 of this report, the important factors discussed in Sections 2.1 through 2.5 have been classified into three major important technology areas, and the information required in each of these technology areas is listed.

2.6.1 Basic Aero/Propulsion Technologies

2.6.1.1 Aerodynamic Characteristics of High Lift Wings

- What are the lift characteristics for clean wing, such as C_{L_α}, α_{Lo}, $C_{L_{max}}$, etc?
- What are the increments due to the addition of trailing edge and leading edge devices?

2.6.1.2 Aerodynamic Characteristics of Other Airframe Components

- What is the influence of stores, nacelles, fuselage, etc. on maximum lift?
- What is the influence of nacelles, fuselage, tail and other components on aircraft stability?

2.6.1.3 Aerodynamic Characteristics of Boundary Layer Control Systems

- What is the change in lift and maximum lift due to application of boundary layer control and what is the influence of BLC parameters (C_μ, V_J/V_∞) on the change in lift?
- What is the influence of BLC on aircraft moments and drag?

2.6.1.4 Operational Envelopes of Propulsion Systems

- What are the operational envelopes of various propulsive devices as influenced by inlet flow distortions, sand and dust environment and recirculation of hot gases ?

2.6.1.5 Behavior of Propulsive Devices at Low Speed

- What is the variation of thrust of the propulsive system with power setting, flight speed and angle of attack ?

- What are the moments and in-plane forces due to the propulsive system ?

- What are the flow fields created by propulsive devices in terms of velocities and downwash angles ?

2.6.1.6 Aerodynamic/Propulsion Interactions

- What are the lift, drag and moment characteristics of the integrated aerodynamic/propulsive system ?

- What is the influence of the high lift wing on the propulsive system ?

- What are the flow fields created by the combined aero/propulsion system ?

2.6.2 Interacting Technologies

2.6.2.1 Stability

- What are the regions of adequacy of linear stability analysis or small perturbation methods ?

- What terms of the stability equation which are normally neglected for conventional aircraft must be included for accurate analysis of STOL aircraft stability and control ?

2.6.2.2 Ground Effects

- What is the influence of $\left\{ \begin{array}{l} \text{proximity to ground,} \\ \text{configuration, airplane} \\ \text{weight, flow deflection,} \\ \text{forward speed, power} \end{array} \right\}$ on $\left\{ \begin{array}{l} \text{flow field, lift, drag,} \\ \text{moments, stability,} \\ \text{propulsive system} \end{array} \right\}$?

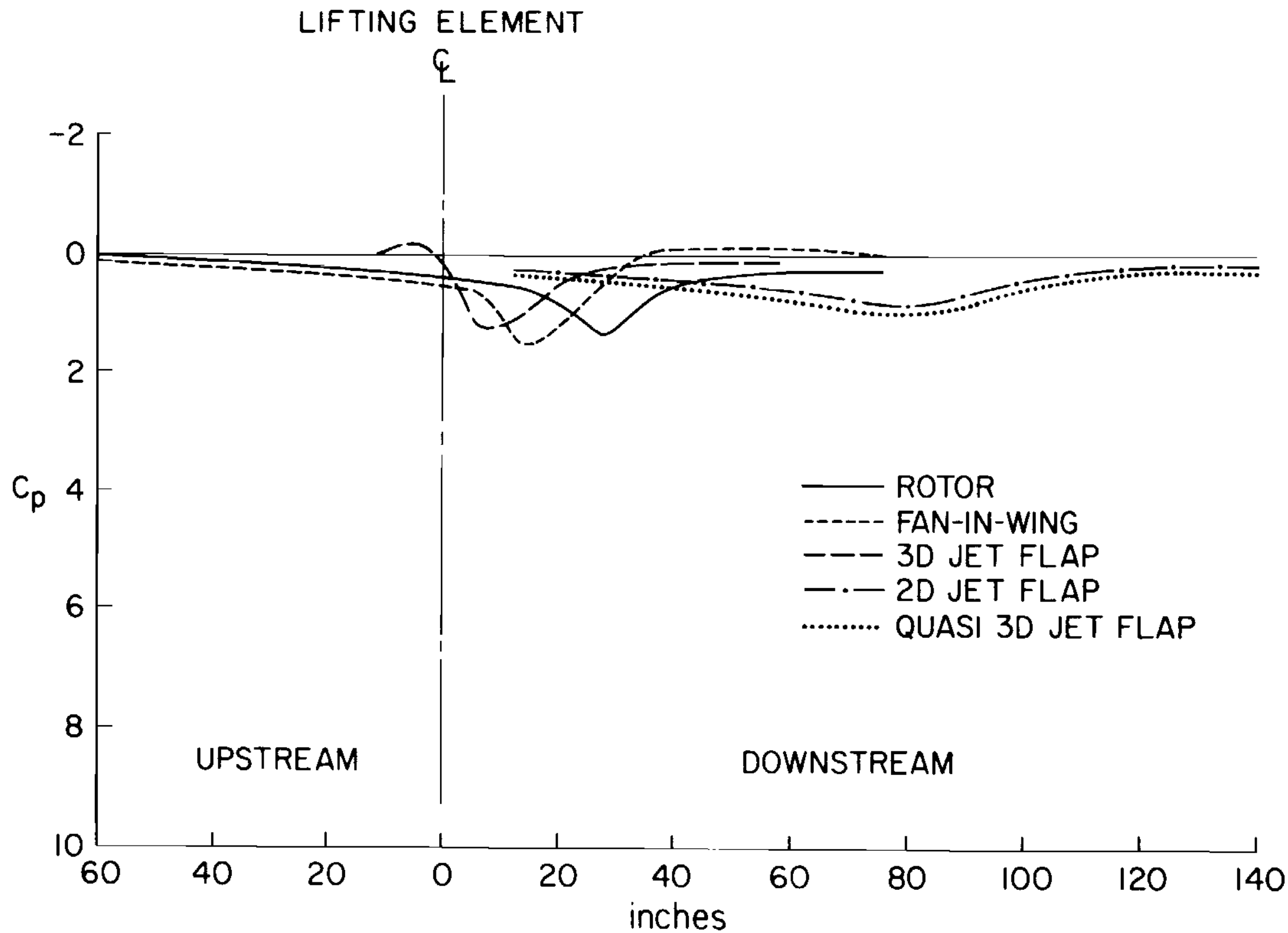


FIGURE 24. PRESSURE DISTRIBUTIONS INDICATING THE PRESENCE OF INCIPIENT STAGNATION

2.6.2.3 Surface Effects

- What is the influence of {disc loading, airplane weight, configuration, forward speed, ground composition} on {pressures, flow velocities, visibility, ground erosion, damage to structure, engine damage} ?

2.6.2.4 Atmospheric Gusts

- What is the sensitivity of various STOL concepts to operation in gusting environment?

2.6.3 Systems Requirements

2.6.3.1 Power Required

- What is the power required to operate various types of BLC systems?

- What are the power requirements for aircraft control?

- What are the power penalties due to slipstream deflection or thrust vectoring?

2.6.3.2 Control Requirements

- What is the control effectiveness of various types of control devices proposed for low speed STOL operation?

- What are the aerodynamic problems associated with the use of these control devices?

2.6.3.3 Margins and Criteria

- What are the appropriate criteria for STOL aircraft?

- What is the aircraft performance sensitivity to the criteria used?

2.6.3.4 Test Requirements and Limitations

- What special model and test techniques can be used to simulate the full scale conditions?

- How do you correct the test data to compensate for the difference between test and full scale conditions?

- What combinations of test conditions could potentially limit the ability to compensate for the difference between test and full scale conditions?

- What techniques can be used to monitor the test to assure the validity of the results?

The above exposition of the necessary information was prepared in the form of a series of questions to serve as a convenient check list for the information search.

The main emphasis in this study has been placed on aerodynamic problems of STOL aircraft operating in the low speed flight regime. The technology associated with cruise and other high speed flight modes has accordingly been de-emphasized. The study has concentrated on the technology of the aerodynamics of lifting systems, propulsion interactions, stability and control, criteria and margins and model test techniques.

3. LITERATURE SEARCH

3.1 SCOPE

The literature search was made in order to identify the available test data and prediction methods relevant to the aerodynamics of selected STOL aircraft concepts.

Seven STOL lift/propulsion concepts were researched:

1. Externally Blown Flap
2. Deflected Slipstream
3. Jet Flaps
4. High Lift Devices
5. Fan-in-Wing
6. Tilt Wing
7. Jet Lift

The areas of aerodynamic technology for which references were sought were:

1. Forces and Moments
2. Flow Fields
3. Ground Effect
4. Stability and Control
5. Handling Qualities and Criteria
6. Testing

In addition to the seven STOL concepts and the six technology areas, a further classification of "general" was included to cover references that were relevant to some or all of the concepts or technology areas.

3.2 SOURCES

The main body of references was obtained by use of automated literature searches carried out by the Defense Documentation Center and by The Boeing Company

Library. Other sources of references were the NASA STAR index, the Royal Aeronautical Society Library Acquisition Lists and a number of AGARD bibliographies.

Of the references that were originally thought to be relevant, by virtue of their title (or in some cases abstract), many were found to be outside the scope of this literature search and were, therefore, discarded. The resulting bibliography contains references to about 900 documents with abstracts from most of them and comprises Volume II of this report.

3.3 CLASSIFICATION OF BIBLIOGRAPHY

The bibliography is classified in three levels. In the first level, the references are classified into I, Prediction Methods and II, Test Data. Within these groups the bibliography is organized, as a second level, into divisions according to the STOL concepts as listed above. The third level of classification, within the 'STOL concept' divisions, consists of subdivisions identified by the technology areas named above. The numbering system employed in classifying the references has four components.

The first, Roman Numeral I or II, indicates whether the data contains prediction techniques or test data.

The second and third components identify the STOL concept and technology area respectively, the numbers employed corresponding with the above lists.

The fourth component is the position of the reference within the technology area subdivision.

4. EXTERNALLY BLOWN FLAPS

4.1 AN ENGINEERING METHOD FOR ESTIMATING THE TAIL-OFF LONGITUDINAL AERODYNAMIC CHARACTERISTICS OF AN AIRCRAFT WITH EXTERNALLY BLOWN FLAPS

4.1.1 Introduction

The externally blown or external-flow jet augmented flap (EBF) is a means of increasing lift in order to achieve short takeoff and landing (STOL) performance. The propulsive jet is directed at the trailing edge flaps in order to augment the wing lift at low speeds. Part of the propulsive jet flows through the flap slot and the entire jet is turned by the flap. Lift is increased due to the vertical momentum of the jet and induced flow around the airfoil. A sketch derived from water tunnel flow visualization work reported in Reference 7 is used to illustrate the concept, Figure 25.

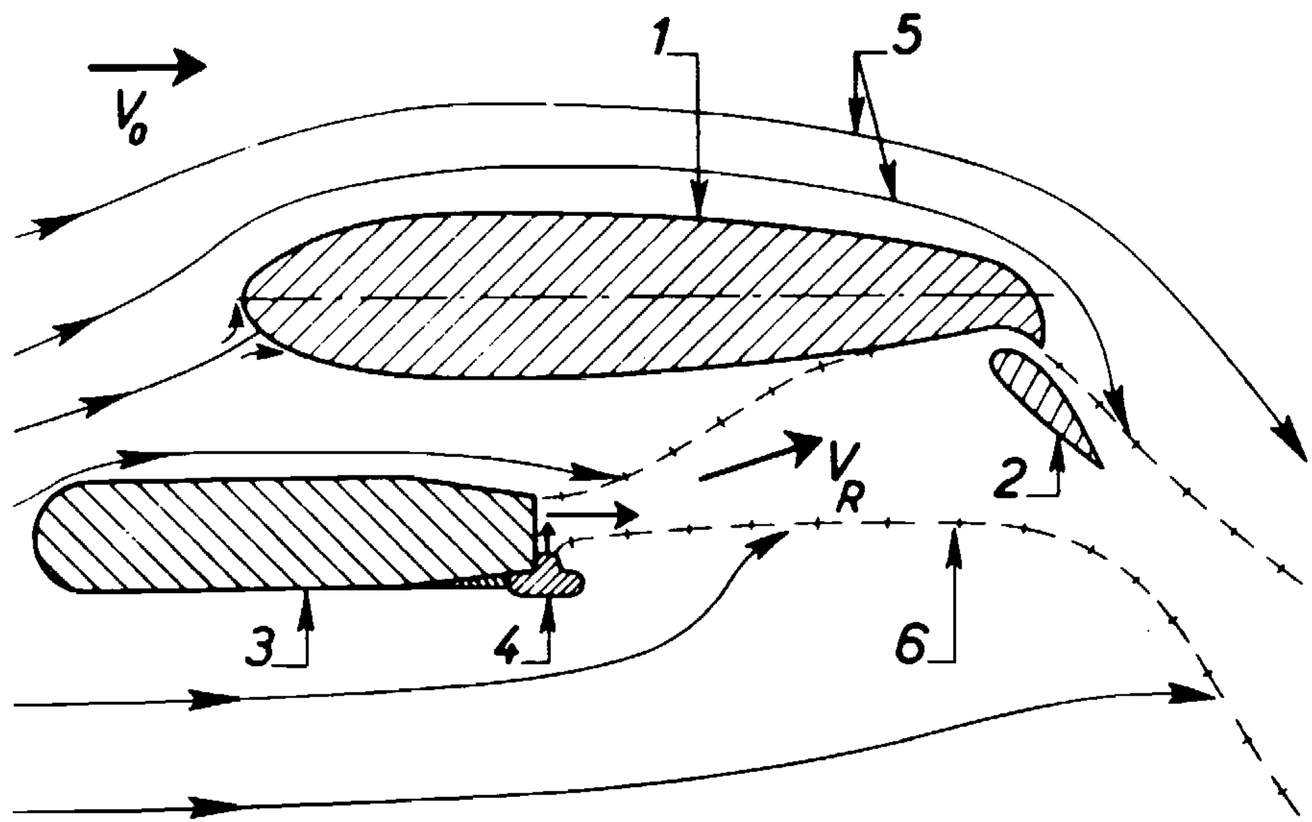
The externally blown flap integrates the propulsion and mechanical high-lift systems to produce a powered-lift airplane. The EBF is related to the jet-augmented flap in which the propulsive jet is expelled as a thin sheet from the trailing edge of the airfoil.

The externally blown flap concept was introduced by the NACA in 1956. However, little interest was shown in incorporating it into flight hardware due to the high jet velocities and temperatures of the pure turbojet engine. The advent of the high bypass ratio turbofan with its low jet velocities and temperatures advanced the concept from wind tunnel feasibility studies to consideration for incorporation into practical flight vehicles.

To evaluate the worth of a particular EBF configuration, a rapid method for estimating its aerodynamic characteristics must be available. Since no such technique was found during the literature search, the major portion of Phase 2 of the study, Analysis of Techniques and Data, was devoted to the development of a suitable method.

In 4.1 of this section, an engineering method for predicting the tail-off longitudinal aerodynamic characteristics of an EBF configuration is formulated and its correlation with existing experimental data is established. In 4.2, the gaps and deficiencies in the available wind tunnel data are defined in detail and programs to correct these gaps and to extend the prediction capability introduced in this report are recommended.

4.1.1.1 Summary. Developing prediction capability requires that the problem be amenable to theoretical treatment, or that a large body of test data be available to develop empirical techniques, or that parametric variations of the significant parameters have been tested. The wind tunnel test data that is available in the



1 WING
2 FLAP
3 ENGINE
4 JET DEFLECTOR
5 STREAMLINES
6 JET BOUNDARY

FIGURE 25. WATER TUNNEL FLOW VISUALIZATION

open literature is limited in quantity and scope and it is impossible to develop a strictly empirical approach.

No theoretical treatment of the externally blown flap per se exists. However, the externally blown flap is a type of jet flap for which some theoretical development has been done. The approach developed in this report is to use jet flap theory to derive incremental effects due to power. It is assumed that the unpowered characteristics are known from wind tunnel testing or can be estimated by standard methods.

The jet flap consists of a sheet of high energy fluid ejected from the trailing edge of the airfoil. Lift, in addition to that given by standard thin airfoil theory and the lifting component of momentum, is induced due to the jet deflecting the streamlines of the flow downward in the vicinity of the trailing edge. An inviscid linearized solution has been obtained for a thin jet of momentum coefficient C_J emerging tangentially at the trailing edge, References 8 and 9.

This treatment of two-dimensional jet flaps has been extended, References 10 and 11, to the case of a flat, finite aspect ratio wing with a full-span zero thickness jet sheet emerging at a small deflection from the trailing edge. The equations have been solved for the special case of large aspect ratio and elliptic loading. Theoretical treatment of part-span jet flaps has not been undertaken.

In order to use jet flap theory to predict the characteristics of the EBF, a rationale for determining an effective jet angle has been developed. Since the spanwise extent of the wing influenced by the propulsive jet cannot be established from existing wind tunnel data, the EBF data have been analyzed to show that the lift and drag can be estimated with fair accuracy without knowing this extent. This is not true for pitching moments on swept wings.

Jet flap theory is manipulated to a form in which the powered lift curve can be expressed as the product of the unpowered lift curve slope and a function of the blowing momentum coefficient. Jet flap theory is a small angle, linear theory, i.e., $\sin \alpha \approx \alpha$. The direct power effect on the lift curve slope is equal to C_J. For the large EBF jet angles normally used, the final expression for lift curve slope has been modified to contain a direct power effect dependent upon C_J and δ_J.

$$C_{L_{\alpha_{\text{powered}}}} = C_{L_{\alpha_{\text{unpowered}}}} K(C_J) - C_J + C_J \cos \delta_J \tag{1}$$

Similarly, the incremental effects of flap lift due to power have been derived from jet flap theory

$$\Delta C_L (C_J) = \Delta C_{L_\delta} (C_J) \; F(A, C_J) \; \delta_J / 57.3° \tag{2}$$

A correction to the direct lift power effect has not been made because the test data correlates better without it.

Maximum lift increment due to power has been correlated as a function of the vertical component of momentum.

The force polars are derived by showing that the theoretical jet flap induced drag polar is equivalent to unpowered elliptical loading induced drag, $C_L^2/\pi A$, with the thrust deflected to the optimum angle for level flight.

The pitching moments can be estimated by extracting a center of pressure of the power-induced flap lift from jet flap theory.

4.1.1.2 Organization of the Method. The effective jet angle, extent of wing influenced by the jet, and static turning efficiency are discussed in Section 4.1.2. Expressions for lift curve slope, flap lift increment due to power, and incremental maximum lift are derived in Section 4.1.3. Force polar estimation is explained in Section 4.1.4. Pitching moment is discussed in Section 4.1.5.

The effects of asymmetric thrust are considered in Section 4.1.6. It is shown that the methods of the prior sections can be used to adequately predict the longitudinal aerodynamic characteristics of the configuration with engine out.

An example illustrating the application of the method is given in Section 4.1.7. A correlation of theoretical and experimental results are contained in Section 4.1.8. Since the method is derived to some extent from this same data, the final proof of the adequacy of this method must come from applying it to data that will be obtained subsequent to publication of this report.

4.1.1.3 Nomenclature

A	Aspect ratio
C	Chord
C'	Developed chord with flaps extended
C_D	Drag coefficient
$C_{DP_{min}}$	Minimum profile drag
C_f	Flap Chord
C_f'	Developed flap chord
$\dot{C}_J$	Engine exhaust momentum coefficient, $\dot{m}\, V_E/qS$

C_J	Jet flap blowing momentum coefficient
C_l	Rolling-moment coefficient
C_L	Lift coefficient
C_{L_c}	Circulation lift coefficient
C_{L_T}	Total lift coefficient
C_{L_α}	Three-dimensional lift curve slope
C_{L_α}, $C_{L_{\alpha 2D}}$	Two-dimensional lift curve slope
$\Delta C_L \ (C_J)$	Increment in lift due to power at zero angle of attack
$\Delta C_{L_\delta} \ (C_J)$	Increment in 2-D flap effectiveness due to jet flap effect
$\Delta C_{L_{max}} \ (C_J)$	Increment in maximum lift due to power
C_m	Pitching moment coefficient
do	Function of flap chord and C_J, Reference 9 (Identical with D_o from Reference 9)
Do	Function of C_J, Reference 11
$F(A, C_J)$	Jet flap finite aspect ratio factor
F_A	Axial force
F_N	Normal force
F_R	Resultant force
$K(A, C_J)$	Ratio of powered to unpowered 3-D lift curve slope for a jet flapped wing
$\mathring{m}$	Engine mass flow
S_{REF}	Reference wing area, ft^2
S_g	Gross wing area with flaps extended, ft^2
S'	Flapped wing area, ft^2
T	Gross thrust

V_E	Engine exit velocity
α	Angle of attack
δ	Jet flap deflection angle
δ_j	Effective jet deflection angle, $\tan^{-1} F_N/F_A$
δ_u	Flap upper surface slope
δ_l	Flap lower surface slope
η	Static turning efficiency, F_R/T
η_1	Effective roll arm of engine lift/wing semispan
η_{eng}	Spanwise location of engine centerline/wing semispan
χ	$2 \sin^{-1}\sqrt{c_f/c}$
X_{CP}/c	Chordwise location of center of pressure

4.1.2 Required Information

Jet flap theory, in its present state of development, requires that the blowing be full span on the wing and that the angle and blowing momentum coefficient of the jet as it leaves the trailing edge of the wing be known in order to estimate the aerodynamic characteristics of a jet flapped wing. In order to adapt jet flap theory to the prediction of wings with externally blown flaps, the effect of part-span blowing must be assessed and an effective jet deflection angle and momentum coefficient must be defined.

4.1.2.1 Extent of Wing Influenced by the Jet. The extent of wing influenced by the propulsive jet of the externally blown flap can be determined only by having detailed wing and flap pressure data from wind tunnel testing. To date, no such pressure data is readily available. A limited amount of Boeing pressure data taken on the flaps only, indicate that the loading on the flap is highly localized in the vicinity of the nacelle centerline. However, following a mechanical trailing edge flap, most of the lift is expected to be induced on the wing. The distribution of the induced lift on the wing cannot be determined from available test data. Therefore, a method was developed which is independent of the extent of the wing influenced by the jet.

The NACA test data indicate that the lift and drag of an EBF configuration are a function primarily of the blowing momentum coefficient and only weakly dependent on engine location. Pitching moment is more sensitive to engine location, especially on swept wings. Figure 26 taken from Reference 12 shows a comparison at zero angle of attack of lift, drag and pitching moment with the engine efflux distributed along the span and with it localized well inboard. On the basis of this and other data, the assumption was made that, for the level of accuracy being sought, a method could be developed which was independent of the extent of wing influenced by the jet.

4.1.2.2 Jet Deflection Angle. It is necessary to be able to determine an effective jet deflection angle. Since the propulsive jet of the externally blown flap is diffuse rather than infinitesimally thin as jet flap theory assumes, there is no well-defined jet angle. A jet angle deduced from static force testing will be taken to be the jet deflection angle. It is then necessary to relate this effective jet angle to the trailing-edge geometry.

If a wind tunnel model is tested statically with engines operating and trailing edge flaps down, the axial and normal forces on the model due to the jet can be measured. Assuming that there is no induced flow around the model and therefore no induced aerodynamic forces, an effective jet deflection angle can be defined by $\tan^{-1} F_N/F_A$, Figure 27. NASA has done a substantial amount of static testing to determine the effective jet angle. These data have been examined to correlate the measured effective jet angle with the flap system geometry. The effective angle can be estimated by

$$\delta_j = 1/2 \ (\delta_u + \delta_l) \tag{3}$$

where δ_u is the upper surface angle and δ_l is the lower surface angle, Figure 28. This correlates well with the NASA static test data as shown in Section 4.1.8.

All of the data used are those in which care has been taken to obtain efficient turning of the jet. Therefore, jet angles may be lower than this if care is not used. The assumption is made that the effective jet angle does not change with forward velocity or thrust level.

4.1.2.3 Static Turning Efficiency. As with any system that redirects the propulsive jet, losses are incurred. Once again from static testing, a turning efficiency can be inferred by knowing the static thrust input to the system and measuring the resultant force, $\eta = F_R/T$, Figure 27.

Static turning efficiency has been correlated, as determined from NASA static tests, as a function of the effective jet angle. The data of Reference 13 indicate that the static efficiency is independent of thrust, Figure 29. Figures 30 and 31 show the correlation for double- and triple-slotted flaps. In general, triple slotted flaps appear to be more efficient thrust vectoring systems although some double-slotted flap systems perform equally well. These data represent the current state-of-the-art for the flap systems considered. Turning efficiencies much

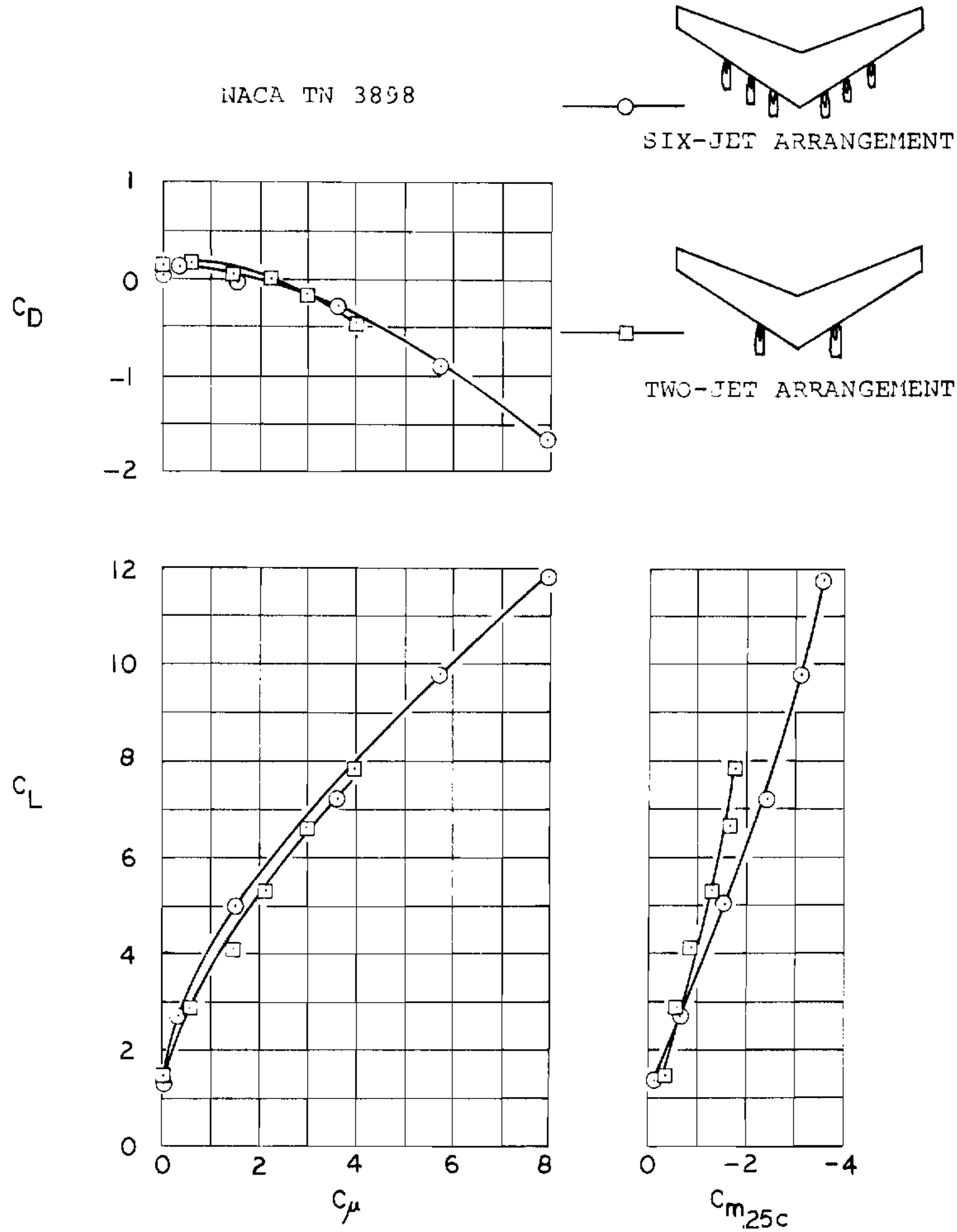
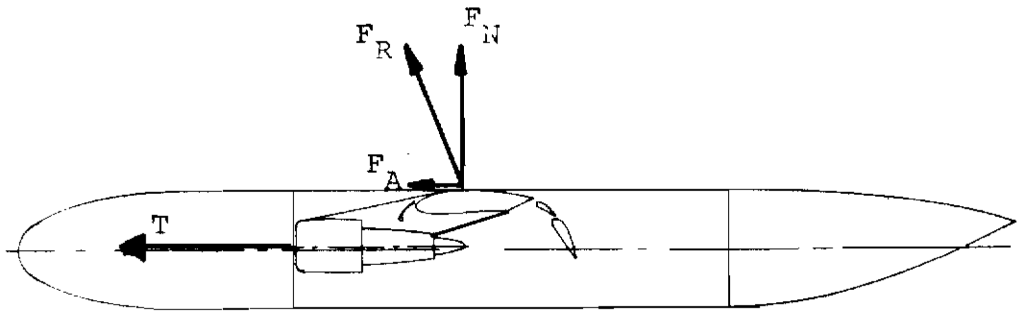


FIGURE 26. COMPARISON OF AERODYNAMIC CHARACTERISTICS WITH SPREAD AND CONCENTRATED BLOWING, $\alpha = 0^{\circ}$



$$\delta_J = \tan^{-1} F_N/F_A$$

EFFECTIVE JET DEFLECTION ANGLE

$$\eta = F_R/T$$

STATIC TURNING EFFICIENCY

FIGURE 27. NOMENCLATURE, EFFECTIVE JET DEFLECTION ANGLE AND STATIC TURNING EFFICIENCY

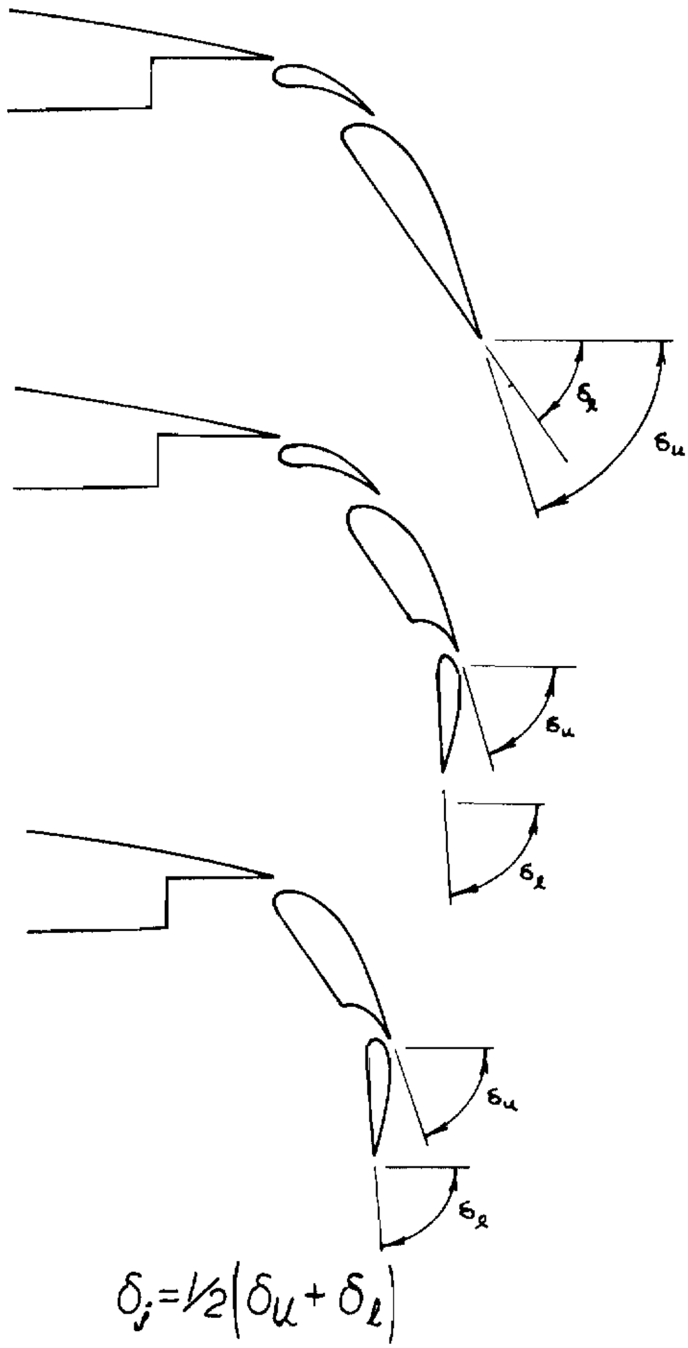


FIGURE 28. NOMENCLATURE, EFFECTIVE JET DEFLECTION ANGLE

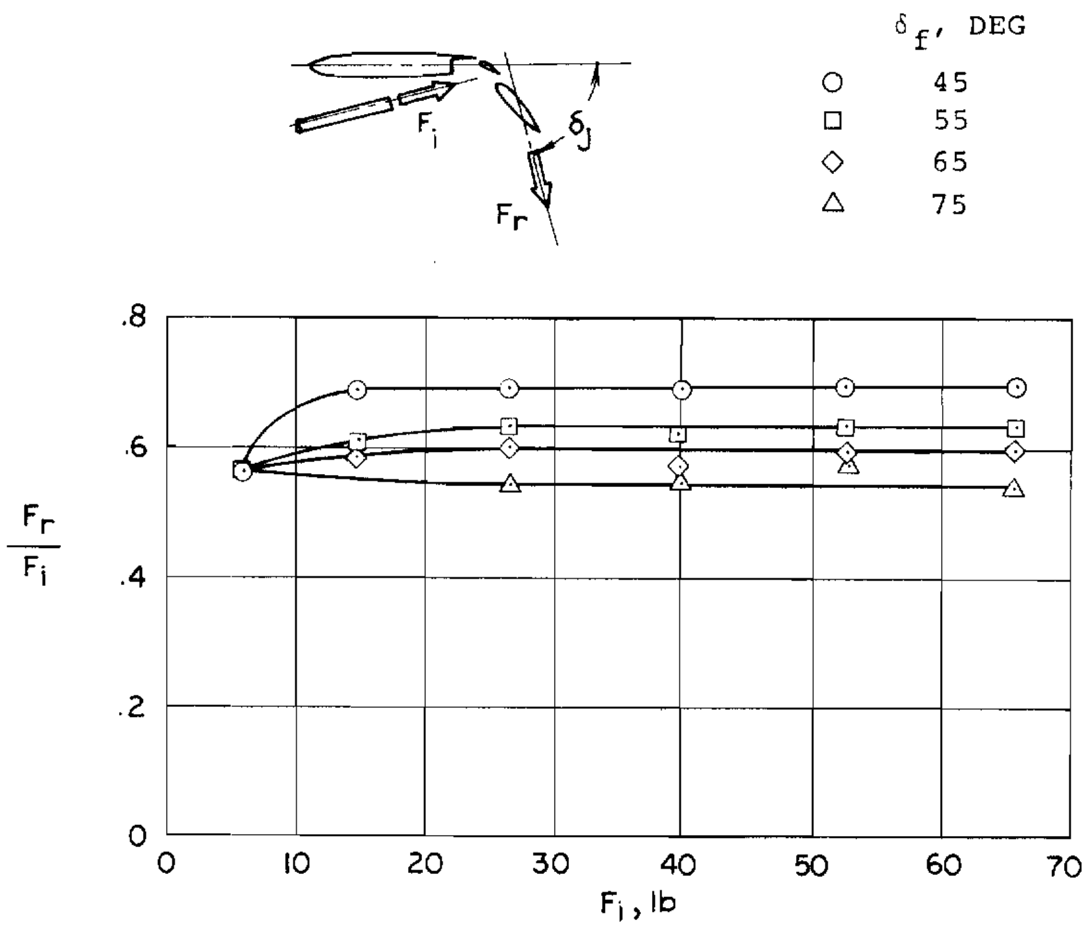


FIGURE 29. EFFECT OF THRUST LEVEL ON STATIC TURNING EFFICIENCY

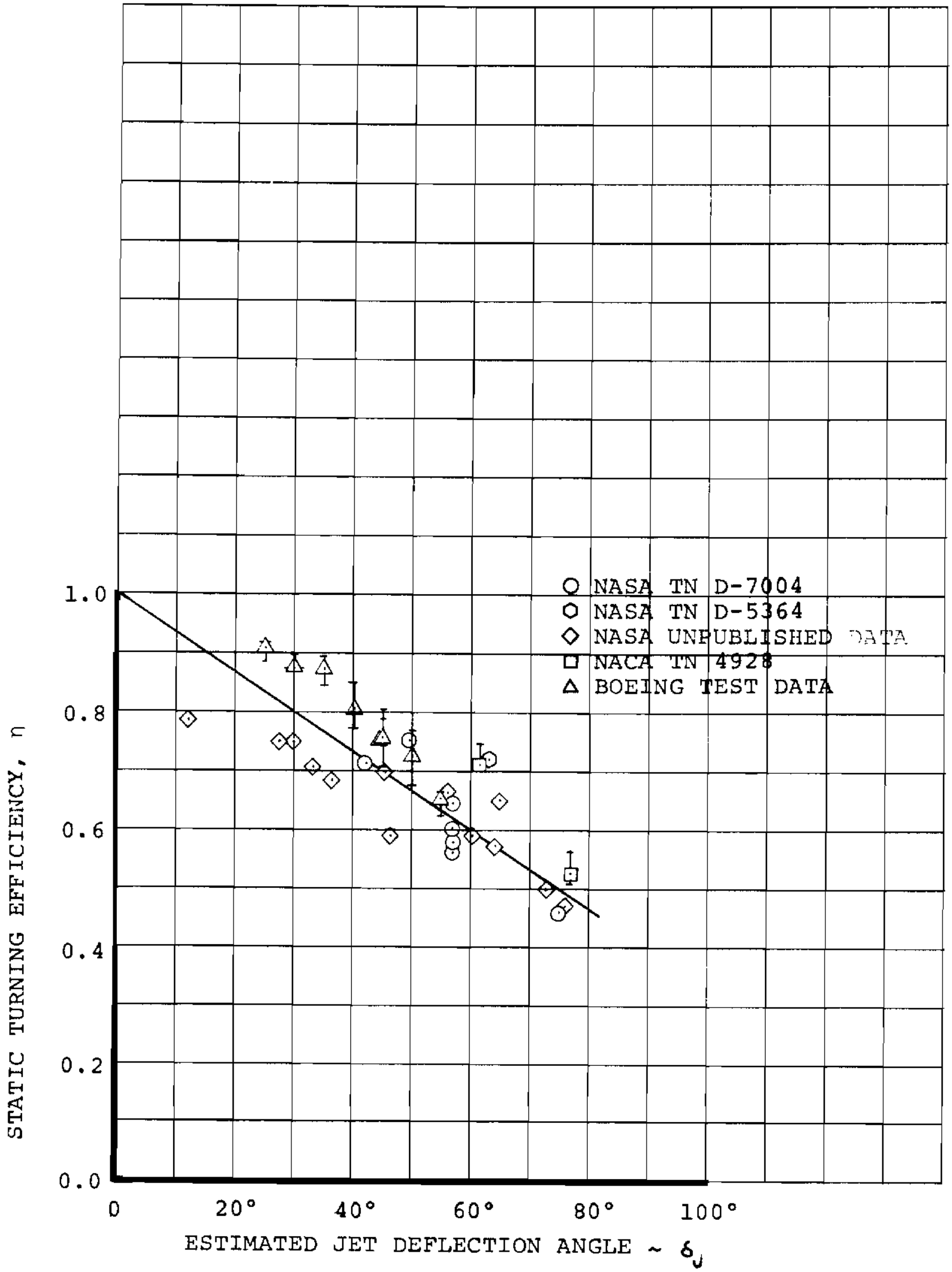


FIGURE 30. FLAP TURNING EFFICIENCY CORRELATION
DOUBLE-SLOTTED FLAPS

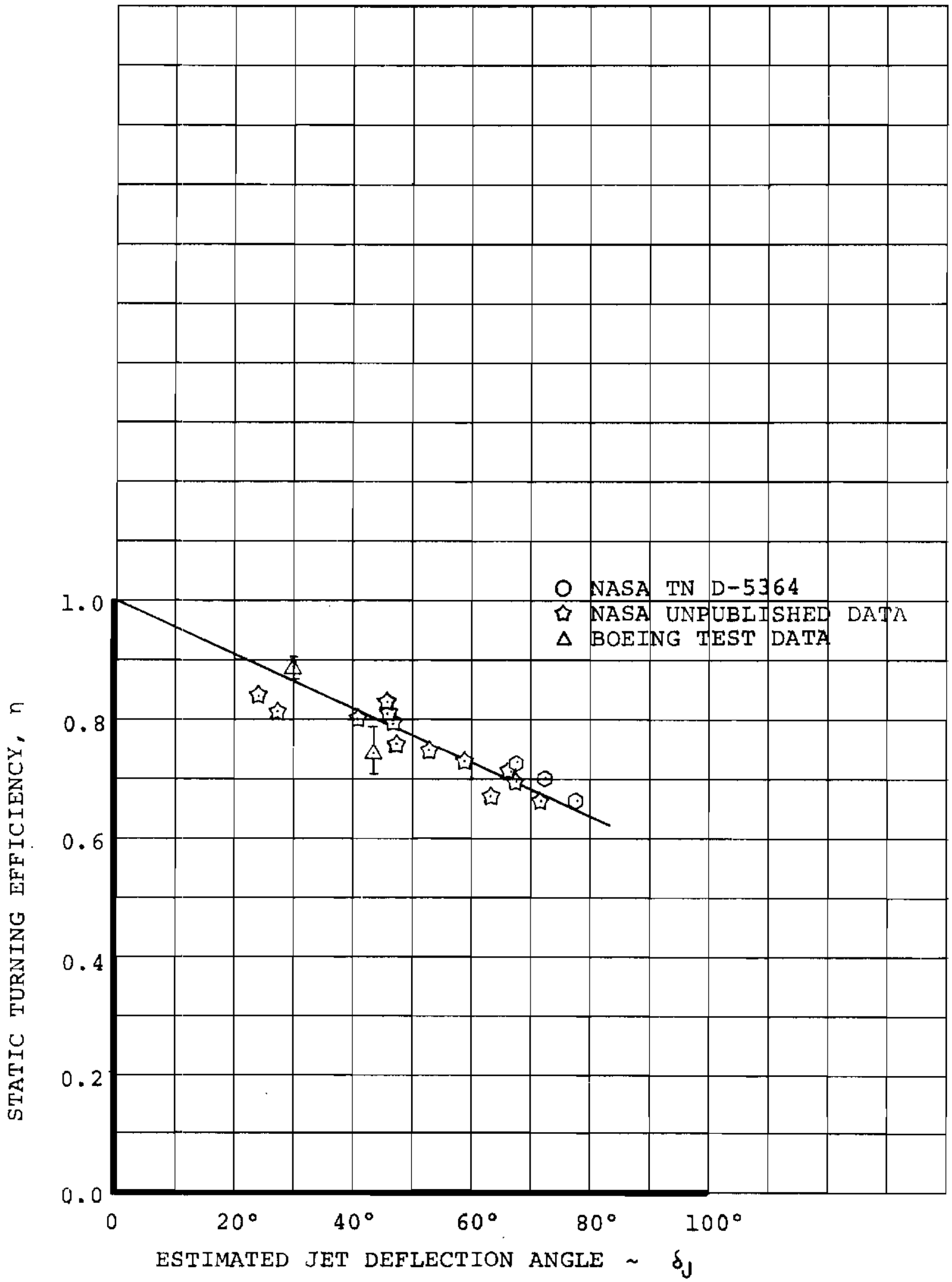


FIGURE 31. FLAP TURNING EFFICIENCY CORRELATION
TRIPLE-SLOTTED FLAPS

worse than those shown can result. This report did not attempt to determine how to ensure good turning efficiencies.

In determining theoretical jet flap lift, the important momentum parameter is the momentum coefficient at the wing trailing edge. Correlating test data, externally blown flap lift correlates best when the engine exhaust jet momentum is used directly and losses are considered only in determining drag.

