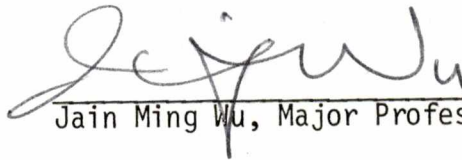
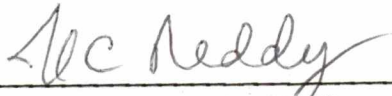
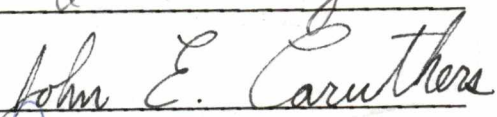
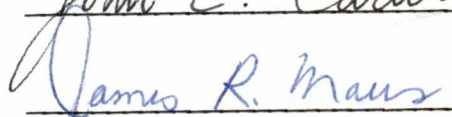
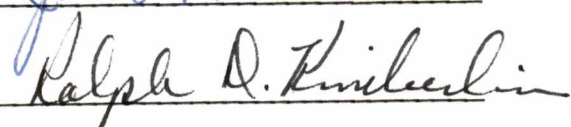


To the Graduate Council:

I am submitting herewith a dissertation written by Uwe Peter Solies entitled "A Study of the Flow Field About an Upper Surface Blown Wing and Its Effects on Longitudinal Static Stability of a Small Jet Airplane." I have examined the final copy of this dissertation for form and content and I recommend that it be accepted in partial fulfillment of the requirements for the degree of Doctor of Philosophy, with a major in Aerospace Engineering.


Jain Ming Wu, Major Professor

We have read this dissertation
and recommend its acceptance:


Accepted for the Council:


Vice Provost
and Dean of The Graduate School

A STUDY OF THE FLOW FIELD ABOUT AN UPPER SURFACE
BLOWN WING AND ITS EFFECTS ON LONGITUDINAL
STATIC STABILITY OF A SMALL JET AIRPLANE

A Dissertation
Presented for the
Doctor of Philosophy
Degree
The University of Tennessee, Knoxville

Uwe Peter Solies
December 1985

TO
MY FATHER
WHO SPARKED MY LOVE FOR FLYING
AND ENCOURAGED MY CURIOSITY
ABOUT THE LAWS OF NATURE
WHICH MAKE IT POSSIBLE

ACKNOWLEDGMENTS

The author is deeply grateful to Dr. Willi F. Jacobs, who introduced him to the exciting subject of Powered Lift Technology and shared with him the results of a lifetime of research work, before he retired from his duties at UTSI.

The author wishes to express his sincere appreciation to the chairman of his committee, Dr. Jain M. Wu, who gave him thoughtful guidance, valuable suggestions and encouragement during the course of this study.

Very special thanks are due to the author's advisor and friend, Professor Ralph D. Kimberlin, who was always willing to pilot the test aircraft, sharing his vast flight testing and engineering experience, offering many helpful suggestions, and giving support and encouragement. Without him, this study would not have been possible.

Special thanks are also due to Donald Freeman, aircraft mechanic, and Robert Jones, instrumentation specialist, who were always willing to help and assist in setting up and calibrating flight test equipment.

The author feels deep appreciation for the many fruitful discussions and valuable suggestions that he received from Dr. John E. Caruthers, Dr. James R. Maus, Dr. Kapuluru C. Reddy, and Dr. H. Joe Wilkerson during the conduct of this research and the preparation of this dissertation.

At this point, the author wants to extend his thankful gratitude to the librarians Helen Mason, Marjorie Joseph, and Mary Lo for their

eagerness in spending time and effort in obtaining numerous references required for this work.

He also wishes to thank Marshall Hutto, manager of UTSI computer systems, and Rhonda Dawbarn, assistant manager, for their helpful attitude and assistance during the often frustrating time of computational work on this research.

The author feels special appreciation for Kathy Hice, upper B-wing secretary, and her valuable help in preparing a preliminary version of this dissertation, and for Lena Farrar and her professional dedication in typing the final script.

Last, but not least, the author wants to thank his fellow students and special friends in Tennessee. Their friendship and support made the years at UTSI a time of joyful growth and fond memories.

ABSTRACT

The effects of powered lift technology on the longitudinal stability characteristics of an upper surface blown (USB) jet-flapped aircraft have been studied. Both theoretical and experimental methods were employed to investigate the flow field in the vicinity of the lifting wing and its influence on pitching moments of the aircraft in motion.

A nonlinear vortex-lattice theory was used to model the lift induced airflow in the near flow field about the wing. Comparison with literature data showed excellent agreement, giving confidence in the method used.

Flight tests were conducted with the "Ball-Bartoe Jetwing" technology demonstration aircraft to visualize flow angles on the fuselage surface and at selected points in the flow field near the horizontal tail. To save cost, simple data acquiring techniques were used. Also, the theoretical flow computations were restricted to a simplified Jetwing model with a relatively small number of panels. Still, there was fair agreement between theoretically and experimentally determined flow angles at the tail of the Jetwing aircraft.

The vortex-lattice routine was then used for a parametrical study of flow angles and longitudinal stability parameters. The results show that the powered lift strongly affects the flow field about the wing. With increasing blowing coefficient the flow field has a more destabilizing impact on the pitch-stability of the aircraft. The data also indicate that for certain conditions the presence of the jet sheet improves the stability contribution of the horizontal tail by reducing

the downwash derivative in this area. This may be attributed to the particular geometry of the trailing wake and tip vortices. Further research is needed to substantiate these findings.

The vortex-lattice routine developed as part of this study is considered a valuable tool for the flow analysis and the preliminary design of powered lift aircraft with jet-flapped wings.

Da steh' ich nun, ich armer Thor,
und bin so klug als wie zuvor!

Johann Wofgang von Goethe
a.d. 1777

TABLE OF CONTENTS

SECTION	PAGE
1. INTRODUCTION	1
1.1 The Need for Powered Lift and Its Advantages	1
1.2 The Ball-Bartoe Jetwing Experimental Aircraft	4
2. FORMULATION OF PROBLEM	12
2.1 The Definition of Longitudinal Stability and Control (LSC)	12
2.2 The Influence of Upper Surface Blowing on LSC	15
2.3 The Need for Research	16
3. EXPERIMENTAL DETERMINATION OF FLOW ANGLES	18
3.1 Preparation and Conduct of Flight Tests	18
3.2 Data Evaluation	24
4. THEORETICAL DETERMINATION OF FLOW ANGLES	34
4.1 Literature Findings	34
4.2 Vortex-Lattice Method and Computer Program	38
4.2.1 General	38
4.2.2 Potential Flow	39
4.2.3 Biot-Savart Law	40
4.2.4 Vector Analysis	40
4.2.5 Vortex Representation of Jet Sheet	47
4.2.6 Vortex Representation of Wing and Free Vortices	53
4.2.7 Shape of Jet Sheet	57
4.2.8 Lift Distribution	62
4.2.9 Computational Process	64
5. DETERMINATION OF LONGITUDINAL STABILITY PARAMETERS	69
5.1 General Derivations	69
5.1.1 Wing	74
5.1.2 Inlet	78
5.1.3 Thrust	80
5.1.4 Gear	82
5.1.5 Fuselage	84
5.1.6 Horizontal Tail	86

SECTION	PAGE
5.2 Determination of Derivatives from Experimental Data . .	87
5.2.1 Flight Test	87
5.2.2 Wind Tunnel	90
5.3 Determination of Derivatives from Theoretical Data . . .	91
6. PRESENTATION AND DISCUSSION OF RESULTS, CONCLUSIONS AND RECOMMENDATIONS	95
6.1 Chosen Geometry of Panels and Vortex Filaments	95
6.2 Comparison of Lift Coefficients	95
6.3 Comparison of Downwash Data	97
6.4 Generation of Downwash Data and Derivatives for Longitudinal Static Stability	109
6.5 Conclusions and Recommendations	116
BIBLIOGRAPHY	119
VITA	124

LIST OF FIGURES

FIGURE	PAGE
1. Comparison of Noise Contours Around Different Airport Types	2
2. Mechanical High Lift Devices	2
3. Examples for Powered Lift Concepts	3
4. Some Jet Flap Schemes	5
5. Jacobs-Hurcamp Jet-Flapped Wing	6
6. Components of Lift for a Jet-Flapped Wing	6
7. Ball-Bartoe Jetwing Experimental Upper Surface Blown Aircraft	7
8. Jetwing Engine Exhaust Ducting Arrangement	9
9. Two-Dimensional Views of the Jetwing Concept	10
10. Principal Forces and Moments on the Jetwing Aircraft Steady Level Flight	13
11. Experimental Set-Up for Downwash Measurements	19
12. Example for a Photographic Picture of the Test Vehicle Taken from a Chase Airplane	21
13. Example for a Photographic Picture of the Tufted Rake Taken by a Jetwing Tail Mounted Camera	22
14. Example for Jetwing Wind Tunnel Data	27
15. Jetwing Thrust Calibration Diagram	29
16. Flow Angles at the Horizontal Tail	31
17. Flow Angle Correction for the Influence of the Horizontal Tail	31
18. Horseshoe Vortex Distribution for Downwash Calculations Used in NACA TR 651	37
19. Parameters for the Application of the Biot-Savart Law . .	41

FIGURE	PAGE
20. Coordinate System and Sample Vortex Filament with Induced Velocity at a Point in the Flow Field	42
21. Different Geometries and Vortex Spins for the Application of the Biot-Savart Law	44
22. Element of a Curved Jet Sheet in Two-Dimensional Nonviscous Flow	48
23. Schematic Illustration of the Singularity Distribution for Modelling the Wing and Various Flow Components . . .	54
24. Example for the Spanwise Distribution of Jet Exhaust Velocities on the Jetwing Determined by LASER Velocimetry	58
25. Parameters for Jet Sheet Description and Change of Jet Sheet Position and Panel Velocities after Iterations . . .	60
26. Plotting Sample of Jetwing Flow Field, Jet Off	67
27. Plotting Sample of Jetwing Flow Field, Jet On	68
28. Major Forces and Moments for Jetwing Longitudinal Stability Considerations	70
29. Relationship between Geometric, Aerodynamic, and Zero Lift Angle of Attack	76
30. Normal Force at the Inlet	79
31. Gear Forces at Different Angles of Attack	83
32. Potential Flow About a Body of Revolution and a Typical Layout for Computing Fuselage Moments	85
33. Example for Jetwing Elevator Positions and Stick Forces versus Airspeed	88
34. Determination of $\partial C_L / \partial \alpha$ Derivatives from Wind Tunnel Data	92
35. Determination of $\partial C_m / \partial \alpha$ Derivatives from Wind Tunnel Data	93
36. Chosen Geometry for Jetwing Panels	96
37. Comparison of Lift Distributions Resulting from Different Theories	98
38. Comparison of Wind Tunnel Data with Results from Nonlinear Vortex-Lattice Theory for a Simplified Jetwing Model . . .	99

FIGURE

PAGE

39.	Comparison of Results from Simple Lifting Line Theory and Nonlinear Vortex-Lattice Theory for the Downwash Angle Distribution Behind Untwisted Wings ($x=-b/2, z=0$)	100
40.	Comparative Sketches of Vortex Arrangement and Downwash Distribution for two Theories	102
41.	Correction of Flight Test Angle of Attack Measurements for Upwash at the Boom Mounted Sensor	104
42.	Jetwing Downwash Angles versus Airspeed at the Top Tuft Rake Position for Zero and 15 Degrees Flap Deflection	105
43.	Jetwing Downwash Angles versus Airspeed at the Top Tuft Rake Position for 30 and 45 Degrees Flap Deflection	106
44.	Jetwing Downwash Angle and $\partial\epsilon/\partial\alpha$ Distribution Along the Fuselage Centerline for Zero Degrees Flap Deflection	110
45.	Jetwing Downwash Angle and $\partial\epsilon/\partial\alpha$ Distribution Along the Fuselage Centerline for 15 Degrees Flap Deflection	111
46.	Jetwing Downwash Angle and $\partial\epsilon/\partial\alpha$ Distribution Along the Fuselage Centerline for 30 Degrees Flap Deflection	112
47.	Jetwing Downwash Angle and $\partial\epsilon/\partial\alpha$ Distribution Along the Fuselage Centerline for 45 Degrees Flap Deflection	113
48.	Comparison of Downwash Derivatives at the Jetwing Tail Resulting from Different Methods	115