4.1.3 Lift

Jet flap theory is still in its early stages of development compared to unblown wing theory. The theories all assume small perturbations of the undisturbed flow. Small jet angles are assumed. Inviscid theories have been developed for the jet flap in two and three dimensions (References 8, 9, 10, and 11). The two-dimensional theory, Reference 9, is derived for bent flat plates. The three-dimensional theories, References 10 and 11, have been restricted to flat, elliptically loaded high aspect ratio wings with full span constant flap deflection and constant sectional momentum coefficient along the span. For a jet-flapped wing, the high aspect ratio assumptions require that the wing plus curved portion of the jet be of high aspect ratio.

Despite these limiting assumptions, jet flap theory compares well with experimental data that lie outside the range of the assumptions made.

Since the externally blown flap is a type of jet-augmented flap, jet flap theory has been used as a base on which to build a semi-empirical lift prediction method.

4.1.3.1 Lift Curve Slope. From Reference 11, the lift curve slope of an elliptically loaded jet flapped wing of high aspect ratio is

$$C_{L_\alpha} = \frac{C_{L_{\alpha_{2D}}}}{1 + \left[\dfrac{C_{L_{\alpha_{2D}}} - 8\pi D_o - 2C_J}{\pi A + 2C_J}\right]} \tag{4}$$

where D_o is a function of the jet momentum coefficient, and is reproduced in Table VI. Rewriting equation 4 and adding and subtracting 2π in the denominator.

$$C_{L_\alpha} = \frac{C_{L_{\alpha_{2D}}} (\pi A + 2C_J)}{\pi A + C_{L_{\alpha_{2D}}} - 8\pi D_o + 2\pi - 2\pi} \tag{5}$$

Factoring out A/ (A + 2)

$$C_{L_\alpha} = \frac{C_{L_{\alpha_{2D}}} \left(\frac{A}{A+2}\right)\left(1 + \frac{2\,C_J}{\pi A}\right)}{1 + (C_{L_{\alpha_{2D}}} / \pi - 8D_o - 2)/(A+2)} \qquad (6)$$

From Reference 9, the lift curve slope of a flat plate airfoil with a jet flap is

$$C_{L_{\alpha_{2D}}} = 2\,\pi\,(1 + .151\,C_J^{1/2} + .219\,C_J) \qquad (7)$$

Substituting this into equation 6 and rearranging

$$C_{L_\alpha} = \left[2\,\pi\left(\frac{A}{A+2}\right)\right]\frac{(1 + .151\,C_J^{1/2} + .219\,C_J)\,(1 + 2C_J/\pi A)}{(1 + 2\,(.151\,C_J^{1/2} + .218\,C_J - 4\,D_o)/(A+2)} \qquad (8)$$

The leading term, $2\,\pi A/(A+2)$, is the lift curve slope of an unpowered elliptically loaded wing so that the powered lift curve slope of a jet flapped wing with small jet deflection can be expressed as

$$C_{L_{\alpha\,\text{POWER ON}}} = C_{L_{\alpha\,\text{POWER OFF}}}\,K\,(A,\,C_J) \qquad (9)$$

$K\,(A,\,C_J)$ can be taken to be a function of blowing momentum coefficient only, with an error of less than 1% for A of 6-10 and C_J up to 10. $K\,(A,\,C_J)$ is given in Figure 32.

Since the theory is linear and assumes small angles, there is a direct thrust effect on the lift curve slope included in equation 9 of $\partial\,[C_J\,(\alpha + \delta)]/\partial\,\alpha = C_J$. To account for large jet deflection angles equation 9 will be modified by replacing $(\alpha + \delta)$ by the correct $\sin\,(\alpha + \delta)$ so that the direct thrust effect is given by $\partial\,[C_J \sin\,(\alpha + \delta)]/\partial\alpha \doteq C_J \cos\,(\alpha + \delta) = C_J \cos\,\delta$ at $\alpha = 0°$

$$C_{L_{\alpha\,\text{POWER ON}}} = C_{L_{\alpha\,\text{POWER OFF}}}\,K\,(A,\,C_J) - C_J + C_J \cos\,\delta_J\ (\text{RAD}^{-1}) \qquad (10)$$

$$C_{L_{\alpha\,\text{POWER ON}}} = C_{L_{\alpha\,\text{POWER OFF}}}\,K(A,\,C_J) - \frac{C_J}{57.3} + \frac{C_J \cos\,\delta_J}{57.3}\ (\text{DEG}^{-1}) \qquad (11)$$

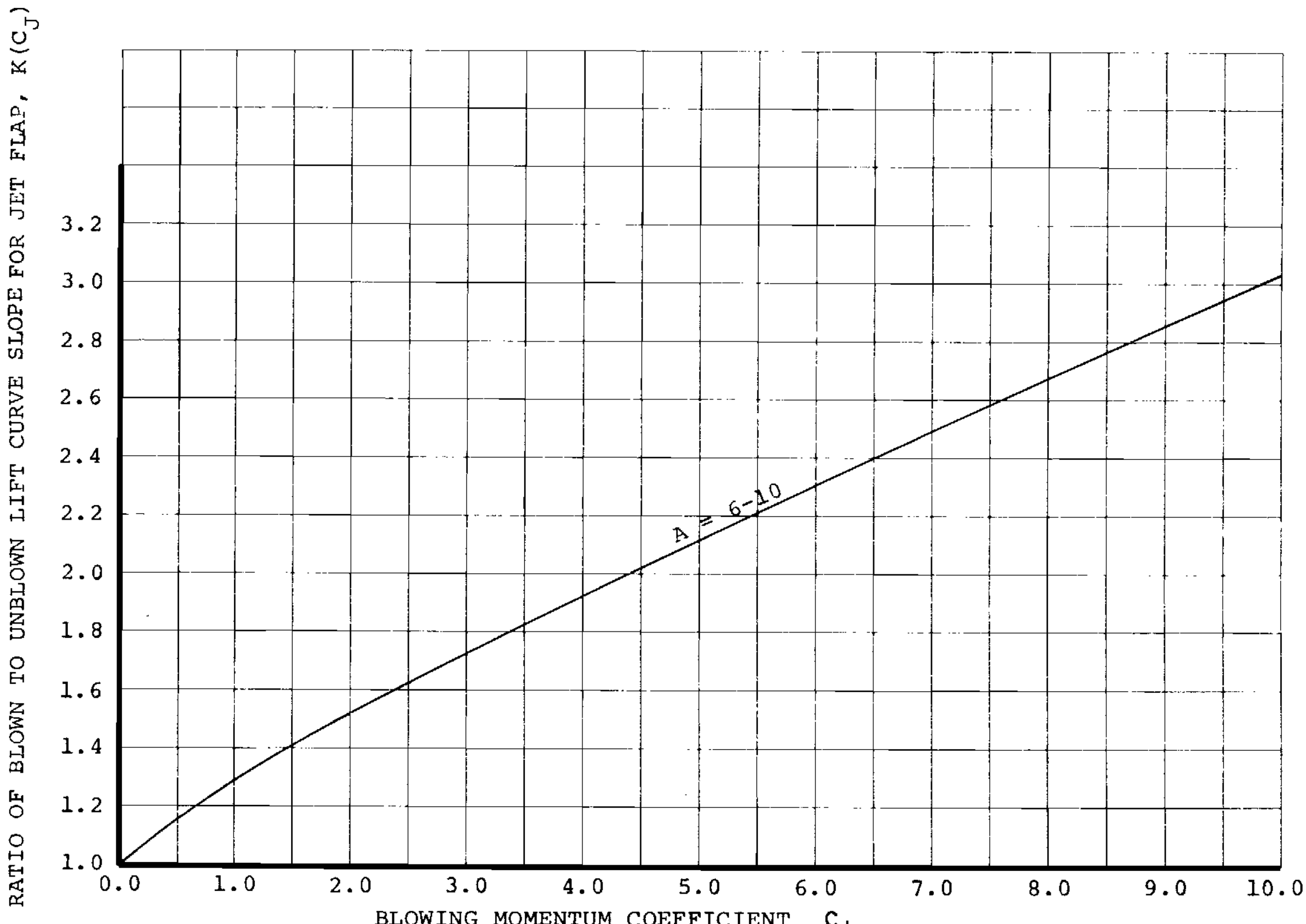


FIGURE 32. JET FLAP EFFECT ON LIFT CURVE SLOPE

TABLE VI

RELATIONSHIP OF C_J AND D_O FOR LIFT CURVE SLOPE CALCULATIONS

C_J	D_o
0.01	-.0008
0.05	-.0040
0.10	-.0080
0.20	-.0158
0.40	-.0318
0.50	-.0398
1.00	-.0798
1.50	-.1198
2.00	-.1600
3.00	-.2402
4.00	-.3204
5.00	-.4008
10.00	-.8034

4.1.3.2 Flap Lift Increment. In Reference 9, the increment in lift on a two-dimensional bent flat plate airfoil with jet augmented flap is derived as

$$C_L = 2(\chi + \sin\chi + 2\pi d_o)\delta \tag{12}$$

where $\chi = 2\sin^{-1}\sqrt{C_f/c}$ and $d_o = d_o\ (Cf/c,\ C_J)$.

Now

$$C_L = 2(\chi + \sin\chi)\delta \tag{13}$$

is just the lift increment of an unblown bent flat plate airfoil. Therefore, the effect of blowing on the derivative of lift with respect to flap deflection is given by

$$\Delta C_{L_\delta}(C_J) = 4\pi d_o \tag{14}$$

This is given in Figure 33 as a function of flap chord ratio and jet momentum coefficient. From Reference 11, the relationship between two-dimensional and

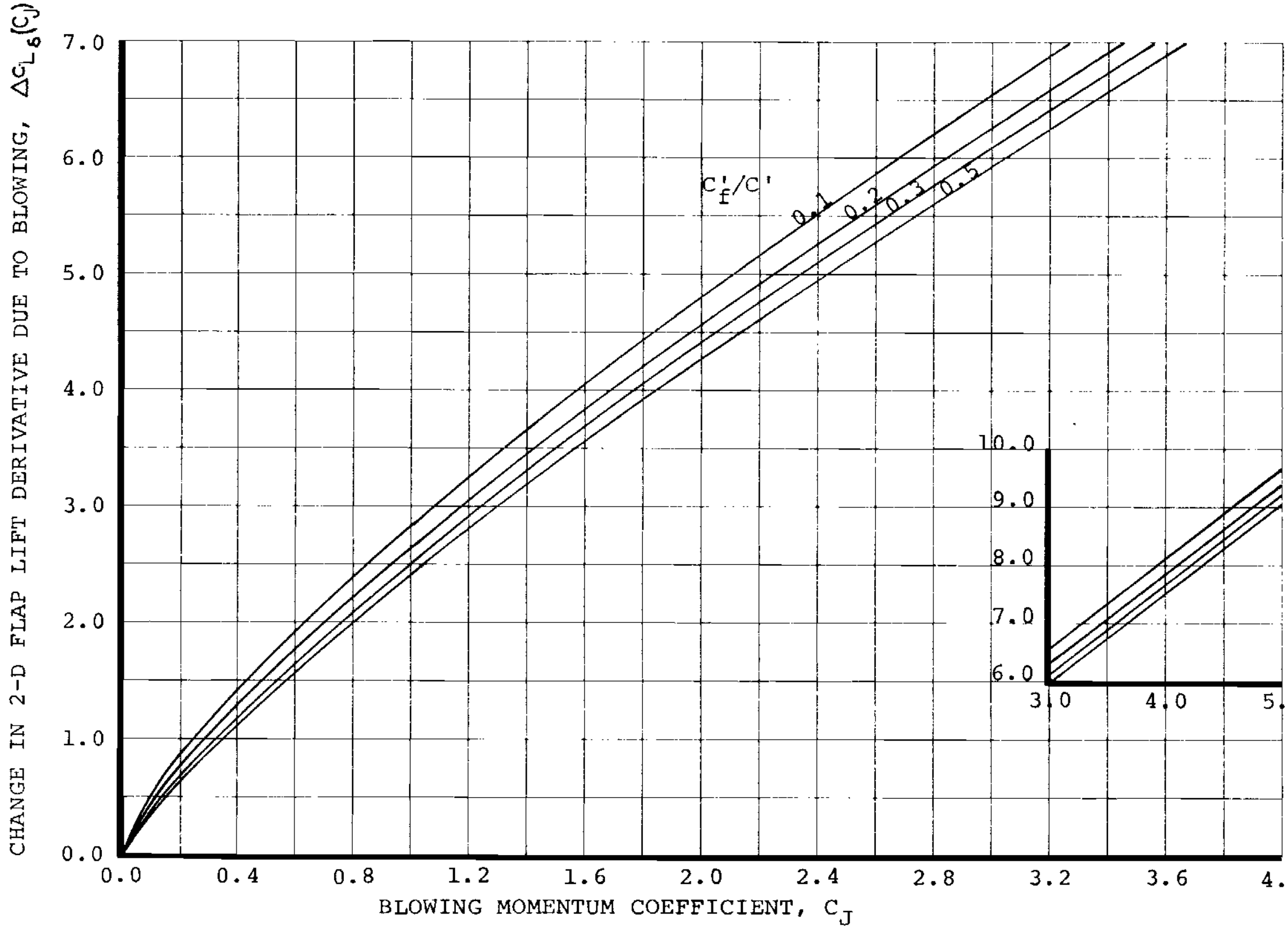


FIGURE 33. TWO-DIMENSIONAL JET FLAP EFFECTIVENESS

three-dimensional lift is given by

$$\frac{C_L}{C_{L_{2D}}} = \frac{1}{\left[1 + \frac{(C_{L_{\alpha_{2D}}} - 8\pi D_o - 2C_J)}{\pi A + 2C_J}\right]} = F(A, C_J) \qquad (15)$$

This has been calculated for A = 6–10 and C_J = 0–5 and is plotted in Figure 34.

The incremental flap lift due to power is given by

$$\Delta C_L(C_J) = \Delta C_{L_\delta}(C_J')\, F(A, C_J')\, \frac{\delta_J}{57.3}\, \frac{S'}{S} \qquad (16)$$

where S' is the flapped wing area and C_J' is reference to S' (Figure 35).

4.1.3.3 Maximum Lift. The estimation of maximum lift coefficient is not amenable to theoretical calculation. Therefore, an empirical correlation approach has been undertaken. It is reasonable to assume that the increment in maximum lift due to power will be proportional to the induced camber. In turn, this induced camber is expected to be a function of the vertical component of the jet momentum, $\eta C_J \sin \delta_J$. The data from a number of tests has been plotted in Figure 36. The data generalizes very well with no discernable trends due to aspect ratio, flap chord, or leading edge device.

This curve should be used with discretion. A configuration having substantial leading edge separation for the unpowered case may yield increments in $C_{L_{max}}$ due to power much higher than these due to the boundary layer control effect of the blowing. Also, as the vertical component of jet momentum increases, a given leading edge geometry would be expected to depart from this curve as the leading edge is no longer able to support the pressure distribution required without separation.

4.1.4 Force Polars

The theoretical induced drag of a finite aspect ratio wing with full span jet flaps and elliptical loading is derived in Reference 10

$$C_{D_i} = C_{L_T}^2 / (\pi A + 2C_J) \qquad (17)$$

where C_{L_T} is the total lift.

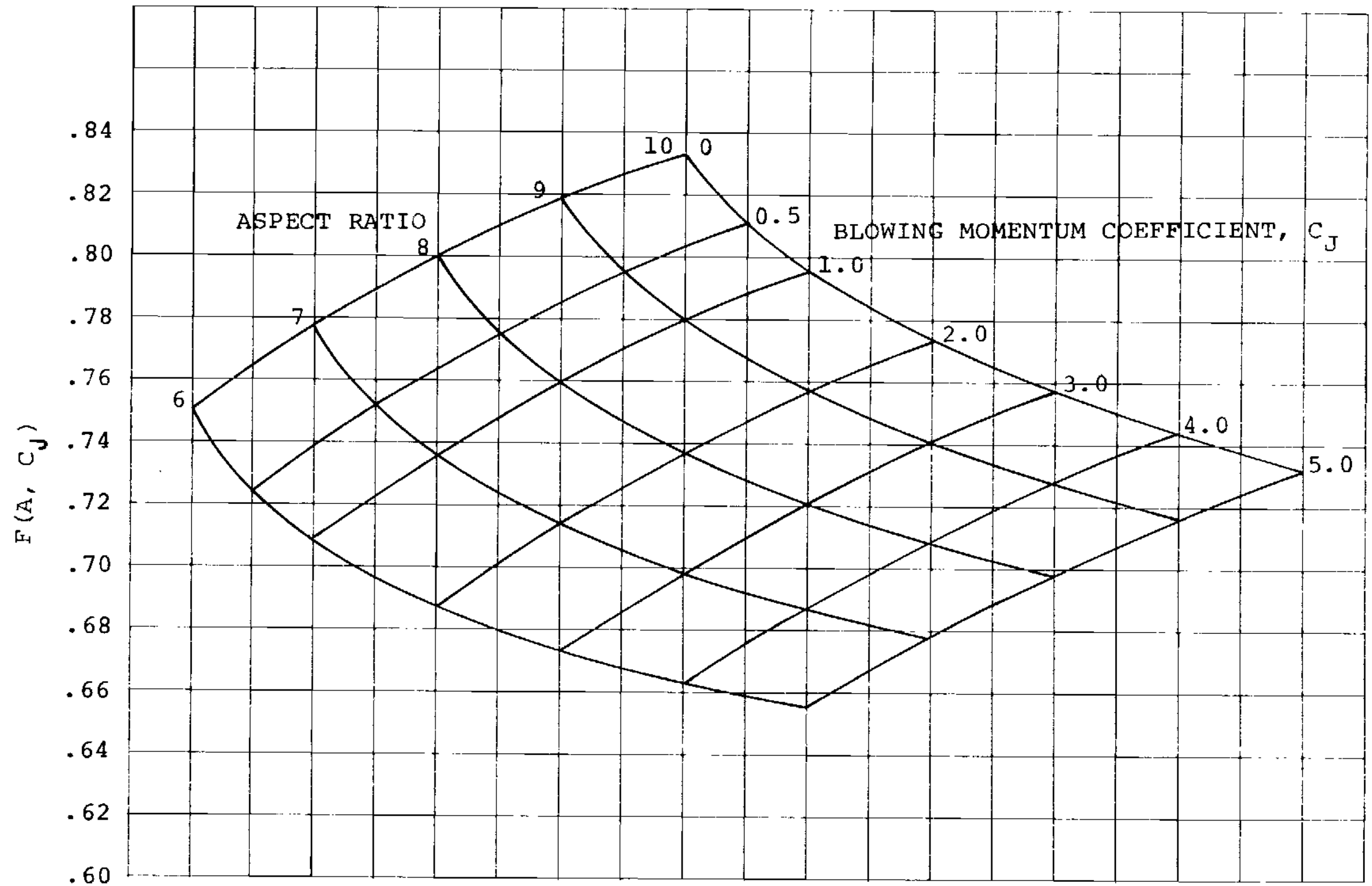
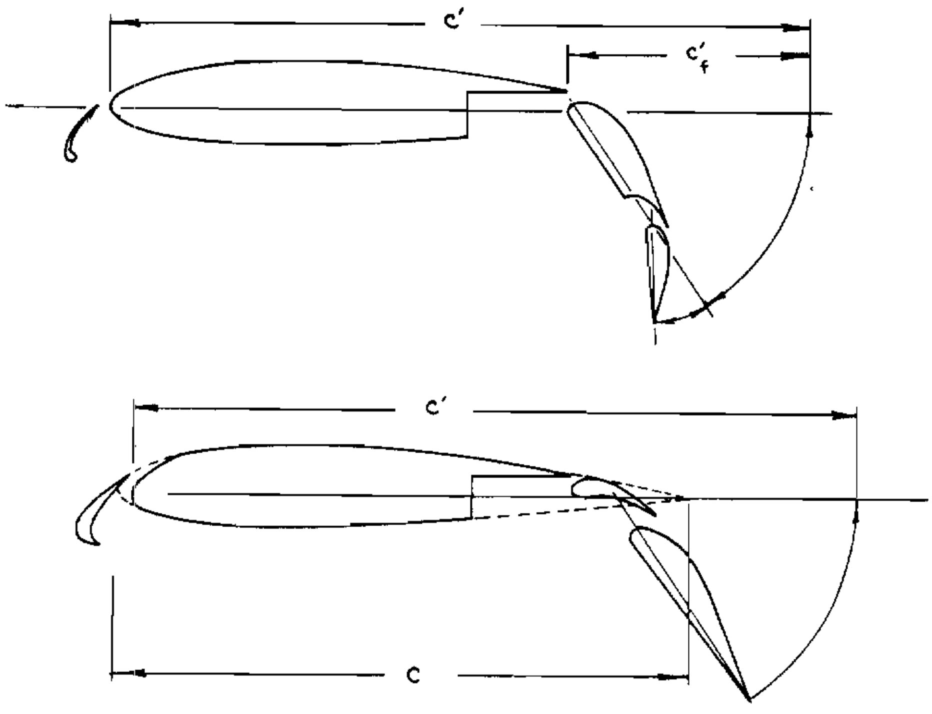
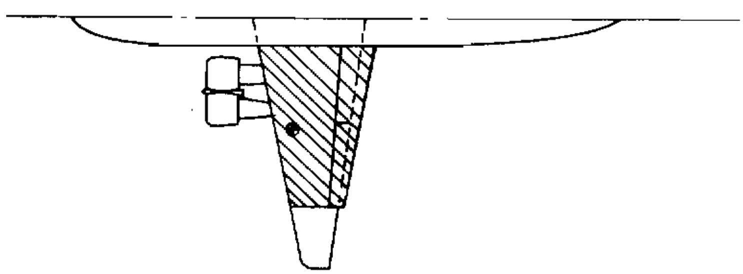


FIGURE 34. JET FLAP FINITE ASPECT RATIO FACTOR



DEVELOPED FLAP AND WING CHORDS



FLAPPED WING AREA, S'

FIGURE 35. NOMENCLATURE, DEVELOPED FLAP AND WING CHORDS AND FLAPPED WING AREA

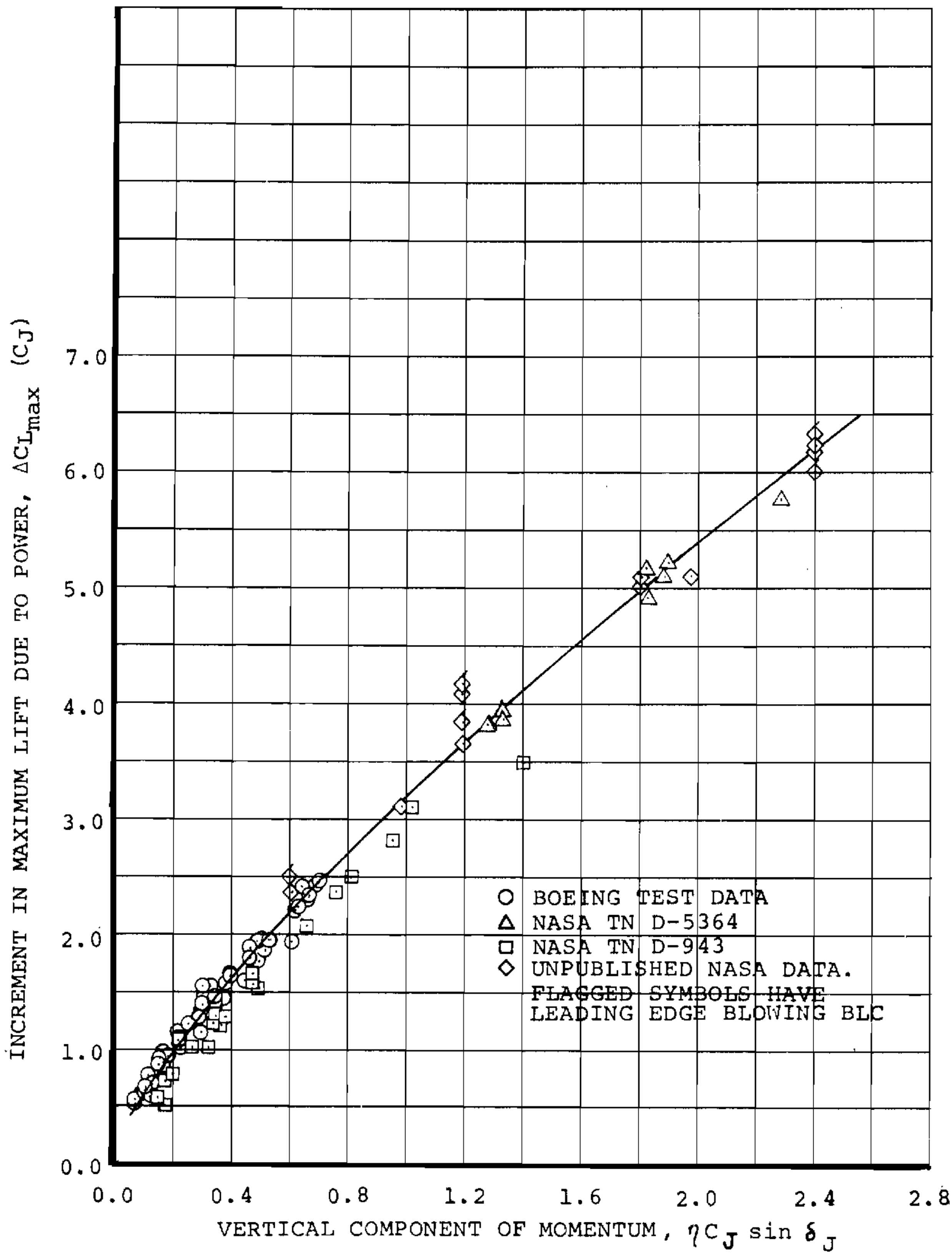


FIGURE 36. CORRELATION OF MAXIMUM LIFT DUE TO POWER

This has been shown, Reference 14, to be equivalent to

$$C_{D_i} = \left[C_{L_c}^2 / \pi A \right] (1 + 2C_J/\pi A) \tag{18}$$

where C_{L_c} is the circulation lift coefficient.

The force polar is

$$C_D - C_{D_{Pmin}} = -C_J + \left[C_{L_c}^2 / \pi A \right] (1 + 2C_J/\pi A) \tag{19}$$

It will be shown that this is equivalent to the polar given by the unpowered theoretical induced drag for an elliptical loaded wing with the thrust deflected to the optimum angle for level flight.

Consider the unpowered induced drag polar given by

$$C_{D_i} = C_{L_c}^2 / \pi A \tag{20}$$

Now the optimum thrust deflection for level flight is given by the vector normal to the polar. Therefore,

$$\tan \theta = -2 \; C_{L_c}/\pi A = \Delta \; C_{L_{c_J}}/\Delta C_{D_{c_J}} \tag{21}$$

Now

$$\Delta C_{L_{c_J}}^2 + \Delta C_{D_{c_J}}^2 = C_J^2 \tag{22}$$

Solving equations 21 and 22 for $\Delta C_{D_{c_J}}$

$$\Delta C_{D_{c_J}} = - \frac{C_J}{\left[1 + \left(\frac{2C_{L_c}}{\pi A} \right)^2 \right]^{1/2}} \tag{23}$$

Therefore, the total drag polar is given by

$$C_D - C_{D_{Pmin}} = \frac{C_{L_c}^2}{\pi A} - \frac{C_J}{\left[1 + (2C_{L_c}/\pi A)^2 \right]^{1/2}} \tag{24}$$

Since the theory considers large A, the denominator of equation 24 can be expanded and only the first-order term kept

$$C_D - C_{D_{Pmin}} = C_{L_c}^2/\pi A \quad - C_J \left[1 - 2C_{L_c}^2/(\pi A)^2\right]$$

$$= \frac{C_{L_c}^2}{\pi A}\left[1 + 2\, C_J/\pi A\right] \quad - C_J \qquad \text{Q.E.D.} \tag{25}$$

There are no theories and little test data for a wing with part-span jet flaps. Most of the externally blown flap test data have part-span flaps and what must be considered unknown spanwise blowing extent.

The force polars for the available externally blown flap tests with spread engines are given closely by the sum of the minimum profile drag, theoretical unpowered induced drag, the ram drag, and the thrust deflected to the optimum angle for level flight, Figure 37. For the externally blown flap, the thrust must be multiplied by the turning efficiency to account for the losses in the system.

Some test configurations have the engines moved well inboard to reduce asymmetric thrust effects on the lateral characteristics. This would be expected to yield a spanwise load distribution which was highly concentrated inboard. These data do not yield polars consistent with the method just outlined. If, however, aspect ratio is based on flap span, then good agreement is again obtained.

4.1.5 Pitching Moments

None of the three-dimensional jet flap theories consider pitching moments. Therefore, two-dimensional theory will be used for a starting point.

Since the theories are linear, the total pitching moment may be expressed as the sum of the pitching moment due to the unpowered potential flow lift and the pitching moment due to power-induced lift. The pitching moment about the leading edge can, therefore, be written

$$C_m = C_{L_T}\, X_{cp}/c\Big|_T = \Delta C_{L_{c_J}}\, X_{cp}/c\Big|_{c_J} + \Delta C_{L_{f_u}}\, X_{cp}/c\Big|_{fu} \tag{26}$$

Solving for the center of the power-induced lift

$$\left.\frac{X_{cp}}{C}\right|_{c_J} = \frac{C_{L_T}\left.\frac{X_{cp}}{C}\right|_T - \Delta C_{Lf_u}\left.\frac{X_{cp}}{C}\right|_{fu}}{\Delta C_{Lc_J}} \tag{27}$$

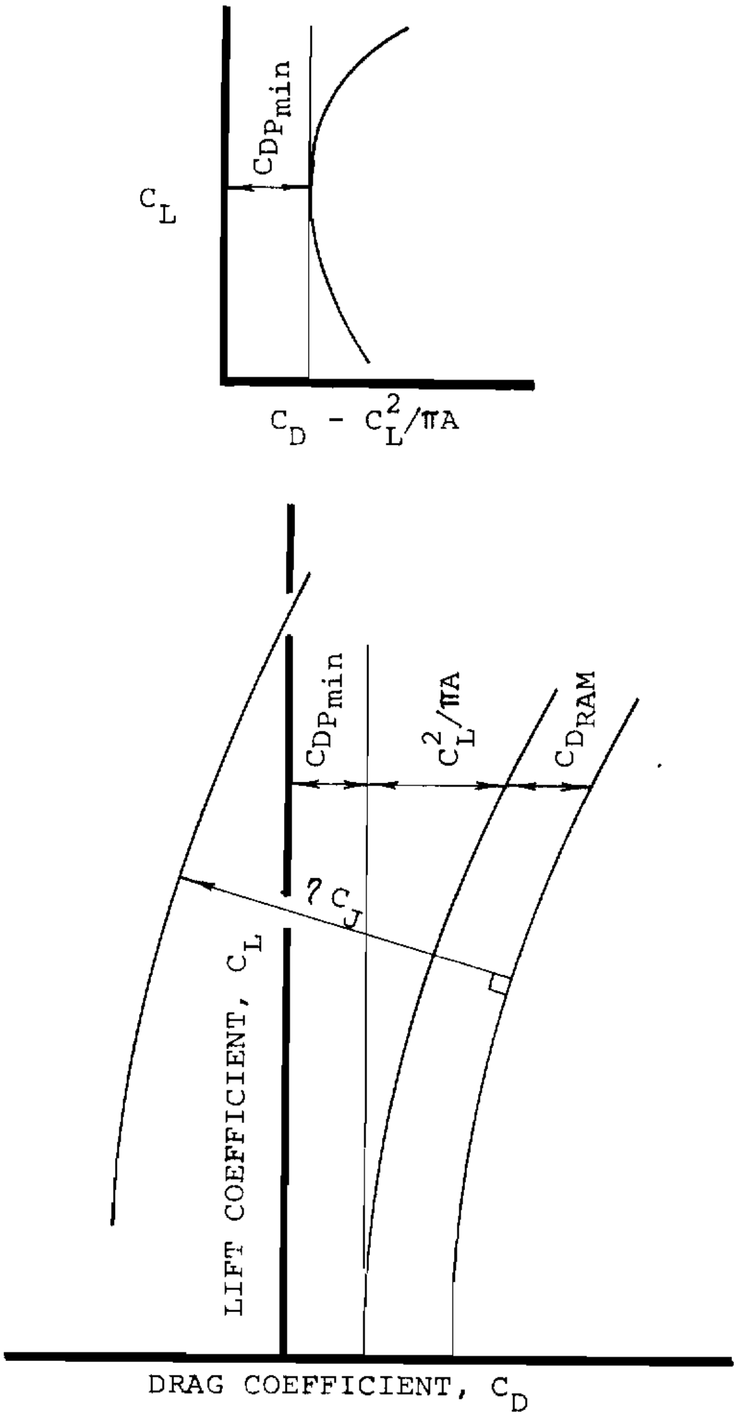


FIGURE 37. FORCE POLAR BUILD-UP

Substituting known functions

$$\left.\frac{X_{cp}}{C}\right|_{c_J} = \frac{\dfrac{\partial C_{m_T}}{\partial \delta} - \dfrac{\partial C_{m_{fu}}}{\partial \delta}}{\dfrac{\partial C_{L_T}}{\partial \delta} - \dfrac{\partial C_{L_{c_J}}}{\partial \delta}} \tag{28}$$

The terms on the right are known from two-dimensional theory, Reference 9. This has been solved for a range of flap chord ratios and momentum coefficients and the results plotted in Figure 38.

For conventional flaps, the effect of finite aspect ratio is to move the flap lift center aft. Finite aspect ratio effects for conventional flaps have been derived in Reference 15 and are presented in Figure 39. It has been assumed that this same effect will be experienced on the jet flap. The spanwise center of load due to flaps is taken at the spanwise location of the mean aerodynamic chord of the flapped portion of the wing.

There is a constant pitching moment coefficient increment at constant angle-of-attack given by

$$\Delta C_m = \Delta C_L\,(c_J)\left(\frac{X_{cg}}{\bar{c}} - \left.\frac{X_{cp}}{\bar{c}}\right|_{cJ}\right) \tag{29}$$

There is also an increment in $\partial Cm/\partial a$ given by

$$\Delta\left(\frac{\partial c_m}{\partial a}\right) = \Delta\left(\frac{\partial C_L}{\partial a}\right)\left(\frac{X_{cg}}{\bar{c}} - \left.\frac{X_{cp}}{\bar{c}}\right|_{a}\right) \tag{30}$$

where $X_{cp}/\bar{c}/a$ is taken to be for a flap chord to wing chord ratio of one $(C'_f/C' = 1.0)$.

Pitching moment appears to be the longitudinal characteristic most sensitive to engine or blowing location and extent, Figure 26.

4.1.6 Asymmetric Thrust

The loss of an engine on an externally blown flap configuration will result in a reduction of high energy air supplied to that wing to augment the lift. The asymmetric thrust condition may result in a loss in lift and a rolling moment due to an asymmetrical lift distribution.

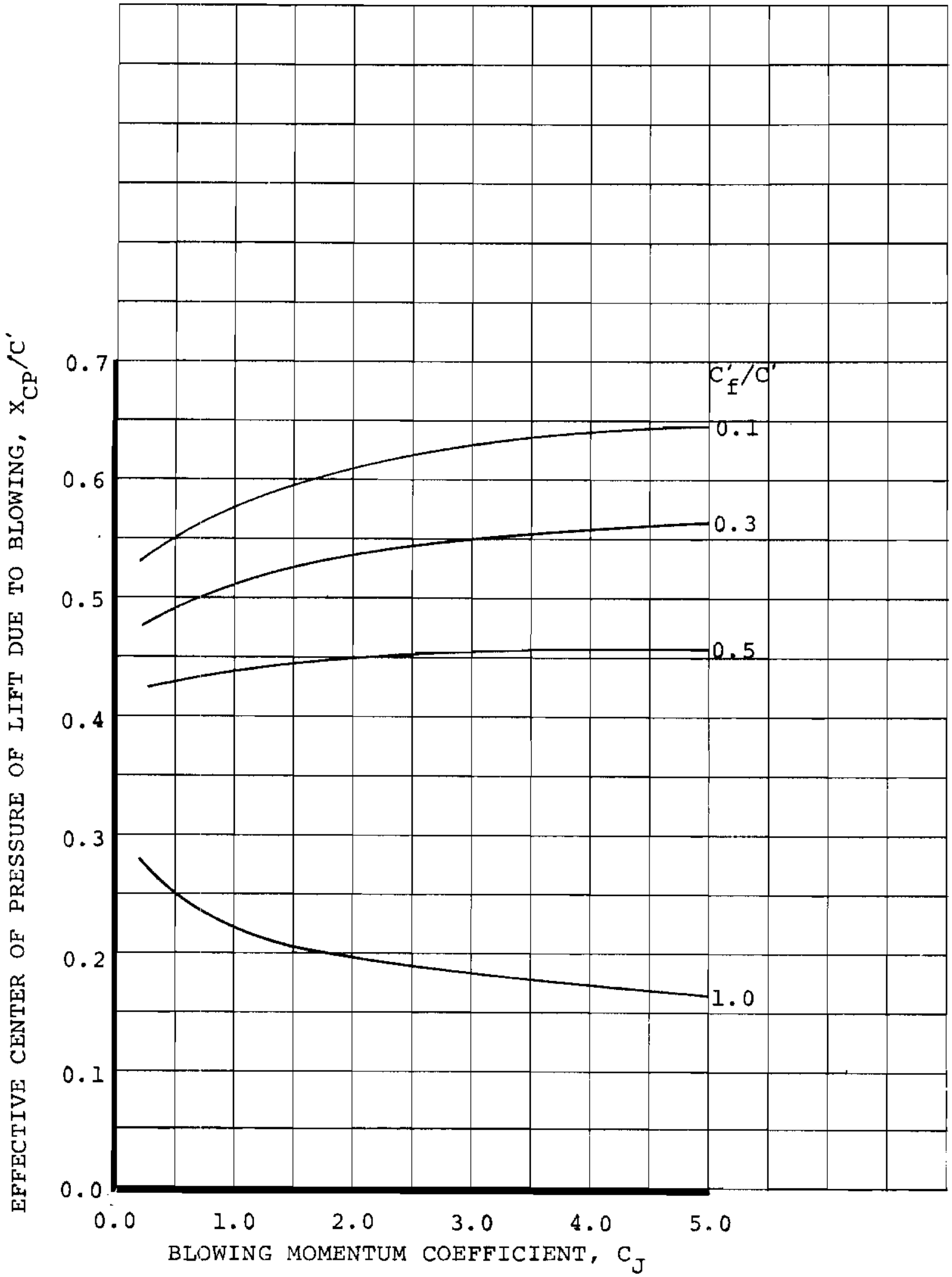


FIGURE 38. CENTER OF PRESSURE OF POWERED LIFT

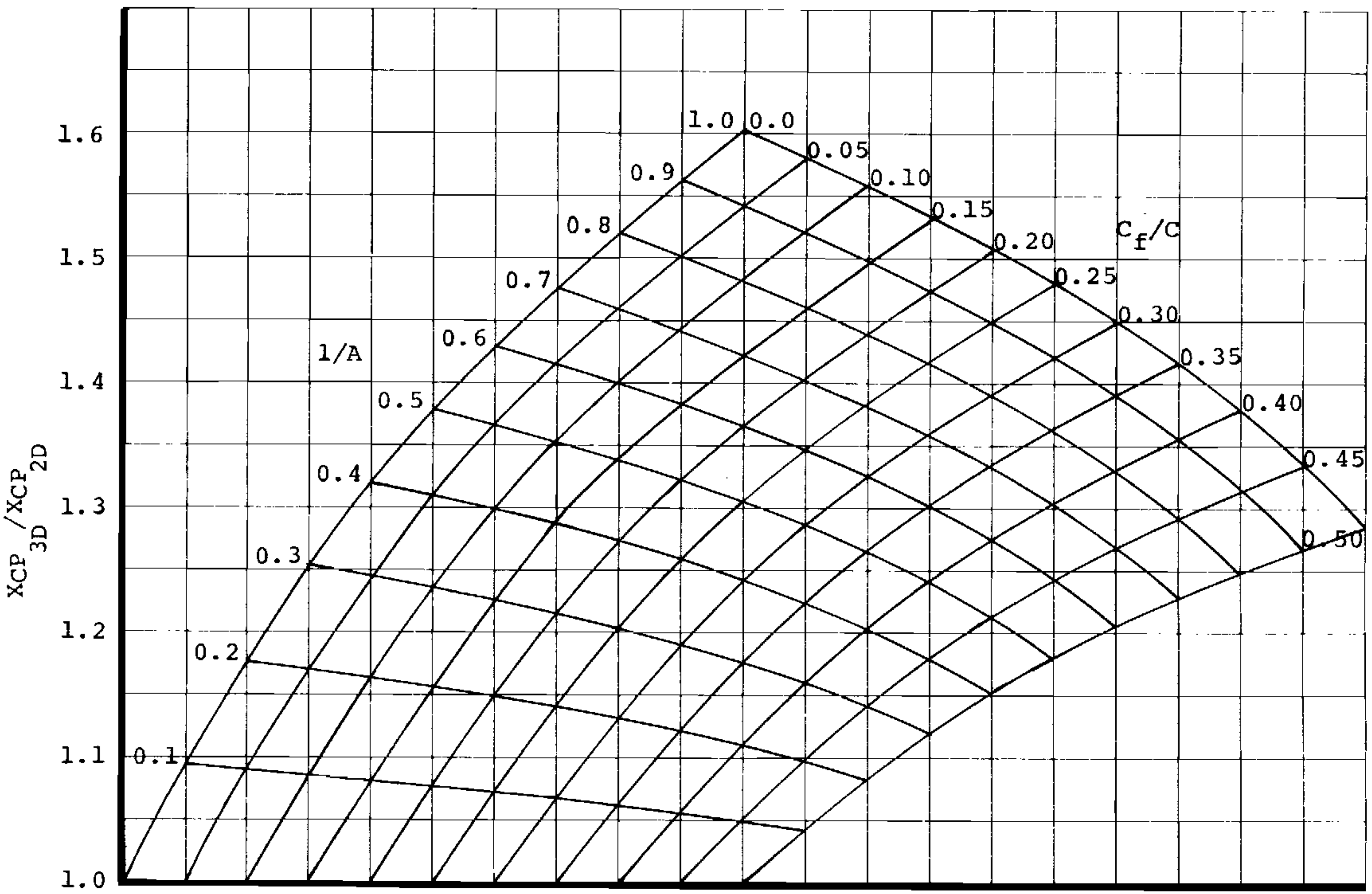


FIGURE 39. FINITE ASPECT RATIO EFFECT ON CHORDWISE LOCATION OF CENTER OF PRESSURE OF FLAP LIFT

Once again, the extent of wing influenced by the propulsive jet must be known in order to compute the aerodynamic characteristics with engine out. NASA test data indicate that the lift and drag of a configuration with asymmetric thrust at a given total jet momentum coefficient level is the same as the symmetric thrust case at the same total C_J, Figure 40 (Reference 16). Tail-on pitching moments are affected due to the changed downwash at the tail. Therefore, the methods of the previous sections may be used without modification.

The rolling moment due to engine-out cannot be established theoretically without knowing the extent of the wing influenced by the jet. Therefore, an empirical correlation considering engine location has been undertaken. The rolling moment can be related to the lift loss by

$$\Delta C_l = \left(\eta_l \big/ 2\right) \quad \Delta C_L \tag{31}$$

where η_l is the effective nondimensional roll arm.

There is substantial scatter in the experimental data, part of which is due to the fact that we must divide increments obtained from test data for which a small absolute error in determining the lift and rolling moments results in large error in determining the effective roll arm.

The spanwise location of the inoperative engine is the primary consideration affecting effective roll arm. Influences which are also important are wing sweep and the spanwise location of the operating engine. Insufficient data is available to determine secondary effects and only the spanwise location of the inoperative engine is considered here.

While there is substantial scatter. the equation

$$\eta_l = \eta_{eng} \tag{32}$$

yields a reasonable first approximation, Figure 41.

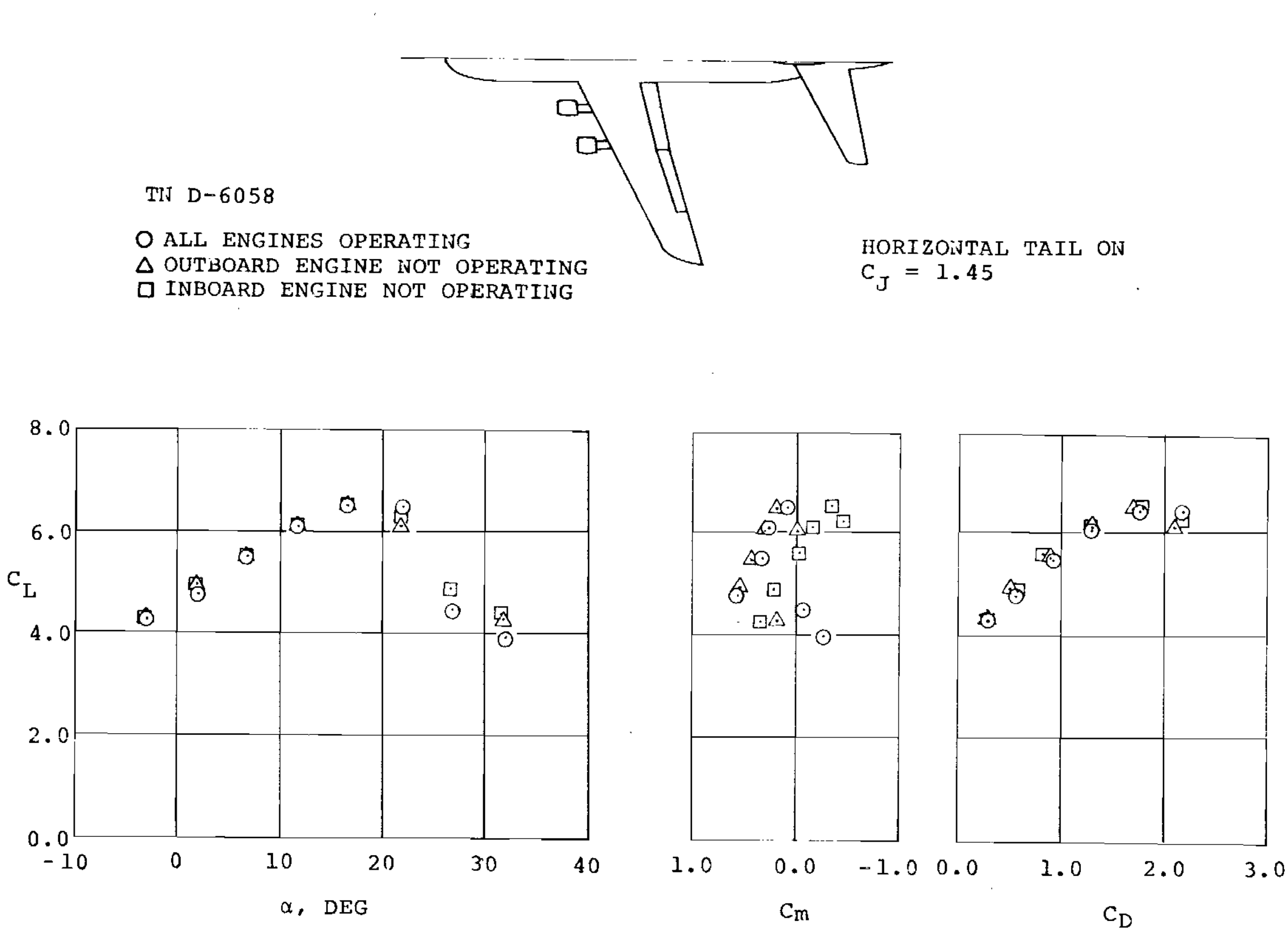


FIGURE 40. EFFECT OF ASYMMETRIC THRUST

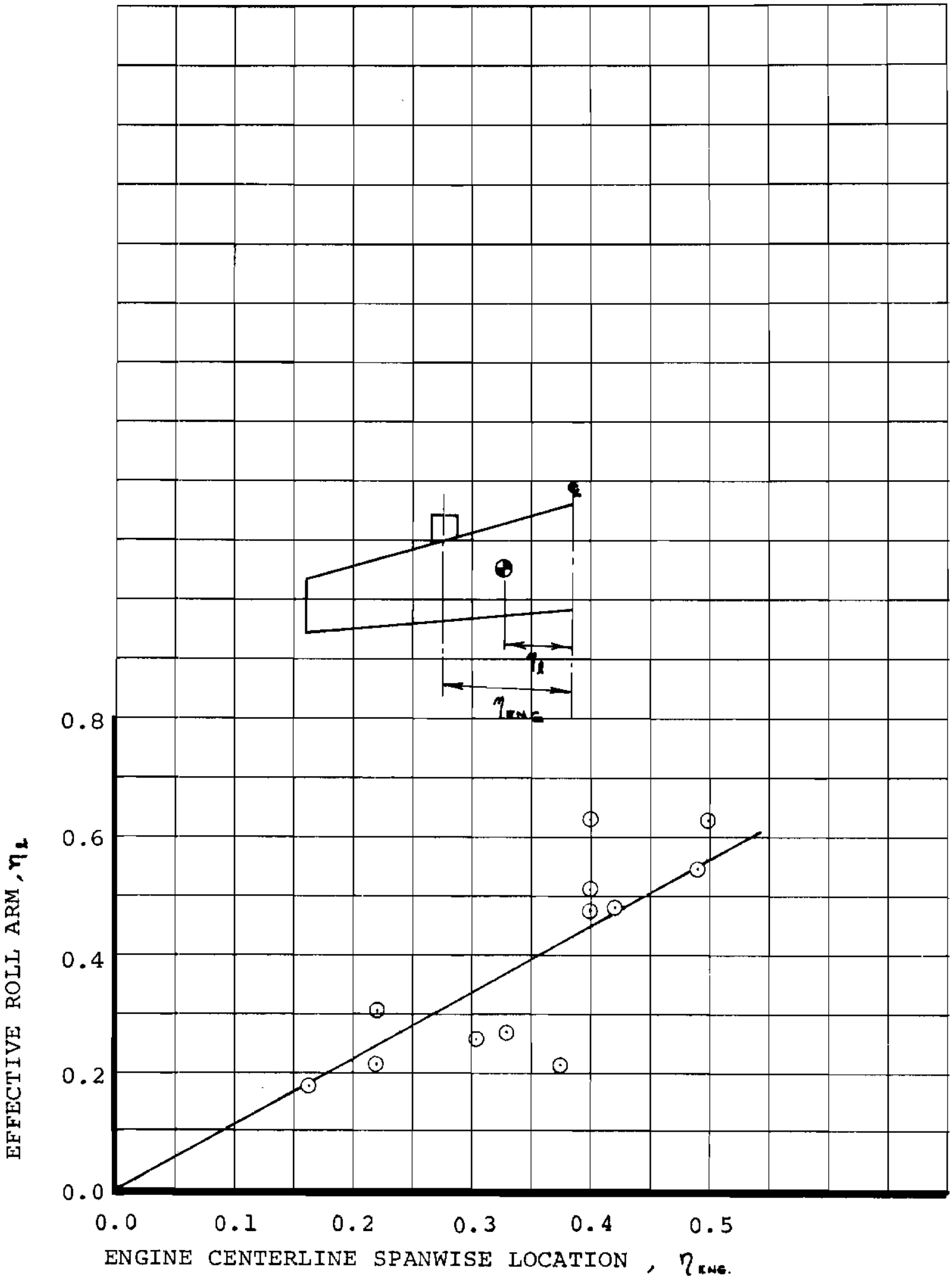
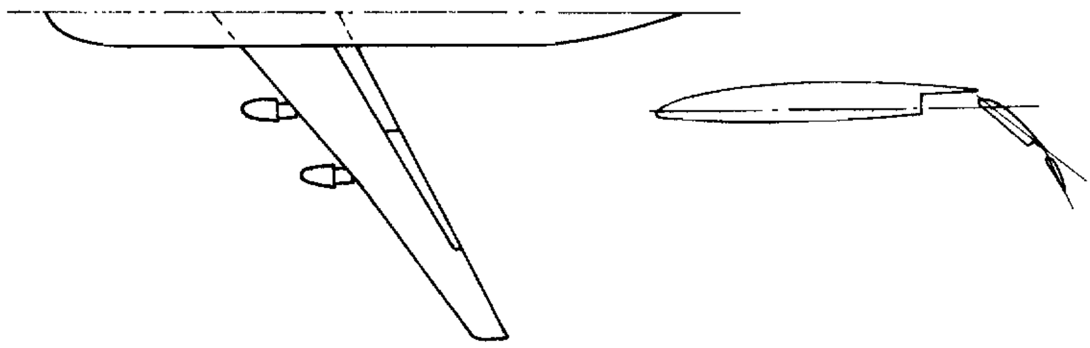


FIGURE 41. ROLL ARM OF FAILED ENGINE LIFT

4.1.7 Application of the Theory



$A_{REF} = 8.4$ $A_{Flaps\ Down} = 7.71$

Flap Span $\eta_{i/b} = .091$ $\eta_{o/b} = .732$

$C'_f/C' = .279$

$C'/C = 1.145$

$S_{REF} = 7.35\ ft^2$

$S_{Flaps\ Down} = 8.0\ ft^2$

$S'/S_{REF} = .697$

$C_J = .59,\ 1.24$

$\delta_f = 30/60$ $\delta_u = 41°$ $\delta_\ell = 59°$

$C_{L_\alpha} = .090\ Deg^{-1}$
Flaps Down
Unpowered

Unpowered wind tunnel data available

STEP 1. Calculate δ_J

$$\delta_J = 1/2\ (\delta_u + \delta_\ell) = 50°$$

STEP 2. From Figure 30 Find η

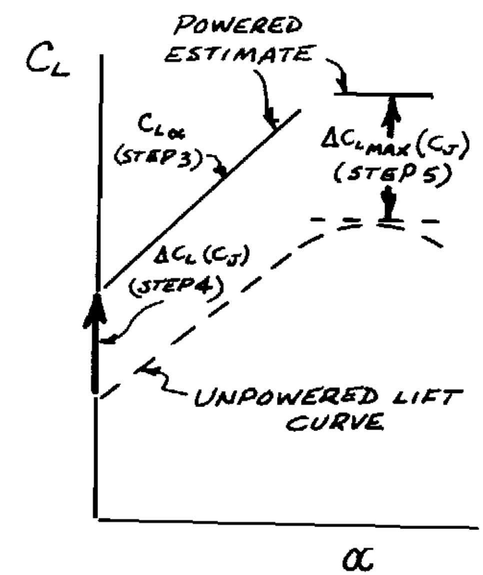
$\delta_J = 50°$, double slotted flaps $\Rightarrow \eta = .665$

STEP 3. Calculate Lift Curve Slope, C_{L_α}

C_J	$C_{J_g} = C_J \frac{S_{REF}}{S_{Flaps\ Down}}$	$K(C_{J_g}, A)$ [Fig. 32]	$C_J\left[1-\cos\ \delta_J\right]/57.3$	$C_{L_\alpha} = .090\ K(C_{J_g}) - C_J \left[\frac{1-\cos\ \delta_J}{57.3}\right]$
.59	.59 (7.36)/8.0) = .542	1.17	.002	.103
1.24	1.13	1.32	.008	.111

STEP 4. Calculate Flap Lift Increment Due to Power, $\Delta C_L\ (C_J)$

C_J	$C_J' = C_J \frac{S_{REF}}{S'}$	$\Delta C_{L_\delta}\ (C_J)$ [Fig. 33]	$F(A, C_J')$ [Fig. 34]	$\Delta C_L(C_J) = \Delta C_{L_\delta}\ (C_J)\ F\ (A, C_J') \frac{\delta J}{57.3} \frac{S'}{S_{REF}}$
.59	.846	2.25	.759	1.04 = (2.25) (.759) (50/57.3) (.679)
1.24	1.780	4.05	.736	1.82



STEP 5. Calculate Increment in Maximum Lift, $\Delta C_{L_{max}}\ (C_J)$

$\eta C_J \sin\ \delta_J$	$\Delta C_{L_{max}}(C_J)$ [Fig. 36]
.665 (.59) sin 50° = .300	1.33
.632	2.25

STEP 6. Estimate Force Polar

From the Unpowered Force Data, Figure 42, find

$$C_{D_{Pmin}} = \min\left[C_{D_{C_J = O}} - \frac{C_L^2}{\pi A}\right] = .095$$

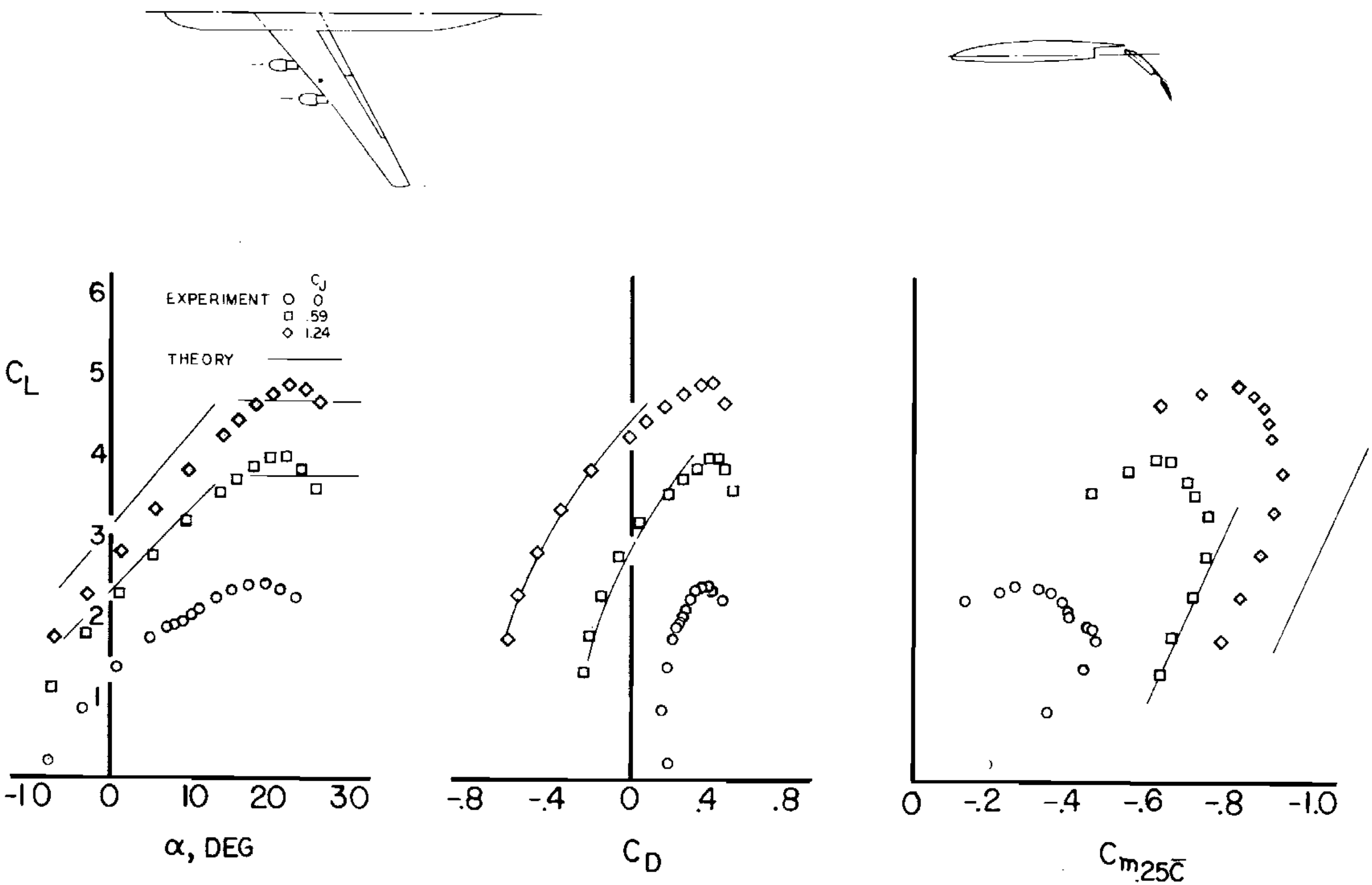


FIGURE 42. THEORETICAL AND TEST DATA ON COEFFICIENT OF LIFT VERSUS ANGLE OF ATTACK, DRAG, AND PITCHING MOMENT AT THE QUARTER-CHORD

Calculate ηC_J

C_J	ηC_J
.59	.392
1.24	.825

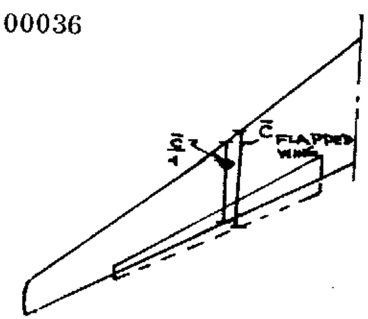
Construct force polars as shown.

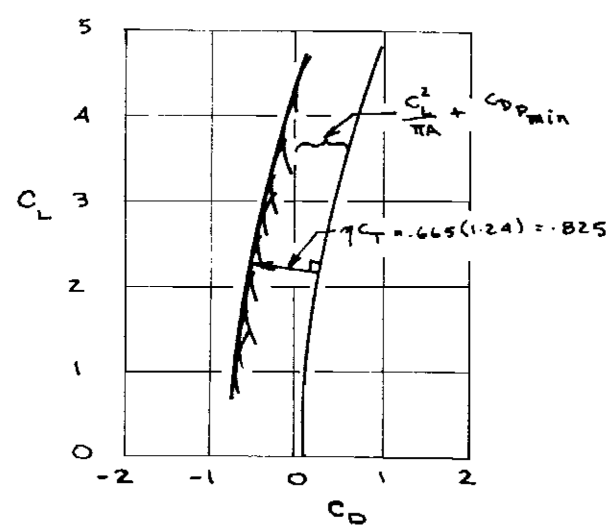
Test used domed inlets so there is no ram drag.

STEP 7. Estimate Pitching Moments

C_J	C'_J	$X_{cp}/C'\big\|_{2D}$ $\frac{C'_f}{C'} = .279$ [Fig. 38]	$X_{cp\,3D}/X_{cp\,2D}$	$X_{cp}/C'\big\|_{3D}$	$\Delta C_{m_\alpha} = o = (.25 - \frac{X_{cp}}{\bar{c}})\Delta C_{L_f}$
.59	.846	.512	1.105	.565	(.25 - .536) (1.04) = -.298
1.24	1.78	.542	1.105	.6	(.25 - .596) (1.82) = -.63

$\frac{X_{cp}}{\bar{c}}$ $C_f'/C' = 1.0$	$\Delta\left(\frac{\partial C_m}{\partial \alpha}\right) = \Delta\left(\frac{\partial C_L}{\partial \alpha}\right)\left(\frac{X_{cg}}{\bar{c}} - \frac{X_{cp}}{\bar{c}}\right)$
.230	.013 (.25 - .264) = -.00015
.203	.021 (.25 - .233) = .00036





4.1.8 Test-Theory Correlation

The validity of a theory must be inferred by comparing theoretical estimates with test results. In this section, test-theory correlation are presented. While scatter exists, it is not sufficiently large to invalidate the methodology developed.

Estimated jet deflection angle is correlated with the jet deflection angle inferred from static power-on testing, Figure 43. Data from eight NASA tests shows that the method correlates within about 10%.

Lift curve slope also correlates within ±10% except for one NASA test which is over-predicted, Figure 44. However, subsequent test data from that model agree well with the technique used.

The largest discrepancy in any estimated increment is in the lift increment due to power, Figure 45. This may be expected since this increment depends on other factors that must be estimated; the extent of wing influenced by the flap and the jet deflection angle. The lift increment should be considered reliable to only ±20%.

The estimated center of pressure of the flap lift due to power agrees well with test data, Figure 46. It would be expected that the flap center of pressure should be significantly different for the same configuration with clustered inboard or spread engines. However, Figure 46 indicates that the difference is no greater than the scatter due to flap angle differences. Actual pitching moment, of course, depends on estimating not only the flap center of pressure but also the flap lift increment.

The longitudinal aerodynamic characteristics of two dissimilar configurations have been estimated using the methods developed, Figures 42 and 47. Figure 42 is the configuration treated in the example of Section 4.1.7. Agreement is fair with the largest discrepancy in pitching moment level which is due to overpredicting the flap lift.

4.1.9 Conclusions

The determination of the lift and drag of a wing with externally blown flaps can be made using jet flap theory with suitable empirical values for jet angle and turning efficiency.

There is a need for more data in order to refine the prediction techniques and determine the influence of some secondary variables. Pressure data is needed in order to determine the influence of the jet on the wing so that part-span loading factors can be developed.

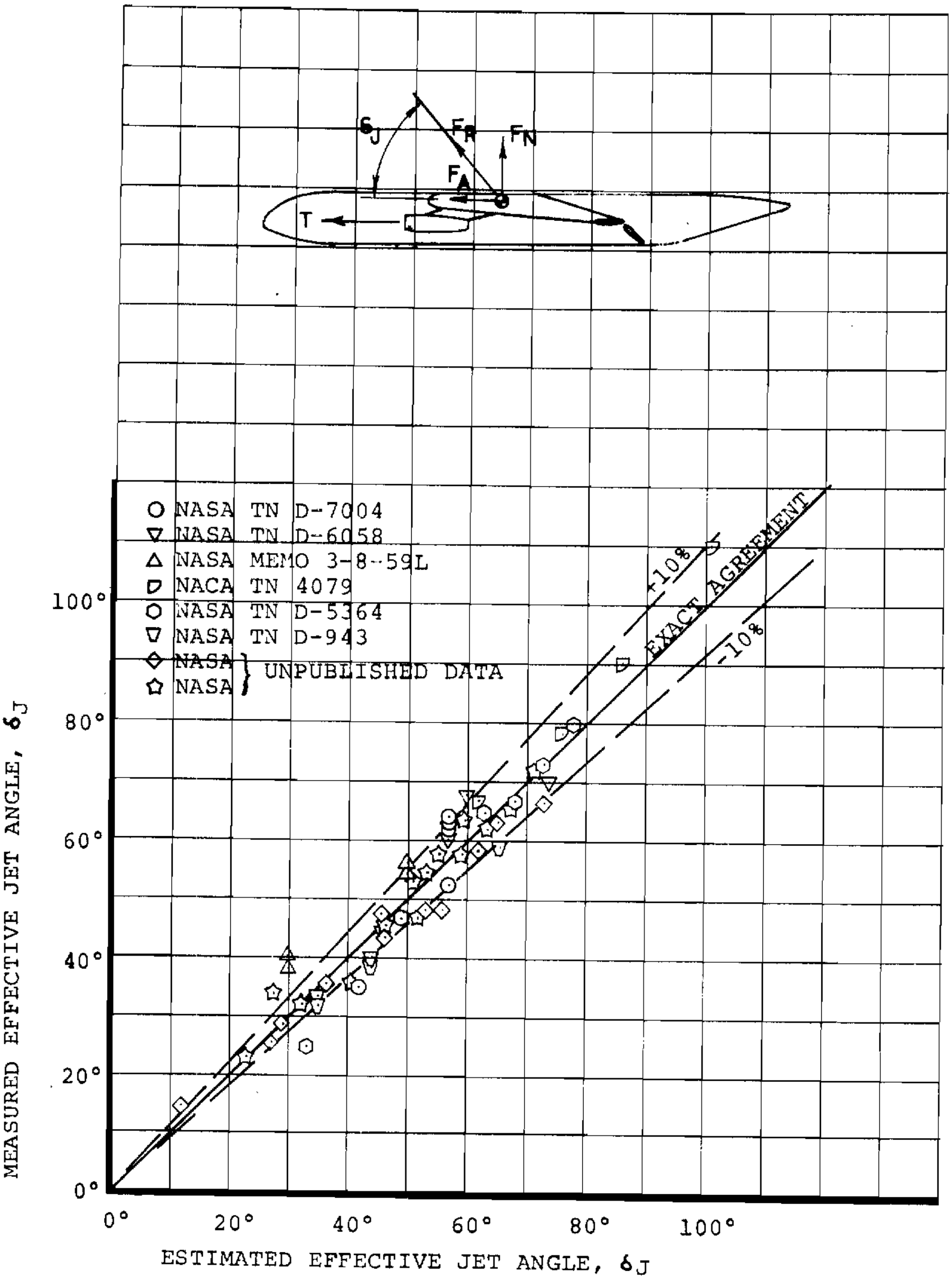


FIGURE 43. EFFECTIVE JET ANGLE CORRELATION

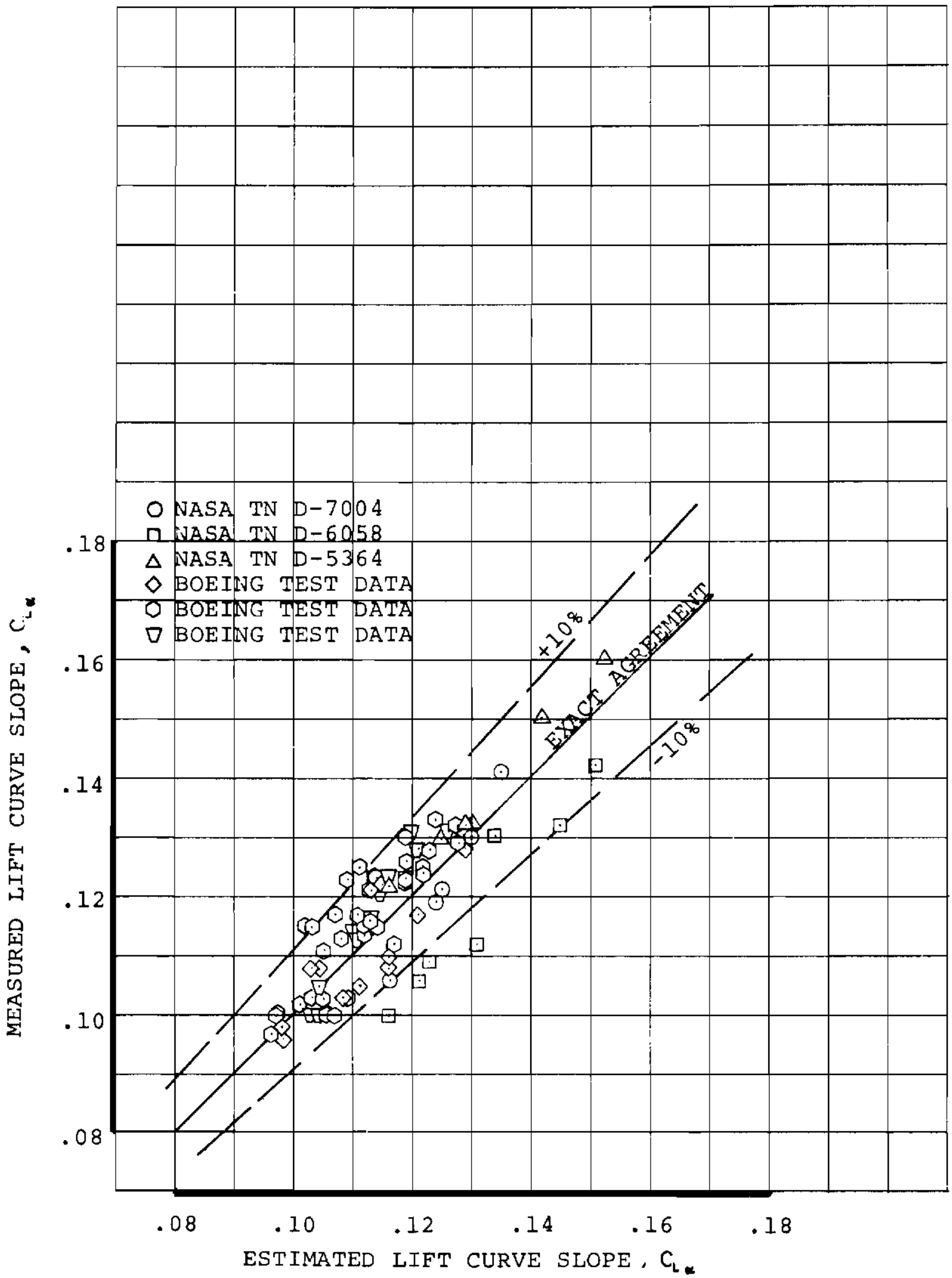


FIGURE 44. LIFT CURVE SLOPE CORRELATION

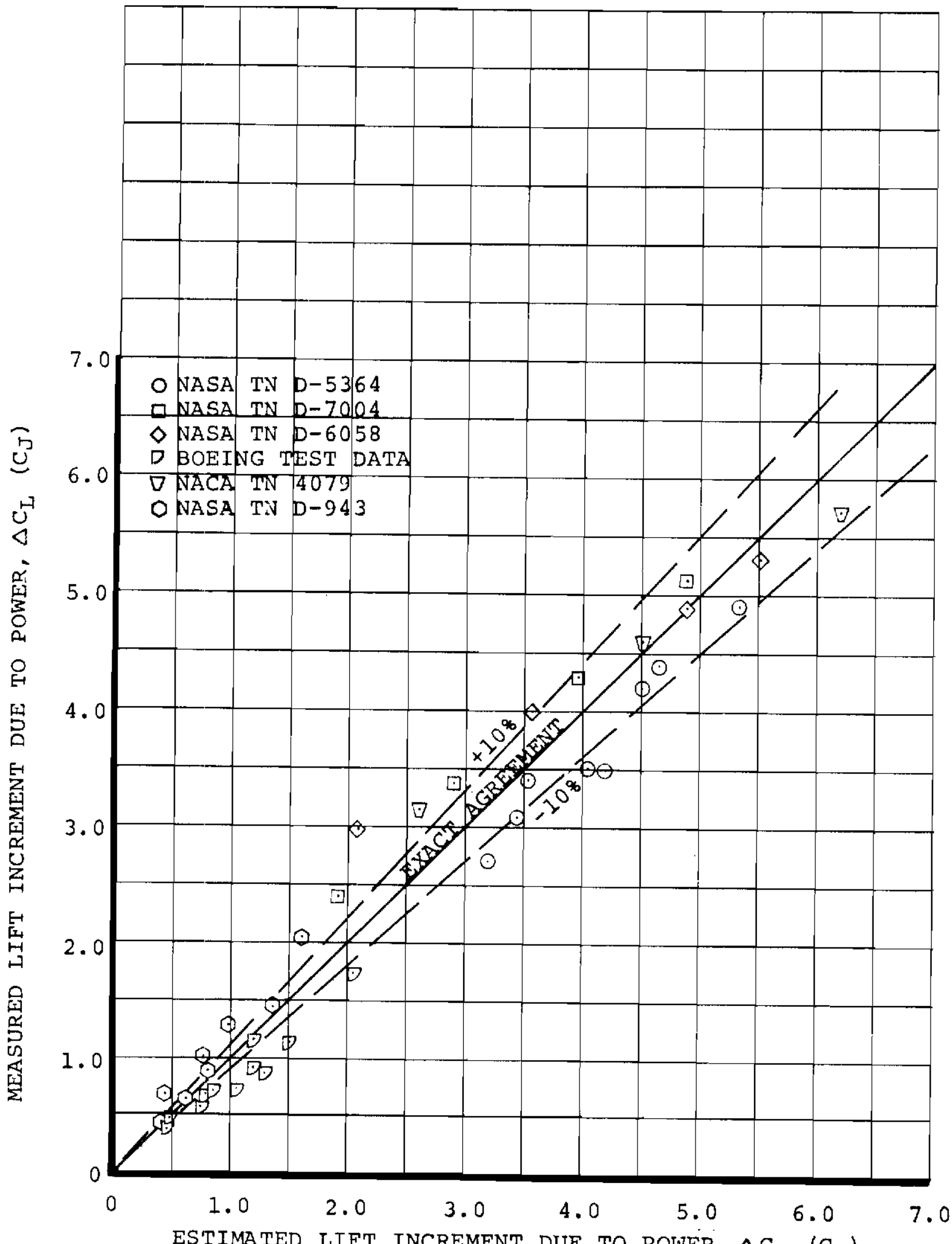


FIGURE 45. CORRELATION OF LIFT INCREMENT DUE TO POWER AT ZERO ANGLE OF ATTACK

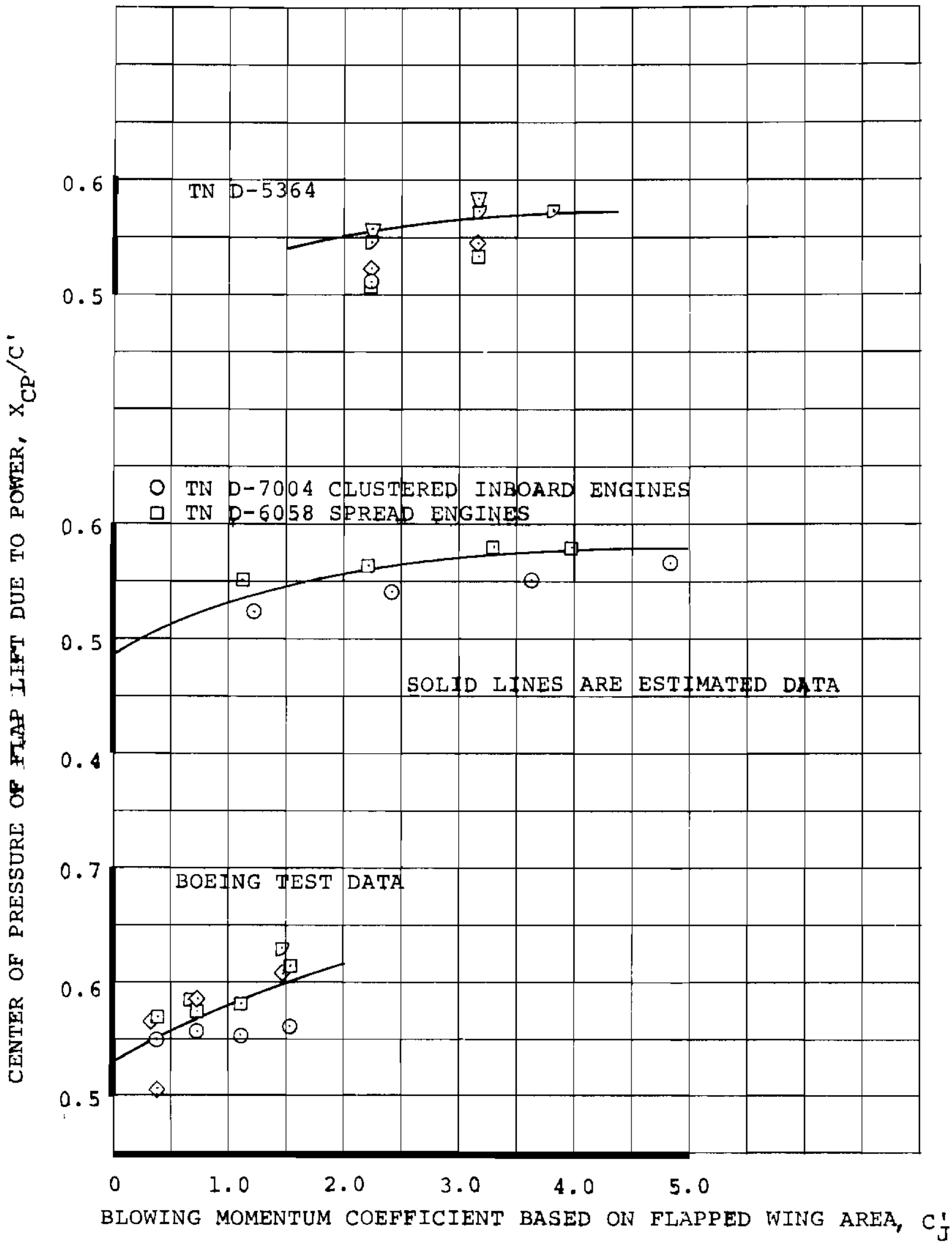


FIGURE 46. POWER INDUCED LIFT CENTER OF PRESSURE CORRELATION

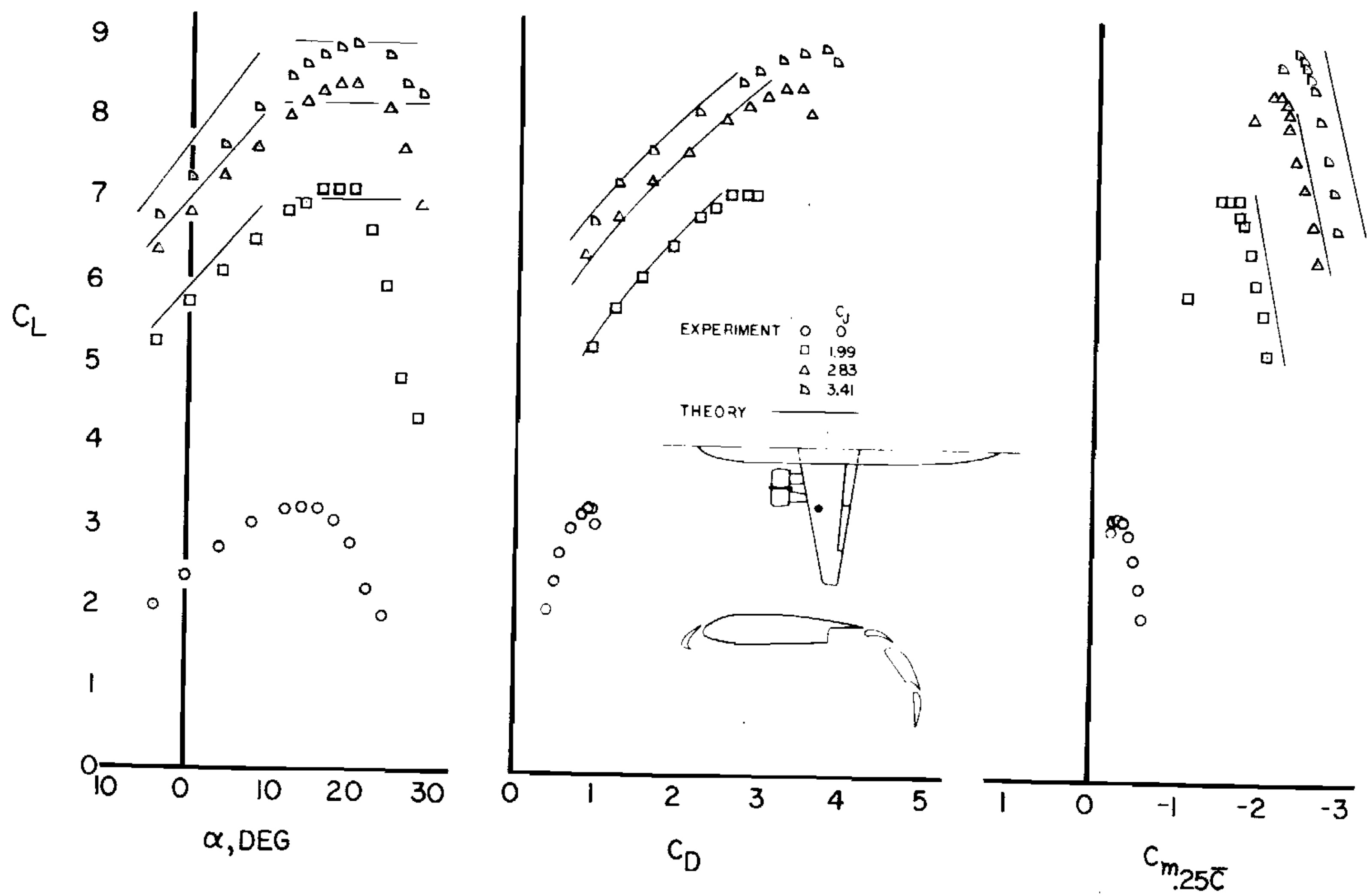


FIGURE 47. THEORETICAL AND TEST DATA ON COEFFICIENT OF LIFT VERSUS ANGLE OF ATTACK, DRAG, AND PITCHING MOMENT AT THE QUARTER-CHORD

4.2 DEFINITION OF DEFICIENCIES AND GAPS IN KNOWLEDGE

4.2.1 Introduction

In the second part of section 4 the deficiencies in the available test and analytical techniques are defined. Programs to fill these voids and to develop improved analytical techniques for externally blown flaps are recommended.

The externally blown flap concept has been tested by the NASA and others. In a recent working paper, the NASA have summarized the objectives of the wind tunnel testing to date and have noted the work they feel needs to be done.

> "Although considerable wind-tunnel research has been conducted on the external-flow jet-flap concept, the objective of most of the work in the past has been mainly to explore the general area of performance and stability and control - with particular reference to problem areas and to finding practical solutions to the problems - such that the overall feasibility of the concept in terms of practical reliable application could be accurately assessed. This research has provided the necessary information to show that the concept is effective for providing high lift on turbofan STOL aircraft but has provided very little information relative to the optimization of the jet-flap parameters involved. Because of the increased interest at the present time in the jet-flap concept, there is now a need for more detailed information for the rational design of such systems. The overall objective of this investigation is to provide the basic design information on the effects of geometric variables such as wing planform, engine location, thrust deflectors, flap span, flap size and type, leading-edge high-lift devices, and horizontal and vertical tail location".

The NASA have established that the externally blown flap is aerodynamically feasible and should be considered a candidate system for a STOL airplane. The risk and cost of such an airplane can be reduced by further rational technology development.

In order to design an airplane using the externally blown flaps, it is necessary to be able to estimate its aerodynamic characteristics with a reasonable degree of accuracy. This requires theoretical, empirical, or semi-empirical estimation techniques substantiated by test data. The gaps and deficiencies in the available test data and analytical methods will be defined.

4.2.2 Test Data

While the NASA have tested a range of representative configurations, a systematic series of tests intended to provide a basis for design of an airplane with externally blown flaps has not been done.

4.2.2.1 Modeling. An externally blown flap wind tunnel model differs from a conventional high-lift model only in the requirement for an engine simulation unit. A variety of thrust units have been used on the existing tests: blowing nacelles with domed inlets, ejector powered nacelles, and rotating machinery. If the engines are located such that the inlet flow has little effect on the wing flow fields, the criteria for a thrust simulation unit must be that the exit conditions be properly simulated; scaled exit geometry and correct exit momentum coefficient. This gives the proper relationship between the jet and the trailing edge flap system. In addition, an exacting calibration of the engine simulation unit must be made.

The available testing, aimed at different test objectives, have not in general used a calibration of sufficiently high standards to obtain accurate design data.

4.2.2.2 Test Conditions. Most of the test data available in the open literature have been obtained at very low dynamic pressures. This has been dictated, in part, by the use of existing thrust simulation units to obtain thrust coefficients representative of high thrust-to-weight aircraft.

The majority of the available data were tested with model-tunnel combinations for which wall correction theory is adequate and flow breakdown should not be a problem. In future testing in other tunnels, model size may be limited by the size of the wall correction that should be allowed. Tunnel flow conditions must also be monitored to assure that the data is not affected by flow breakdown.

4.2.2.3 Ground Effect. There is as of this writing, no published ground effect data for the externally blown flap. This is a serious deficiency, since ground effect will affect takeoff and landing performance as well as stability and control characteristics. The effect of lift level and height above the ground must be assessed. The significance of the ground effect on lift can be seen by looking at jet flap data. Figure 48 indicates that as the jet flap approaches the ground, there is a substantial loss in lift, Reference 17. The figure also shows that care must be taken to provide a realistic ground simulation since there is a substantial difference between fixed board and moving belt data.

Ground effect testing as it is normally done yields static data. Ground effect as it is encountered in flight is a dynamic phenomenon. In order to provide realistic inputs to airplane simulation, it would be advisable to be able to evaluate the aerodynamic lag or $\dot{h}$ derivatives. This will be discussed further in Section 4.3.1.

4.2.2.4 Flow Visualization. Understanding of force test results can be facilitated, insights necessary for theoretical development gained, and configuration improvements defined from the use of flow visualization. This flow visualization can range from simple tuft studies, to smoke, or the use of a separate model in a water tunnel. The value of flow visualization is indicated in Reference 18 where smoke was used in understanding the destabilizing influence of a horizontal tail.