LIST OF SYMBOLS AND ABBREVIATIONS

Latin Symbols

a	shortest distance of point from vortex line
a_o	coefficient in function describing jet sheet
a_{ij}	influence coefficient describing downwash at panel i due to vortex at panel j
a_W	lift curve slope $\partial C_L / \partial \alpha$ of the wing
a_T	lift curve slope $\partial C_L / \partial \alpha_T$ of the horizontal tail
A	with aspect ratio b^2/S
A_T	horizontal tail aspect ratio b_T^2/S_T
$[A]$	$n \times n$ matrix of influence coefficients
A.C.	Aerodynamic center
AC	(index) aerodynamic center
b	wing span
b_o	coefficient in function describing jet sheet
c	(index) corrected
$\vec{c}$	vector with direction of vortex induced velocity
c_o	coefficient in function describing jet sheet
$\bar{c}$	mean aerodynamic chord of wing
$\bar{c}_B$	mean geometric chord of blown wing section
$\bar{c}_j$	mean geometric chord of wing strip j
C	general term for coefficient
C	(index) tangential force vector

C_j	mean sectional blowing coefficient (two-dimensional)
C_J	overall blowing coefficient (three-dimensional)
C_c	coefficient of tangential force
C_D	drag coefficient
C_{DI}	coefficient of induced drag
C.G.	center of gravity
CG	(index) center of gravity
C_G	tangential force at gear
C_ℓ	local lift coefficient (two-dimensional)
C_L	overall lift coefficient (three-dimensional)
$C_{m_{CG}}$	pitching moment coefficient about the center of gravity
$C_{m_{CG\dot{\theta}min}}$	coefficient of minimum required pitching moment for sustaining pitch rate $\dot{\theta}$
$C_{m_{CGTO}}$	tail off pitching moment coefficient about the C.G.
C_N	normal force coefficient
C_T	thrust coefficient or tangential force at tail
C_μ	theoretically determined blowing coefficient
d	differential
D	drag
D_G	gear drag
D_T	tail drag
e	constant, base of natural logarithm ($e=2.71828$)
e	(index) elevator

f	(index) fuel
F	force
F	(index) fuselage or flap
F_G	gross thrust
g	constant, standard gravitational acceleration on earth ($g=9.81\text{m/s}$)
h_p	pressure altitude
i_T	incidence angle of horizontal tail stabilizer, referred to fuselage axis
i	(index) indicated or induced or dummy variable
I	(index) inlet or induced
j	(index) dummy variable
J	jet momentum
J	(index) jet
J'	sectional jet momentum per unit span
k	lift induced drag factor; ratio of actual induced drag over minimum induced drag for elliptical lift distribution
k	(index) dummy variable
k_{CO}	carry over factor, accounts for the influence of the spanwise blown portion of the wing on the unblown portion
k_F	empirical factor for the determination of fuselage pitching moment derivatives
ℓ	(index) local lift
L	lift
L_W	wing lift force

L_T	horizontal tail lift force
m	(index) moment or measured
$\dot{m}$	mass per unit time, mass flow
$\dot{m}_\mu$	theoretical mass flow in jet exhaust
$\dot{m}_j$	actual mass flow in jet exhaust
M	Mach number or moment
M.A.C.	mean aerodynamic chord
MAC	(index) mean aerodynamic chord
M_o	wing pitching moment about its aerodynamic center (.25 $\bar{c}$)
M_{oT}	horizontal tail pitching moment about its aerodynamic center
M_{CG}	pitching moment about the center of gravity
$M_{CG_{min}}$	minimum required pitching moment about the center of gravity
M_{CGTO}	tail off pitching moment about the center of gravity
M_F	pitching moment of the fuselage
n	number or dummy variable
n	(index) number or normal
N_I	normal force at engine inlet
N_1	rotational speed of engine primary rotor (fan spool)
N_G	normal force at gear
N_T	normal force at tail
NLVLP	nonlinear vortex-lattice program

O	origin of coordinate system
OAT	outside air temperature
$\overrightarrow{OT}$	vector from the origin to point T
p	pressure
p_o	static pressure at sea level on a standard day ($p_o=1013.25$ mbar)
Δp	pressure difference
PT	scalar distance between points P and T
$\overrightarrow{PT}$	vector from point P to point T
q	dynamic pressure
r	(radial) distance
$\vec{r}$	distance vector
r	(index) resulting
R	radius of curvature of the jet sheet
S	wing reference area
S	(index) strip
S_B	wing reference area relating to the spanwise extent of blowing
ΔS_j	reference area of wing strip j
S_T	horizontal tail reference area
t	(index) test
th	(index) theoretical
te	(index) trailing edge
tr	(index) true
T	absolute temperature or thrust force
T	(index) tail

T_0	absolute temperature of standard day at sea level ($T_0=288K$)
TH	(index) thrust
u_∞	free stream velocity
v_c	calibrated airspeed
v_i	indicated airspeed
v_1, v_2	velocities of jet sheet at its boundaries in potential flow
v_j	average jet velocity $(v_1+v_2)/2$
v_μ	theoretical jet velocity
v_{tr}	true airspeed
V_f	fuel volume
V_T	horizontal tail volume
w_n	normal velocity
$\overline{w}_n$	average normal velocity
W_t	weight of aircraft at test time
W_{maxto}	maximum takeoff weight of aircraft
x_A	horizontal distance of wing aerodynamic center from aircraft center of gravity
x_G	horizontal distance of gear A.C. from aircraft C.G.
x_T	horizontal distance of horizontal tail A.C. from aircraft C.G.
x_{te}	x-coordinate of wing trailing edge
XT,YT,ZT	x,y,z components of vector from origin to point T in Cartesian coordinates
XPT,YPT,ZPT	x,y,z components of vector $\overline{PT}$
Δy_j	spanwise width of wing strip j

z_A	vertical distance of wing A.C. from aircraft C.G.
z_G	vertical distance of gear A.C. from aircraft C.G.
z_T	vertical distance of tail A.C. from aircraft C.G.
z_{TH}	vertical distance of thrust line from aircraft C.G.
z_{te}	z-coordinate of wing trailing edge

Greek Symbols

α	angle of attack
α_0	angle of attack for zero lift
α_g	geometric angle of attack
α_a	aerodynamic angle of attack
α_I	engine inlet flow angle
$\alpha_{i\infty}$	induced downwash angle at infinity
α_T	angle of attack of the horizontal tail
β	flow angle of attack at the fuselage centerline, positive like free stream angle of attack
γ_J	linear strength of the vortex representing a two- dimensional jet sheet
Γ	total circulation of the wing or a vortex filament
Γ_J	circulation of the jet sheet
Γ_{JS}	circulation of a streamwise strip of the jet sheet
Γ_{Wj}	circulation of wing strip j
Γ_{Wij}	circulation of wing panel (i,j)
δ	thickness of jet sheet or static pressure ratio p/p_0
δ_e	elevator deflection angle referring to the horizontal stabilizer chord line; positive for trailing edge down