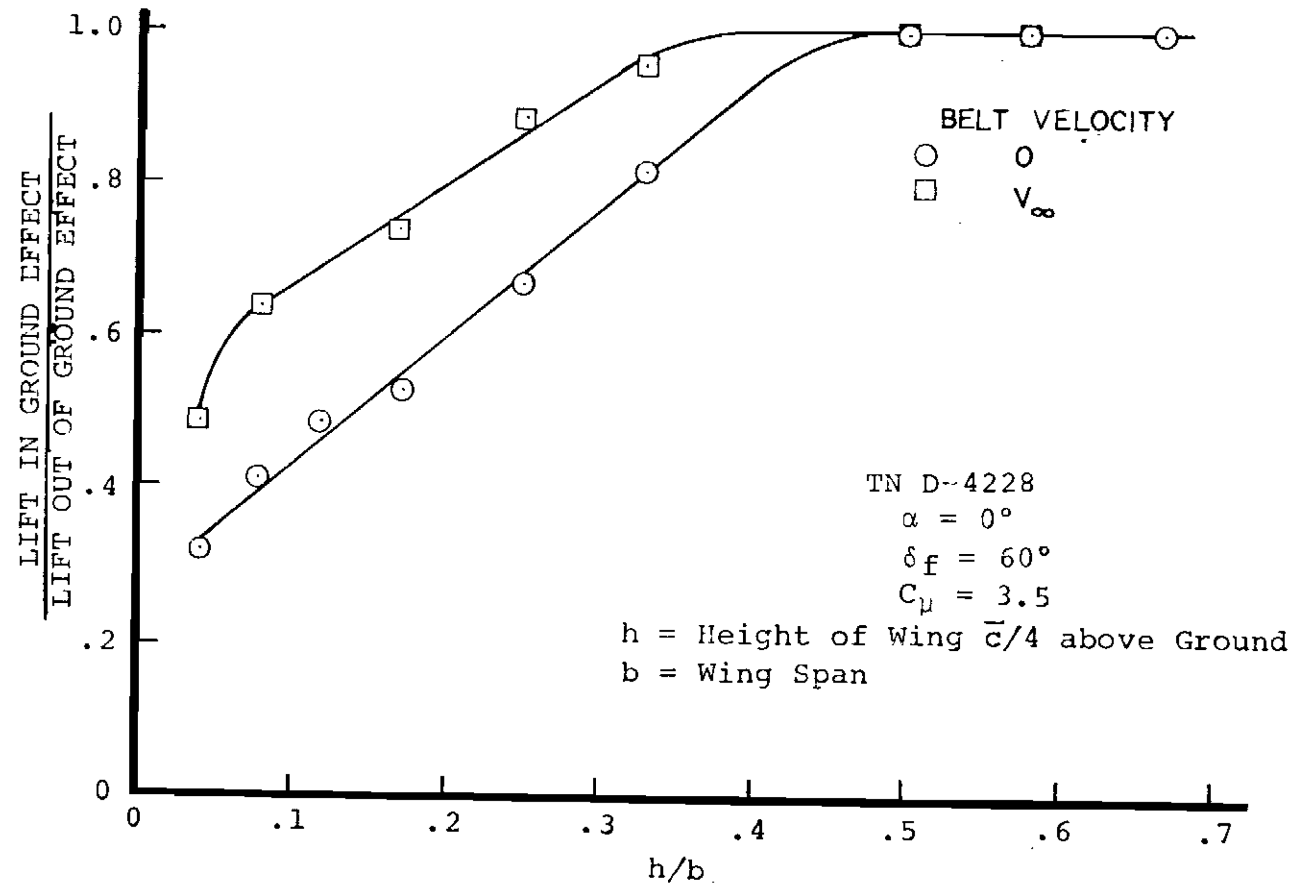


FIGURE 48. GROUND EFFECT OF JET-FLAPPED WING

Very little flow visualization has been accomplished. The flow pattern of the engine exhaust needs to be made visible to help determine the extent of the wing influenced by the propulsion jet. This would also aid in optimizing engine-flap arrangement. Boeing and NASA have used water injection into ejectors very successfully to make visible exhaust jet location.

Water tunnel flow visualization requires a separate model. It is a powerful technique, however, which ONERA has used to understand power and vortex effects. Figure 25 shows a sketch made from a photograph of water tunnel visualization of an externally blown flap in free air.

Flow visualization techniques complement a force test program and allow more intelligent interpretation of the force data.

4.2.2.5 Pressure Data. Unlike the pure jet flap, the extent of wing influenced by the propulsive jet of the externally blown flap is not known a priori. Wing and flap static pressures would be needed to determine the extent and distribution of the induced loading due to power. No pressure data is available in the open literature. The NASA has a limited amount of pressure data that will be available in a future technical note.

4.2.2.6 Parametric Force Testing. The available externally blown flap data contains little parametric test data. In order to develop empirical or semi-empirical estimation methods and to stimulate theoretical development, consistent, high-quality, parametric test data is required. A summary of the force data available with the deficiencies noted is given in Table VII.

4.2.3 Analytical Methods

The deficiencies in the methods developed in 4.1 will be defined. Also, the deficiencies in jet flap theory on which the methods of 4.1 were developed will be considered.

The approach and some of the approximations made in developing the engineering method of this report were dictated by the limited range of data available and the state of development of jet flap theory.

The method developed estimates incremental effects due to power. This assumes that a good method is available for estimating the unpowered characteristics of an unpowered wing. Another problem is that a part of the blowing goes into boundary layer control before excess momentum is available to induce lift through the jet flap effect. Thus, a wing that is badly separated unpowered may show much larger increments due to power but lower overall levels than a wing that has minimum separation. Using data generated from various sources for different purposes, factors such as these cannot be determined.

TABLE VII

SUMMARY OF GAPS AND DEFICIENCIES IN KNOWLEDGE OF EXTERNALLY BLOWN FLAPS

Parameters	Available Data	Gaps and Deficiencies
Trailing Edge Flap		
Flap Type Shape	Existing data are for mechanical flap designs with external blowing.	Best EBF design not necessarily best mechanical design.
No. of Slots	Limited amount for 0, 1, 2, 3 slots.	Effect of number and location of slots not defined.
Chord Split	No data.	
Flap Deflection	Limited deflections on a given configuration.	Need sufficient data to optimize EBF systems. Very high deflections, $\delta_f \geq 90°$ are lacking.
Flap Gap + Overlap	No data.	Need to establish criteria for EBF.
Flap Chord	No consistent data.	Flap chord would be expected to have a significant effect on the thrust vectoring performance.
Flap Span	Two tests with span variations $\eta = .7$ and 1.0; $\eta = .44$ and .89	Insufficient data to determine span effects. Engine location must also be considered. Location of the trailing vortex sheet. Downwash.

TABLE VII - Continued

Parameters	Available Data	Caps and Deficiencies
<u>Leading Edge Device</u>		
Mechanical or BLC	Available data is very limited on LE effects.	Leading edge device must be designed for the peak pressures expected. Tested leading edges probably not adequate above $C_{Lmax} = 2.5\text{-}3.0$.
Shape	Available data is very limited on LE effects.	
Deflection		
Chord		
Blowing Quantity + Location	One test.	Leading edge BLC needs to be explored with leading edge designed specifically for blowing.
<u>Wing</u>		
Aspect Ratio	No parametric data — isolated points at AR = 6, 6.5, 7, 7.75. Sweep not constant.	Consistent, parametric aspect ratio effect not available. AR affects jet flap lift, induced drag.
Sweep	No parametric data — isolated points at Λ = 0, 8, 25, 30, 25 AR varying	Sweep effects on C_{Lmax}, pitching moment, engine-out rolling moment undefined.

TABLE VII - Concluded

Parameters	Available Data	Gaps and Deficiencies
<u>Engine Nacelle</u>		
Location	No parametric variation.	Effect of spanwise, chordwise and height variation needed on lift, pitching moment, force polars, trailing edge vortex sheet location, downwash, engine location on engine-out stall characteristics, rolling moment and sideslip derivatives.
Orientation	No data.	Spanwise and vertical angular variation.
Single vs Double Pods	Limited.	Need more data to establish trades.
Bypass Ratio	Insufficient data.	As bypass ratio and T/W increase ratio of flap chord to jet diameter decreases.
Exhaust Jet Shape	Insufficient data.	Jet deflectors, rectangular nozzles, etc.
Lateral Control System	Limited.	Insufficient flaperon, blown aileron, etc.

The method employs an effective jet deflection angle and static tuning efficiency. How to ensure good angles and efficiencies was not determined. Obviously, such parameters as engine orientation relative to the flap system, exhaust jet shape, the relationship between flap chord and engine diameter, and engine bypass ratio may have an effect on both the turning losses and the angle to which the flap system effectively turns the jet.

Insufficient data was available to develop a method to account for the effects of flap span and blowing location on the aerodynamic characteristics. Fortunately, the data indicated that the incremental effects on lift were only weakly dependent on engine location. A force polar method for the two extremes of spread engines and concentrated inboard loading only was developed. The pitching moments on swept wings also depended on engine locations since the loading distribution is important. Methods more accurately treating flap span and engine location must depend on acquiring data giving detailed load distribution on the wing.

Methods for estimating lateral-directional characteristics were not considered. These are extremely important and further work is indicated. Also, ground effect must be considered.

4.3 RECOMMENDED PROGRAMS

An indication will be given of the program required to develop an estimation and evaluation base for configurations using externally blown flaps.

4.3.1 Wind Tunnel Test Programs

To avoid a proliferation of unrelated and uncorrelatable test data, a consistent, parametric wind tunnel test program is required to eliminate the deficiencies noted in Table VII. This would increase confidence in detailed design of an externally blown flap configuration. Any wind tunnel program should include, in addition to basic free-air force data, flow visualization, flow field investigations, wing and flap pressure distributions, and ground effect testing.

As discussed previously, ground effect as encountered by an aircraft is a dynamic process. Aerodynamic lag may alleviate the adverse effects on lift at high lift coefficients. It would be desirable to measure aerodynamic lag or (h/c) derivatives. This can be done in several ways. NASA has used the tow tank at Langley with the model on a carriage moved over a ground board simulating several approach angles. The Princeton Dynamic Track could also be used with the carriage programmed to simulate an approach and flare. A conventional wind tunnel with a sting mounted model and moving ground belt and a data system equipped to take dynamic data could also be used by placing the model at a given angle of attack and traversing the model from free air toward the ground plane while measuring a time history of the forces on the model.

4.3.2 Analytical Development

As additional test data becomes available the engineering method can be extended. Since it relies on jet flap theory some additional jet flap theoretical work is indicated. The effect of part span flaps, part span blowing, and high flap deflections need to be theoretically evaluated. Theories to adequately estimate the flow fields generated by jet-flapped wings must be evaluated.

In addition, tools for evaluating complete configurations using jet-flapped and externally-blown-flapped wings need to be developed. The possibility of extending existing general inviscid three-dimensional computer programs should be investigated. These programs are potentially powerful tools to understand powered lift configurations. Since the basic programs have already been developed, the intention would be to try to improve the simulation of engine operation and its interactions with the rest of the aircraft. The process will be difficult and is not assured of success. The risk involved is high but the potential payoff is higher.

4.4 CONCLUSIONS

The feasibility of the externally blown flap concept has been determined from wind tunnel testing. The testing has not, however, been intended to generate design information so that there are a number of deficiencies in the data. These deficiencies can be eliminated by a parametric wind tunnel test program. Concurrent analytical development spurred by the accumulation of test data would improve the present very limited design and evaluation confidence level.

5. THE DEFLECTED SLIPSTREAM CONCEPT

5.1 INTRODUCTION

The feasibility and practicality of the deflected slipstream concept has been clearly demonstrated by model test and by flight hardware both in flight test and simulated operations.

The development of the deflected slipstream type of configuration has been a fairly gradual process based on the wing-propeller system of early aircraft. The factors that have made such a concept suitable for STOL operation are the development of the turboshaft engine and the high thrust levels available from modern propellers. Additional refinements such as cross-shafting, boundary layer control, and flaps with high turning effectiveness add to the possibilities of the full exploitation of this concept.

Large amounts of model test data have been obtained since the early nineteen fifties. In recent years, interest in the tilt-wing concept has provided considerable amounts of data applicable to the deflected slipstream configuration. Most of this data was obtained during tests designed for project evaluation or demonstration of the feasibility of the concept. Because of the lack of a systematic variation of parameters, it is difficult to apply much of this data to the development of empirical methods (see, for example, Reference 19).

Flight test data have been obtained on a number of configurations. Of particular interest are the Breguet 940 and 941. The latter first flew in 1961 and has since been tested by the FAA and NASA and has been flown in simulated operations by the commercial airlines. Other configurations that have been flight-tested include the Shin Meiwa STOL Seaplane, the Convair Charger, the Stroukoff YC-134A and the Ryan 92 (VZ3-RY). The last named configuration, though, was a V/STOL rather than a STOL aircraft and the emphasis on the VTOL performance compromised the STOL considerations.

Tables VIII and IX show a brief summary of model test data and flight test data respectively. The quoted reference numbers refer to the Bibliography, Volume II.

A number of analytical and empirical methods have been developed for the prediction of the aerodynamic characteristics of deflected slipstream configurations. None have been found, however, that can satisfactorily predict all the necessary information required to assess a given configuration.

TABLE VIII

DEFLECTED SLIPSTREAM MODEL TEST DATA

Reference	Configuration	Areas of Investigation			Comments
		Lift, Drag Pitch Mom.	Stability & Control	Prop Data	
II. 2. 1. 1	Wing in Jet	X	-	-	Wing had b. l. c.
II. 2. 1. 3	Prop-wing-flap, body	X	Static	None	Measured wing root loads, pressure in slipstream.
II. 2. 1. 4	Prop-wing-flap, body			None	
II. 2. 1. 5	Prop-wing-flap	X		Torq. RPM	Static test
II. 2. 1. 6	Model like C-123	X			Wing with b. l. c.
II. 2. 1. 7	As above but 4 props	X		Approx	Measured downwash at tail & spanwise loading
II. 2. 1. 8	Prop-wing-flap	X		Thrust	Static test
II. 2. 1. 9	Prop-wing-flap	X			Uncorrected data

TABLE VIII - Continued

Reference	Configuration	Areas of Investigation: Lift, Drag Pitch Mom.	Areas of Investigation: Stability & Control	Areas of Investigation: Prop Data	Comments
II. 2. 1. 10	Wing in prop slipstream	X	Center of pressure		Measured q and down-wash in slipstream
II. 2. 1. 11	4-prop model	X			Mobile test rig, b. l. c. on wing
II. 2. 1. 12	Wing in free jet	X			B. l. c. on wing
II. 2. 1. 13	4-prop model	X	Longitudinal	Approx	Three different values of AR .
II. 2. 1. 15	Prop-wing-flap	X		X	Static test, various props
II. 2. 1. 16	Prop-wing-flap	X		X	Static test
II. 2. 1. 17	Prop-wing-flap	X		X	Static test
II. 2. 1. 18	Various				Review paper
II. 2. 1. 19	VZ3	X	Longitudinal & lateral	Approx	Full-scale airplane in wind tunnel
II. 2. 1. 20	Wing-prop			X	Pressure measure-ments

TABLE VIII - Continued

Reference	Configuration	Areas of Investigation: Lift, Drag Pitch Mom.	Stability & Control	Prop Data	Comments
II. 2. 1. 21	As above				Plotted data from above
II. 2. 1. 22	2-prop-wing-flap	X		X	Wing with jet flap pressure measurements
II. 2. 1. 23	2-prop-wing-flap	X		X	
II. 2. 1. 24	2-prop-wing-flap	X		X	Various prop locations
II. 2. 1. 26		X			Not reviewed
II. 2. 1. 27	4-prop model	X			Mobile test rig (Part II of II. 2. 11)
II. 2. 1. 29	Prop-wing-flap	X		X	Static tests, varied prop locations
II. 2. 1. 30	Model VX5-FA	X	Longitudinal	Approx	
II. 2. 1. 31	2-prop-wing flap	X		T, N_p, M_p	Includes some tests with one prop

TABLE VIII - Continued

Reference	Configuration	Areas of Investigation			Comments
		Lift, Drag Pitch Mom.	Stability & Control	Prop Data	
II. 2. 1. 32	2-prop-model	X	Longitudinal & lateral	X	Inverted V-tail
II. 2. 1. 35	Wing in jet				Not reviewed
II. 2. 1. 37	Prop-wing-jet flap	X			Not reviewed
II. 2. 1. 38	Wing in jet				Review of data
II. 2. 1. 39					Static; effect of prop location
II. 2. 1. 40	Ducted prop-wing-flap	X		X	
II. 2. 1. 41	Prop-wing-flap	X		X	
II. 2. 1. 42	6-prop model	X			Pressure distribution on wing
II. 2. 1. 43	6-prop model	X			Static tests, some in ground effect

TABLE VIII - Concluded

Reference	Configuration	Area of Investigation			Comments
		Lift, Drag Pitch Mom.	Stability & Control	Prop Data	
II. 2. 1. 44	Prop-wing-flap	X			Static test
II. 2. 1. 45	4-prop wing	X	Lateral		
II. 2. 1. 46	6-prop model				Static test, pressure distribution
II. 2. 1. 52	Various	X		X	Surveys of prop slipstream pressure distribution on wing.
II. 2. 3. 1	Model VZ5-FA	X	Lateral	Approx	
II. 2. 3. 2	6-prop model	X	X	X	

TABLE IX

SUMMARY OF FLIGHT TEST DATA

Reference	Airplane	Area of Investigation							Comments
		Lift, Drag	Trim	Opera. Envir.	Other Perf.	Long. Stab.	Lat. Stab.	Hand. Qual.	
II. 2. 1. 2	VZ3-RY			X	TO, L	X	X		Poor lateral-directional control in approach config.
II. 2. 1. 33	UF-XS			X		X	X		Low static longitudinal stability, unstable spiral mode, low directional control in approach.
II. 2. 1. 25	STOL Research airplane	X			L	X			Modified D. H. Otter
II. 2. 1. 28	Br 941	X			TO, L				Review paper.
II. 2. 1. 34	Br 941				TO, L	X	X		Review paper.
II. 2. 1. 36	Br 941								
II. 2. 4. 1	Br 941	X	X	X	TO, L	X		X	Neutral longitudinal stability at approach
II. 2. 5. 1	UF-XS	X		X			X	X	

TABLE IX - Concluded

Reference	Airplane	Area of Investigation							Comments
		Lift, Drag	Trim	Opera. Envir.	Other Perf.	Long. Stab.	Lat. Stab.	Hand. Qual.	
II. 2. 6. 1	VZ-3-RY				GE		X	X	
II. 2. 1. 47	NC-130				TO, L			X	Review paper.
II. 2. 1. 14	Convair Coin	X		X	TO, L			X	
II. 2. 1. 48	YC-134A	X	X			X	X	X	Poor directional control at low speeds
II. 2. 1. 49	NC 130	X	X	X	TO, L	X	X	X	
II. 2. 1. 50	Br 941			X		X	X	X	Stability Augmentation Test
II. 2. 1. 51	VZ3		X						
II. 2. 2. 1	Various								Downwash at tail

5.2 REVIEW OF PREDICTION METHODS

5.2.1 General Considerations

The available techniques for predicting the aerodynamic characteristics of deflected slipstream configurations fall between two extreme approaches.

One extreme is to represent the configuration by a purely mathematical model of the flow processes involved and solve the resulting equations either analytically or numerically. In principle, this leads to an evaluation of pressure and velocity distributions everywhere and from that aspect is most desirable. However, a comprehensive mathematical model of the real flow is difficult to solve analytically and requires large amounts of digital computer time to obtain numerical solutions. In addition, it is not clear that a purely potential flow model will adequately represent the real fluid effects.

The other extreme is the empirical approach based on the evaluation of test data and guided by consideration of the physical phenomena involved and by the simpler theories. However, lack of systematic test data hampers the development of such methods.

Most techniques combine the two approaches.

Traditionally, the approach to the evaluation of the characteristics of any aerodynamic configuration has been to first evaluate the characteristics of all the major components in isolation. Then the effect of adding each additional component to the major one (usually the wing) is computed until the whole configuration has been "assembled." In the case of propeller-wing aerodynamic characteristics, it is necessary to add another term of major importance - representing the mutual interaction between wing and propeller slipstream - including the induced circulation lift on the wing due to the prop and the inflow distortion to the prop due to the airplane structure.

In view of the above, the following contains a brief discussion of the characteristics of the isolated components of a deflected slipstream configuration, mainly propellers and wing-flap systems and a discussion of the combined propeller-wing-flap system.

The major components discussed are the propeller (including spinner, hub, blade shanks and cuffs)and the wing-flap system.

Other components that affect the aerodynamics are the nacelles (including intake and exhaust apertures and jet), the fuselage, the tail, etc. These items are discussed below.

5.2.2 Propeller Methods

It is not intended to discuss the state of the art of propeller aerodynamic technology but rather to indicate what slipstream predictions are available to the STOL aerodynamicist who needs to evaluate the aerodynamic characteristics of a propeller-wing-flap system.

The prediction of propeller aerodynamic characteristics has been under consideration for a very long period with a considerable degree of success. In the last two decades the interest in VTOL and STOL applications has brought to light shortcomings in the accepted prediction methods. Traditionally, the propeller has been used in axial flow conditions and the effects of small deviations from axial flow were easily estimated to a sufficient degree of accuracy. In STOL applications, propellers experience very high disc loadings and can be subject to large deviations from the axial flow condition. Under these conditions the prediction of forces, moments and flow fields is less accurate than is necessary to give reliable performance predictions for STOL configurations. Because of the high disc loadings associated with deflected slipstream configurations, the flow in the neighborhood of the propeller is subject to very high pressure and velocity gradients.

The large deviations from axial flow that can occur due to, for example, gusts of the same order of magnitude as the flight speed, give rise to in-plane forces and moments that are no longer negligible. Even at fairly small angles of sideslip or incidence, when the in-plane force may be small compared with the axial thrust of the propeller, the induced velocity due to the in-plane force can cause a significant change in the direction of local total velocity vectors in the slipstream.

In view of the importance of the flow field behavior of the propeller, the following propeller related effects should be taken into consideration when analyzing a deflected slipstream configuration:

(i) Thrust, side force and hub moment on the propeller in an originally uniform, non-axial flow

(ii) Changes in thrust, side force and hub moment caused by changes in the uniformity of flow due to

  (a) wing-flap system upwash field

  (b) nacelle-interference

  (c) interference due to other airframe components

(iii) Velocity and pressure distribution of the isolated propeller's slipstream. This includes

(a) slipstream rotation, axial and radial velocities

(b) periodicity and unsteadiness of velocities and pressures

(iv) Interference effects on the slipstream caused by Items (ii) (a), (b) and (c) above.

Various mathematical models of the propeller and slipstream have been proposed.

5.2.2.1 Actuator Disc. The actuator disc is a mathematical device replacing the propeller that gives an impulse to the stream tube flowing through the propeller disc. The simplest form assumes that in that streamtube there is no rotational or radial velocity component anywhere in the field of flow and that the axial component is uniform at any given axial distance from the propeller. With this model the slipstream axial velocity can be related to the specified thrust of a propeller of given size.

This model of the slipstream has been used in a number of prediction methods and the resulting definition of slipstream dynamic pressure has been widely adopted for the purpose of non-dimensionalizing test data. A convenience stemming from the use of slipstream dynamic pressure is the fact that coefficients based on it do not approach infinity as flight speed approaches zero.

5.2.2.2 Inclined Actuator Disc. This approach as used in Reference 20 is the same as the above except that the actuator disc affects only that component of the freestream that is normal to the disc. Depending upon the choice of assumptions about the nature of the flow, two slightly different values of slipstream dynamic pressure and induced velocity are obtained.

5.2.2.3 Modified Actuator Disc. Glauert in "Aerodynamic Theory" (Ed. Durand) outlines two modifications to the 'uniform' actuator disc, allowing radial variations of axial velocity and introducing a rotational velocity term that is obtained by relating the propeller torque to change of angular momentum of the slipstream.

Reference 21 proposed two basic modifications:

A solid body rotation of the slipstream was assumed, the angular velocity being a function of propeller speed and attitude and thrust and torque levels. The angular velocity was expressed in terms of the propeller torque and thrust coefficients and advance ratio.

Variations in the axial velocity of the slipstream are represented by constructing the slipstream of a number of concentric annular streamtubes having different values of the velocity potential and therefore axial velocity. The values of some of the axial velocities must be specified and only a study of real slipstreams will indicate their relative magnitudes.

Figure 45 of Reference 21 gives an indication of why such a process is necessary. The figure shows the variation along a radius of the disc of the dynamic pressure in the slipstream of a real propeller in the static thrust condition, at two different blade settings.

5.2.2.4 Theodorsen's Wake Model. This was an attempt at simulating the real flow in the wake while still retaining a potential flow model. The wake was represented by a helical sheet of vorticity shed from each propeller blade. The sheets were assumed to the rigid and rotating with the blades. The flow field was obtained by solution of the potential flow equations.

Such a model gave fairly good results for lightly loaded propellers. For highly loaded propellers the distortion of the nearby portion of the wake is an important factor in determining the flow conditions in the neighborhood of the propeller.

5.2.2.5 Vortex Wake Representation. Various models have been proposed that use a discrete vortex representation of both propeller blades and the wake. Such methods are amenable to numerical solution by digital computer.

Two basic approaches are used, both of which represent the propeller blade with a lifting line trailing a finite number of discrete vortex lines.

The first approach, called the free wake technique, is to allow the wake to take up the shape imposed by its own induced velocity field.

The second, called the fixed wake analysis, forces the wake to conform to a shape predetermined by empirical methods.

The second approach is known to provide accurate predictions of propeller performance in forward flight conditions but in the very low speed range accuracy decreases.

No attempts at the correlation of test data with a free vortex model have been discovered during this study.

A refinement sometimes used in the above models is the avoidance of singularities by using vortices with viscous cores rather than potential line vortices.

The latter type of models offer the best promise of providing accurate predictions of forces and flow fields since they have the scope of relating slipstream properties to blade geometry and of being expanded to include factors heretofore omitted. Such factors as spinner and nacelle interference should probably be included. One factor common to all of the above models is that the results given are a time average of the flow conditions and not instantaneous values of the flow conditions.

5.2.2.6 Empirical Methods. Numerous approximate methods are available for the prediction of propeller forces and moments in uniform flow fields. The literature search has not revealed any empirical method for predicting the slipstream properties.

5.2.3 Wing-Flap System Methods

In general, analytical methods for the prediction of aerodynamic characteristics of wing-flap systems in uniform flow, are inadequate for all but "rough cut" work. The usual technique for evaluating the aerodynamics of a wing-flap system is to evaluate the characteristics of the unflapped wing and then evaluate the changes in the wing characteristics due to the addition of the particular flap configuration under consideration.

Frequently the first step and almost invariably the second are empirical approaches such as are outlined in USAF DATCOM or RAES Data Sheets. These lead to evaluation of forces and moments and simple methods can be used to evaluate the downwash field at locations not too close to the wing-flap system. These methods are not very accurate and are used only in the absence of test data on the configuration under investigation or on a similar configuration. The most reliable procedure is to use test data from a configuration similar to or closely related to the configuration being studied. Empirical trends and theoretical predictions are used to account for small differences in configuration. However, such a technique introduces the necessity of also taking consideration of differences between test and real conditions, wall effects, scale effects in addition to differences of geometry.

The following wing-flap system methods can be used.

5.2.3.1 Horseshoe Vortex. This is the most rudimentary model and as such is probably the least accurate. The horseshoe vortex model is capable of predicting downwash fields only near a streamwise line through the center of the span of high aspect ratio wings.

5.2.3.2 Lifting Line. In this technique the wing is again represented by a line vortex. But instead of the two trailing vortices of the horseshoe vortex model there is a flat vortex sheet extending downstream from the lifting line. This model is better than the horsehsoe vortex but is still limited to high aspect ratio wings. The flat wake assumption is a restriction to fairly low values of lift coefficient for at high lift coefficients the wing wake is highly curved so that the downwash field is not accurately modelled. Thus this approach is effectively limited to wings of high aspect ratio and with no high lift devices.

5.2.3.2 Lifting Surface. This approach represents the attempt to simulate the downwash field in the vicinity of the wing. The technique uses a 'mesh' of horseshoe vortices located at the wing. The strength of the vortices is determined by 'tailoring' the local downwash velocity at each of a number of strategic points on the wing surface to the slope of the wing surface there. If a sufficiently large number of 'node points' is used presumably a good simulation of the wing-flap could be obtained. Unfortunately, the amount of calculation required increases with the number of node points specified so that usually a compromise between accuracy and computation time must be made.

5.2.3.4 Vortex Lattice. This approach is similar to the lifting surface technique but is more ambitious and not limited to bodies of zero thickness. Here, the vortex lattice is used to represent a complete surface (upper and lower surface of a wing of finite thickness, for example) and local velocities are constrained to follow the surface at node points.

This method has the prospect of being able to simulate almost any potential flow but is subject to the same compromise between accuracy and computation time as the lifting surface method.

For example, the vortex lattice simulation could be used for a slotted flap configuration whereas the lifting surface theory would not be able to predict the significant changes due to small variations in the slot configuration.

5.2.3.5 Empirical Methods. A variety of empirical methods is available - mostly of a 'building-block' nature. The usual procedure is to calculate each of the following items in turn, starting with the unflapped configuration:

Airfoil section characteristics:

lift curve slope

angle of zero lift

maximum lift coefficient

aerodynamic center

profile drag, pitching moment coefficient at zero lift

Wing planform characteristics:

airfoil characteristics subjected to the effects of finite aspect ratio, taper, sweepback, twist; induced drag

Wing-flap characteristics:

wing planform characteristics subjected to the effects of flap deflection, span and chord, extension, gap, etc.

5.2.4 Effects of Nacelles and Fuselage

Analytical methods for the effect of nacelles and bodies on wings exist and are (in principle) reasonably simple to apply (slender body theory). However, empirical charts based on wind tunnel data are readily available and simple to use, employing only a few parameters such as wing incidence, body length and width.

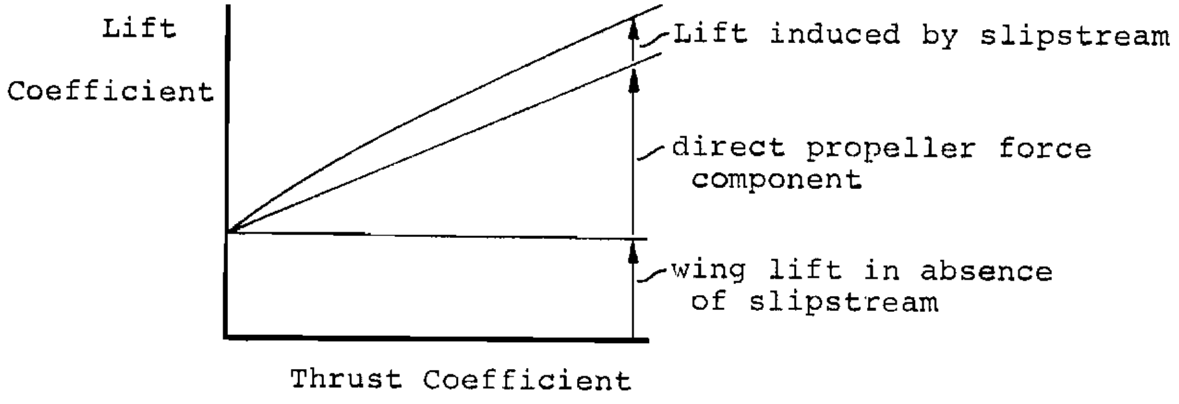
Nacelles for turboshaft engines need special consideration because they lie in a slipstream, and the location of intake and exhaust apertures has an influence on the pitching moment of the system.

5.2.5 Combined Propeller-Wing-Flap System

The prediction of the characteristics of this system depends on the evaluation of propeller characteristics, wing-flap system characteristics, mutual interference, and the influence of the propeller-wing-flap system on the rest of the airframe and the flow field of the combination.

The approach can be made in two ways. First, the isolated characteristics could be evaluated and then changes due to mutual interference between the two systems could be calculated. Secondly, a mathematical model of the combination could be postulated and used to evaluate its total characteristics.

5.2.5.1 Lift. The high lift of this combination can conveniently be viewed as the sum of three major components, as illustrated:



The following observations can be made about the three components:

Lift of the wing out of influence of the propeller has already been discussed in 5.2.3.

The direct propeller force component is not necessarily the same propeller force as would be obtained by the propeller (or propellers) in isolation at the specified attitude and freestream condition. A major factor influencing the propeller is the upwash field ahead of the wing due to the high value of the circulation at the high lift condition. This means that in general there will be a non-axial and non-uniform flow incident at the propeller. This in turn introduces a propeller in-plane force. The net direct propeller force then is composed of both thrust and in-plane components. Some authors include in the 'direct thrust' term the lift due to the turning of the propeller slipstream by the wing-flap system. This is the same as assuming that the thrust vector is rotated by the action of the wing-flap system, and the amount of rotation (called "static turning effectiveness") is usually an empirical value obtained from tests at zero forward speed. The assumption that the turning effectiveness at forward speed is the same as at static conditions is suspect. The drag of the wing in the slipstream produces a loss of momentum of the slipstream so that the effect of turning the slipstream is less than might be predicted by an amount depending on the deflection. Because of the upwash field of the wing it is seen that the direct propeller force component of lift depends on the wing-flap system to some extent even if the lift gained by turning the slipstream is not included as part of the direct force term.

The lift induced by the slipstream is due to a number of different effects, the main ones being

(a) increased dynamic pressure in the slipstream,

(b) change of local angle of attack of that portion of the wing within the slipstream. This is caused by the increased mean velocity vector, the spanwise variation in downwash resulting from rotation of the propeller slipstream and the spanwise variation of axial induced velocity resulting from non-uniform blade loading, and

(c) the deflection of the propeller slipstream by the wing-flap system.

Other factors that can affect the lift induced by the slipstream include

(d) unsteady and/or periodic fluctuations of velocity components in the slipstream. Test data indicates that this effect is negligible except perhaps at high incidence. It is a well known phenomenon that cyclic pitching or heaving motions of an airfoil result in changes of maximum C_L, and that the magnitude of the change depends upon the frequency of the motion. This may explain the difference between the lift curves obtained without propellers and those obtained with propellers producing zero net thrust.

(e) spanwise spreading (and contraction) of the slipstream effect due to spanwise pressure variations near the edges of the immersed portions of the wing,

(f) increase of local Reynolds Number and Mach Number within the slipstream,

(g) effects of regions of velocity shear, and

(h) entrainment of the freestream by the slipstream.

5.2.5.2 Drag. The drag can also be thought of as being composed of three major components.

Drag of the airframe in the absence of the slipstream can be calculated by standard methods as discussed in 5.2.3.

The direct propeller component of drag is subject to the same remarks as the propeller component of lift.

The slipstream induced drag is subject to the same considerations as the corresponding lift term. In addition, it should be noted that the spanwise variation of lift directly affects the drag; due to the large spanwise variations in lift, relatively high values of induced drag are likely to occur.

In addition to the above, there are significant contributions to the net drag from the ram drag and jet efflux of the nacelles that are normally placed behind the propellers in the slipstream.

5.2.5.3 Pitching Moment. Pitching moment of the propeller-wing-flap system can be analyzed into three components. They are:

The isolated wing-flap pitching moment.

The direct propeller term. This includes moments arising from thrust, in-plane force and hub moment. Also, the ram drag and jet efflux of the engines will cause moments of magnitude dependent on the location of nacelles with respect to the reference point.

The induced moment due to the effect of the slipstreams on the wing. It is to be expected (and is shown by test data to be so) that a large nose down pitching moment is the penalty paid for turning the slipstreams through large downward angles by means of the flaps. The necessary download required at the tail to trim these moments may considerably reduce the gain in lift obtained by turning the slipstream.

5.2.5.4 Downwash. Because of the complex flow resulting from the interaction of the propeller slipstreams with the wing-flap system and the high deflection of the airflow by the flaps the downwash field cannot be accurately calculated by the simple horseshoe vortex or flat vortex sheet models that are suitable for low lift levels.

5.2.5.5 Lateral-Directional Characteristics. The aspects of lateral-directional aerodynamics that are of primary interest are control response, sideslip and yaw characteristics and engine (or propeller) failure. It is probable that the forces resulting from lateral control deflections and those due to engine failure could be evaluated by consideration of the local changes of lift, drag, and moments on the wing. The sideslip and yaw dependent forces will be due in part in the in-plane forces and hub moments of the propellers; other components will be as for the isolated airframe unless the sideslip is so great that propeller slipstreams will impinge upon the fuselage or vertical tail.

5.2.5.6 Stability and Control. The necessity of trimming the large nose down pitching moments due to the use of flaps has already been noted. Calculation of the required restoring moment provided by the horizontal tail depends upon a detailed knowledge of the local dynamic pressure and the downwash in the vicinity of the tail. The changes in trim due to flap deflection and due to power variation can be large because of the fact that the tail lies close to and sometimes within the propeller slipstreams and therefore is subject to large and rapid variations of downwash and dynamic pressure.

The tail provides the nose up restoring moment by means of a download which reduces the total lift of the aircraft significantly at high flap deflections. Prediction of this effect is impossible without a thorough knowledge of the details of the flow field in the vicinity of the tail and the variations in the flow field due to geometric and power changes.

No special methods for predicting dynamic stability have been discovered relating to deflected slipstream configurations and in the STOL operation doubt has been cast on classical dynamic stability analysis. Because of the low flight speeds involved in STOL operations the assumptions of small disturbances and linear equations are possibly not valid and may even be misleading. Also, since the takeoff and landing operations are acceleration maneuvers, the assumption of an initial steady state may also lead to erroneous stability predictions. Time lag terms that result from the finite time required for slipstream to the tail are of importance since the moment provided by the tail may be very sensitive to changes of slipstream properties and location.

5.3 A DISCUSSION OF SELECTED METHODS FOR THE DEFLECTED SLIPSTREAM CONCEPT

5.3.1 General

Section I.2 of the Bibliography (Volume II of this report) contains 38 references that include prediction methods or that include discussions of the problems associated with the development of methods.

Of these references that include prediction methods, a representative sample has been reviewed below. The methods considered include both the purely analytical and empirical types and the table below indicates the nature of each of the reviewed methods.

The methods listed in Table X are reviewed in detail in the order shown.

TABLE X

PREDICTION METHODS FOR THE DEFLECTED SLIPSTREAM CONCEPT

Reference	Description
"A Preliminary Theoretical Investigation of the Effects of Propeller Slipstream on Wing Lift", Douglas Rep. SM 14991, E.W. Graham et al (Reference 22)	Combines actuator disc representation of prop with lifting line or slender wing theories. Also discusses lifting surface theory.
Series of reports on wings extending through slipstreams by Vehicle Research Corporation V.R.C. Reports 1, 8, 9, 9a, 10. and "Aerodynamics of non-uniform flows as related to an airfoil extending through a circular jet". J. Aero. Sci. Vol. 25 No. 1 S., Rethorst (References 23 through 27 and 39)	Combine uniform jet with Weissinger lifting surface theory to consider lift of wings with multiple jets, inclined jets, at high angles of attack and in separated flow conditions. Only theory is included, no comparison with data except in Reference 39.
"Semiempirical procedure for estimating lift and drag characteristics of propeller-wing-flap configurations for vertical-and-short-take-off-and-landing airplanes". NACA Memo 1-16-59L R.E. Kuhn (Reference 28)	Uses actuator disc theory to calculate slipstream properties. Evaluates lift and drag by estimating change of momentum of prop, slipstream and 'wing stream-tube'. Not limited to unflapped wings.

TABLE X - Concluded

Reference	Description
"A stability analysis of tilt-wing aircraft (analytical)". Princeton U. Rep. 477, C.H. Cromwell and H.E. Payne (Reference 29).	Inclined actuator disc theory is used to calculate local flow conditions at wing. Mass flow correction used to account for the fact that only part of the wing may be inside the slipstream.
"Effects of propeller slipstream on V/STOL aircraft performance and stability". TRECOM TR 64-47, Dynasciences Corp. (Reference 20)	Uses inclined actuator disc theory and slender wing theory. Includes consideration of flapped wings.
"An investigation of propeller slipstream effects on V/STOL aircraft performance and stability". USAAVLABS TR 65-81 L. Butler et al. (Reference 21)	Extends Reference 20 to predict onset of stall, including slipstream rotation and uneven axial flow.
"Lifting surface theory for V/STOL aircraft in transition and cruise". USAAVLABS TR 68-67 E. Levinsky et al. (Reference 30)	Combines an inclined actuator disc with a wing represented by a lifting surface.

5.3.2 Discussion of Specific Methods

5.3.2.1 A Preliminary Theoretical Investigation of the Effects of Propeller Slipstream on Wing Lift (Reference 22). This report discusses three different methods for calculating the increase of lift on a wing due to interaction between it and a propeller slipstream. In each case the propeller slipstream is represented by a uniform, straight tube of fluid of specified velocity. The three methods differ in their representation of the wing.

First, the lifting line model as originally proposed by Koning (Reference 31) is used and has been solved by solution of Laplace's equation in the Treffz plane and the adoption of a Fourrier sine series for the circulation on the parts of the wing in-and outside the slipstream. The boundary conditions at the edge of the slipstream are satisfied by the adoption of an "image" horseshoe vortex system of the appropriate strength located at the inverse points corresponding with the horseshoe vortex system representing the wing. It is stated that the aspect ratio of the portion of the wing in the slipstream, D/C, (ratio of slipstream diameter to wing chord) is an important parameter and that the lifting line approach is of doubtful value when D/C becomes small (less than 6, say).

The second approach was to employ the slender wing theory of R. T. Jones (Reference 32). By assuming a separated solution for Laplace's equation in terms of a perturbation potential in polar coordinates at the trailing edge of the wing a closed form solution is obtained for the lift increment at any spanwise station.

It is acknowledged by the authors that the slender wing approach is likely to be valid only for small values of D/C (less than about 1.0, say) and so a third approach is taken in order to bridge the gap between large and small values of D/C.

The third approach was the application of Weissinger's lifting surface theory (Reference 33). The only case for which a solution was obtained for this model was that of an infinite span wing with a sinusoidally varying angle of attack spanning a slipstream of finite width and infinite height.

A comparison is made between the three methods for the case of a wing in a slipstream of infinite height and is shown in Figure 49. Also a comparison with some test data from Reference 34 is made using the lifting line and slender wing theories in Figure 50.

Figure 49 indicates, as might be expected, that the Weissinger lifting surface approach approximates the slender wing theory at low values of D/C and the lifting line theory at high values of D/C.

Figure 50 illustrates the relative merits of the lifting line and slender wing approaches in predicting the spanwise variation of lift on a wing spanning a slipstream with constant velocity distribution and no rotation. It is seen that the slender wing approach is more accurate than the lifting line, at least for the portion of the wing that lies in the slipstream. Some doubt is cast on the accuracy of this test data because of the apparent influence of the jet on the wing at large distances from the center of the jet.

The report also includes a comparison of predictions using the slender wing method with test data from Reference 35.

The comparison shows poor agreement between predicted and test values of lift increment due to slipstream.

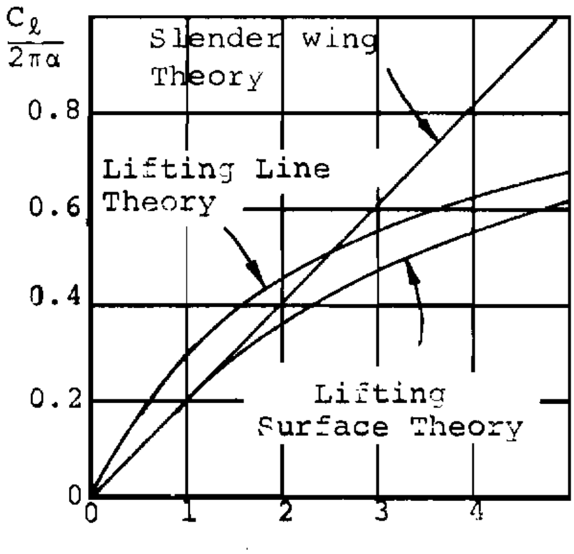


FIGURE 49. COMPARISON OF LIFTING LINE, SLENDER BODY AND LIFTING SURFACE METHODS OF REFERENCE 22 FOR A WING WITH SINUSOIDALLY VARYING ANGLE OF ATTACK IN AN INFINITE PLANE-SIDED JET

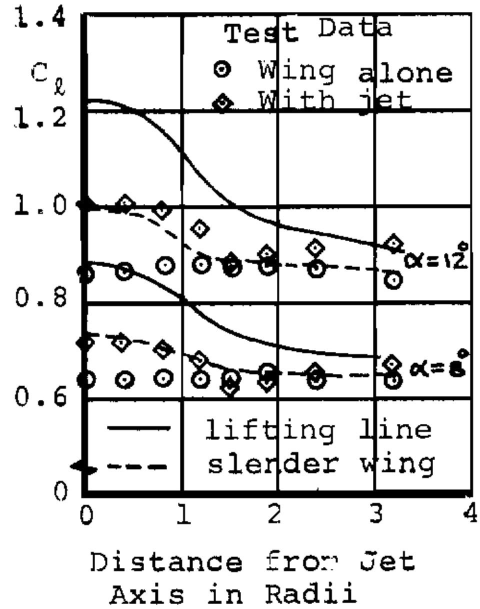


FIGURE 50. COMPARISON OF LIFTING LINE AND SLENDER WING PREDICTIONS OF REFERENCE 22 WITH TEST DATA FROM REFERENCE 34 FOR A WING EXTENDING ACROSS A CIRCULAR JET

The methods of this report contain the following general assumptions:

The freestream and slipstream are composed of an ideal, incompressible, non-viscous fluid;

The propeller is an actuator disc producing a uniform pressure discontinuity across its surface and introducing no rotation to the slipstream;

The axis of the slipstream intersects the wing;

Interference effects of wingtips, fuselage, nacelles, etc. are ignored;

The boundary conditions of equal pressure and zero normal velocity at the edge of the slipstream must be satisfied.

In addition, the following assumptions apply to the lifting line model:

The wing is represented by a straight vortex normal to the axis of the slipstream;

The trailing vortex system is represented by a series of 'elementary' horseshoe vortices (or vortex doublets).

The two following assumptions apply to the slender wing and lifting surface models respectively:

Flow at the trailing edge of the wing is tangential to the wing;

The wing is replaced by straight lifting line located at the quarter chord of the wing and the flow is forced to match the local airfoil slope at the three-quarter chord of the wing.

Slipstream rotation does not greatly affect the overall lift except when the wing is partly stalled as is seen in References 36 and 37. However, slipstream rotation does affect the local spanwise lift variation (Reference 38). This effect should be included so that the onset of stall can be predicted.

5.3.2.2 Vehicle Research Corporation. A series of reports published by the Vehicle Research Corporation (References 23 through 27) presents the development of a lifting surface theory for wings extending through propeller slipstreams. Special aspects of the aerodynamics of the deflected slipstream concept that are considered include:

Wings extending through multiple jets, (Reference 24);

Wings at high angle of attack extending through multiple jets, (Reference 25);

Wings at high angle of attack extending through inclined jets, (Reference 26);

Wings extending through multiple jets in separated flow conditions, (Reference 27);

Highly cambered wings "as used in deflected slipstream V/STOL arrangements" (Reference 25).

The work described in References 23 through 27 was the result of a program designed to generalize and extend the basic lifting surface theory of Reference 39.

The basic lifting surface theory of Reference 39 is an application of Weissinger's lifting surface theory. The representation of the propeller slipstream in the analysis, as in the analysis of Reference 22, is a circular jet of uniform flow properties and no rotation. The approach to satisfying the boundary conditions at the edge of the slipstream differs, however. The wing is assumed to consist of "even" and "odd" parts of horseshoe vortices with their bound vortices at the wing quarter chord. The even parts of the system are pairs of parallel vortices of strength $\Gamma/2$ extending to infinity both upstream and downstream of the bound vortex. The odd parts consist of a conventional horseshoe vortex of strength $\Gamma/2$ extending to infinity downstream and another of strength $-\Gamma/2$ extending to infinity upstream. The effect of the slipstream on the wing is represented by a perturbation velocity potential, ϕ , that is expressed as either of a pair of infinite series in Bessel functions depending on whether points are being considered inside or outside of the jet. The value of ϕ is obtained by imposing the boundary conditions of constant pressure and tangential flow at the jet boundary and the downwash condition at the three-quarter chord line of the wing.

Predictions of spanwise lift distribution obtained by the use of the method of Reference 39 show good agreement with the test data of Figure 50 for that part of the wing lying in the slipstream.

The special aspects of the deflected slipstream concept noted above are all treated by modifications of the method of Reference 39.

High angles of attack are treated in Reference 26 by satisfying the Weissinger downwash condition at the three-quarter chord points of the wing rather than at the corresponding points in the horizontal plane through the bound vortex.

Multiple jet effects are accounted for in Reference 24 by the application of image systems for each jet as before with the addition of further image systems related to the original image systems. Logically this leads to an infinite set of image vortices but to simplify the problem the author of Reference 25 used only the basic vortices, their images as required for each jet and the images of those images as required for each jet.

Wings extending through inclined jets (Reference 26) are treated by applying the Weissinger downwash condition to an "effective" local airfoil slope at the three-quarter chord point. The effective slope is simply the difference between the airfoil local slope and the jet slope inside the jet and the difference between the local slope and a modified jet slope outside the jet. The jet slope is calculated by an approximate inclined actuator disc theory.

Separated flow conditions are allowed for in Reference 27 by placing the bound vortex of the separated portion of the wing at the one-third chord location and imposing the Weissinger downwash requirement at the mid chord position or by employing actual two dimensional airfoil characteristics in separated flow conditions.

In some of the above work a term is included to describe rotation in the propeller slipstream.

All of the work presented in References 23 through 27 and 39 requires a large amount of computation to evaluate the lift distribution.

References 23 through 27 do not contain any numerical calculations or comparisons between prediction and test data.

5.3.2.3 Semiempirical Procedure for Estimating Lift and Drag Characteristics of Propeller-Wing-Flap Configurations for Vertical-and-Short-Take-off-and-Landing Airplanes. The approach taken in Reference 28 is different from those considered above in that no attempt is made to model the details of the airflow. Only the gross effects of wing-slipstream interaction are considered.

The propeller slipstreams comprise one body of air and their change of momentum is the result of the increment of velocity due to passing through the propellers and turning downwards by the action of the wing and flap.

The effectiveness of the flaps in redirecting the propeller slipstream is assumed to be the same as that achieved in static tests. Reference 28 includes correlations of the flap effectiveness for a number of different configurations, together with data indicating the loss of momentum due to the turning action of the flaps.

The second body of air considered is that flowing through a circular stream tube of diameter equal to the wing span; a correction term is included to allow for the fact that the propeller slipstreams are inside this larger one.

Simple actuator disc theory is used to relate slipstream velocity to propeller thrust.

The lift and drag of the propeller-wing-flap combination are evaluated by computation of the gain of downward and forward momentum of the combined stream tubes. The expressions developed for the lift and drag coefficients contain three distinct terms:

(a) the "power-off" coefficient

(b) a term representing the rotation and modification of the propeller thrust vector and

(c) a so called "augmentation" term

A correction factor of 1.6 is applied to the augmentation term as a result of comparing predictions with several sets of test data.

Figure 51 shows a comparison of predictions obtained using this method with test data from Reference 40.

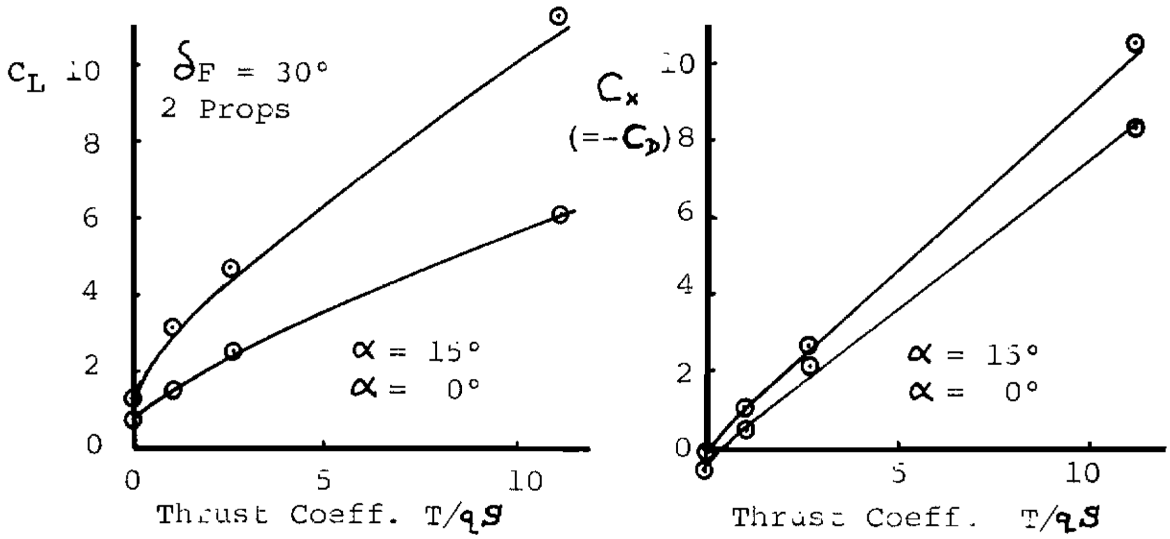


FIGURE 51. COMPARISON OF PREDICTIONS OBTAINED USING THE METHOD OF REFERENCE 28 WITH DATA FROM REFERENCE 40

This method has the merit that by the judicious use of correction factors it could be adapted to fit a related family of configuration.

The method uses the "power-off" characteristics of the configuration as a basis for calculation of the slipstream effects. This indicates that such considerations as the effect of aspect ratio are automatically accounted for a least in part.

The possibility exists of extending the method to include, for example, the effects of propeller slipstream rotation, without the need for lengthy analysis or computation.

Further comparisons are made in Section 5.4 of test data with predictions made using this technique.

5.3.2.4 A Stability Analysis of Tilt Wing Aircraft (Analytical). Although the prediction method for lift and drag outlined in Reference 29 is intended for tilt wings it is also applicable to deflected slipstream configurations. Also, it represents a third type of approach to the problem of assessing wing-slipstream interactions which is different from the purely analytical approach of References 22, 23 through 27, 39, and from the momentum approach of Reference 28.

First, an approximate inclined actuator disc theory is used to relate the velocity and inclination of the propeller slipstreams. The angle between the wing chord-line and the slipstream velocity vector is chosen as an effective angle of attack for the wing and the assumption is made that the wing possesses the same lift and drag coefficients as it would in a free stream of speed equal to the slipstream velocity and placed at the effective angle of attack. It is recognized that only part of the wing may be immersed in the slipstreams and so a mass flow correction factor is calculated, which is the ratio of the "actual" mass flow to the "assumed" mass flow. The "assumed" mass flow is the product of the velocity through the propeller disc and the momentum area of the wing. The "actual" mass flow is the sum of the mass flow through the propeller disc and the mass flow through the momentum area of the wing (which is adjusted to allow for the presence of the slipstreams within it.)

Figure 52 shows predictions of the lift and drag coefficients (referred to slipstream dynamic pressure) as a function of "effective" angle of attack for the model test of Reference 29. Correlation appears to be poor especially at high values of effective angle of attack.

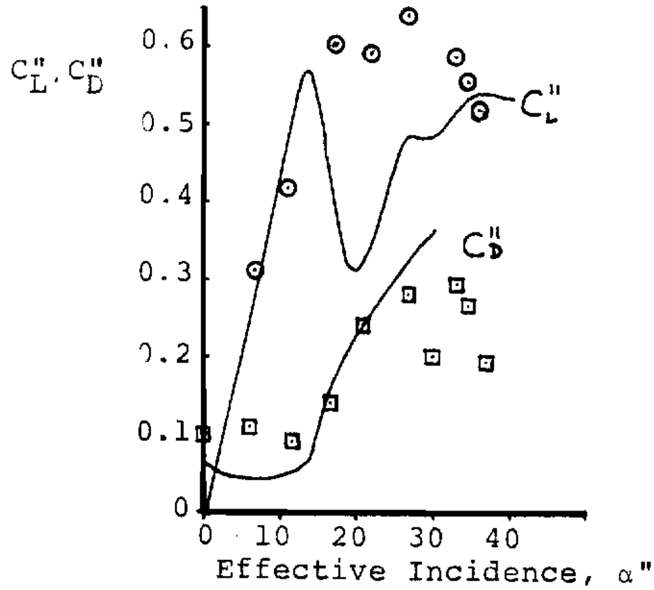


FIGURE 52. COMPARISON OF PREDICTION BY METHOD OF REFERENCE 29 WITH TEST DATA FROM THE SAME SOURCE

The method, in effect, assumes that the whole of the wing is at the effective angle of attack experienced by that part of the wing that is in the slipstream.

The inclined actuator disc theory employed is good only for small angles of attack.

The report continues, having evaluated lift and drag, to study the stability and control of tilt wing aircraft. The equations of motion are formulated and the conventional assumptions of uncoupled longitudinal and lateral motion and small perturbations are made. The stability derivatives are defined and evaluated for a specified configuration. Derivatives that could not be evaluated were assumed negligible. The response of the configuration to various perturbations is shown in the report.

5.3.2.5 Effects of Propeller Slipstream on V/STOL Aircraft Performance and Stability (Reference 20). The lift and drag of a deflected slipstream (or tilt wing) airplane are calculated and expressed in simple form.

The increment of lift on the part of the wing immersed in a slipstream is calculated using the slender wing theory of Reference 32. Other contributions to lift and drag calculated are, propeller thrust and normal force and the "free stream" lift and drag of the wing.

Inclined actuator disc theory is employed to calculate the inclination and velocity of the slipstream at the wing.

Figure 53 shows a comparison of predictions with test data from Reference 40. Note that the coefficients are referred to slipstream dynamic pressure, $C_{L_s} = L/q+T/_A)$ etc.

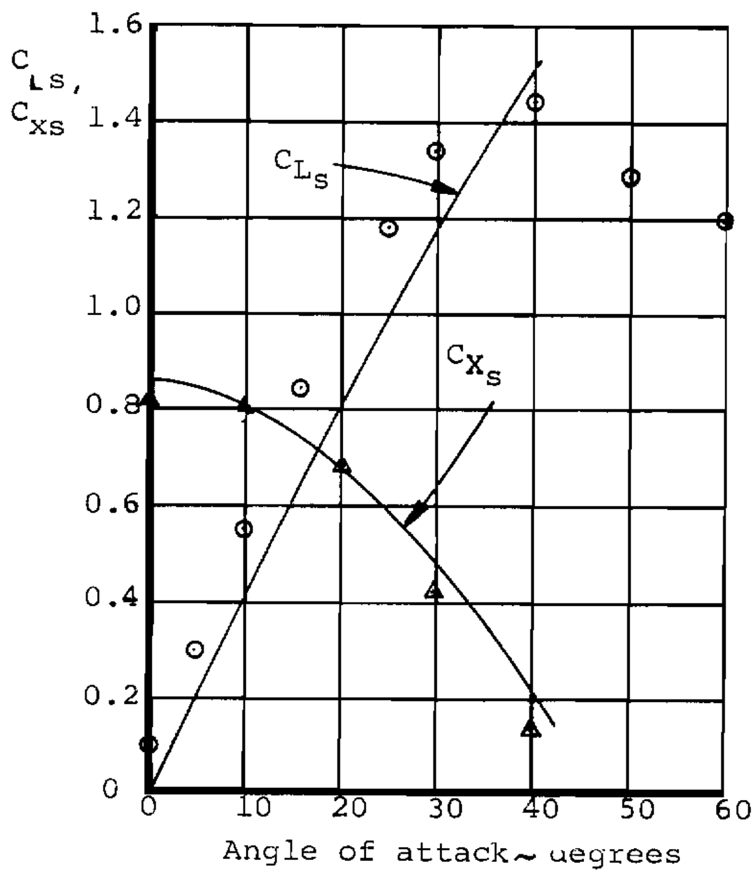


FIGURE 53. COMPARISON OF PREDICTIONS USING METHOD OF REFERENCE 20 WITH TEST DATA FROM REFERENCE 36

Comparisons are included of test data from Reference 36 and calculations predicting lift and drag for the model test of Reference 36. Generally, correlation appears to be good at low angles of attack and provided flaps are not deflected.

The effect of flap deflection has been assumed to be equivalent to a change of angle of attack of the same magnitude as the flap deflection.

5.3.2.6 An Investigation of Propeller Slipstream Effects on V/STOL Aircraft Performance and Stability (Reference 21). The prediction method of Reference 21 is based on that of Reference 20. The lift increment due to the slipstream interacting with the wing is calculated as before by the slender wing theory of Reference 32. Reference 21 differs in the evaluation of the local angle of attack in the slipstream by allowing for a rotational term.

Using Schrenk's spanwise loading approximation the prediction of the onset of stall is demonstrated. Figure 54 illustrates predictions of stall angle of attack in comparison with test data from Reference 41. Onset of stall was considered to occur at the angle for which the lift curve first became non-linear.

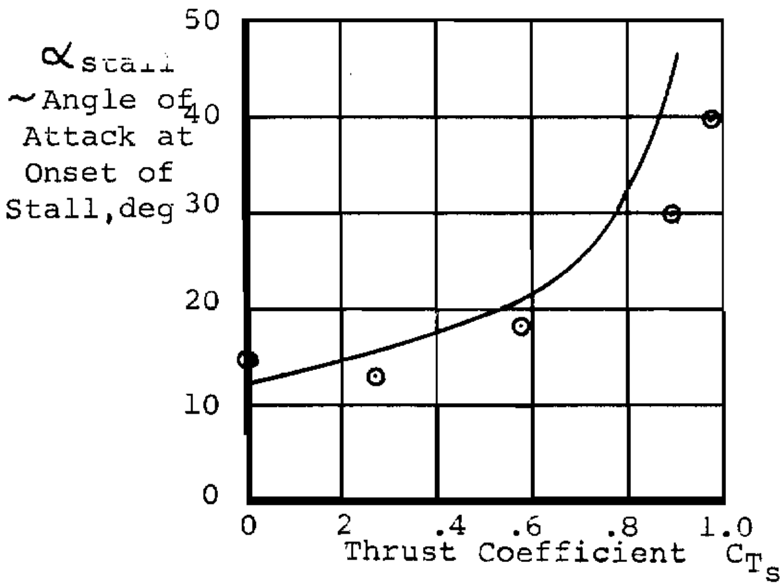


FIGURE 54. PREDICTION OF ONSET OF STALL

Two further modifications are made to the basic method of Reference 20. First the effect of flap deflection on the wing effective angle of attack is modified by a factor depending on the flap deflection and based on test data comparisons. This resulted in improved predictions for large flap deflections but deteriorated the predictions for flap deflections of less than 20°.

Secondly, a scheme for taking into account radial variations in the axial velocity in the slipstream was indicated. A sample case was calculated using this scheme and it was shown that if the peak velocity in a propeller slipstream is close to the axis the gain in lift due to wing-slipstream interaction will be higher than if the peak velocity is near the outside of the slipstream or if the slipstream is uniform.

The rotational term included was in the form of a solid body rotation whereas a vortex type of rotation may be more applicable.

5.3.2.7 Lifting Surface Theory for V/STOL Aircraft in Transition and Cruise (Reference 30). The prediction method of Reference 30 is developed in two parts.

First, an inclined actuator disc theory is developed in incompressible flow, producing a circular slipstream that is simulated by a set of ring vortices coaxial with the slipstream and by vortices, sources and sinks distributed along the length of the slipstream. The usual boundary conditions are imposed to evaluate the velocity field of the slipstream. Representation of the propeller by a set of bound vortices with a trailing vortex extending helically downstream from the tip and another extending axially from the blade root allows of the introduction of rotational velocity terms inside the slipstream. The authors illustrate that the actuator disc theory can predict the downwash at the centerline of the slipstream fairly well when compared with test data showing average downwash in the vicinity of a tail.

The second part of the development of this prediction method is the simulation of the wing-slipstream combination. The wing is represented by a system of discrete small horseshoe vortices with the bound vortex elements on the quarter chord line of the wing.

A system of horseshoe vortex elements is distributed around the slipstream in order to satisfy the "equal pressure" boundary condition. The tangential velocity boundary condition is satisfied by the assumption of a 'reduced potential' inside the slipstream. The Weissinger downwash condition at the three quarter chord point is imposed. Satisfaction of the boundary conditions leads to set of linear simultaneous equations whose unknowns are the strengths of the wing and slipstream vortices.

Several sample calculations were carried out and comparison with test data from Reference 42 shows fairly good correlation with measurements of spanwise lift variation at small angles and attack. Downwash measurements, however, are considerably larger than the method predicts. An improvement in the prediction was achieved by assuming the slipstream had been deflected by only a half of the value calculated from the inclined actuator disc theory.

All trailing vortices (from the wing and from the propeller) are inclined at the same angle to the freestream direction.

The propeller slipstream starts at the wing quarter chord location.

The actuator disc theory may be used to predict flow fields only at large distances from the actuator disc.

5.3.3 Summary Comments

The prediction methods reviewed above are a sample of the available methods. Those available concentrate on the problem of the interaction between the propeller slipstream and the wing.

The mathematical methods consisting of distributions of vorticity 'tailored' to fit certain boundary conditions appear to have employed models that are oversimplified particularly with regard to flapped-wings. The fact that all such methods employ an actuator disc to represent the propeller and ignore the effect of the wing on the propeller performance sheds doubt on their validity.

The early empirical or semi-empirical methods (Reference 28) were somewhat oversimplified but later methods have indicated that the semi-empirical approach has several advantages:

They can easily be forced to fit the data from tests of a family of configurations in order to predict the change in characteristics caused by alteration of important parameters;

They are easily extended to include the effects of phenomena that were originally omitted;

They require a minimum of computation time;

The more complicated analytical methods have not been shown superior in their ability to predict lift.

It is to be noticed that of the methods available only lift, drag and the induced velocities due to the wing-propeller slipstream interaction are evaluated, and these only for the symmetrical flight condition.

Reference 21 indicates a method for the prediction of wing pitching moment by demonstrating that the center of pressure of the wing, when expressed as a fraction of extended chord, does not shift when propeller thrust coefficient is changed except near the static thrust condition ($C_{T_S} > 0.8$). Reference 21 also includes a survey of data related to propeller normal force and hub moment including the changes in these items due to the presence of a wing. This aspect is mentioned here because in most of the methods reviewed above these items, although of importance, have been ignored.

In all of the above, the characteristics of real propellers have not been taken into consideration. Various methods exist for the prediction of propeller loads in axial and inclined flow conditions. Again, there are two basic approaches, analytical and empirical. In the past, analytical methods have been of reasonable accuracy for axial flow conditions at low thrust coefficients but in the near static case and in inclined flow conditions these methods are inadequate. Recent analytic attempts, Reference 43 for example, taking advantage of the large capacity digital computational facilities now available have been encouraging. The approach taken has been to represent each propeller blade as a bound vortex with a system of trailing vortices. The trailing vortices are either constrained Reference 43 to fit in a wake of given shape determined empirically (as in Reference 43) or allowed to follow the motion imposed upon them by the induced velocity field of the whole system. Such a technique is necessary in the high thrust conditions achieved in STOL operations because of the large distortions of the trailing vortices from the usually assumed regular helical form.

Theoretical predictions of the stability and control of deflected slipstream configurations are, in principle, easily obtained by traditional methods if the problem of defining the forces and moments on the airplane in all relevant flight conditions has been solved. This area of study has been hampered by the lack of knowledge about the trajectory of propeller slipstreams and the induced flow fields about the aircraft, particularly in the vicinity of the tail.

The validity of classical stability methods applied to V/STOL aircraft has been questioned (Reference 44) for variety of reasons. Aspects upon which doubt has been cast include: the assumption of small perturbations, the validity of linearization of the equations of motion and the assumption of an equilibrium steady state. Evidence of non-linearities in the longitudinal characteristics of tilt wing airplanes has been reported in Reference 19.

No methods have been found that predict ground effects for deflected slipstream configurations. The best approach to date has been to employ test data (for example, Reference 45) or use the general method of Heyson (Reference 46).

5.4 CORRELATION OF TEST DATA WITH PREDICTION

The method of Reference 28 has been used to predict ligt and drag coefficients for two deflected slipstream configurations. In the first case the method is compared with flight test data for the Breguet 941 (Reference 47). Secondly, the method is compared with wind tunnel data from NASA TND4448 (Reference 48).

5.4.1 Prediction Method of Reference 28

The equations developed for lift and drag coefficients by the method of Reference 28 are

$$C_L = C_{Low} + \left(\frac{F}{T}\right) T_c \sin(\alpha_w + \theta) \left[\frac{K}{\sqrt{1 + \quad 1 + T_c (S/A)}}\right] \tag{33}$$

and

$$C_D = C_{Dow} + \left(\frac{F}{T}\right) T_c \left[\frac{K\left\{1 - \cos(\alpha_w + \theta)\right\}}{\sqrt{1 + T_c (S/A)}} - \cos(\alpha_w + \theta)\right] \tag{34}$$

where:

C_{Low}, C_{Dow}	are	"power-off" lift and drag coefficients at angle of attack α_W
$\frac{F}{T}$	is	thrust recovery factor (Figure 3 of Reference 28)
T_c	is	thrust coefficient, T/qS
q	is	freestream dynamic pressure
S	is	wing area
α_W	is	wing angle of attack relative to the freestream direction
θ	is	flap turning effectiveness (Figure 2 of Reference 28).
A	is	total propeller disc area
K	is	an empirical factor used to make the above equations fit test data used in the formulation of the method. A value of K=1.6 was recommended by the author of Reference 28 and will be used for the predictions calculated below.

5.4.2 Flight Test Data and Calculations

The data used for this comparison was obtained from Reference 47, Figure 29(a). Reference 47 is a report of flight test carried out by NASA on the Breguet 941 deflected slipstream airplane. Figure 29(a) contains the flight test lift-drag polar for the aircraft in the take-off configuration with settings of 45° on the inboard flaps and 30° on the outboard flaps.

Since the method of Reference 28 predicts the lift and drag of the propeller-wing-flap system the basic flight test data was adjusted to allow for the tail contributions required to trim the airplane. The tail characteristics and the downwash at the tail were estimated using the methods of Appendix B of Reference 49.

The basic data used for the predictions were the lift and drag corresponding to $T'_c = 0$ in Figure 29(a) of Reference 47.

The values of C_{Low}, C_{Dow} were obtained from the trimmed values of C_L, C_D at $T' = 0$ using the following expressions:

$$C_{Low} = C_L - C_{LT}\frac{S_T}{S} \tag{35}$$

$$C_{Dow} = C_D - C_{DT}\frac{S_T}{S} \tag{36}$$

$$C_{LT} = a_{1T}(\alpha_F + i_T - \epsilon - \alpha_{oT}) + a_{2T}\,\delta_e \tag{37}$$

$$C_{DT} = C_{LT}\sin\epsilon + C_{LT}^2/\pi \mathit{AR}_T e_T \tag{38}$$

where

ST	is	tail area	
a_{1T}	is	tail lift curve slope	obtained from Appendix B of Reference 49
a_{2T}	is	elevator effectiveness	
ϵ	is	downwash at the tail	
α_F	is	fuselage angle of attack	
i_T	is	horizontal tail setting	
α_{oT}	is	zero lift angle of attack of tail	
δ_e	is	elevator deflection	
C_{LT}	is	tail lift coefficient required for trim	

C_{DT}	is	tail drag coefficient required for trim
AR_T	is	tail aspect ratio
ϵ_T	is	Oswald efficiency of tail

A sample calculation for this set of test data is shown below for the case of $\alpha_w = 5°$ at $T'_c = 1.0$.

The following data are required for the calculation:

Wing Area, S	889 ft^2
Wing Incidence, i_W	3° relative to fuselage reference line
Wing MAC, $\bar{c}$	12.15 ft
Wing span, b	76.1 ft
Wing chord at ℄ of inboard prop, c	13.2 ft
Wing chord at ℄ of outboard prop, c	10.1 ft
Extended flap chord at 30°, cf	0.42 local chord
Extended flap chord at 45°, cf	0.43 local chord
Spanwise extent of inboard flap	0.56 semispan
Spanwise extent of outboard flap	0.44 semispan
Propeller diameter, D	14.76 ft
Propeller total disc area, A	684 ft^2
Tail area, S_T	320 ft^2
Tail aspect ratio, AR_T	3.3
Tail setting, i_T	5.6°

The following values will be required for the calculation of "power-on" lift and drag coefficients:

$(c_f/D)\, i = 0.43 \times 13.2/14.76 = 0.385$ inboard

$(c_f/D)\, o = 0.42 \times 10.1/14.76 = 0.287$ outboard

Figure 2 of Reference 28 gives corresponding values of flap turning effectiveness of:

$(\theta/\delta)\, i = 0.65$

$(\theta/\delta)\, o = 0.55$

and for inboard and outboard flap deflections of 45° and 30° respectively the flap turning angles are obtained

$\theta\, i = 29.2°$

$\theta\, o = 16.5°$

A mean of these two values, weighted in proportion to the spanwise extent of the respective flaps gives:

$\theta = 0.56 \times 29.2° + 0.44 \times 16.5° = 23.6°$

Corresponding with this value of θ, Figure 3 of Reference 28 gives

$\frac{F}{T} = 0.98$ for the thrust recovery factor.

We have selected the condition $\alpha_w = 5°$.

The corresponding fuselage angle, α_F, is therefore, 2°. This angle must be related to α_u, the angle of attack indicator reading as this is the parameter against which the test data are plotted. Figure 30 of Reference 47 is a calibration of α_u vs α_F at various values of T_c'.

At $T_c' = 0$ $\alpha_F = 2°$ gives $\alpha_u = 3.4°$

Corresponding values of trimmed lift and drag coefficients are then obtained from Figure 29 (a) of Reference 47.

$$\left.\begin{aligned} C_L &= 1.72 \\ C_D &= 0.15 \end{aligned}\right\} \text{unpowered}$$

Values obtained for the trim lift and drag of the tail use the following data:

a_{1T} = 0.058/deg.

a_{2T} = 0.075/deg.

α_{oT} = 4° (this is an assumption based on the information given in Reference 47)

e_T = 0.7 (value suggested in DATCOM)

Downwash, ϵ, has been evaluated using the method of Appendix B of Reference 49 and is illustrated in Figure 55 as a function of wing lift coefficient.

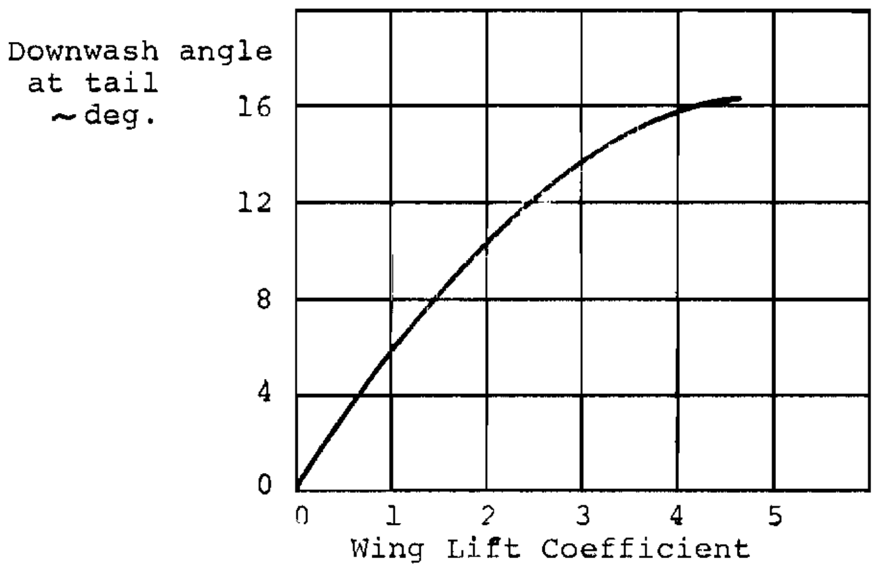


FIGURE 55. ESTIMATED DOWNWASH AT THE TAIL OF BREGUET 941 IN TAKE-OFF CONFIGURATION

At C_L = 1.72 we obtain ϵ = 9.2° and from Figure 20 of Reference 47 the value of δ_e is estimated to be -0.5°. Evaluating the expressions for tail lift and drag we get C_{LT} = -0.36 and C_{DT} = 0.04.

Thus the values of C_{Low} and C_{Dow} at α_w = 5° are

$$C_{Low} = 1.72 - (-0.36)\,\frac{320}{889} = 1.85$$

$$C \quad = \;.15 - (-0.04)\,\frac{320}{889} = 0.164$$

Now, all the information required to calculate C_L, C_D at the "power-on" condition has been obtained and we substitute in the appropriate equations, as follows:

$$C_L = 1.85 + 0.98 \times 1.0 \times \sin(5.0° + 23.6°) \times \left[\frac{1.6}{1 + \sqrt{1 + 1.0 \times (889/684)}}\right] \quad (39)$$

and

$$C_D = 0.164 + 0.98 \times 1.0 \times \left[\frac{1.6 \times \{1 - \cos(5.0° \times 23.6°)\}}{\sqrt{1 + 1.0 \times (889/684)}} - \cos(5.0° + 23.6°)\right] \quad (40)$$

$$C_L = 2.81 \text{ and } C_D = -0.57 \text{ at } \alpha_W = 5°, \; T'_c = 1.0 \quad (41)$$

The values calculated here must be compared with the test data suitably adjusted for the trim lift and drag, in the same manner as the $T'_C = 0$ data was corrected. At $T'_C = 1$ $\alpha_F = 2°$ corresponds with $\alpha_u = 4.8°$ which corresponds with $C_L = 2.76$ and $C_D = -0.52$ in Figure 29 (a) of Reference 47. Figure 20 of Reference 47 gives $\delta_e \simeq -5.6°$ and from Figure 5 the downwash $\epsilon = 13°$.

Hence $C_{LT} = -0.965$ and $C_{DT} = -0.89$ and this results in $C_L = 3.11$ and $C_D = -0.49$ compared with predictions of $C_L = 2.81$ and $C_D = -0.57$. Comparison of predictions with the test data is shown in Figures 56, 57, and 58.

5.4.3 Wind Tunnel Test Data

The test data used for this correlation was obtained during wind tunnel tests of a large scale model of a four propeller deflected slipstream configuration that had no horizontal tail. Thus no trim corrections were required. The data was obtained from Reference 48, Figure 11(e).

Predictions were made for "power-on" lift and drag at thrust coefficients of 1.0 and 3.0 based on the test data obtained at $T'_C = 0$ which is approximately equivalent to the "power-off" condition.

Relevant data for this correlation are:

Wing Area	329 ft^2
Propeller diameter	9.3 ft
Flap type	Triple slotted
Flap deflection	60°
Extended flap chord	0.38 local chord

Correlation between predictions and test data is illustrated in Figure 59.

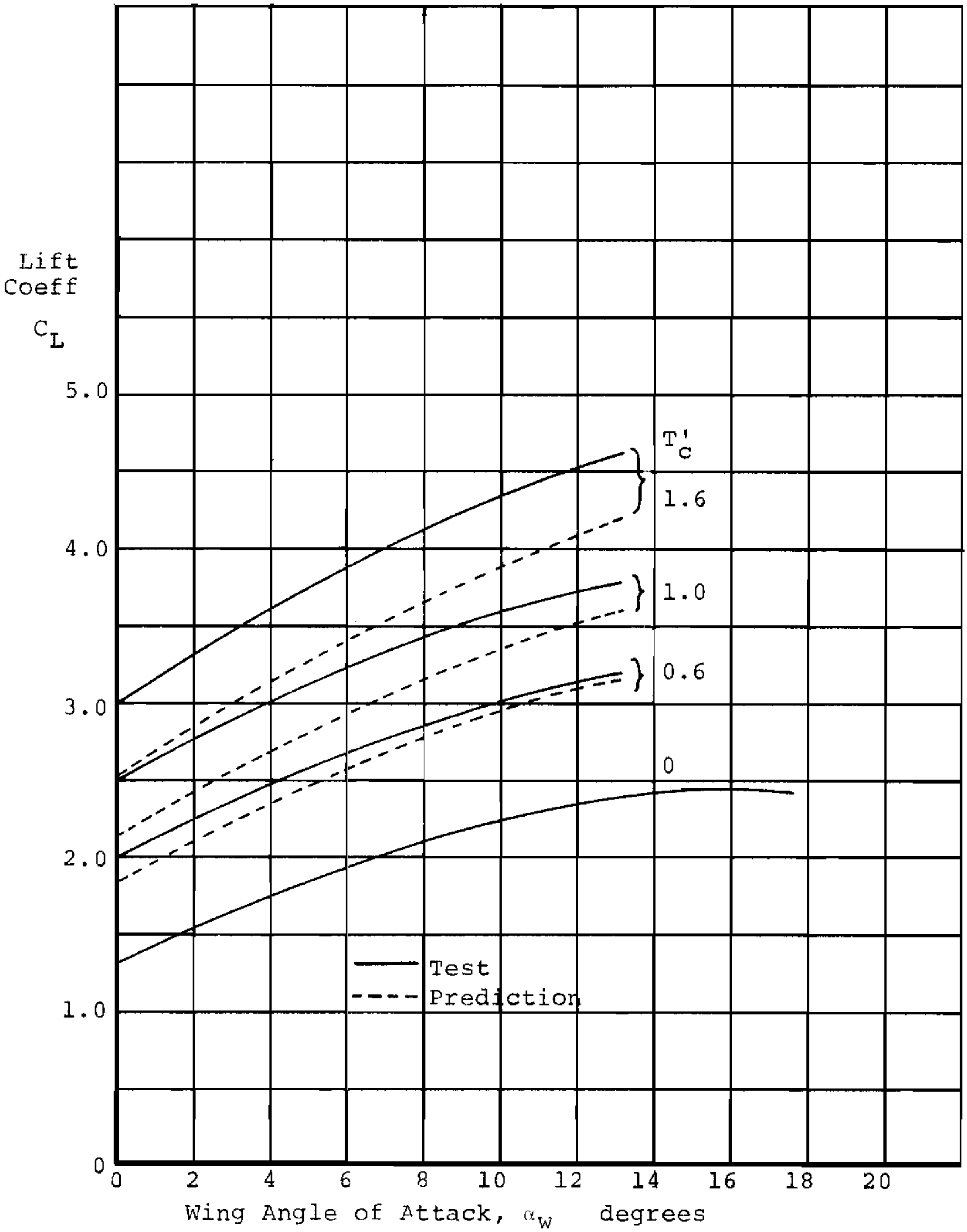


FIGURE 56. COMPARISON BETWEEN LIFT COEFFICIENT AND TEST VALUES FROM REFERENCE 47

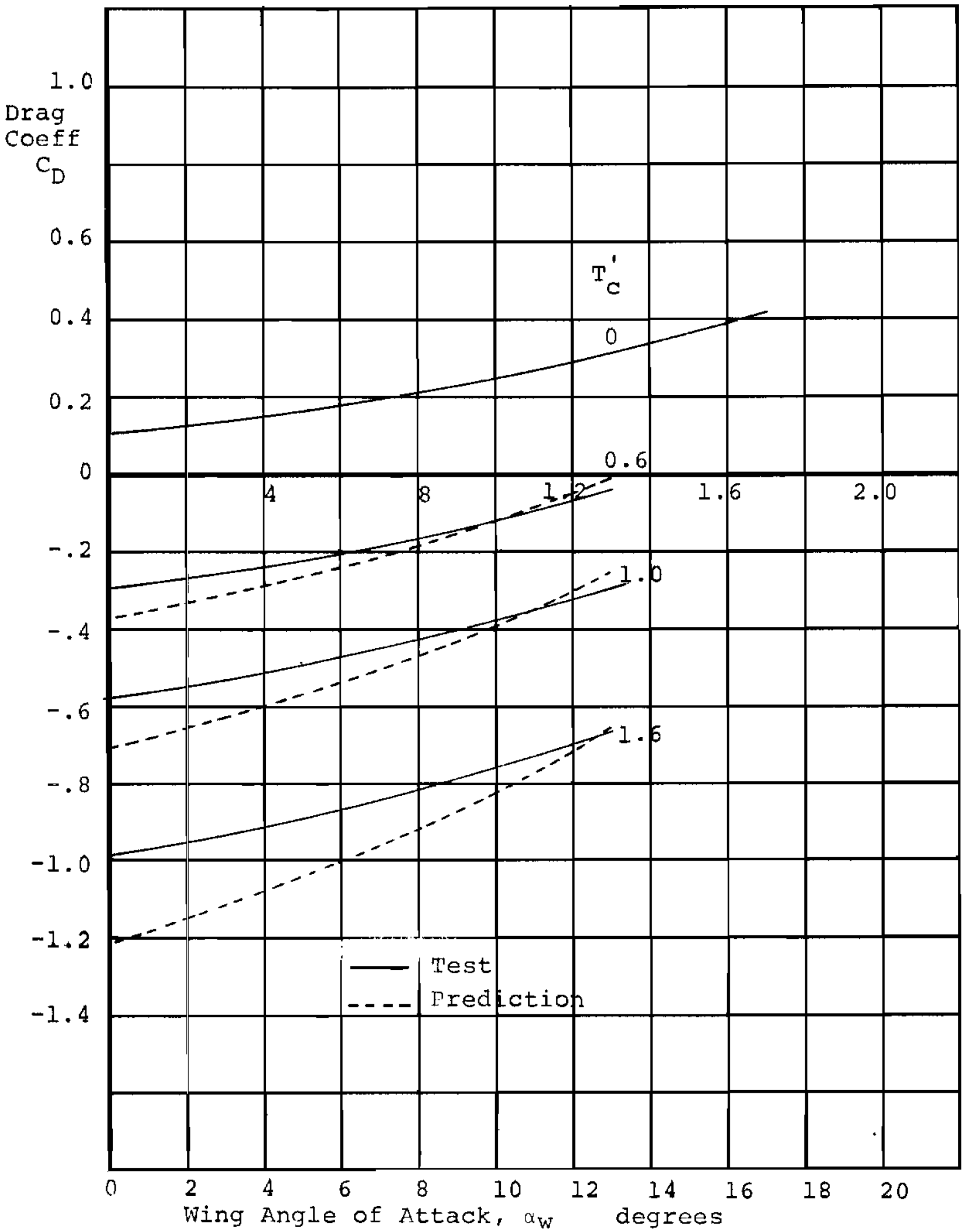


FIGURE 57. COMPARISON BETWEEN PREDICTED DRAG COEFFICIENT AND TEST VALUES FROM REFERENCE 47

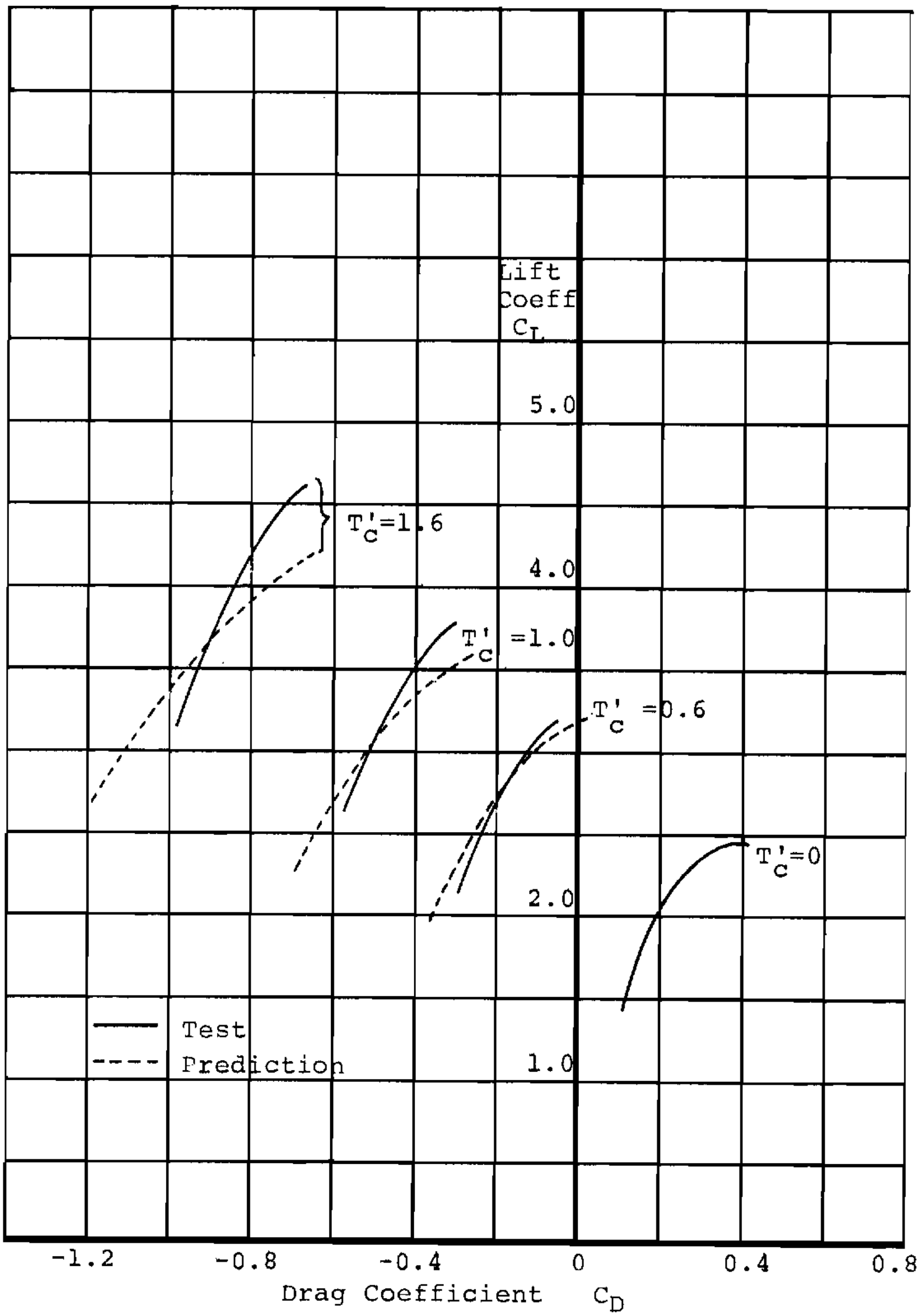


FIGURE 58. COMPARISON BETWEEN PREDICTED FORCE POLAR AND TEST VALUES FROM REFERENCE 47

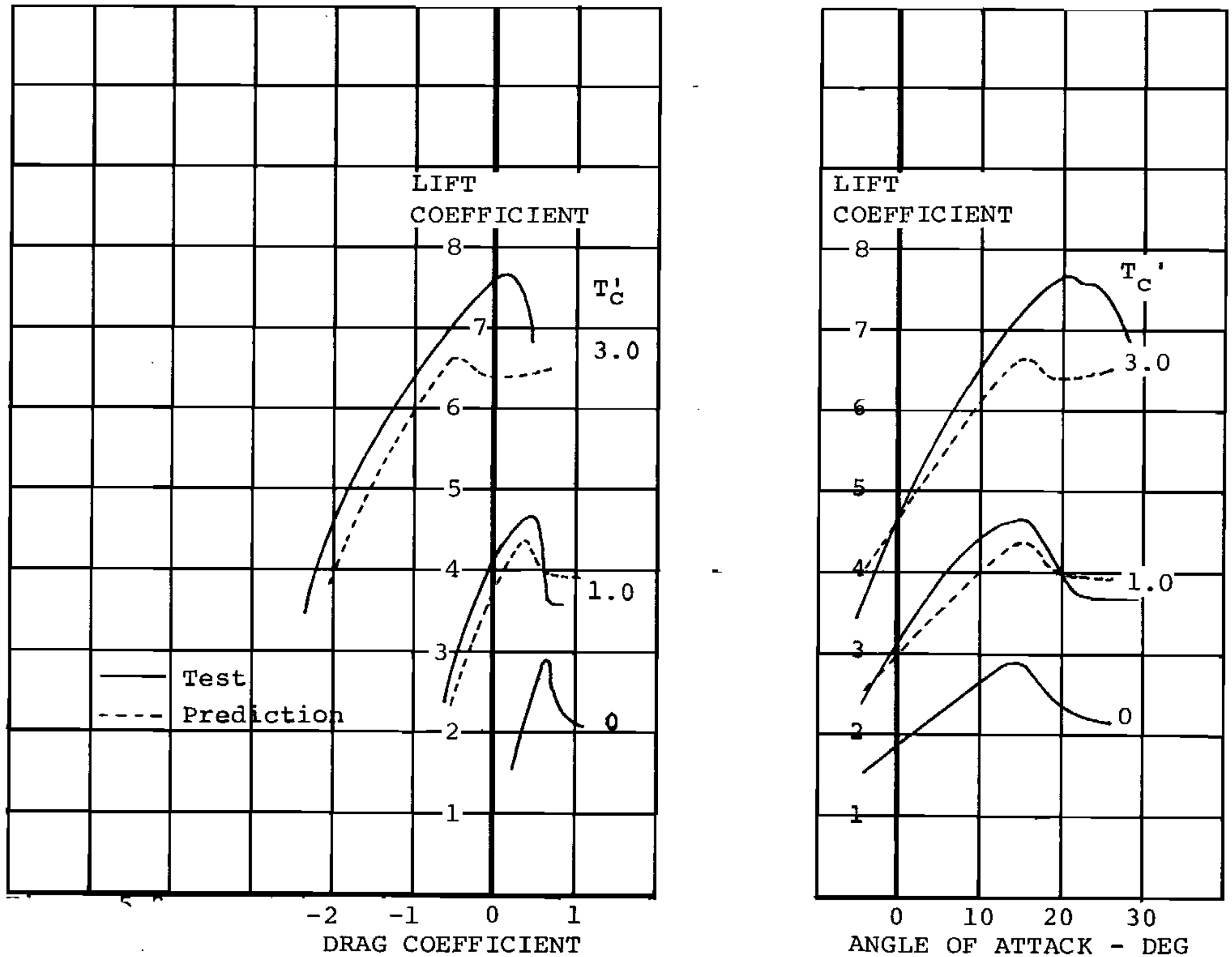


FIGURE 59. COMPARISON BETWEEN PREDICTED LIFT AND DRAG COEFFICIENTS WITH DATA FROM REFERENCE 48

5.4.4 Comments on the Predictions

The comparison of the flight test data from Reference 47 with predictions (Figures 56, 57, and 58) shows poor agreement at the higher values of thrust coefficient. Errors of 0.5 in lift coefficient and 0.2 in drag coefficient occur at zero angle of attack when the thrust coefficient is 1.6. These numbers represent at 10% overestimate of take-off speed, an overestimate of drag of about 40% and an error in climb angle of about 6° for the airplane in question.