δ_F	wing flap deflection angle referring to the wing chord line; positive for trailing edge down
δ_J	jet sheet deflection angle referring to the wing chord line; positive for trailing edge down
Δ	finite difference
∂	partial differential
ϵ	lift induced flow angle referring to free stream direction; positive for "downwash"
$\bar{\epsilon}$	correction angle, upwash angle of horizontal tail referring to stabilizer chordline; positive for leading edge up
ϵ_I	upwash angle at inlet entrance, positive like angle of attack
ϵ_m	measured downwash angle
ϵ_{mc}	measured downwash angle, corrected for the influence of the horizontal tail
ϵ_{th}	theoretical downwash angle
ϵ_T	wing lift induced downwash angle at the horizontal tail A.C.
η_T	tail efficiency; ratio of average dynamic pressure at the tail to free stream dynamic pressure
θ	pitch angle or ratio of actual absolute temperature to sea level absolute temperature on a standard day (T/T_0)
$\dot{\theta}$	rate of change of pitch angle, pitch rate
λ	freely variable parameter in vector equation
μ	measured tuft angle referring to stabilizer chord line, positive for tuft end up
μ	(index) theoretical values
π	constant for the quadrature of the circle ($\pi = 3.1415927$)

ρ	density
ρ_{∞}	free stream density
ρ_0	air density at sea level on a standard day ($\rho_0 = 1.224\text{kg/m}^3$)
ρ_f	fuel density ($\rho_f = 0.78\text{kg/lit}$)
σ	ratio of actual air density to air density at sea level on a standard day

1. INTRODUCTION

1.1 The Need for Powered Lift and Its Advantages

The increase of air travel connecting urban areas generates the need for a transport aircraft that can operate from short runways and is quiet and able to approach and depart the airports on steep flight paths in order to reduce the noise impact on the adjoining communities (Figure 1).

In the military there exists a need for an aircraft that is capable of operating from short, unimproved airfields close to the battle lines or from decks of smaller ships and is also capable of combat maneuvering. In each of these cases, the aircraft should have a large useful load and characteristics that allow fuel efficient operation at high cruise speeds. While these characteristics require small wing areas and low structural weight, the conventional designs for short takeoff and landing (STOL) operations require large wing areas and heavy flap systems (Figure 2). In recent years these problems have lead to several research and development efforts that incorporated "powered lift" technology. In addition to the basic aerodynamic lift generated by conventional airfoils and flap systems, powered lift is generated by either turning the jet exhaust downward in order to achieve a vertical thrust component, or by blowing pressurized engine air along airfoil, flaps, trailing edge nozzles or some combination of these (Figure 3).