Although this represents poor correlation the fault may not lie entirely with the basic prediction method. The method of calculating the downwash at the tail is certainly in error to some extent as it is based on the assumption that the wing wake is not "rolled-up" and neglects all but the gross lift effect in calculating local downwash.

The comparison of predictions with wind tunnel test data (Figure 59) shows slightly better agreement than with the flight test data. In this case no trim corrections were required.

It is evident that the prediction method of Reference 28 gives only fair predictions of lift and drag (at least for the test data considered here) and further improvements are necessary before it could be used with confidence.

5.5 DEFINITION OF DEFICIENCIES AND GAPS IN KNOWLEDGE

5.5.1 General

Successful development of the deflected slipstream configuration requires a thorough understanding of these aerodynamic effects.

The important mutual interference effects between wing and propeller including stall onset, lateral control power, and the effects of power changes on forces and moments.

The wake characteristics including position and thickness of the wake core and distribution of strength and direction of flow within the wake.

These are also the most difficult characteristics to predict accurately and require detailed attention to develop an adequate test program.

5.5.2 Test Data

A large body of force test data has been obtained on a number of basic configurations representing the deflected slipstream concept. These data are valuable in the design and analysis of similar configurations. However, many of the test programs had omissions that limit the usefulness of applying the data to the development of basic aerodynamic technology. In particular, the lack of instrumentation for the measurement of propeller loads has limited the usefulness of some of the data. Much of the data has been obtained at specific design-point conditions and, though useful for certain design work, lacks the systematic variation of parameters which is important to the development of methodology.

5.5.2.1 Modeling. Considerable advances have taken place in the development of small model motors. However, careful attention must be paid to the matching of motors and propellers to the airplane model.

Power/thermal limitations of electric motors may restrict the thrust coefficient attainable or require testing at extremely low tunnel speeds to obtain the required thrust coefficient. Heating of the electric motors requires that special attention be paid to cooling the motors and the ducting of coolant lines (usually water) must be carefully integrated within the model. The motor heating can also cause inaccuracies in the measured propeller forces through temperature drift in the balances, which must then be compensated.

Hydraulic motors have been developed which do not have the inherent power limitations of the electric motors but the routing of hydraulic lines complicates the model design and fabrication and the heating and expansion of the lines can cause serious interference with the propeller balances.

Pneumatic motors tend to be smaller per unit horsepower developed than electric motors and therefore lend themselves more effectively to use on small models. The routing of the air supply around balances and to the motor, however, requires that unusually detailed attention be paid to model design and to accurate, thorough calibration for data corrections.

Size and shape of nacelles should be compromised to the minimum extent possible in order to avoid premature flow separation, particularly with power-off and to minimize interference with flaps and control surfaces. This includes both physical interference which restricts control surface size and/or deflection and the aerodynamic interference, when using air motors, created by the impingement of tail pipe exhaust flow on the wing and trailing edge surfaces or its influence on the trailing wake. At high torque conditions, the air motors can produce sizable tailpipe thrust - much larger, relative to the prop thrust, than would be encountered on the full scale airplane. Since it may be important to simulate not only the correct total thrust coefficient but also the correct thrust-split between prop and tailpipe, careful attention should be paid to the selection of motor rpm during the test.

Motor rpm is also influenced strongly by the model dynamics - with the rpm often specified by the requirement to avoid resonance bands - and by the relationship between tunnel speed, required thrust, rpm, and prop blade angle.

Relatively little has been done, by analysis or experiment, to show what the effect might be on a wing-flap-slipstream system of simulating more accurately the propeller span loading and slipstream swirl content. These variables should be evaluated.

Excited by the motors and propellers, the model's dynamic motion can influence data collection accuracy, even with high sampling rates and digital recording equipment. It may be necessary to experiment with tuning systems which compensate for these effects.

As previously stated, it is of the utmost importance to measure propeller forces and moments - both to accurately calibrate the thrust coefficient and to extrapolate the data to full scale. The changes in both prop thrust and normal force as the propeller is pitched through an α -sweep can cause a significant error in the airplane induced drag if prop instrumentation is not available to account for these effects.

5.5.2.2 Test Conditions. As noted, the development of small, powerful air motors has reduced one problem which has plagued powered testing for years-namely, the requirement to produce high thrust coefficients without testing at very low tunnel speeds (and correspondingly-at low Reynolds number).

In addition, modern V/STOL tunnels now exist, providing a wide range of controlled tunnel speeds, with large test sections and with the ability to adapt the wall treatment to the test conditions - from open to slotted to closed.

5.5.2.3 Ground Effect. Some ground effect data has been obtained mainly, however, on tilt wing configurations. These data may be incomplete in that they appear to show some inconsistencies and contradictory trends. The significance of ground effect to the performance and stability of deflected slipstream airplanes in landing is shown by flight test (Br 941 for example, Reference 47). The necessity for realistic ground effect simulation has already been demonstrated for various configurations and a criterion has been evolved (NASA SP-116) to determine the conditions for which a moving belt facility is required.

Dynamic ground effects have been noticed, primarily in the flight testing and tunnel testing of the XC-142. Very little systematic testing has been done to explore the dynamic ground effects of deflected slipstream configurations, but the exploratory work on tilt wings may be applicable.

5.5.2.4 Flow Visualization. A number of different techniques have been used to visualize the flow from a prop or wing or complete model including smoke, bubble, tufted screens, schlieren (using local heat addition), etc. None of these have provent universally acceptable and no standard, practical method of flow visualization exists today. The development of a useable technique is of paramount importance - both to configuration developmental testing and to development of more realistic mathematical models to aid in theoretical studies.

5.5.2.5 Pressure Data. Pressure distributions on wings have been measured on a number of configurations but mainly in static flow conditions. Such measurements have indicated the large variations of spanwise loading in the part of the wing in the slipstream due to slipstream rotation. The amount of data is sufficient to give a broad understanding but is inadequate to support an analytical effort or to verify theoretical prediction techniques.

5.5.2.6 Parametric Force Testing. As previously discussed, a large body of force test data has been obtained. However, limitations in test instrumentation or range of parameters has severely reduced the usefulness of much of the data for the purpose of creating new configurations or assembling analytical methodology. In addition, it is expected that there is a large amount of data, obtained during contractor's configuration development, and unpublished NASA testing which has never been properly analyzed for the development of new methodology.

5.5.3 Analytical Methods

The review of the analytical methods reveals that the basic mathematical models used to simulate the real flow characteristics are inadequate. Furthermore, it is not clear that the boundary conditions imposed to solve the resulting equations are valid. It is clear that a much more realistic model must be used including real propeller effects and the influence of the wing flow induced at the propeller.

The empirical methods available are generally capable of predicting the trends in aerodynamic characteristics due to change of the relevant parameters though the accuracy is not sufficient without appropriate test data for back-up. In addition, inconsistencies have been found in the correlation of these methods with test data - giving good agreement with one set of data and poor agreement with another.

The methods available cover only overall lift and drag and spanwise loading variation. The calculation of lateral-directional behaviour has not been possible to any level of accuracy because of the ignorance of the characteristics of the wake and the influence of prop direction of rotation. For the same reason, it is difficult to predict the power effects on longitudinal stability and specifically - to predict analytically the horizontal tail height required to minimize these power effects.

The direction of propeller rotation influences both the wing's spanwise loading (and therefore the onset of stall) and the lateral-directional behavior with asymmetric thrust. Yet, it is not possible to predict these effects.

5.6 RECOMMENDED PROGRAMS

Outlined below are the necessary programs required to correct the voids in deflected slipstream aerodynamic technology.

5.6.1 Wind Tunnel Test Programs

The main areas in which test data are required are STOL propeller characteristics and the combined propeller-wing-flap flow field.

The first area requires parametric testing of typical STOL propellers in axial and inclined flow conditions over the range of speeds from static to maximum cruise speed at the appropriate thrust levels. The tests should include measurement of thrust normal force and hub moment and comprehensive studies of the slipstream structure and trajectory. In addition, the effects on the propeller and slipstream characteristics of non-uniform inflows (such as may be induced by high lift wings) should be evaluated.

The second area of importance is the study of the flow field consisting of the combined effects of the propeller slipstreams and the wake of the wing-flap system. Detailed flow surveys are required for the basic wing-flap system and for the combined propeller-wing-flap system with variable propeller position, overlap, etc. Half model tests are generally agreed to be suitable for testing incremental configuration differences in longitudinal characteristics of wing-body-nacelle configurations, but unsuitable where fuselage aerodynamics or downwash may be important. Complete model tests should be carried out in order to evaluate characteristics of the flow field for sideslip and yawed flow conditions.

The lateral tests are required because it is necessary to assess the strong influence of the slipstreams on the lateral characteristics that result from the proximity of the slipstreams to the fuselage and vertical tail. This is especially true of testing for "engine-out" stability and control.

Dynamic testing is required to assess the time lag effect as disturbances at the propeller are convected in the slipstream since such changes could have strong effects on the stability and handling characteristics of the configuration.

All of the above testing should be carried out both in and out of ground effect.

5.6.2 Development of Prediction Methods

The lack of a suitably complete and accurate mathematical model of the propeller slipstream is at least partly responsible for the inability of the resulting wing-flap-slipstream methods to accurately predict the flow field characteristics and, in particular, lift and drag of the wing-flap-slipstream system.

It is important to find a practical technique to predict the propeller forces and moments in the presence of a non-uniform inflow, as can occur at high angles of attack and large values of circulation in the proximity of a wing-flap system. Of equal importance with the propeller forces and moments is the corresponding slipstream vector distribution within which the wing and flap must work.

For an analytical solution to the problem of predicting the flow fields two approaches are possible.

The mathematical model for the first approach should consist of a bound vortex representation of the propeller with trailing vortices to simulate the wake. For the flapped wing a vortex lattice representation is recommended rather than the Weissinger lifting surface which is probably insufficient for all but the simplest plain flapped configuration.

The second approach is similar to the first, except that some of the propeller wake parameters (eg; wake shape, contraction and vortex spacing downstream) may be specified. This approach must await sufficiently detailed test data not now available on the wake characteristics and on the identification of parameters which most determine the wake shape.

Empirical techniques for the calculation of overall lift and drag characteristics have been developed from the simplest up to a fairly comprehensive level without making large gains in the level of accuracy. Items that have typically been omitted but which should be included in these methods include:

The effect of propeller normal force on the induced velocity in the propeller slipstream. Even when the normal force is sufficiently small to be negligible the induced velocity resulting from it can make a significant change in the downwash angle in the slipstream.

The effect of propeller rotation on the variation of downwash in the slipstream. This has been considered (in Reference 21) by assuming a solid body type of rotation of the slipstream with some measure of success;

The effect of the location of the slipstream relative to the wing. This aspect is worthy of study particularly if the case arises that the axis of the propeller slipstream is far away from the wing chord line (of the order of half a diameter, say);

The spanwise influence of propeller-induced flow over the wing. These data may be obtainable empirically from test data or analytically by vortex-based analysis of the flow about wing tip panels outboard of a slipstream, or about wings operating behind overlapped propellers with gaps between them.

5.7 CONCLUSIONS

The feasibility of the deflected slipstream concept has been demonstrated by a considerable amount of model test data and by full scale flying hardware both in test flights and in simulated operations.

Some important test data required for the accurate prediction of performance and stability characteristics are not available and accurate analytical methods of prediction have not yet been developed. Present empirical methods are of sufficient accuracy to permit conceptual design studies with confidence only when model test data is available that is not too dissimilar from the configuration being studied.

No technical problems are foreseen to prevent the acquisition of the required test data.

Development of a realistic mathematical model for the analytic solution of the propeller-wing-flap system and its flow field is, in principle, a fairly simple matter. However, the solution of the equations resulting from a truly comprehensive mathematical model will require large amounts of computer time and storage, and empirical methods which can produce solutions quickly and inexpensively are still needed in the preliminary design process.

6. A COMPARISON OF LIFT/PROPULSION CONCEPTS APPLIED TO A MEDIUM STOL TRANSPORT

6.1 INTRODUCTION

The externally-blown flap concept is a leading contender for a medium-sized STOL transport (MST) because of the relative simplicity by which it achieves powered lift. In this section the technology of the EBF concept is compared to that of four other lift propulsion concepts which could be applied to an MST configuration:

Internally Blown Flaps
Augmentor Wing
Direct Jet Lift
Mechanical High Lift Devices

For each of these, the material is assembled here as a guide to the comparative evaluation of proposed activity in the design, analysis or testing of a medium STOL transport. Section 2 contains a description of the factors which are common to all of these concepts in that application. This section treats each lift-propulsion concept separately in terms of data credibility and limitations and criteria for a comparative evaluation.

The externally blown flap is treated in detail in Section 4. Those data are not repeated here, but only referred to in comparison of the externally blown flap with other concepts.

6.2 DATA CREDIBILITY AND LIMITATIONS

6.2.1 Externally Blown Flaps

The state of the technology of external blown flaps as well described in Section 4. Briefly, the status is this: there are recent wind tunnel tests of good quality which may be applied to specific configurations of the concept; parametric test data are still needed, test facilities and techniques are available which can produce high quality data; this report contains a method of calculating lift, drag and pitching moment to fill a void noticed in the literature; and the feasibility would now appear to be only contingent upon a development of current turbofan transport design.

Typical NASA data are shown in Figure 60 to illustrate the large increase in lift forces available with externally blown flaps. These untrimmed lift coefficients are achieved with thrust coefficients representative of all-engine operation during short takeoff and landing. Pitching moments which result from this lift are large but are consistent with the chordwise centers of pressure of mechanical flap systems. The lift coefficient as used here includes the large vertical thrust component as defined in Figure 61. Some effects of configuration geometry upon these components are

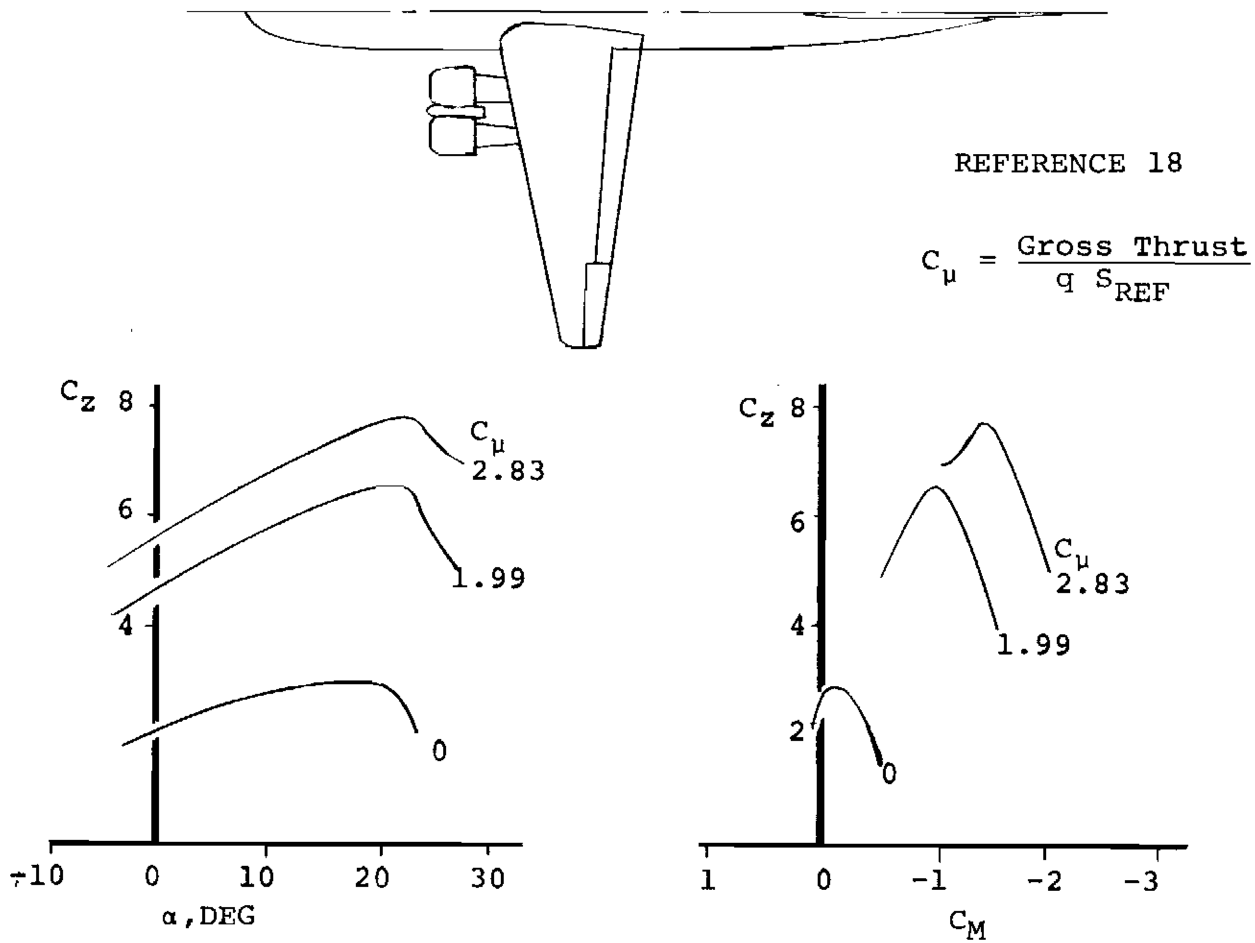


FIGURE 60. EXTERNALLY BLOWN FLAPS LONGITUDINAL CHARACTERISTICS

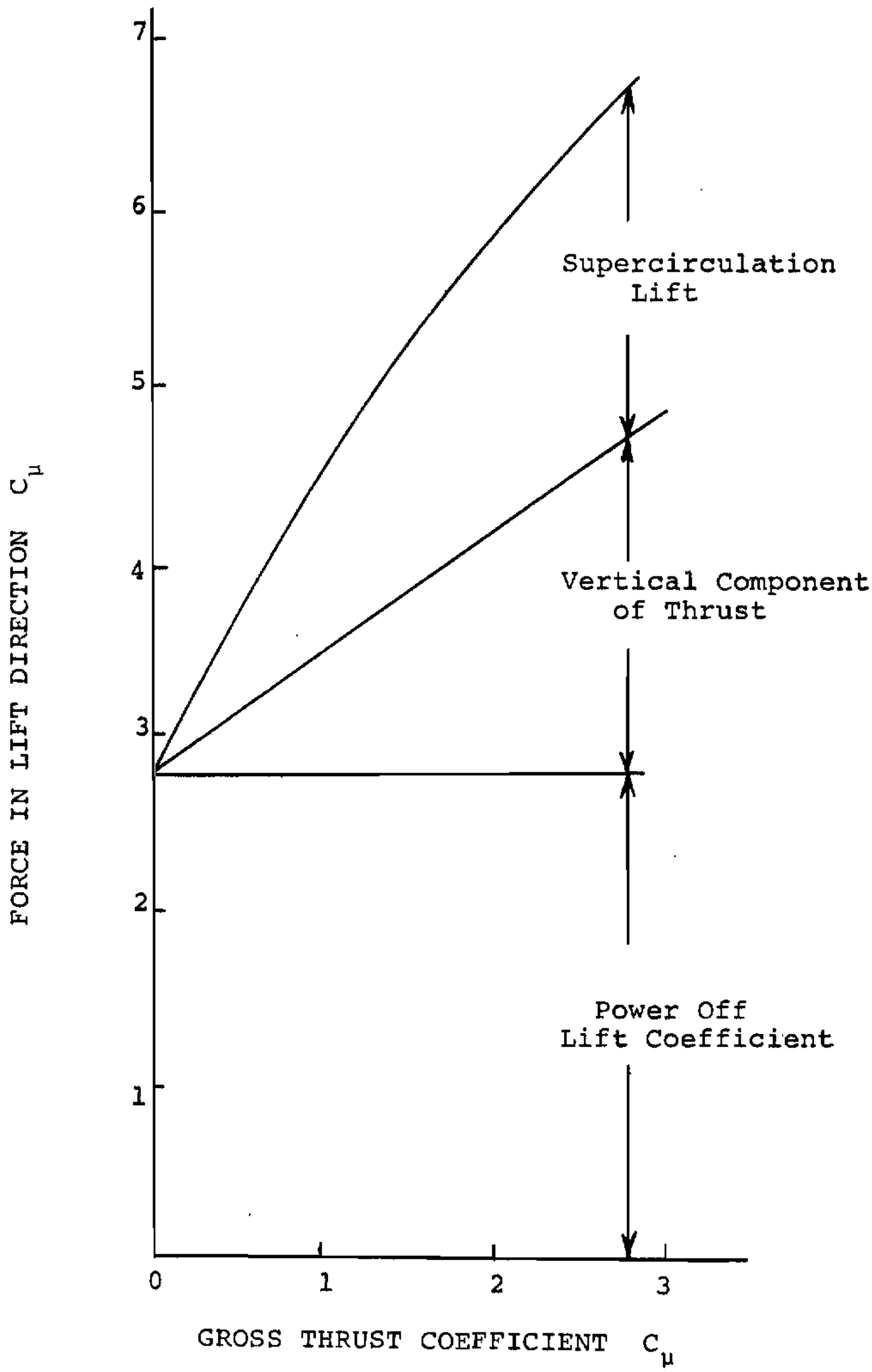


FIGURE 61. COMPONENTS OF LIFTING FORCE

shown in Figure 62. It is seen that the more even spanwise distribution of thrust results in improved lift increments. However, in Section 2, it is noted that failure of engines placed at outboard stations results in large rolling moments. These considerations affect the spanwise positioning of engines for an actual aircraft design.

The engine-out condition on the externally blown flap results in the loss of the lift associated with the dead engine, as well as a rolling moment associated with that lift loss. The rolling moment problem will be most severe at or past stall since there will be an asymmetric stall with the "dead" wing stalling first as shown in Figure 63. Lateral control power may place design limitations on the permissible spanwise location of engines. Engine-out roll can be minimized by moving the engines well inboard or by using a number of smaller engines distributed along the span.

Section 2 indicates that ground effects are unknown but potentially very important. The ability to flare in ground effect and reduce rate of sink to an acceptable level at touchdown may limit the usable lift level.

Although, conceptually, lift coefficient with power is limited only by the amount of thrust we choose to use, the level that can be used operationally is limited. For conventional takeoff and landing (CTOL) aircraft, usable lift has been limited by speed (or resulting "g") margins, minimum control speed criteria, climb capability, longitudinal trim capability, etc., as discussed in Section 2. For the externally blown flap configuration, appropriate margins and criteria must be determined for safe flight operation and it is necessary that their effect on performance be determined. Since C_L is dependent on engine operation, consideration of engine-out lift loss and lift loss due to trimming engine-out rolling moment must be made in determining the permitted operational lift level.

6.2.2 Internally Blown Flaps

Internally blown flaps have been extensively treated in the literature on boundary layer control, both analytically and experimentally in both model and full scale. The many service applications have demonstrated feasibility of the primary concept. For STOL aircraft, both leading edge and trailing edge blowing (see Figure 64) are likely to find application. Figure 65 shows the progress which has been made in blown flap design and the increase in section lift coefficient due to various amounts of flap blowing coefficient. The most efficient amount of blowing is that which is just sufficient to prevent flow separation and which is applied to the best available mechanical high lift design. Increases in lift beyond the knee of the curve are due to the jet reaction component and supercirculation effects. Experiments have shown that to obtain maximum effectiveness from the larger blowing coefficients, the blowing should be applied not only to the trailing edge, but a portion of the blowing should be used at the leading edge, even if mechanical leading edge devices are used. Figure 66 shows that an increase of maximum lift coefficient of

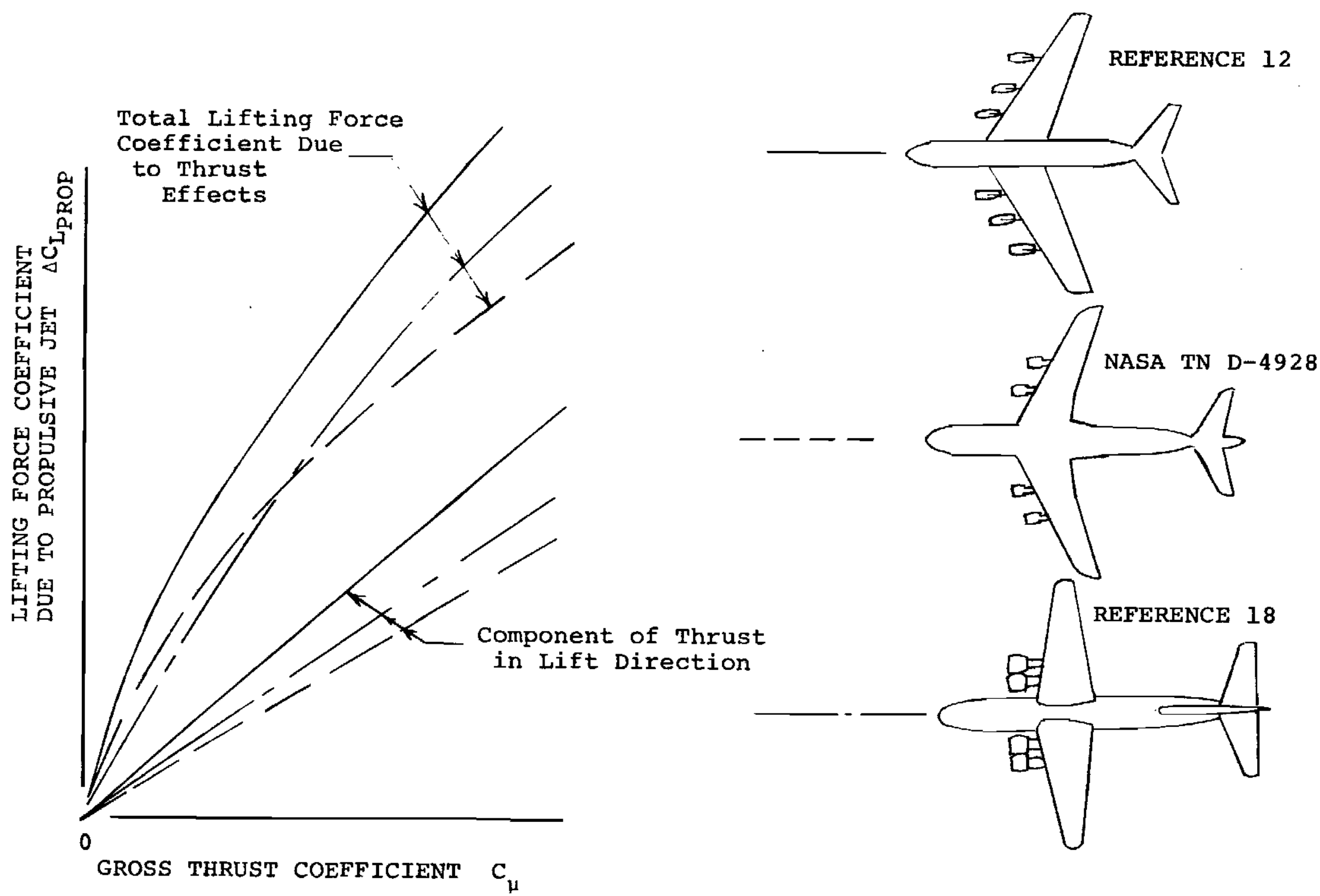


FIGURE 62. LIFT DUE TO THRUST

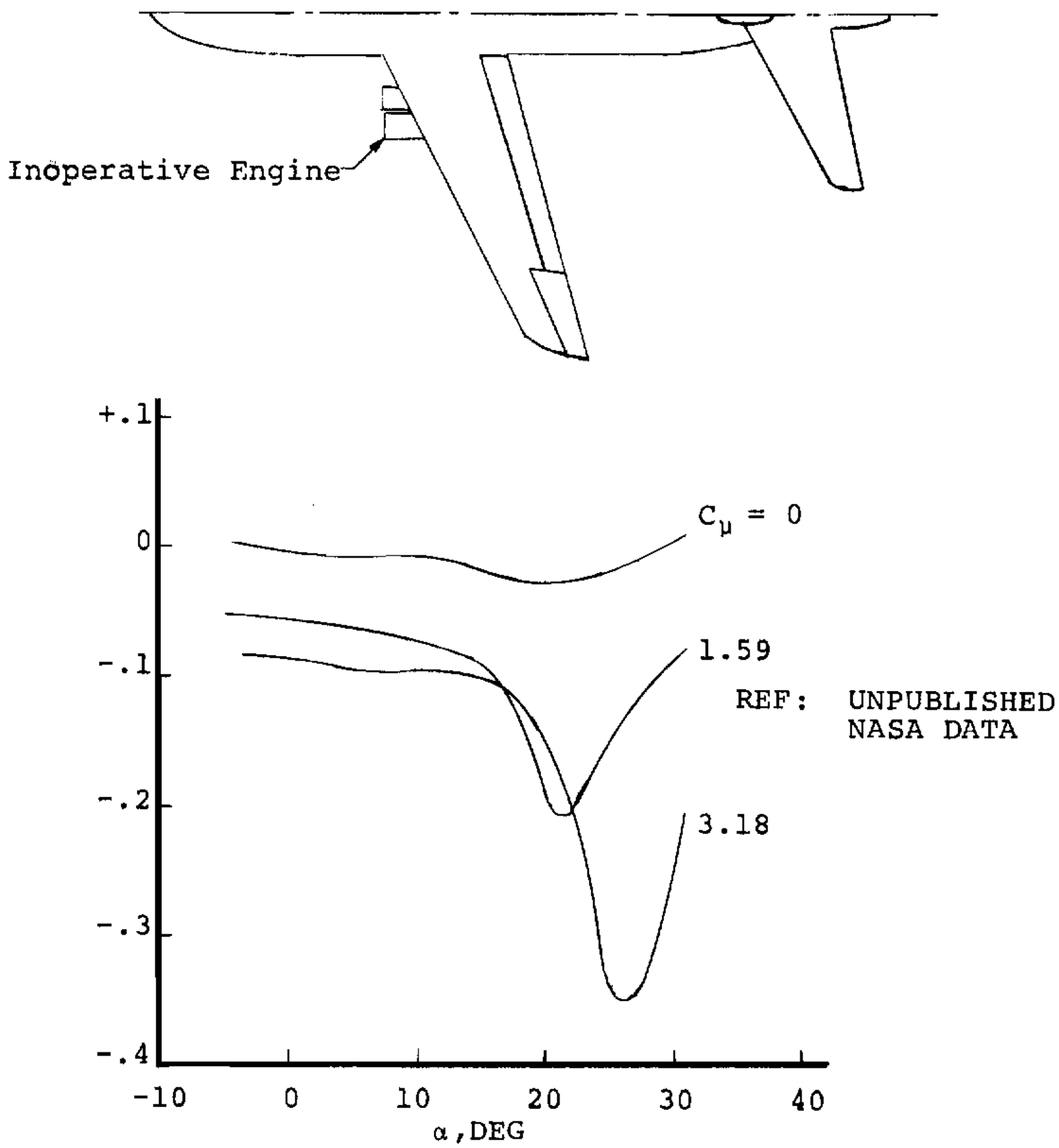


FIGURE 63. ENGINE OUT ROLLING MOMENT

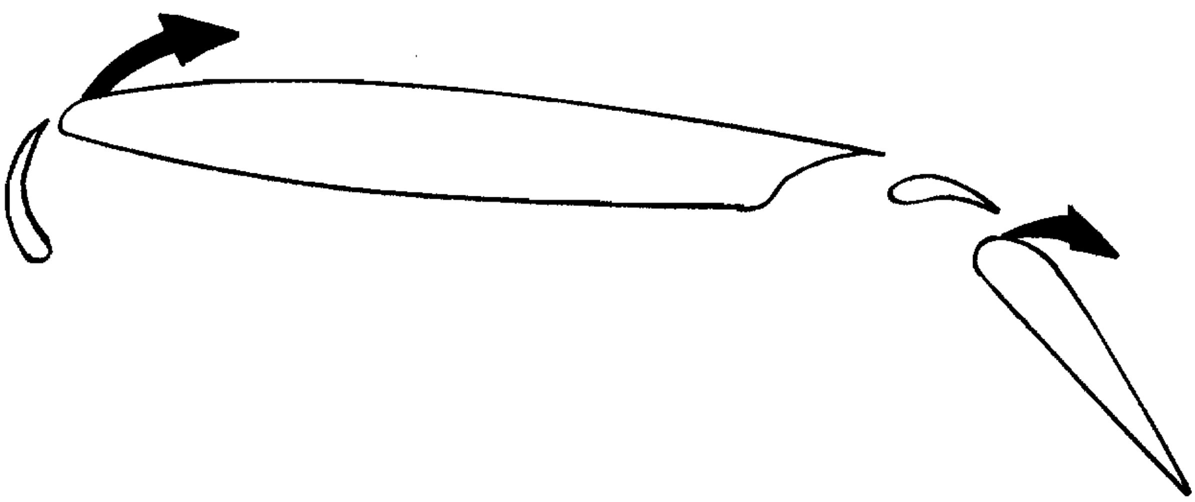


FIGURE 64. INTERNAL BLC

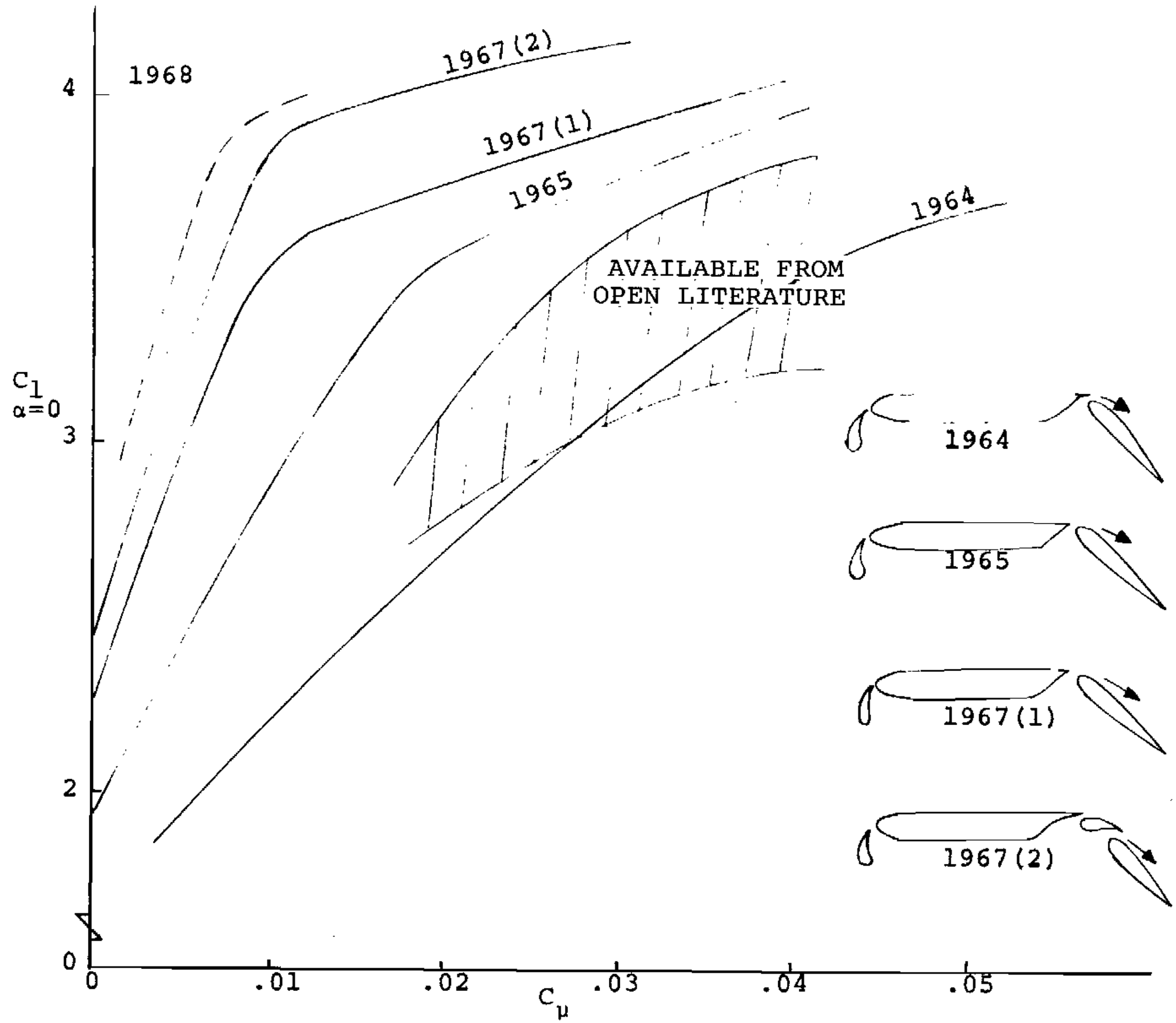


FIGURE 65. BLC FLAP EFFECTIVENESS

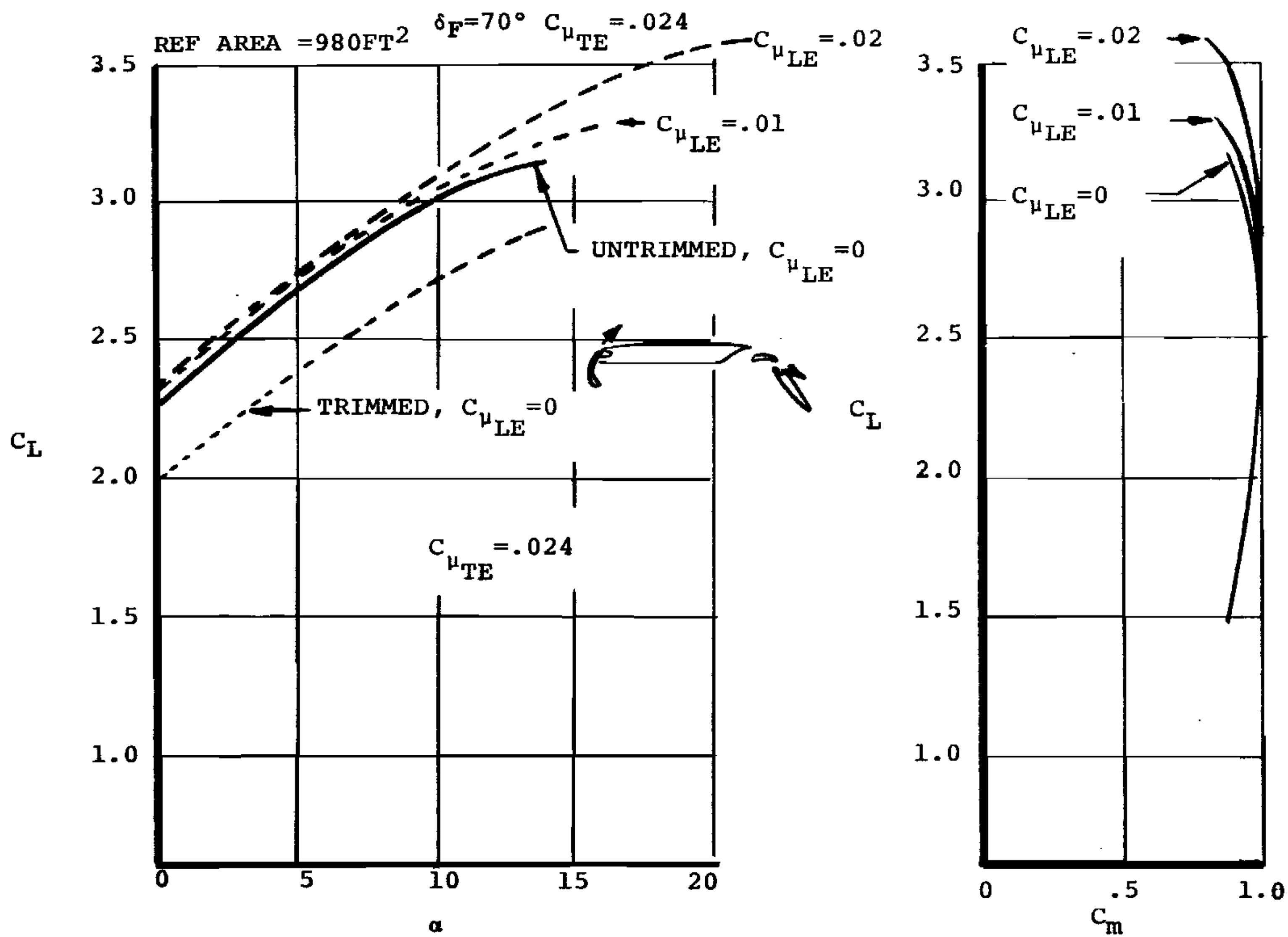


FIGURE 66. AERODYNAMIC CHARACTERISTICS FULL SPAN, DOUBLE SLOTTED FLAP

0.4 resulted from leading edge blowing behind a leading edge slot when used with a blown, double-slotted trailing edge flap.