The underlying principle for these configurations is the basic

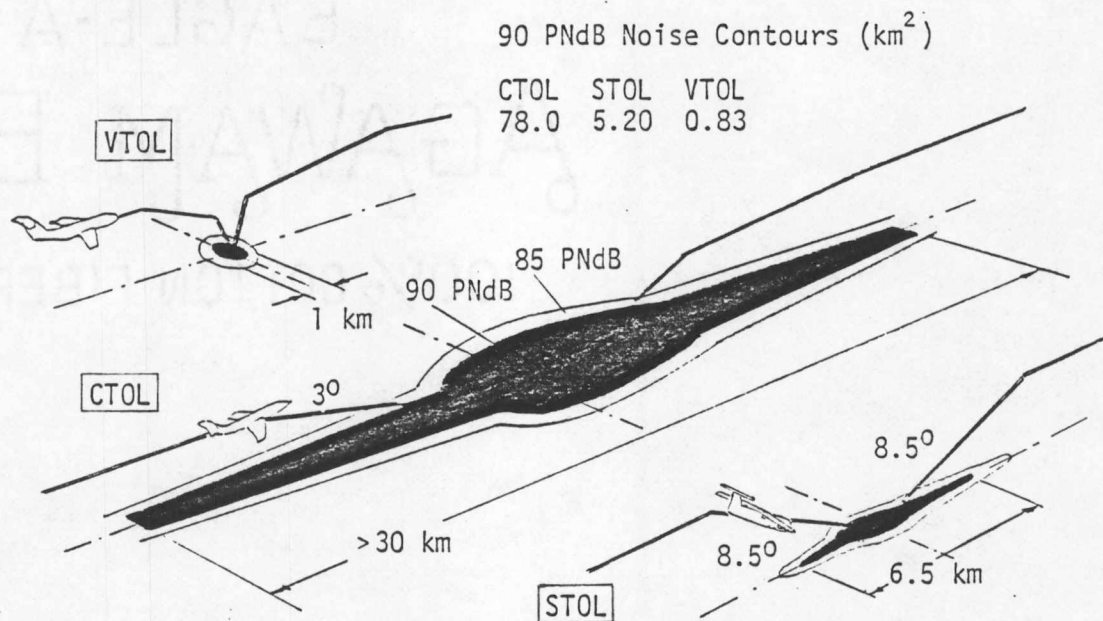


Figure 1: Comparison of Noise Contours Around Different Airport Types

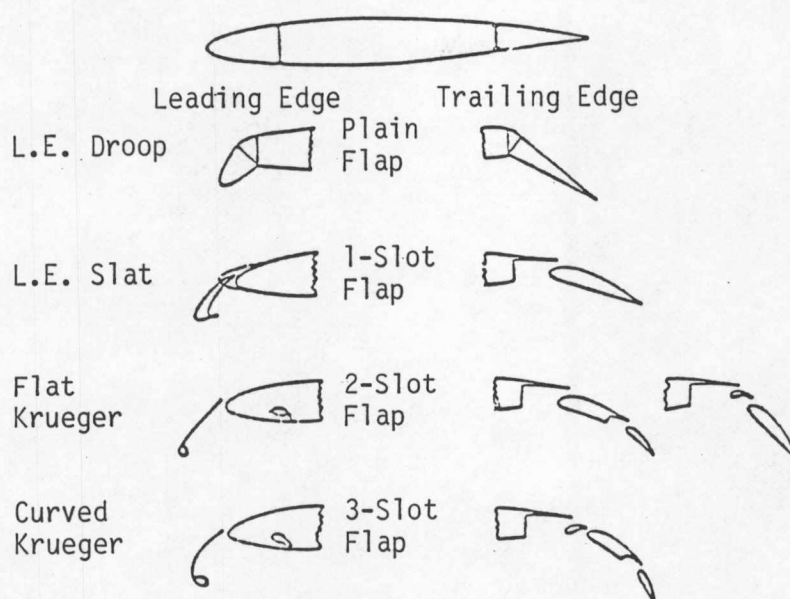


Figure 2: Mechanical High Lift Devices

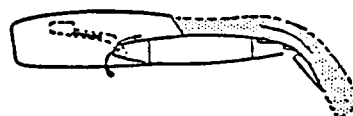
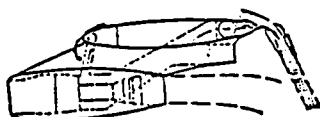
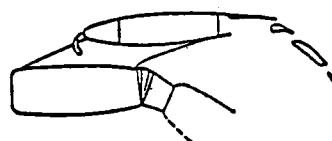
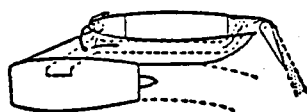
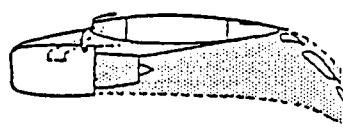
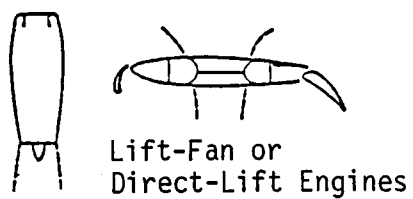


Figure 3: Examples for Powered Lift Concepts

jet-flap as first proposed by Schubauer [1],* and Hagedorn and Ruden [2] in 1938 (Figure 4).

An example of a modern jet-flap design which incorporates features of a mechanical flap and blowing boundary layer control (BLC) was developed and patented by Jacobs of Lockheed-Georgia in 1971 (Figure 5). While for jet aircraft with vertical takeoff and landing capabilities (VTOL) thrust vectoring is the only means of powered lift, for STOL vehicles the jet-flap is much more promising, since a substantial amount of additional aerodynamic lift is generated by increased flow circulation (Figure 6).

A properly designed jet-flapped aircraft combines capabilities for STOL, steep approaches and departures, and high maneuverability, with small and light wings for high speed and efficient cruise. More detailed descriptions of various powered lift concepts can be found in Reference [3].

1.2 The Ball-Bartoe Jetwing Experimental Aircraft

In 1976, the Ball Aircraft Company of Colorado developed an experimental jet aircraft as a technology demonstration vehicle for powered lift [4,5,6]. The "Ball-Bartoe Jetwing" represents a low cost, single engine, small flight test bed with low aspect ratio and uniform blowing over major portions of the wing. This is the first and only single engine upper surface blown (USB) aircraft built to date (Figure 7). The Jetwing achieves upper surface blowing from a single Pratt & Whitney JT15D-1 turbofan engine by ducting all air through the

*Numbers in brackets refer to similarly numbered references in the Bibliography.