The use of trailing edge and leading edge blowing together permit the achievement of a desirable range of pitch attitudes and angles of attack for use during takeoff and landing. The use of very high blowing coefficients and correspondingly high circulation lift coefficients requires that ground effects be investigated. When the large blowing coefficients are used, the problem of providing sufficient lateral control becomes acute, and is often compounded by the desire to use a full span high-lift system. Blown lateral control devices, other than those demonstrated already on such aircraft as the NC-130 and augmentor wing demonstrator will require wind tunnel and functional tests to prove effectiveness and practicality.

6.2.3 Augmentor Wing

Augmentor wing technology has developed rapidly, drawing upon jet flap and BLC background, although distinct from either, and drawing upon large-scale wind tunnel testing. Geometry of the augmentor wing flap is depicted in Figure 67, which shows that the jet which issues from the nozzle is directed not to attach to a surface as in the usual BLC, but to mix with entrained air from the slots provided, so as to obtain ejector action. At the present time there appear to be no analytical methods especially applicable to augmentor wing configurations.

Typical longitudinal characteristics of an augmented jet flap wing are shown in Figure 68. These are unpublished NASA test data and were obtained in the Ames 40 by 80 foot wind tunnel on a 44.15 foot wing span model that geometrically simulated a CV-7A aircraft with an augmented jet flap extending over 55 percent of the wing span. Blown ailerons extended from the flap to wing tip. The compressed air for the augmentor was supplied by axial flow compressors with their turbines driven by exhaust gases of a jet engine. A J85 turbojet was installed under each wing with an exhaust diverter valve simulating a rotating type nozzle with capacity for vectoring thrust aft for takeoff and cruise and downward for approach and landing. With the augmentor flap deflected but with augmentation flow off, the $C_{L_{MAX}}$ is about 2.3. With jet augmentation the $C_{L_{MAX}}$ is 5.7 untrimmed and with the addition of 1500 lbs. of thrust vectored 85° downward the $C_{L_{MAX}}$ becomes 6.8 or an increase in C_L of 1.1. This is about 35 percent higher than just the 1500 pounds of thrust would produce as a vector thus indicating an increase in circulation lift due to the vectored jets.

The two curves labeled "early flap design" are for a more complicated augmentor flap design and show that the performance of the present simplified flap is superior. Blowing the knee of the early flap showed insufficient improvement in lift to warrant the added complexity.

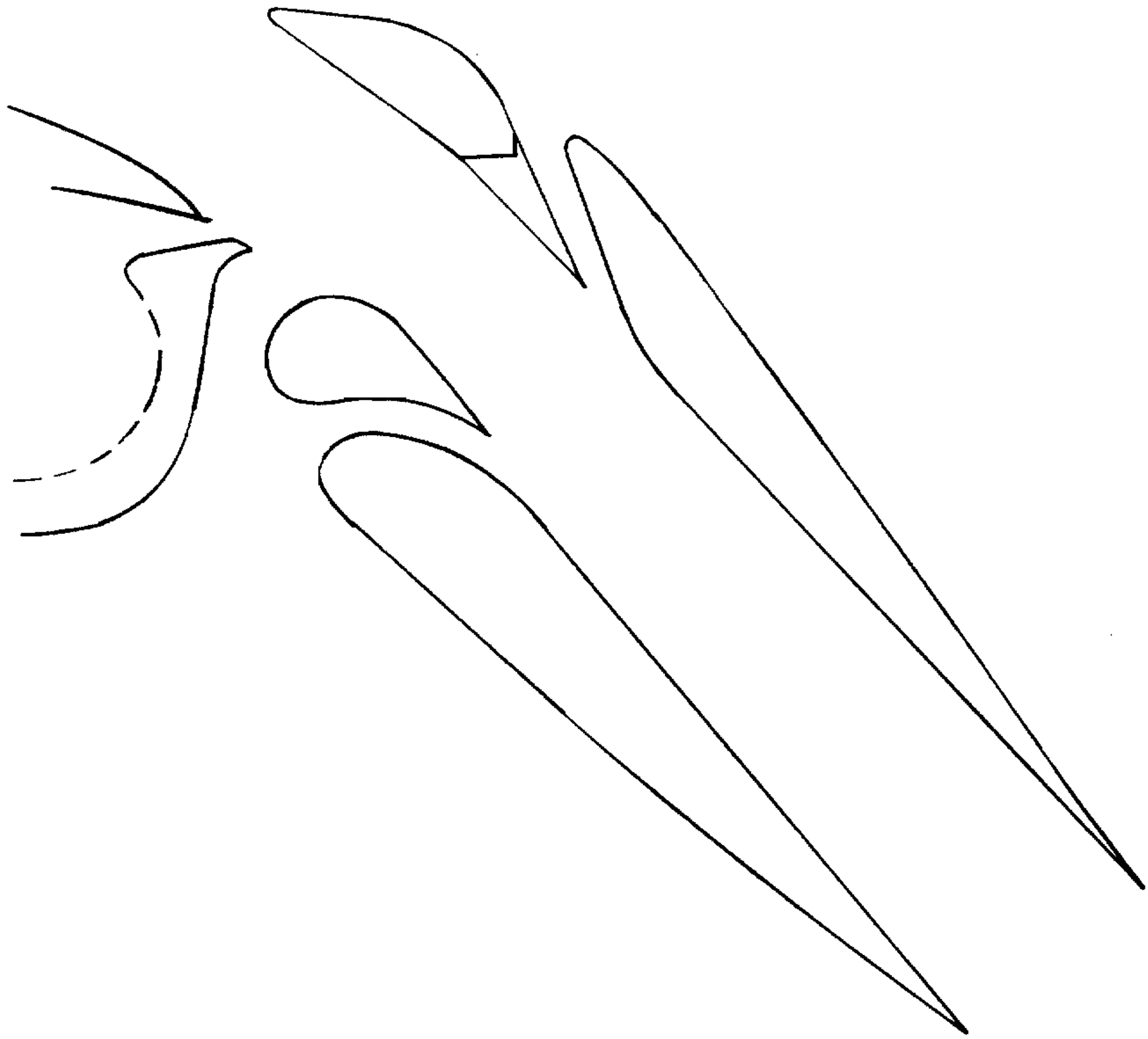


FIGURE 67. AUGMENTOR WING

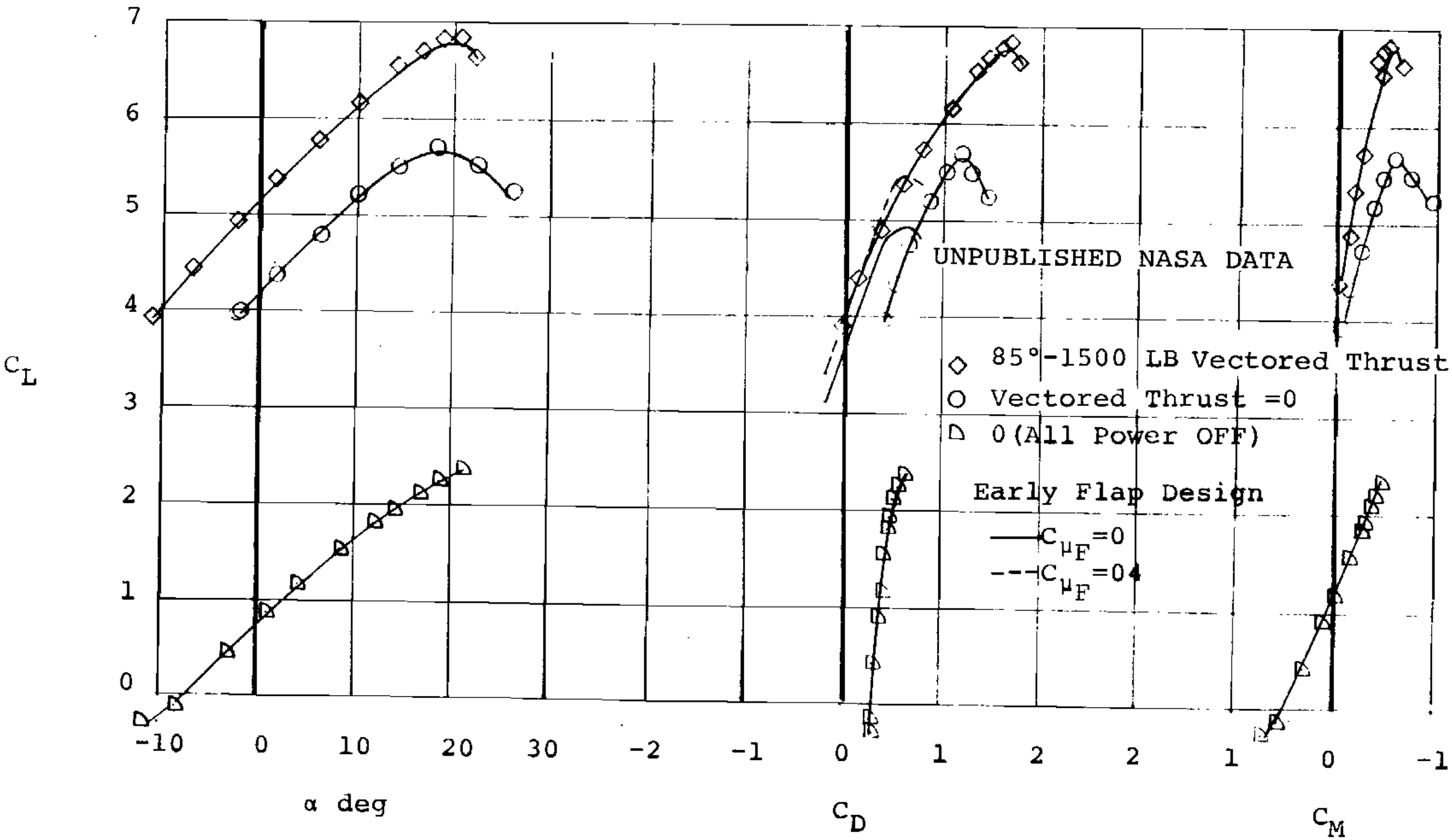


FIGURE 68. AUGMENTOR WING LONGITUDINAL CHARACTERISTICS

One of the strongest points favoring the augmentor wing is that it produces high lift coefficients with relatively low increases in pitching moments compared to most other high lift concepts. For example, at constant α, as the amount of blowing is varied to increase lift coefficient the ratio of $\Delta C_L/\Delta C_M$ is 8 to 10 while the $\Delta C_L/\Delta C_M$ for a slotted flap is about 3. This means that for a given increase in wing lift coefficient the amount of lift that must be sacrified to trim the airplane with an augmentor wing is about one third of that for a slotted flap. Therefore the useable lift is higher.

Using elevator effectiveness from the above test data, the pitch acceleration capabilities of a typical 40,000 pound aircraft employing an augmentor wing were calculated for a sea level standard day as shown in Figure 69. The aircraft is in the landing configuration with the augmentor flap deflected 75 degrees. The line drawn through the calculated points represents the control power from a conventional tail that is capable of a $\Delta C_{L_{TAIL}} = .55$ at full deflection. Indications are that the requirements of Reference 1 would be satisfied above a velocity of about 69 knots, which is a speed constant with STOL operation from field lengths of about 2000'.

The augmentor wing lateral control data presented in Figure 70 are also from the unpublished NASA tests and are for the landing configuration with the flap at 75 degrees and the starboard aileron drooped to 45 degrees. The port aileron deflection is varied from its normal droop of 45 degrees landing position to 0 and 65 degrees.

With the ailerons operating in this manner at all times data are presented showing C_ℓ, C_n and C_y when inboard spoilers are applied and when the augmentor on one wing panel is throttled to produce roll control. Two cases of blockage were tested-25 and 50 percent of the outboard flap semispan. In both cases 75 percent of the augmentor exit area was blocked by means of wedges.

The data show that a maximum rolling moment coefficient of about .115 could be attained with the ailerons alone. The ailerons and inboard spoilers combined produce $C_\ell = .2$.

Throttling the augmentor 25 and 50 percent of its span, and applying aileron, produced maximum rolling moment coefficients of about .21 and .26 respectively.

Using the lateral control data of the unpublished NASA test previously referred to, the rolling angular acceleration provided by each individual roll control device was calculated for a 40,000 pound airplane at sea level with the results shown on Figure 71. The aileron was operative when the augmentor was throttled and also when the spoilers were tested so that the rolling effectiveness of aileron plus spoiler and aileron plus throttled augmentor can be obtained by adding the components. The spoilers were not tested while throttling was applied so their interference effects on each other are not known. However, it will probably be sufficiently accurate to

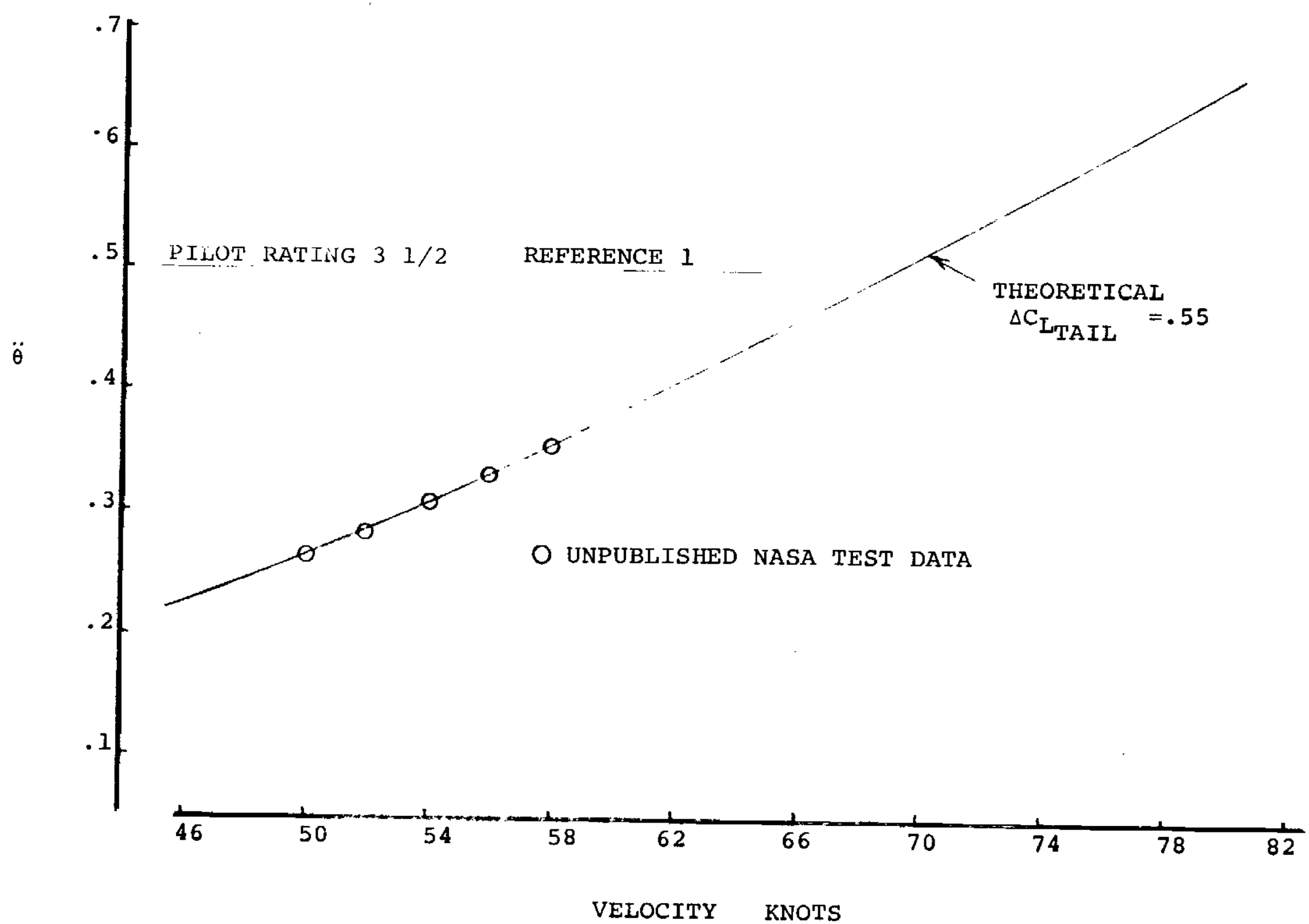


FIGURE 69. AUGMENTOR WING LONGITUDINAL CONTROL - CONVENTIONAL ELEVATOR TYPICAL 40,000 LB. AIRPLANE

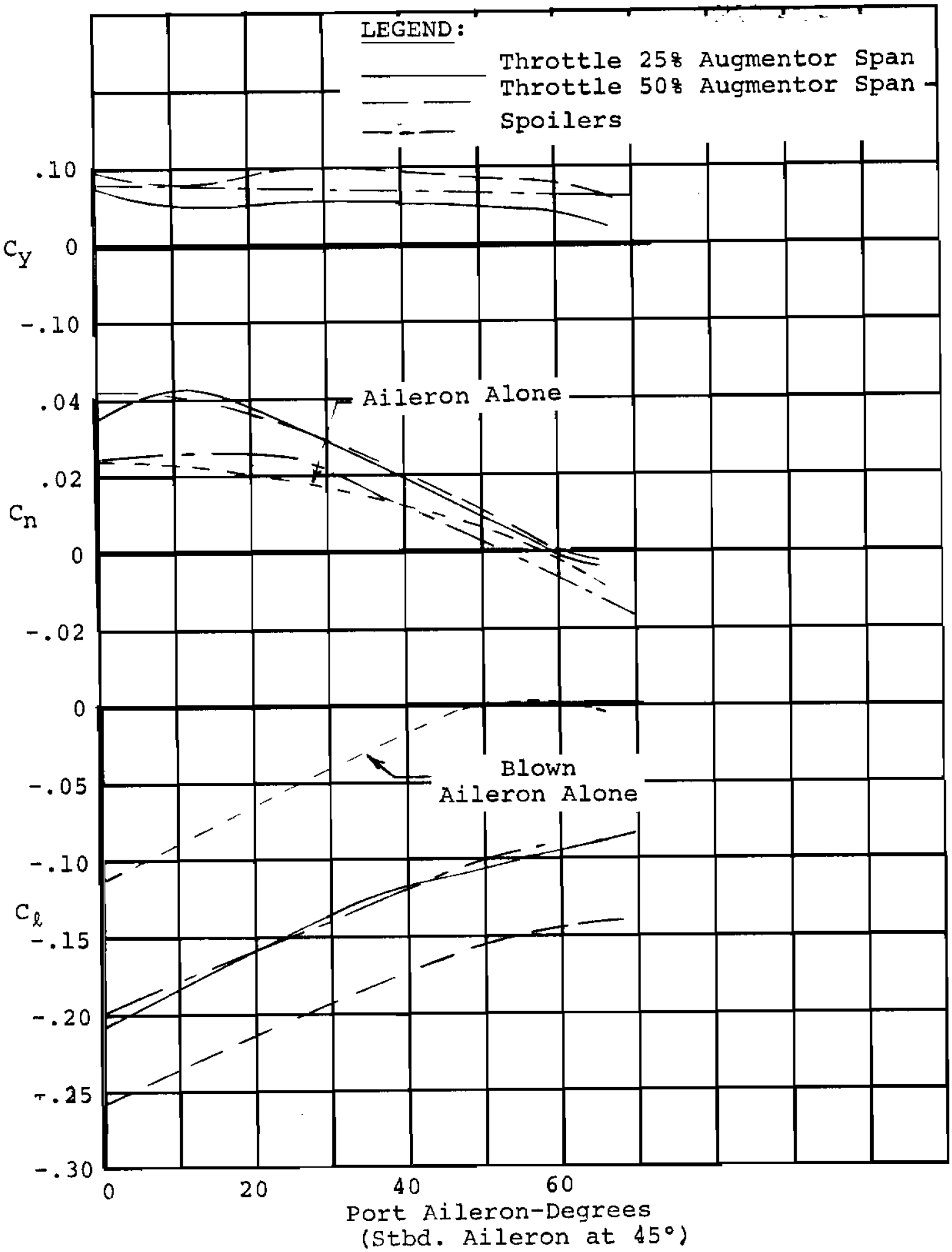


FIGURE 70. AUGMENTOR WING - LATERAL CONTROL
(UNPUBLISHED NASA TEST DATA)

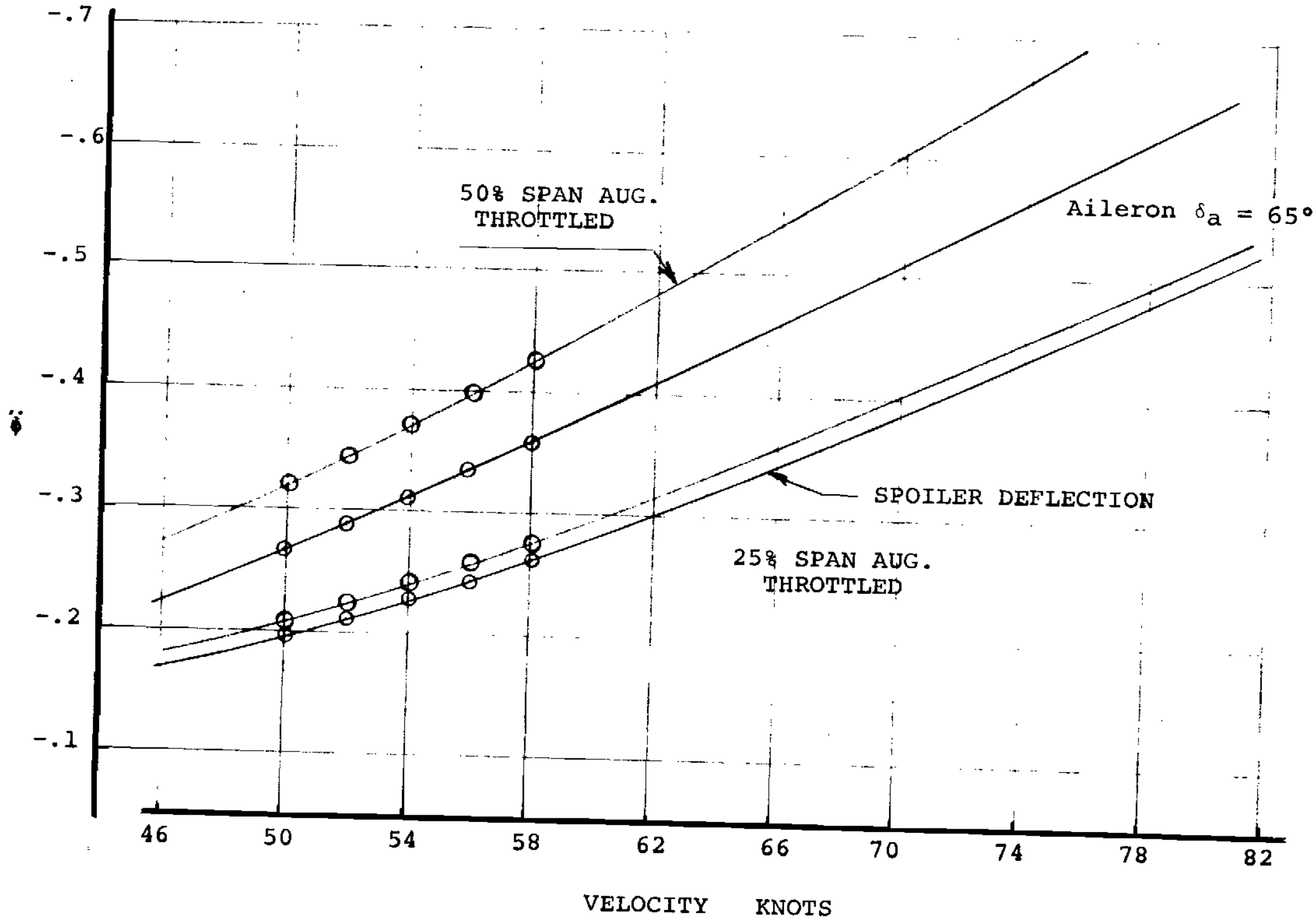


FIGURE 71. AUGMENTOR WING LATERAL CONTROL TYPICAL 40,000 LB. AIRPLANE

obtain the rolling acceleration of aileron, spoiler and throttling by simple addition. To obtain sufficient roll control, all three devices will probably have to be used.

STOL minimum speed margins of augmentor-wing aircraft must allow for reductions in maximum lift consistent with provision of lateral control. Lift losses obtained from the wind tunnel tests discussed above showed these values as increments:

	$\Delta C_{L_{MAX}}$ LOSS
0°/65° blown aileron	.6 to .8
Spoilers outboard	.1
Spoilers inboard	.4
25% span augmentor throttled	.25
50% span augmentor throttled	.9

Ground effect data on lift are available for one augmentor wing configuration and are shown in Figure 72.

The data were obtained using a fixed ground board. It should be noted from the criteria of Figure 18, that for the $h/\bar{c} = 1.3$ and lift coefficients of 3.85 and 4.4 at which these tests were run, that a fixed ground board is sufficient.

Noise is of concern with all powered lift devices, including the augmentor wing. In this case the source of noise is expected to be primarily the mixing region between the air from the primary nozzle and that entrained through the ejector slots. It is expected that the presence of surfaces on both sides of the mixing region will make the augmentor wing amenable to acoustic treatment.

Tests were conducted in 1970 at NASA Lewis to evaluate the acoustic characteristics of an augmentor wing configuration and an externally blown flap configuration. Large scale models were used, as depicted in Figure 73, to investigate the near and far field noise and azimuthal pattern above and below the flaps. The tests were planned to obtained additional data on turning effectiveness, panel flutter and thermal effects.

6.2.4 Direct Jet Lift

The direct lift engine concept appears to be a simple selection to the problem of achieving high lift, but is also subject to the same two major difficulties that affect most of the other STOL concepts: low-speed control and ground effects. practical solutions have already been demonstrated for both of these areas.

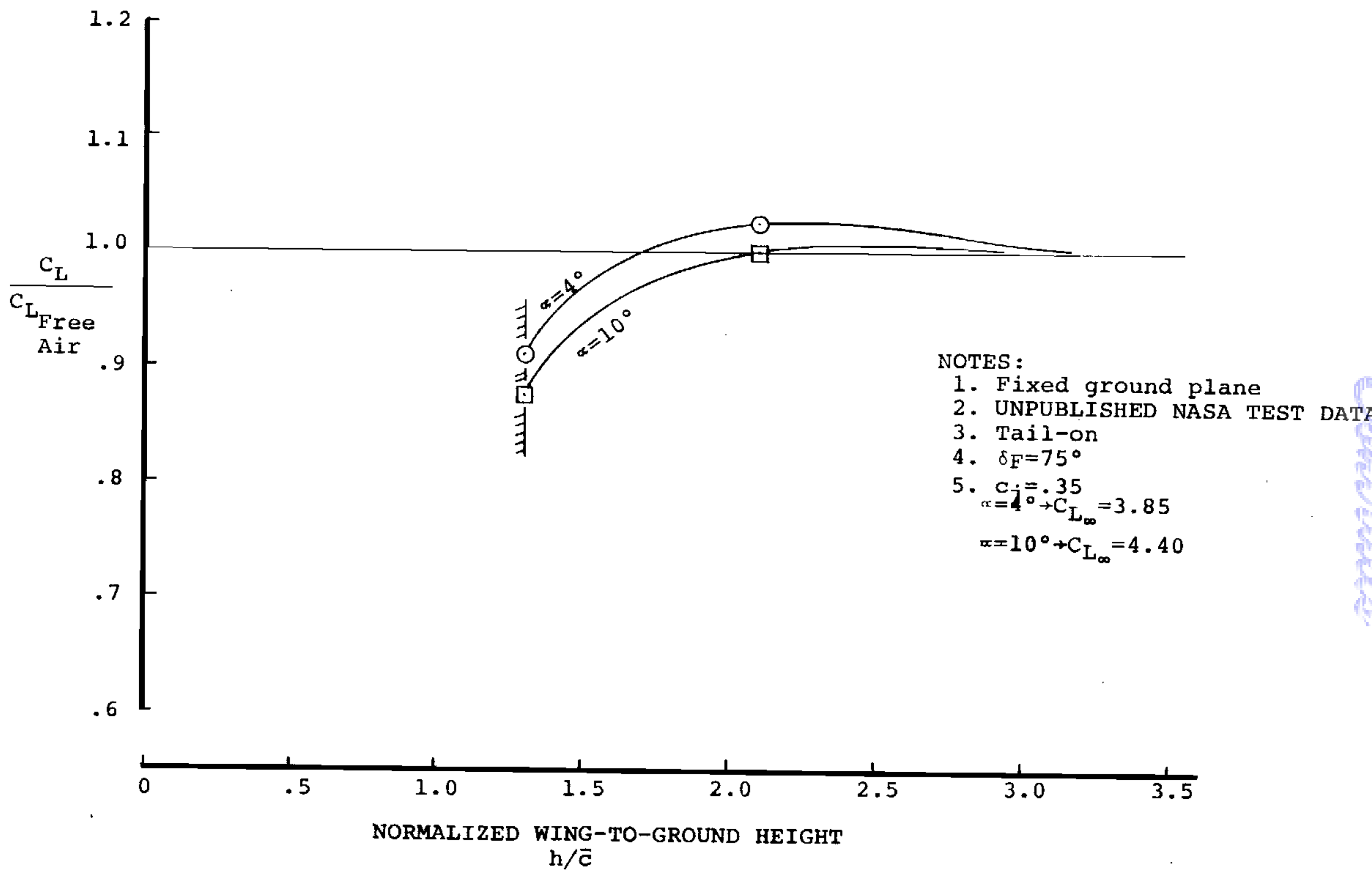


FIGURE 72. AUGMENTOR WING GROUND EFFECT

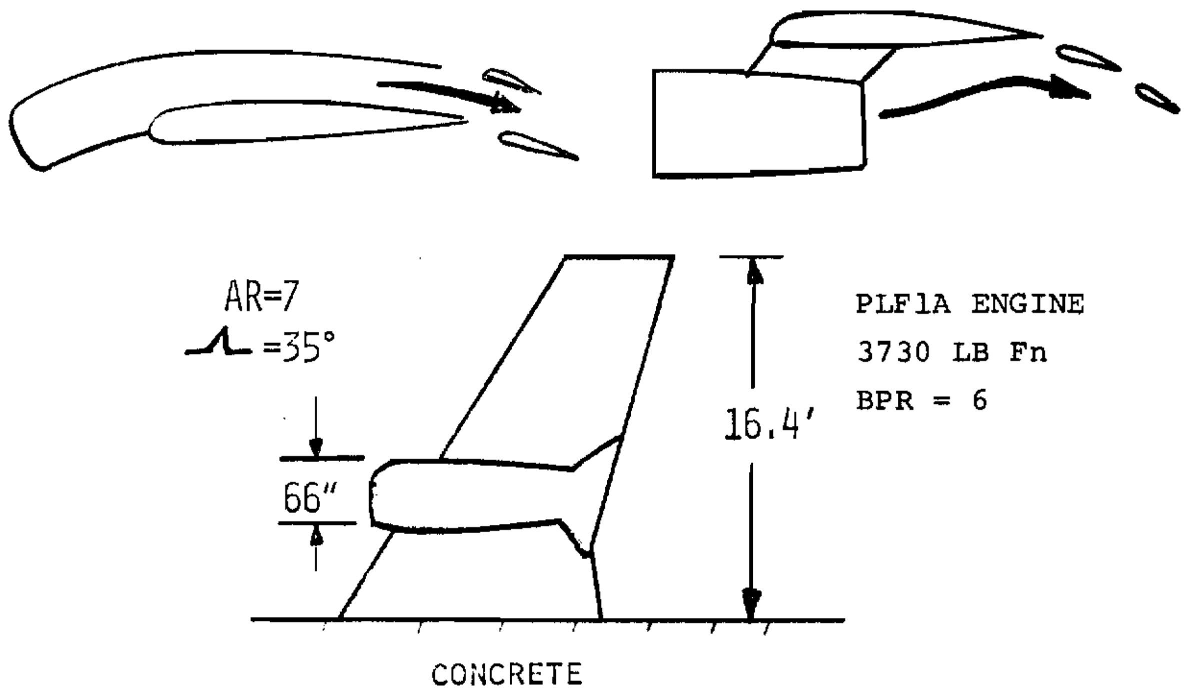


FIGURE 73. NASA LEWIS NOISE TEST MODELS

Satisfactory roll control for VTOL (and that should be sufficient for STOL) has been demonstrated by using the main lift engines for lateral control as on the Dornier Do-31 or by using auxiliary reaction controls as on the Hawker-Siddeley P-1127. The category of "ground effects" as applied to lift engine configurations, includes not only the interaction of the lift engine flow upon the aerodynamic lift, drag and moments, but also the recirculation of hot gases or debris from the ground into the engine and the effect on ground erosion.

Direct lift engines can heavily influence the flow field and the aerodynamic characteristics of the vehicle depending upon location of the engine relative to fuselage, wing, aerodynamic high-lift devices and tail surfaces.

For example, there are three major ways in which lift engines cause changes in the pitching moments of an aircraft even if they are concentrated at the aircraft center of gravity as shown in Figure 74. The intake momentum drag and the exhaust thrust component tend to pitch the airplane up. This pitching moment varies with the engine length, thrust produced and aircraft forward speed.

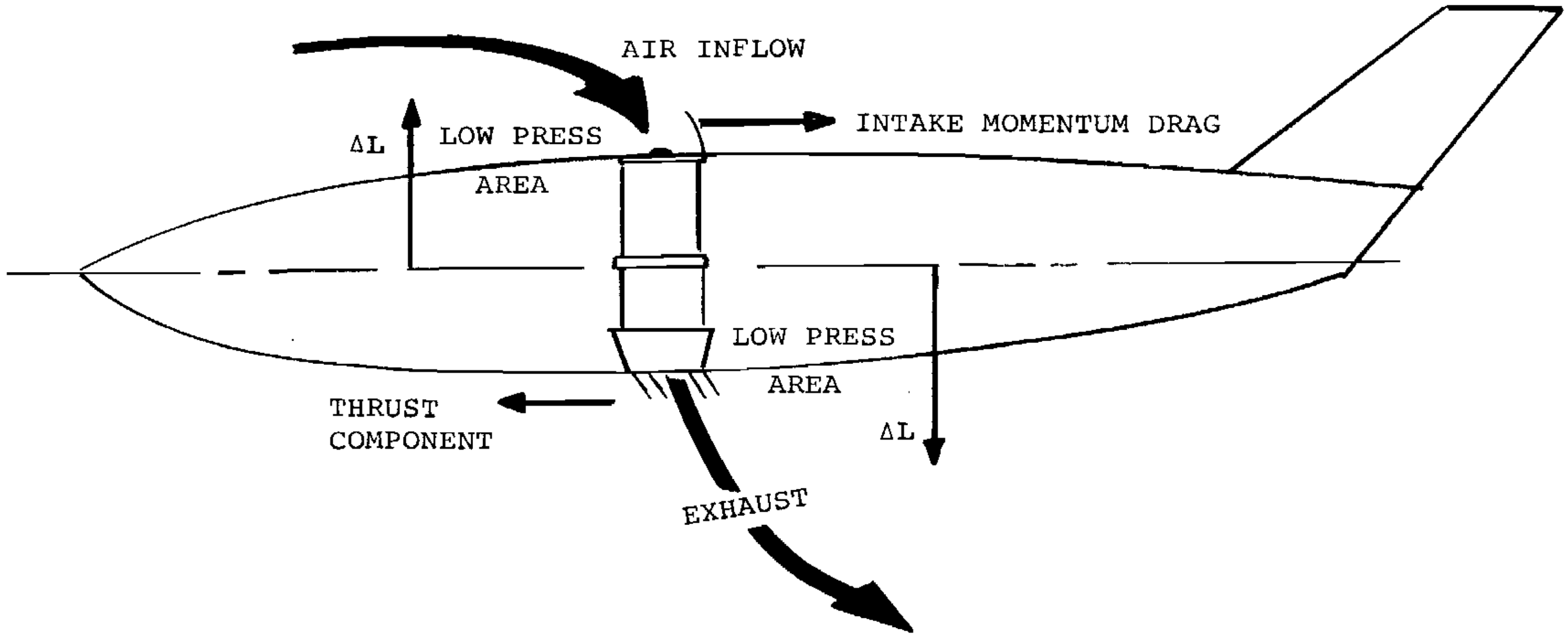
Changes in lift are produced by jet induction on the surfaces fore and aft of the engine inlet and outlet which vary with engine location, forward speed and engine size. The shape and size of the surfaces surrounding the inlet and outlet are also important.

Vorticity generated by the exhaust may complicate stability and trim of the vehicle by causing irregularities in the flow field of the tail. These effects must be considered not only in the aggregate, as when considering a complete configuration, but also in the individual component contributions as when designing and calibrating a wind tunnel model.

Physically, lift engines offer a wide choice of places where they can be mounted. They can be placed in the fuselage or mounted in pods on the wing or fuselage. Because of this versatility in mounting, many things must be considered before selecting a location. If they are placed far from the aircraft's center of gravity they may seriously affect moments of inertia yet this location is desirable if they are used also for longitudinal, lateral and directional control. The pitch and roll moments resulting from engine failure must be considered as well as reingestion and aerodynamic interference in ground effect.

The problem of mechanical installation of engines, and fuel and control lines and of providing thermal protection for structure and other components as well as their maintenance, may be a deciding factor in chosing one location rather than another.

Though the lift engines are only used during takeoff and landing, cabin noise levels should be considered especially if commercial use is anticipated.



- Flow at inlet and exhaust produce pitching moment varying with engine length, thrust produced, aircraft forward speed.
- Changes in lift are produced on surfaces fore and aft of engine inlet and outlet which vary with engine location, forward speed, engine size.
- Vortices generated by the exhaust may strike tail.

FIGURE 74. LIFT ENGINE PITCHING MOMENTS

Work is being done to reduce the noise of lift engines by acoustical treatment. Theoretical predictions of possible noise reduction are encouraging but full scale tests are required for verification.

Figure 75 shows the noise levels for a bypass ratio 0 turbo-jet engine, a bypass ratio 5 cruise engine and a bypass ratio 12 lift fan. The noise level of these three sources shows that the treated cruise engine is still slightly noisier than the auxiliary lift systems. The dominance of fan generated noise tends to make noise levels constant with bypass ratio for a given state of the art. Possibly the future will offer improvements in basic fan noise for cruise engines that will allow use of bypass ratios above 5 in cruise.

The direct lift concept has been proved feasible down to zero speed by flight test of the Do-31.

The addition of lift engines cannot be considered as just "strapping on extra cans of lift" because their high velocity exhausts can cause serious losses in the circulation lift of the high lift wings with which they are associated. This interference is so dependent upon configuration that wind tunnel tests must be conducted for a particular design. The model testing is complicated by the high lift engine exhaust velocities and it may be that inlet velocities should be simulated.

Ground effects can be very critical and they are so dependent upon the particular configuration that tunnel tests with a ground board and perhaps a moving belt are required.

Although theoretical analysis indicates a good noise reduction potential exists for lift engines, a full scale test demonstration should be conducted.

6.2.5 Mechanical High Lift Devices

Mechanical high lift devices - that is, the non-powered-lift category which includes the leading and trailing edge flaps, vanes, slats and similar articles - are well founded in theory and experiment.

Figure 76 portrays the improvement in flap design that has occurred during the years from 1947 to the present. The improvement in maximum lift coefficient of the 737 from 2.6 to 3.53 indicates what a well planned and intense design effort can accomplish. Based on this experience it is estimated that with reduced sweep and a triple slotted flap with about 30 percent Fowler action a $C_{L_{MAX}}$ of 4.0 is attainable. With a full-span high lift device, the provision of lateral control may require some ingenuity, such as the development of mechanisms which produce flaperon action with a triple-slotted flap.