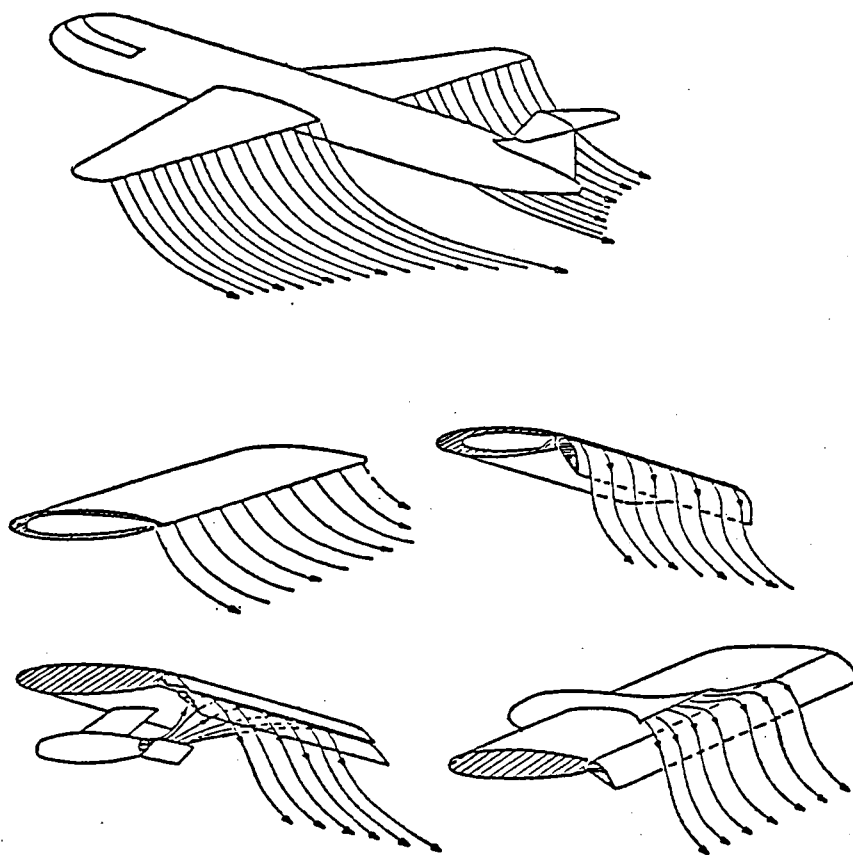


Figure 4: Some Jet Flap Schemes

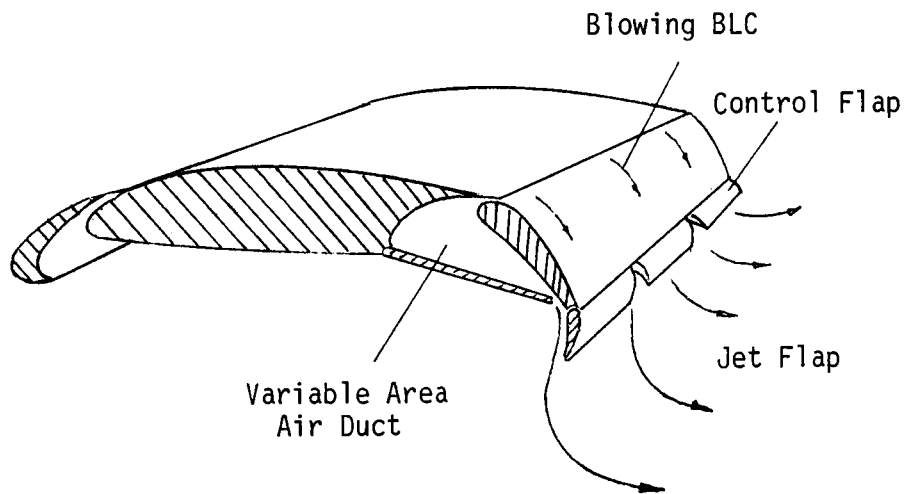


Figure 5: Jacobs-Hurcamp Jet-Flapped Wing

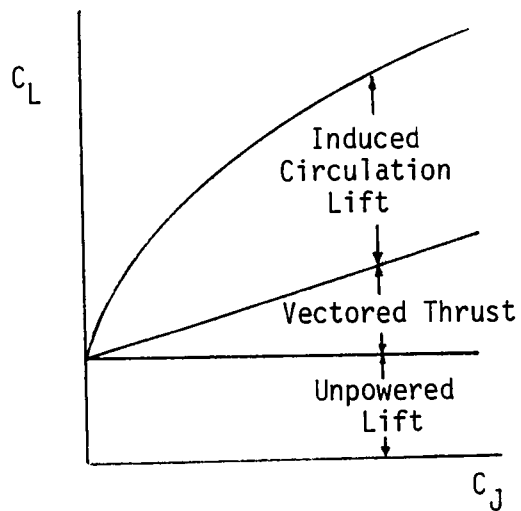


Figure 6: Components of Lift for a Jet-Flapped Wing

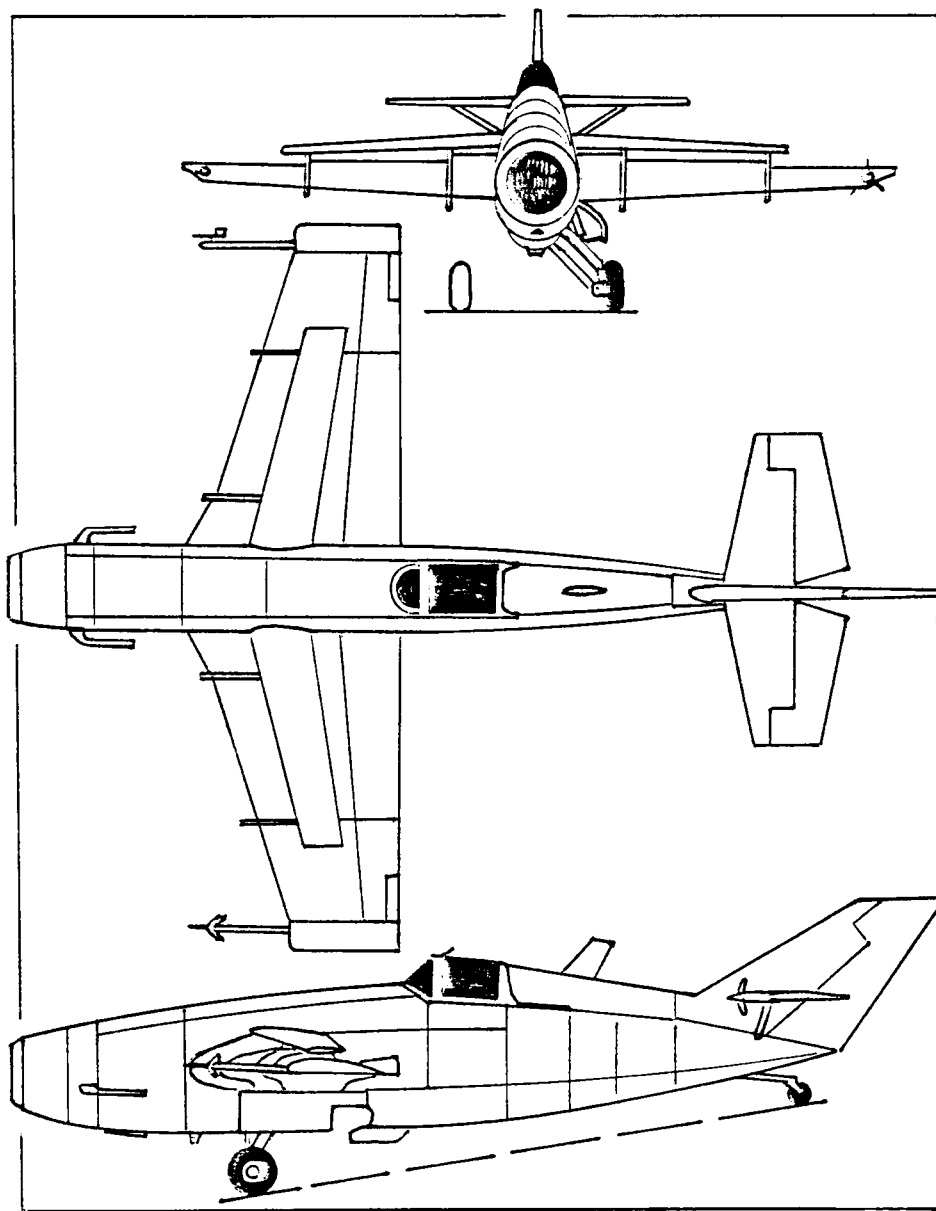


Figure 7: Ball-Bartoe Jetwing Experimental Upper Surface Blown Aircraft

front portion of the wing and ejecting it over the upper wing surface through a slot nozzle. The nozzle is located at 33% of the wing chord and extends along 70% of the wing span, approximately. Bypass air is ducted to the outboard portion of the wing while the core exhaust is ducted to the inboard portion (Figure 8). A removable small wing surface is located above the slot nozzle and acts as an ejector or thrust augmentor. A Coanda type, single element flap is located at the trailing edge of the wing along the blown portion of the wing span. Figure 9 shows a two-dimensional sketch of the Jetwing concept. The top of the bypass air nozzle can be rotated to act as a thrust reverser.

The Jetwing research aircraft and its conceptual patents are now owned by The University of Tennessee. It has undergone extensive full scale wind tunnel tests at NASA Ames Research Center [7] and was flown approximately 70 hours by Ball-Bartoe prior to being donated to The University of Tennessee. At UTSI the aircraft was subjected to 97 hours of flight testing under contract to Naval Air Systems Command. The aircraft's performance and handling characteristics were evaluated with and without the upper wing [8,9].

These test programs and analytical studies [9] confirmed the Jetwing concept's capabilities for STOL operation, low speeds, steep approaches, tight maneuvers and low noise levels. The full potential of the concept could not be demonstrated in flight test due to longitudinal instability of the research aircraft and the loss of longitudinal control at low airspeeds. These characteristics are the results of poor design of the test vehicle rather than an inherent flaw in the concept. A summary of the Jetwing characteristics is shown in Table 1.

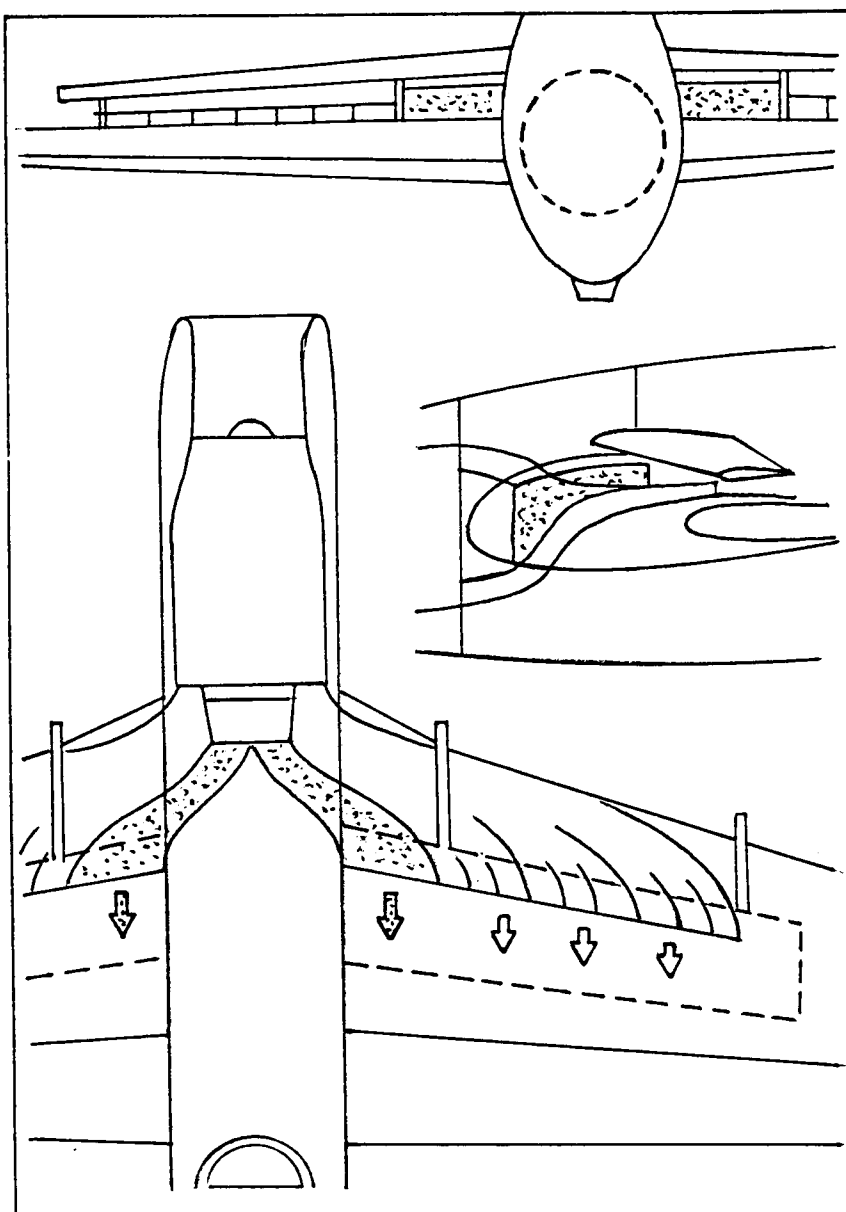


Figure 8: Jetwing Engine Exhaust Ducting Arrangement

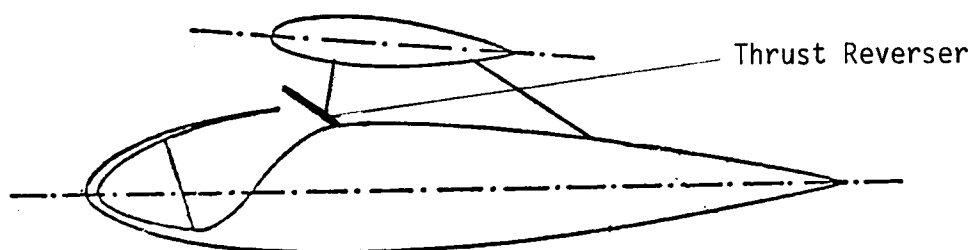
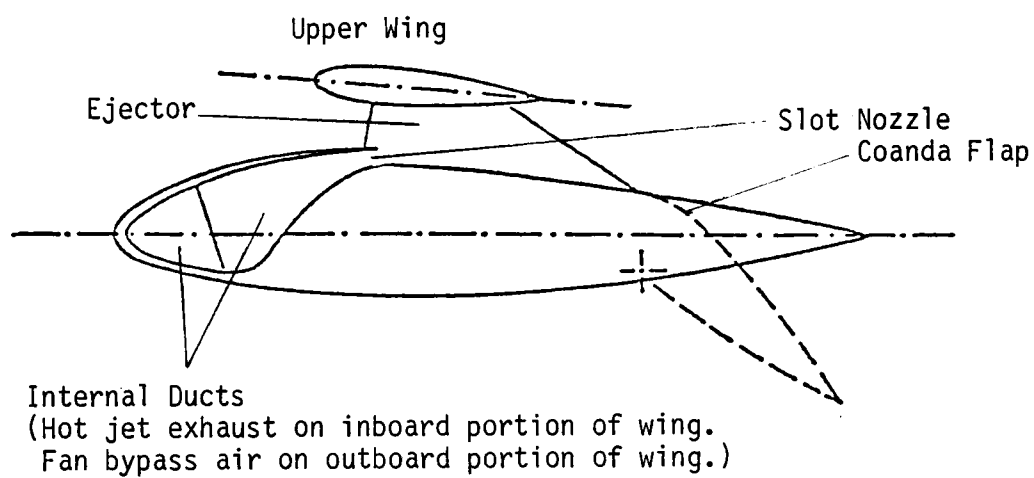