Figure 77 shows data obtained on a straight wing semispan model with a full span, large Fowler action, triple slotted flap. A leading edge slat of 15% chord was

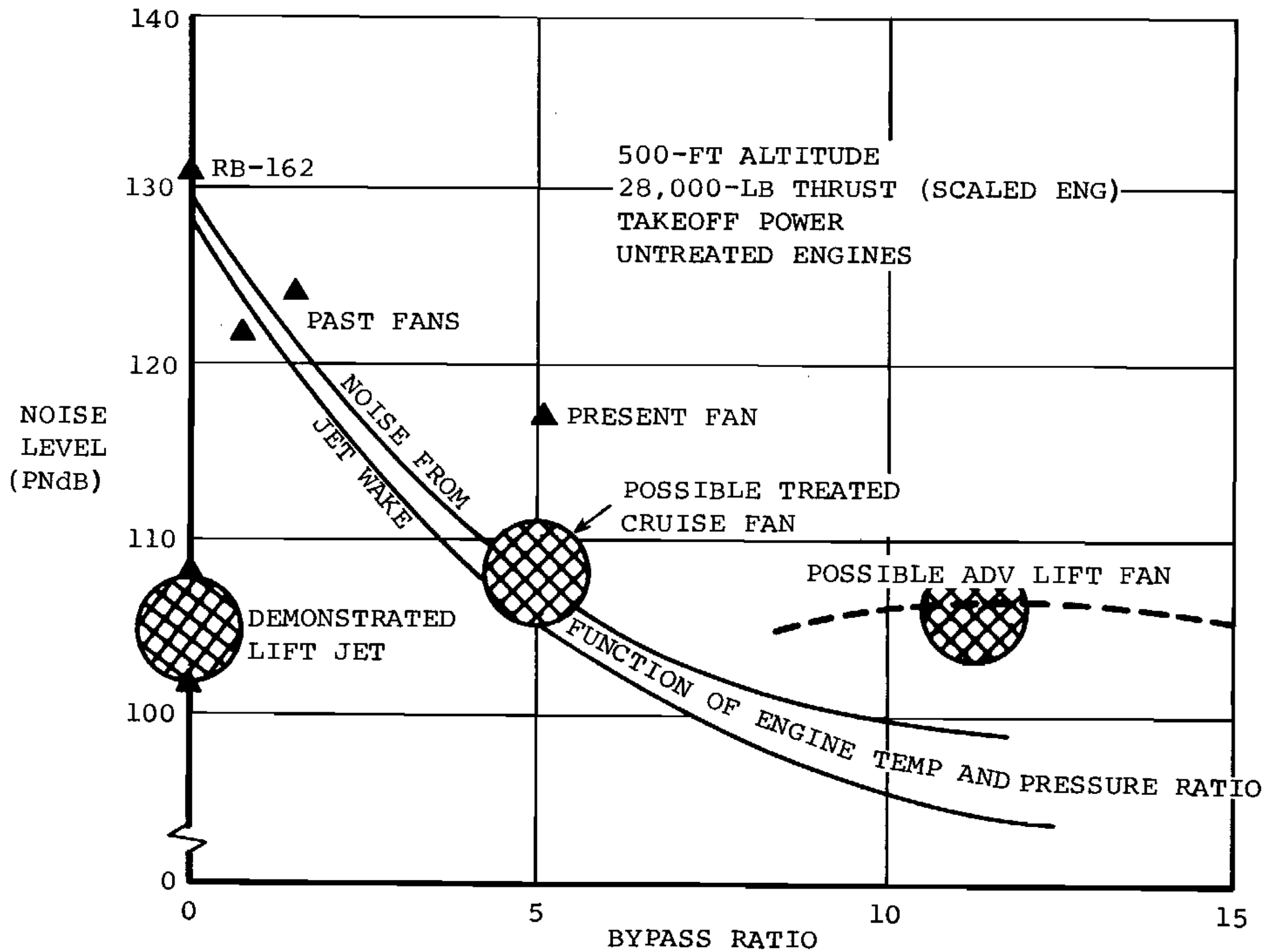


FIGURE 75. NOISE LEVELS - BYPASS RATIO

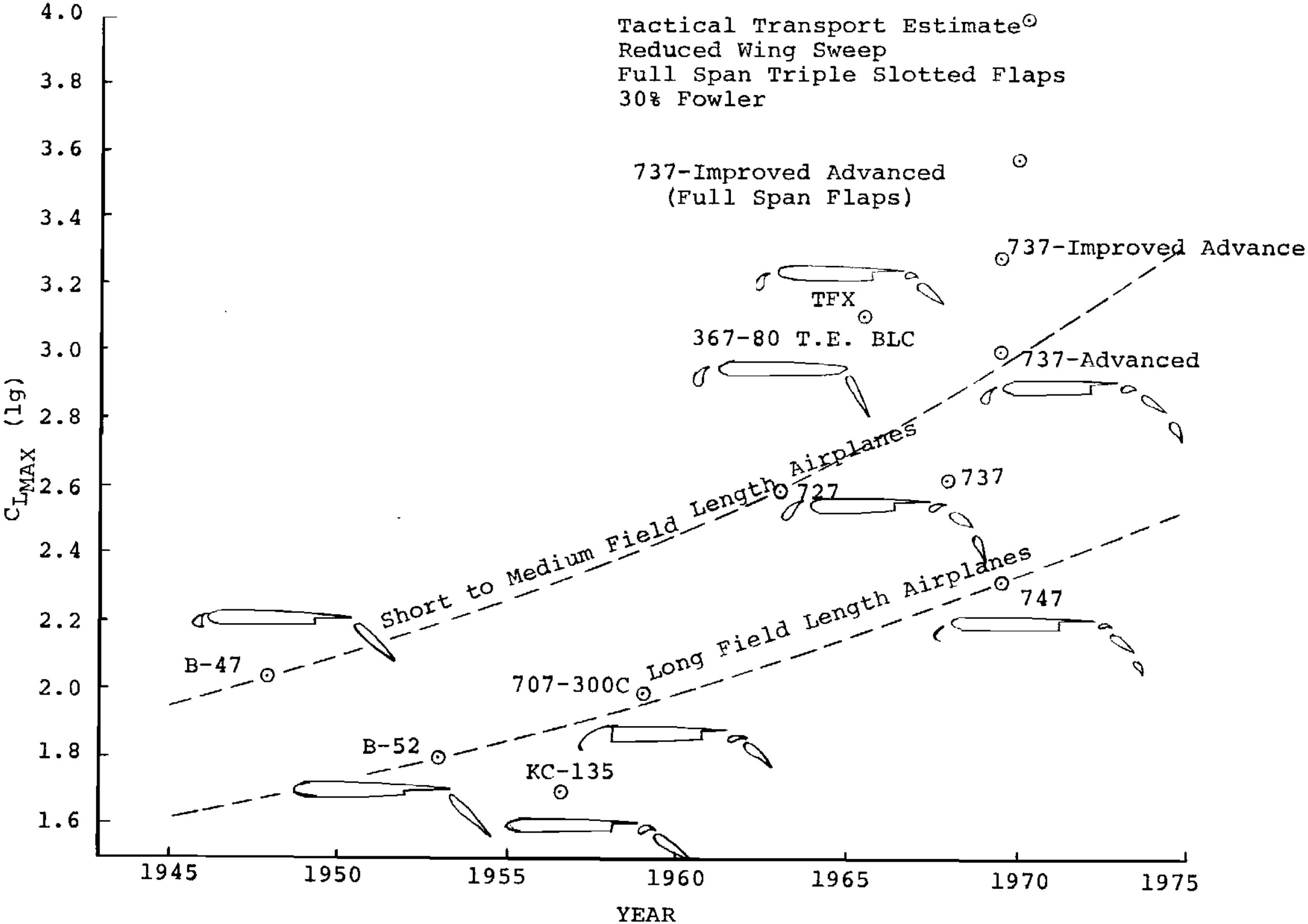


FIGURE 76. MAXIMUM LIFT CAPABILITY OF MECHANICAL HIGHLIFT DEVICES

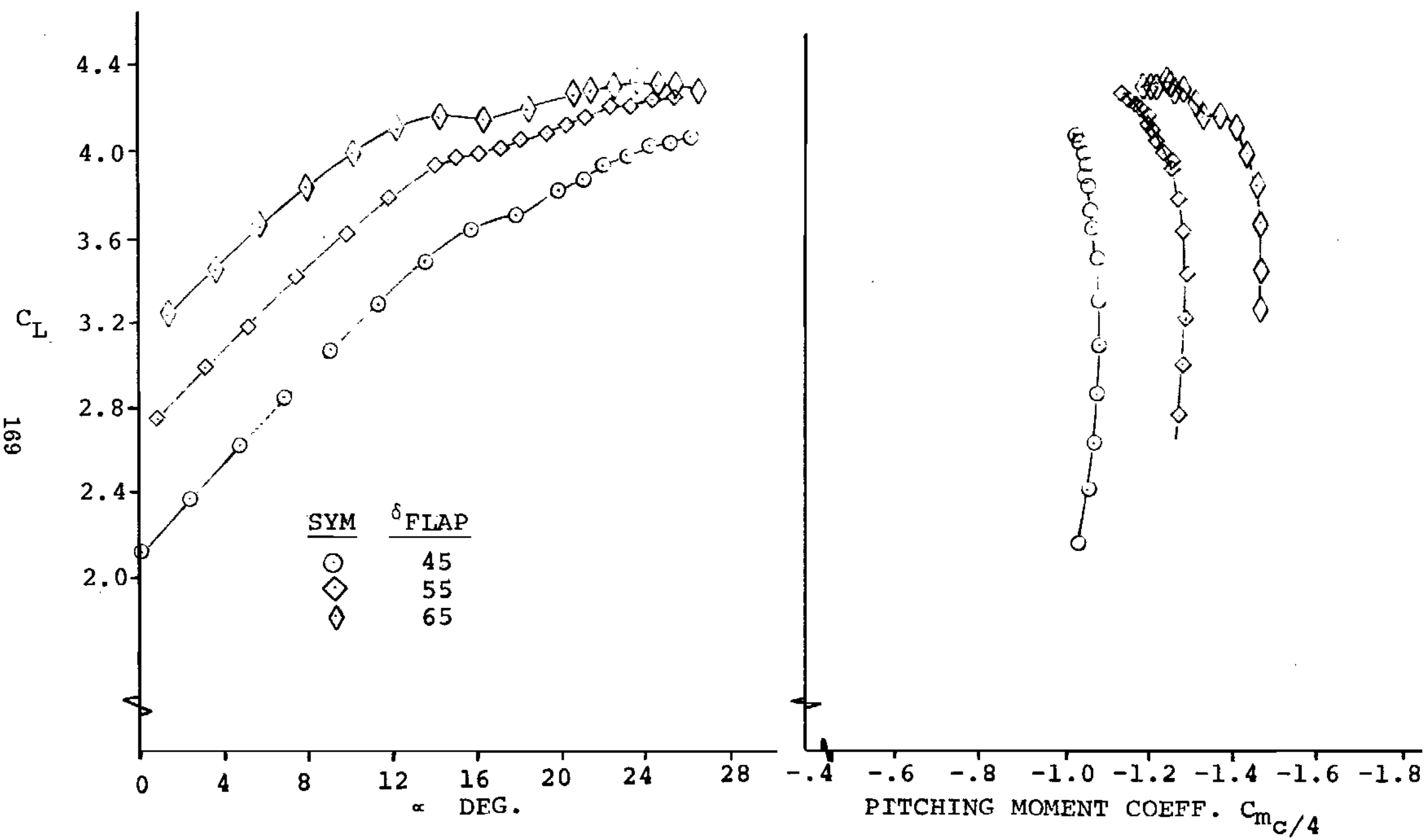


FIGURE 77. TRIPLE SLOTTED FLAPS WITH 20 DEGREES SLAT DEFLECTION

deflected 20 degrees. A maximum untrimmed lift coefficient of about 4.2 was obtained partly due to the 42% chord increase from Fowler action. As is typical with flaps of this type, the large nose down pitching moments result in a large reduction in useable lift when trimmed.

For comparison, lift data were obtained with the same model operated as a double-slotted flap. Model geometry is as shown on Figure 78. The same wing panel was used, but a 35%-Fowler-action double-slotted flap configuration was used instead of a triple-slotted flap. The high untrimmed maximum lift coefficient shown on Figure 79 will be considerably reduced when the high pitching moments are trimmed.

Lift coefficients of this magnitude are achievable with complete aircraft configurations. Figure 80 shows data from Reference 48 which reports test results from a large scale, four propeller STOL airplane with a triple slotted flap and a leading edge slat.

To obtain the high lift coefficients required for STOL operation from a mechanical high lift device will probably require large full span flaps with much Fowler action. This leads to a serious lateral control problem. The lateral control would have to be obtained by using a combination of lateral controls. Sections of the flap could be used as flaperons. Spoilers would undoubtedly be required. If a portion of the wing were reserved for ailerons they would be blown for maximum effectiveness and, for very low-speed operation, it may be necessary to use reaction controls.

The vast amount of data that has been accumulated through the years is for flaps that develop lift coefficients of about 2.8, but very little is available for the higher lift coefficients required for STOL. Test data are needed on control power - especially lateral control and yaw coupling. Ground effects on the aerodynamic coefficients at high values of lift are a subject for further testing.

There is very little ground effect data on mechanical high lift wings at the high values of lift coefficient required for STOL aircraft. Tests have shown that the influence of ground effect increases at higher lift coefficients and also as the trailing edge of the flap gets closer to the ground at high flap deflections.

Wind tunnel tests of mechanical high lift models capable of producing lift coefficients of about four are required. These tests should be made using a moving rather than a fixed ground board where possible.

6.3 CRITERIA FOR A COMPARATIVE EVALUATION

In evaluating the relative merits of STOL aircraft designs based upon the several lift/propulsion concepts, important differences in the concepts will be apparent from an appraisal of cost effectiveness, reliability, maintainability and a host of

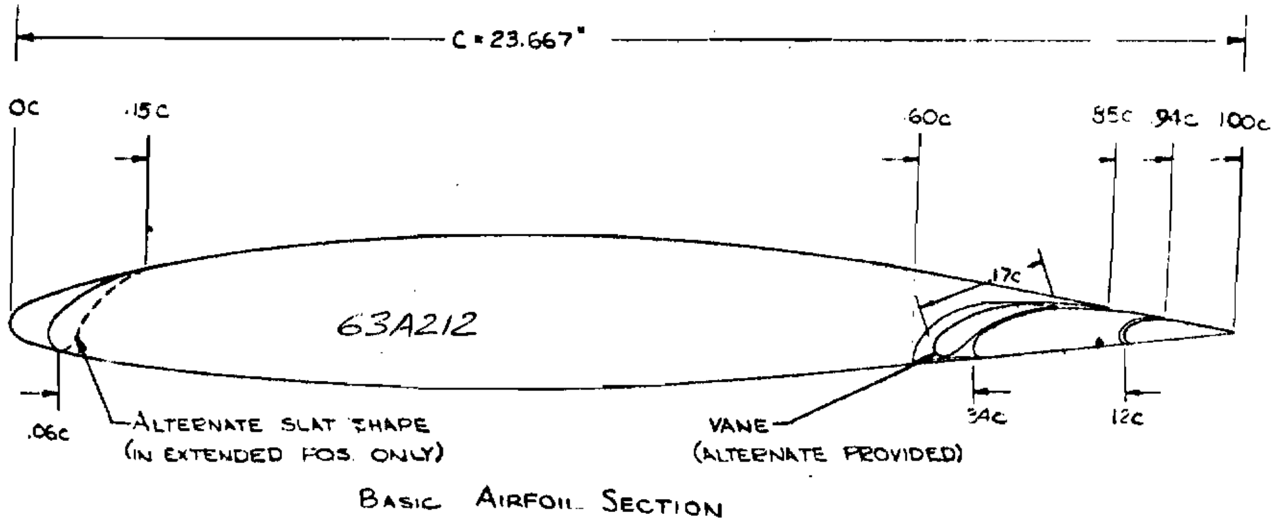


FIGURE 78. GEOMETRY OF DOUBLE AND TRIPLE SLOTTED FLAP MODEL

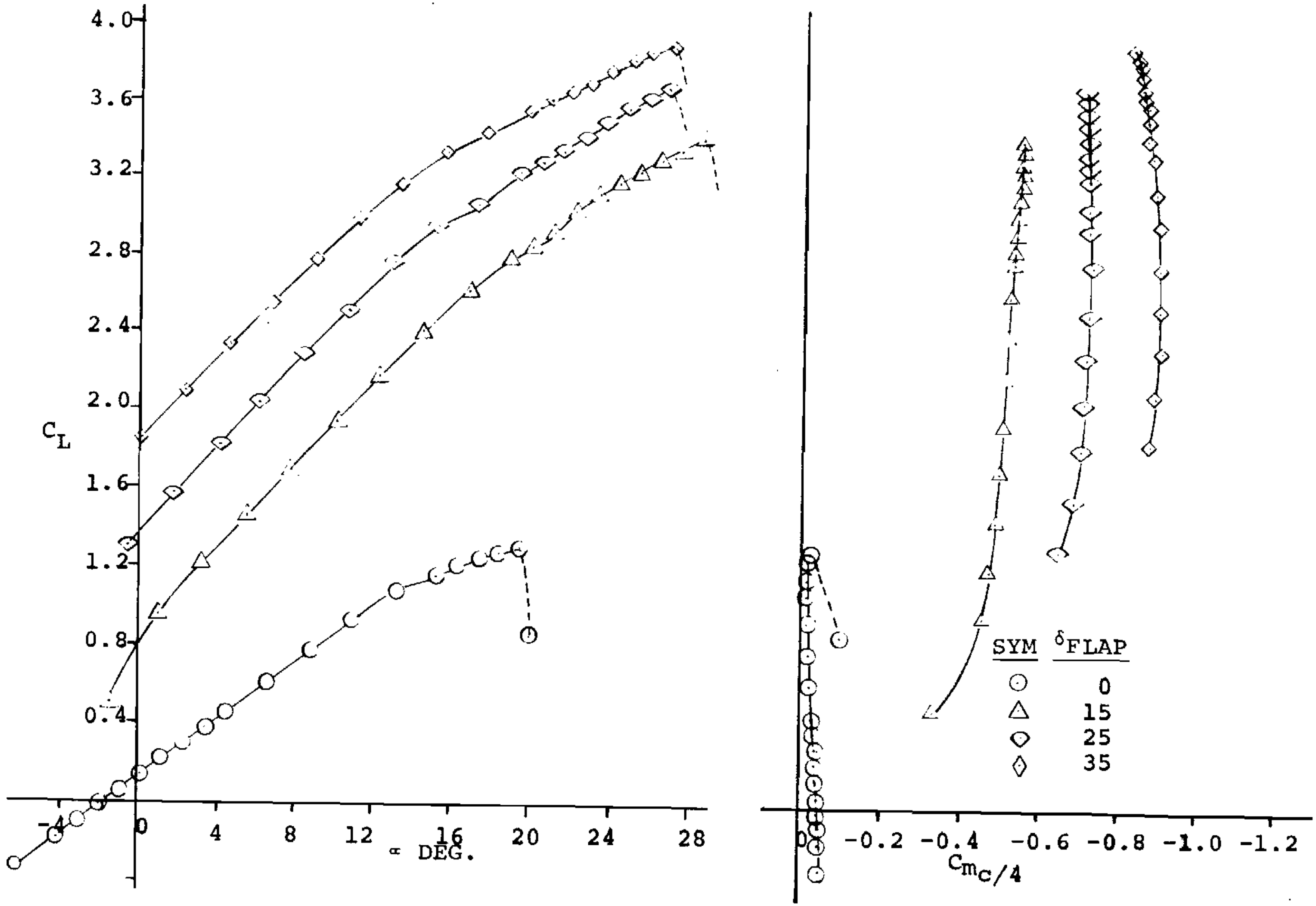
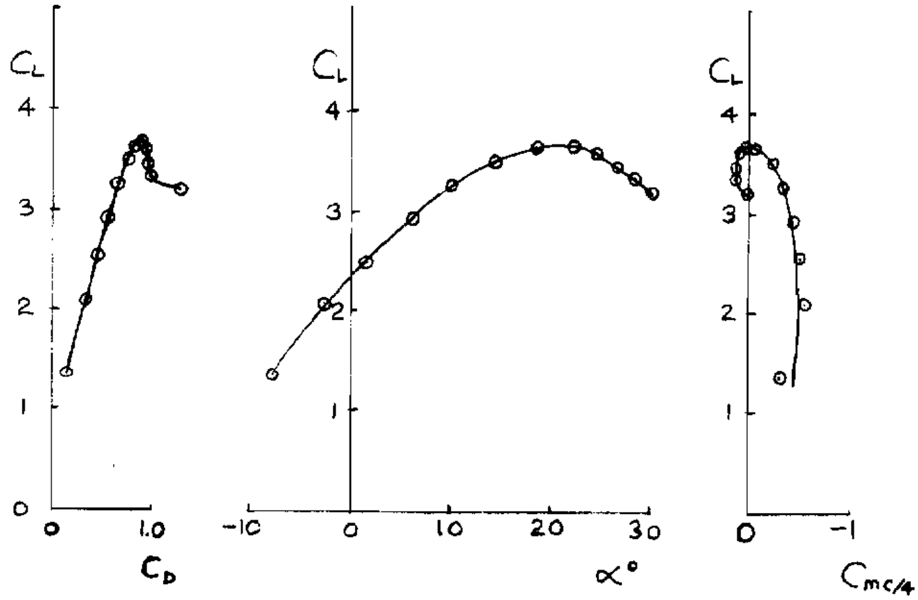


FIGURE 79. DOUBLE SLOTTED FLAPS WITH 20 DEGREES SLAT DEFLECTION



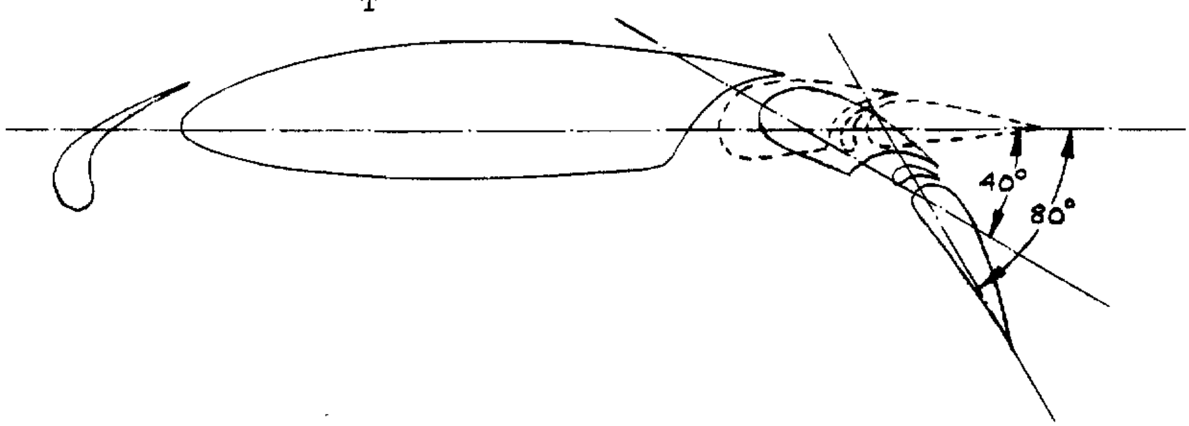
REFERENCE: NASA TND 4448

NOTES: 1. R_e $3x10^6$

2. $AR = 8.06$ b=56ft. $\bar{\bar{c}} = 7.3$ft

3. 63_2416 airfoil

4. $C_T' = 0$



GEOMETRY OF LEADING EDGE SLAT AND TRIPLE SLOTTED FLAP

FIGURE 80. MECHANICAL HIGH LIFT DEVICES ZERO THRUST LONGITUDINAL CHARACTERISTICS OF 4 PROPELLER DEFLECTED SLIPSTREAM STOL AIRCRAFT WITH LEADING EDGE SLAT AND TRIPLE SLOTTED FLAP

other factors as well as the technological factors which affect the STOL aerodynamic performance. The latter are treated here in six headings:

Performance
Handling qualities
Ground effects
Ride quality
Noise
Failure characteristics

These factors, some of which are discussed in more detail in Section 2, are recapped here.

The performance flight envelope of the aircraft must be clearly defined for the all-engines-operative mode. Flight path angle and maneuver G capabilities as a function of speed and altitude should be compared for competing aircraft.

The same type of comparison should be made for the most-critical-engine out condition.

Another important evaluation item is the sensitivity of the aircraft to configuration changes such as flap setting, blowing and thrust deflection, and to what extent these configuration changes may be permitted during normal takeoff or landing operations, abort operations, or transitions between the two. It is important to know the consequences of flight beyond the normal flight envelope boundaries, in order to establish margins of safety. These should be converted into equivalent minimum speeds. Reverse thrust should be available with flow directed to result in the least disturbance to the aircraft itself and to surrounding objects, so as to provide a useful reverse taxi capability.

In evaluating the flying qualities of STOL aircraft, in addition to the conventional stability and control considerations, it is important to consider the rate of change of characteristics with speed. At the low speeds of STOL operations, the changes of control power $\frac{d\theta}{dV}$, $\frac{d\phi}{dV}$, $\frac{d\psi}{dV}$ and $\frac{dT_{REQ}}{dV}$ (which will probably be on the back side of the thrust curve) may have an important bearing on minimum speed requirements.

The sensitivity of the aircraft to gusts during approach and landing is critical in STOL aircraft because the gust can be such a large percentage of the low approach speed. Aircraft should be compared on the basis of the forces and moments resulting from a foot per second of gust velocity. Vertical, lateral. longitudinal and asymmetrical gusts should be considered.

The approach attitude should provide good visibility to the pilot.

If a flare is required, the amount of rotation and the pitch acceleration available to obtain it should be evaluated and considered in establishing field length factors. The ability to develop lift without rotation is a strong asset in a STOL aircraft.

The presence of control cross coupling in any portion of the flight envelope should be accounted for, particularly when the use of lateral control may affect the usable lift.

The susceptibility of STOL aircraft to being tipped over while parked or during ground handling should be considered. The relatively low wing loading high-aspect-ratio straight wings with large flaps with the trailing edge near the ground may make them subject to tip over by side angled gusts. Landing gears should be designed to compensate for this and to provide a desirable turn radius during taxiing. To provide good STOL characteristics the reverse thrust mechanism and basic braking system must be reliable and effective and any possible asymmetries should be easily controllable.

The ability to taxi backwards by means of reverse thrust must be provided, especially for use on short, unprepared, fields that do not provide tractors for reverse taxi. On fields of this type the design of the reverse thrust unit must preclude the possibility of the reverser flow causing reingestion problems.

During all ground handling operation accomplished by application of the aircraft's own power, the resultant noise levels of the competing aircraft should be evaluated.

Ground effect on the high lift systems of STOL concepts cannot be predicted theoretically but must be obtained by wind tunnel tests on the specific configuration proposes. The evaluation of any STOL concept should demand of the proposer sufficient test data to be convinced that the capabilities of the concept in ground effect are known with a reasonable degree of confidence.

Force and moment effects and instabilities induced in ground effect must be evaluated as well as changes in control power.

The recirculation and foreign object damage possibilities of some concepts such as direct lift engines may be greater than those of another concept such as mechanical high lift devices.

The ground signature or trailing vortices resulting from very high lift concepts during takeoff and landing could prove potentially dangerous for another aircraft and may require dimensional separation of STOL aircraft in ground operations or in the traffic pattern.

The STOL aircraft will likely be used for rather short missions and therefore will spend a relatively large portion of its flight time at low altitude in turbulent air.

A STOL configuration with low wing loading, high aspect ratio, low sweep and a high lift curve slope embodies features which are detrimental to smooth ride qualities. Also, the need for large flaps which reduce the size of the wing structural box may affect the elastic gust response in both cruise flight and during STOL operation. The aircraft should be evaluated on the basis of acceleration per ft. per second of gust velocity for not only the rigid body but also the elastic case.

Gust alleviation systems may be provided and their effectiveness and reliability should be assessed especially if credit is taken in weights trend calculations for reduced load factor history.

STOL aircraft have high power loadings and, achieving lift by directing high velocity gases, can have high potential noise levels. These noise levels, both internal (in the cockpit and cabin) and external, should be evaluated and compared for different proposed designs. This should be done for takeoff, approach and cruise conditions.

When noise criteria are specified for takeoff and landing in terms of distance and azimuth, the power levels and flap settings and other configuration variables may be limited and the flight path and STOL distance may be affected.

Evaluation of the failure characteristics of an aircraft involves determination of the effect of the failure on the STOL performance of the aircraft and its ability to complete the mission.

It is important to know the effect of an engine failure on trim and control power in evaluating performance.

Other critical component failures could occur in interconnects including ducting, valves, control links, etc. Some of these are inherent possibilities in some concepts and not in others. Air duct failures would be suspect in augmentor wings and internally blown flaps and not in mechanical and externally flaps systems. SAS, electrical, hydraulic, brakes and thrust reversal systems are common to all concepts.

It is not only necessary to evaluate the results of failures, especially those inherent to particular lift concepts, but it is important to assess the relative probability of their occurrence and the severity of the consequences after occurrence.

6.3.1 Internally Blown Flaps

High lift levels are attainable with internally blown flaps much of it due to supercirculation. As with any high aerodynamic lift device and especially when large chord highly deflected flaps are employed, ground effects are large. A high wing configuration seems preferable in order to reduce ground effect and for other reasons.

Engine development work is required to efficiently furnish the flap blowing air from high-bypass ratio engines which will produce the large thrust to weight ratios needed for STOL operation.

A design effort is needed to quantify the weight and configurational effects of the large high temperature air ducts in the wing and to verify their fatigue life.

Blown flaps require small aircraft rotation angles for takeoff but do produce large pitching moments. By cross ducting a portion of the engine blowing air from each engine to the flap on the opposite wing rolling moments resulting from an engine failure can be kept small.

The initial cost of internal blown flap STOL aircraft may be higher than for conventional aircraft due to additional cost for engines with blowing capability and for ducting, however the aerodynamic benefit in performance and controllability may be substantial.

6.3.2 Augmentor Wing

The augmentor wing is capable of producing very high lift coefficients with much of the lift due to supercirculation. Because of its high lift, ground effects are large and a high wing design is indicated.

To efficiently supply the augmentor blowing air some engine development work will be required. Design work will also be needed to insure that the large high temperature gas ducts are efficient and have a long fatigue life.

Some takeoff thrust augmentation will be available.

The noise level may be potentially low due to the high equivalent bypass ratio and rapid mixing of the augmentor air. Also, by the nature of its design, it is amenable to acoustic treatment.

Engine out rolling moments will be small if a proper portion of the blowing air from each engine is ducted to the augmentor on the opposite wing.

The initial cost of the ducting and flap system will be high as will its maintenance.

6.3.3 Direct Lift Engine Concept

The effect of the lift engines on the lift of the basic wing is unknown in ground effect. Wind tunnel tests on a particular proposed configuration must be conducted to determine this effect.

Lift engines are in limited development and have been used successfully on the Dornier D0-31, which is the best example of a lift-jet transport configuration.

Engine reingestion problems must be avoided during landing and when reverse thrust is applied.

The installation of lift engines with their fuel and control lines is complex as is the addition of the required inlet and exhaust louvers.

The use of lift engines is a potentially simple and effective method of providing STOL performance from a strictly aerodynamic point of view. By obtaining lift this way, smaller wings can be used which will result in higher wing loadings in cruise and good ride comfort.

Lift engine developments promise favorable noise characteristics.

In locating lift engines on the aircraft care must be taken to keep pitching and yawing moments resulting from an engine out as small as possible.

To obtain the best aerodynamic performance, STOL configurations with lift engines must bear the additional cost of lift engine development and procurement in the initial cost and of the maintenance of these multi-engine combinations in the direct operating cost.

6.3.4 Mechanical High Lift Devices

There is a large quantity of wind tunnel and flight test data available on mechanical high lift devices. As a result the technical risk on this concept is low. Also no new engines need be developed specifically for this concept.

A STOL aircraft depending completely on mechanical high lift, rather than powered lift devices would probably have a low wing loading and high gust reaction response unless a gust alleviation system is used.

It is difficult to prevent interaction of the lift and propulsion systems, however, and the moments resulting from engine failure may require consideration of special control system or interconnected propulsion (as with a cross-shafted deflected slipstream propeller driven vehicle).

6.4 CONCLUSIONS

Figure 81 compares the maximum or stall lift coefficient and the usable lift coefficient for five lift concepts. The usable lift reflects the approach C_L rules that provide margins necessary for maneuver, gusts and instrument or operation errors. The externally blown flap with an all engine operative C_L stall at lg of 6.2 represents a C_L stall of 5.4 with one engine out. This results in a usable C_L of only 3.7.

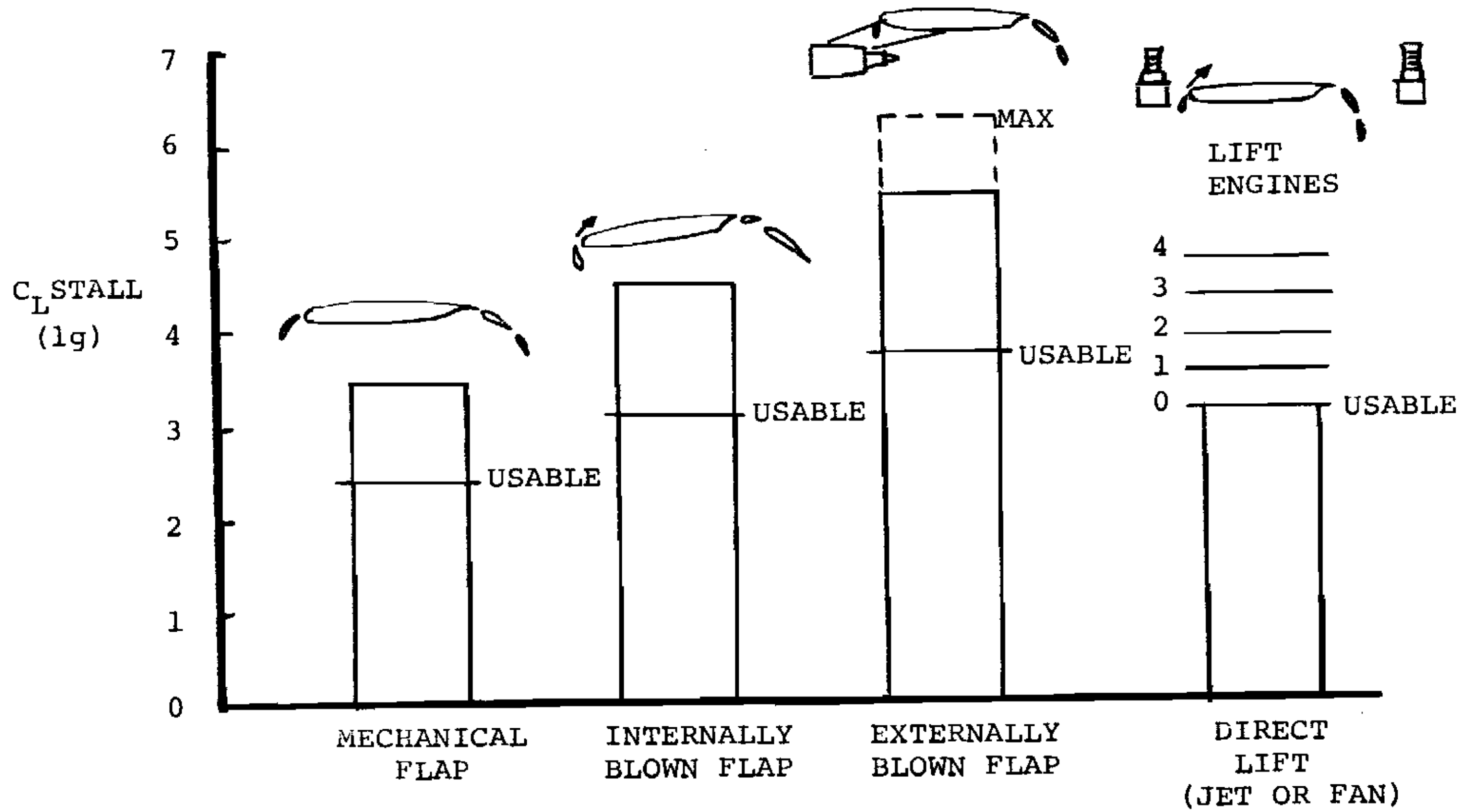


FIGURE 81. LIFT COMPARISONS

If the line marked usable on the direct lift engine bar represents the usable lift of the aircraft with the lift engine thrust set at zero, the lines above it represent increments in "effective C_L stall" that is available by adding lift engines. New stall and maneuver margins might be considered in this design because the lift engine thrust is insensitive to gusts.

Figure 82 compares the noise levels of various STOL aircraft types designed for the same mission and takeoff distance. The basis is the external blown flap which had the highest noise level in takeoff, partly due to a focusing effect of the flap on the noise. Each of the other high lift concepts has a noise level lower than that of the external blown flap in both takeoff and approach.

In takeof, the turbojet or turbofan direct lift designs are 12 to 13 PNdB quieter than the externally blown flap and 6 to 7 PNdB quieter than the conventional mechanical flap or internally blown flap designs. The advantage of the direct lift designs is that less propulsion is lost overcoming the high drag associated with high aerodynamic lift. On the design with high C_L's considerable potential climb thrust is sacrificed to overcome the high drag and the rate of climb is reduced with a resultant loss in altitude and increase in community noise.

The approach case indicates that unpowered lift designs are significantly better than powered designs. The externally blown flap aircraft is noisy because it requires about an 80 percent power setting to maintain the desired descent gradient and aerodynamic lift. Improvements in direct lift systems in approach do look promising as more is learned about inlet fan noise suppression techniques.

Recent studies indicate that the differences shown between concepts are extremes and can be reduced by engine cycle changes or increased bypass ratio, but the increase in engine size, aircraft weight and fuel weights reflect a large penalty of noise constraints upon configurations with STOL performance.

Table XI summarizes the availability of analytical prediction methods, criteria and test data for the five lift concepts being considered.

It will be noted that there is a need for an analytical method for determining ground effects for every lift concept. It should be added that the same need exists for wind tunnel data on ground effect.

Another field where information is lacking for each STOL concept is basic criteria for performance, stability and control, and design. The main reason for this deficiency is that there is such a small experience background available on operating STOL aircraft. Many years and many aircraft have contributed to the evolution of these criteria for CTOL aircraft, both civil and military, and it is desirable to evolve STOL criteria from operational experience with powered-lift aircraft.

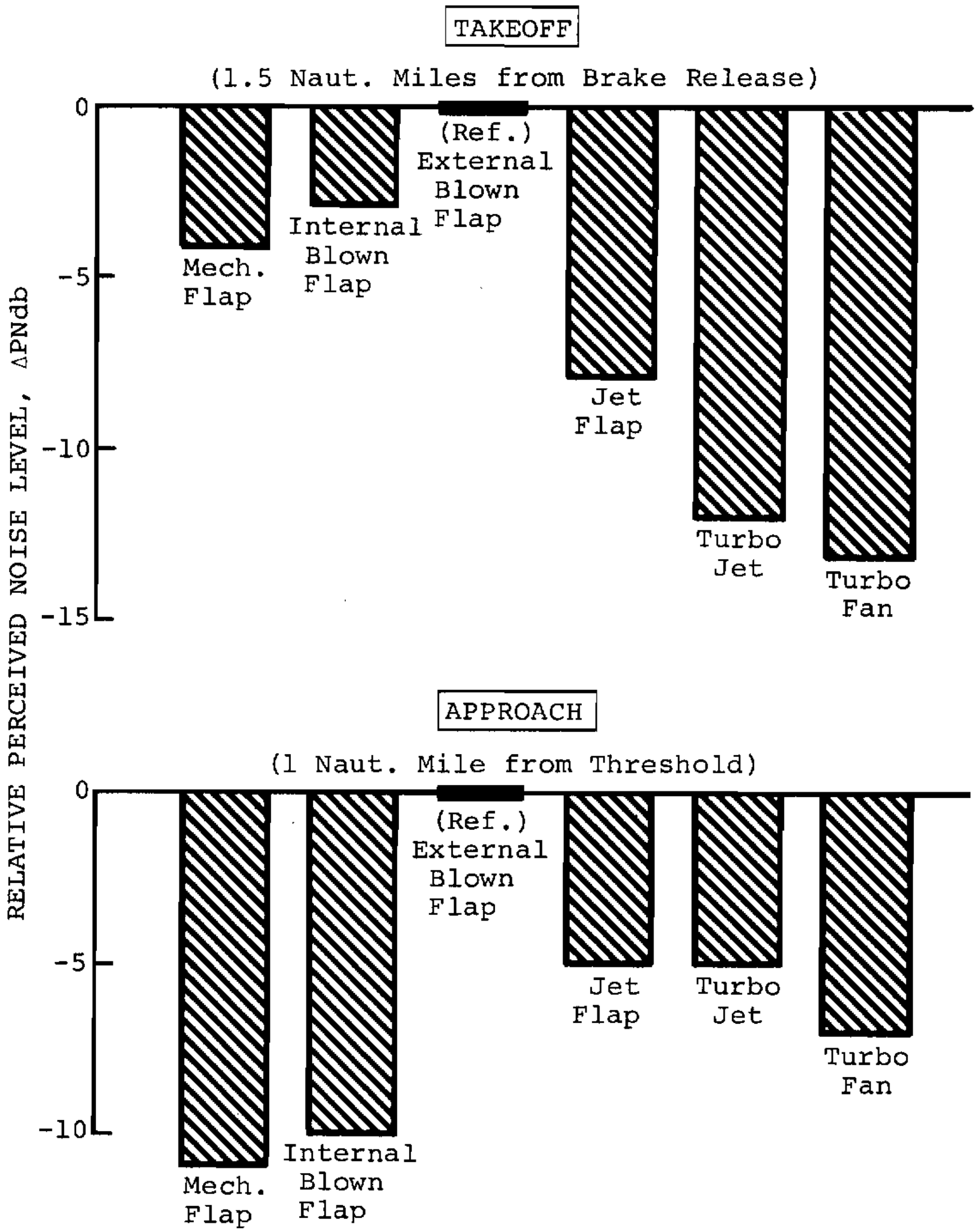


FIGURE 82. NOISE COMPARISON

It is concluded that, if the MST takeoff and landing distance is not less than 2,000 feet, each of the five high lift concepts is feasible. It may be desirable to combine concepts. For example, it may be desirable to add lift engines to the mechanical high lift wing to enable use of a smaller wing. Internal BLC may be needed to provide lateral control for the externally-blown configuration.

The wind tunnel data that are available are valid.

Lateral control limits the STOL performance of the five high lift systems. Powered lift systems such as blown flaps and the augmentor wing are control limited for the critical-engine out condition. To obtain STOL performance with mechanical high lift will require full or near full span flaps. Lateral control will probably have to be obtained by using a portion of the flap as a flaperon (which is difficult to mechanize), plus spoilers. The lateral control engine out problem of the augmentor wing can be alleviated to some degree by ducting a portion of the blowing air from each engine to the flap on the opposite wing so that in the event of an engine failure neither wing flap will be completely unblown.

The most serious deficiency in aerodynamic knowledge of high lift devices is ground effect on the aerodynamic coefficients and particularly on lift. Theory and available tests show that the higher the lift coefficient the more sensitive is the high lift device to ground effect. Very little ground effect test data are available at the high lift coefficients required for STOL performance.

Ground effect is so dependent on configuration that detailed tests must be conducted on the particular design selected.

A flight demonstration is recommended for several reasons. The development of CTOL aircraft has been an evolutionary process through the years. This process has not been available to STOL aircraft. Some of the aircraft that have been used to obtain STOL experience were basically V/STOL types - not STOL and a background is needed for STOL operational and design criteria.

TABLE XI

AVAILABILITY OF METHODS AND DATA

	Lift - Propulsion Concept				
	Mech High Lift	External Blowing	Internal Blowing	Augmentor Wing	Direct Jet lift
Analytical					
Lift and Drag	Good	Moderate	Moderate	Moderate	Good
Moments	Good	Need	Need	Need	Good
Ground Effects	Need	Need	Need	Need	Need
Criteria					
Performance Stability and Control Design	Need	Need	Need	Need	Need
Test					
Wind Tunnel Data	In Hand	Moderate	In Hand	Moderate	Moderate
Adequacy of Wind Tunnel	Good	Good	Good	Good	Good
Flight Test Data	In Hand	One Example	Little	None	One Example

7. CONCLUSIONS

Of all the STOL concepts studied, the technology base is greatest for the deflected slipstream, tilt wing, and mechanical high lift concepts and smallest for the externally-blown-flap and jet flap types. Nevertheless, there are sufficient technical data for all of the concepts to show that each could be developed into a feasible STOL aircraft system.

The low speed operation required to take off or land in short distances must be obtained by increasing the lifting capability through propulsive-lift augmentation or low wing loadings. Either course creates additional problems. Powered lift tends to increase sideline noise, aerodynamic ground effects, and system complexity. Low wing loading reduces comfort of ride quality. Techniques exist in today's technology to solve all of these problems.

STOL aircraft require a set of performance and flying qualities criteria which differ from those applied to conventional aircraft. Many different sets of criteria have been proposed and studied. None has been adopted. Additional operational experience with STOL flight demonstrators is a necessary prerequisite to such an action.

Wind tunnel testing of STOL aircraft is more complex and requires more detailed attention than that required for conventional aircraft. The following factors require special attention: matching of model size to tunnel size, ground effect testing including requirements for a moving ground plane, model motor selection and installation, instrumentation, and flow visualization.

Much of the available test data have limited applicability resulting from incomplete calibration or the omission of measurements of parameters later found to be important. The development of analytical methods for prediction of aerodynamic characteristics of STOL aircraft would benefit from systematic testing of these configurations.

Methodology for deflected slipstream includes several analytical and empirical force prediction methods. These have limited applicability. No such methodology was found for the externally blown flap. A preliminary method has been developed and is presented in Section 4 of this report.

Additional method development is required for all STOL concepts to improve the understanding of the complex aerodynamic effects of these configurations.

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13. ABSTRACT

The state of the art of STOL aerodynamic technology for selected lift/propulsion concepts has been surveyed to identify the available test data and prediction methods in the literature. The report consists of two volumes.

In Volume I important areas of technology and information necessary for the evaluation of STOL aircraft aerodynamics are listed; the aerodynamic test data and prediction methodology relevant to the deflected slipstream and externally blown flap concepts are assessed, with emphasis on the latter; an empirical method for the prediction of the longitudinal aerodynamic characteristics of externally blown flap configurations is presented; and high-lift technology for five lift/propulsion concepts is assessed in application to a medium-sized STOL transport.

Volume II consists of a bibliography that resulted from a literature search for aerodynamic information related to seven lift/propulsion concepts suitable for STOL aircraft. The bibliography contains references to approximately 900 reports classified by concept and by technological area.

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