Figure 9: Two-Dimensional Views of the Jetwing Concept

TABLE 1. JETWING TECHNICAL DESCRIPTION

Item	value in British units	value in SI units
<u>Powerplant</u> Pratt & Whitney JT 15 D-1		
Maximum Takeoff Thrust (Static on Test Stand)	2200 lbs	9790 N
Maximum Continuous Thrust (dto.)	2050 lbs	9123 N
Maximum Continuous Thrust (Installed in JW-1)	1750 lbs	7788 N
<u>Weights</u>		
Maximum Takeoff Weight	3750 lbs	16688 N
Empty Weight, with Instrumentation	2730 lbs	12149 N
Ballast	400 lbs	1780 N
Fuel Capacity, Weight	689 lbs	3066 N
Fuel Capacity, Volume	106 gal	397 lit
Typical Center of Gravity Position		35.5 %MAC
<u>Geometry</u>		
Length	28.60 ft	8.717 m
Height	6.10 ft	1.859 m
Wing Span	21.75 ft	6.629 m
Wing Area	105.78 ft ²	9.827 m ²
Mean Aerodynamic Chord	5.08 ft	1.548 m
Aspect Ratio		4.48
Taper Ratio		0.46
Wing Incidence		0 deg
Airfoil at Root	NACA 23020	(modified)
Airfoil at Tip	NACA 23015	
Flap Type	Coanda Single Element	
Flap Span (each)	69 in	1.573 m
Flap Area (each)	10.6 ft ²	0.985 m ²
Flap Deflection		0 to 55 deg
Horizontal Tail Airfoil	NACA 0008	
H.T. Span	9.33 ft	2.844 m
H.T. Area	27.50 ft ²	2.555 m ²
H.T. Aspect Ratio		3.16
Elevator Area	13.25 ft ²	1.231 m ²
Elevator Deflection		+29 to -25 deg
Stabilizer Trim Deflection		+20 to -2 deg
Elevator Flap Chord Ratio		40%
H.T. Volume		0.74

2. FORMULATION OF PROBLEM

2.1 The Definition of Longitudinal Stability and Control (LSC)

Stability and control are usually mentioned together, for they are concerned with similar design criteria, e.g. the sizing of a tail surface, and in a general sense, the loss of stability is equivalent to the loss of control, and vice versa. On the other hand, the two terms aim at inherently conflicting goals, forcing the designer to compromise.

Stability is concerned with maintaining an aircraft's attitude, while control mainly looks at its ability for desired attitude changes. Aircraft that show strong stability tend to resist attitude changes and are therefore less controllable (transport aircraft), while highly maneuverable airplanes generally show only marginal stability (fighters).

The dynamics of steady flight require the sum of moments about the center of gravity of an aircraft to be zero (Figure 10):

$$\sum_i M_{CG_i} = 0 \quad (1)$$

Positive static longitudinal stability exists, if in case of a small departure from a trimmed angle of attack, the aircraft generates restoring moments that move it back to the trimmed condition. Therefore, all pitch-up moments that decrease in magnitude with increasing angle of attack are stabilizing, and so are all pitch-down moments that increase with increasing angle of attack. Conversely, all pitch-up moments that increase with increasing angle of attack, and all pitch-down moments that decrease with increasing angle of attack, are destabilizing.

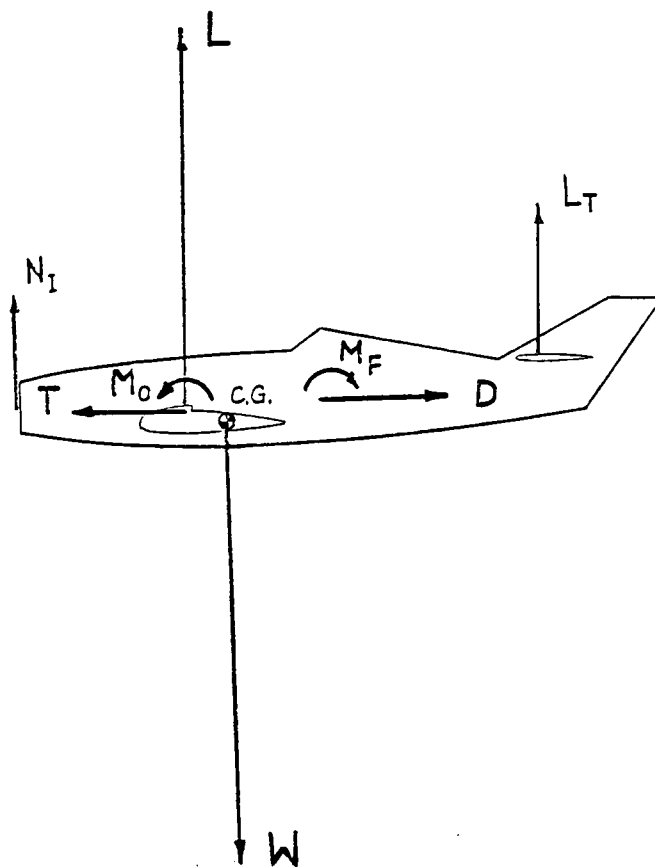


Figure 10: Principal Forces and Moments on the Jetwing Aircraft in Steady Level Flight

A study of longitudinal stability has to consider the contributions of all major aircraft components to the resulting moment about the center of gravity. If defining those moments as positive, which tend to increase the angle of attack (pitch-up), then the mathematical condition for positive static longitudinal stability can be written

$$\frac{\partial M_{CG}}{\partial \alpha} < 0 \quad (2)$$

where M_{CG} is the resulting moment about the center of gravity. In non-dimensional coefficient form this condition writes

$$\frac{\partial C_{m_{CG}}}{\partial \alpha} < 0 \quad (3)$$

Since the angle of attack, α , is rather difficult to measure in flight, this equation is often expressed in terms of the lift coefficient, C_L , which is itself a linear function of α for the small angles that are considered here. The stability condition writes then

$$\frac{\partial C_{m_{CG}}}{\partial C_L} < 0 \quad (4)$$

A study of the longitudinal control power has to investigate whether an elevator (and/or stabilizer) deflection will generate a large enough pitching moment to balance all the moments about the center of gravity according to Equation (1) and, in addition, to allow the pilot to rotate the airplane about its lateral axis for all desired configurations and airspeeds. The limiting condition for the loss of longitudinal control can be expressed as

$$\frac{\partial C_{mCG}}{\partial \delta_e} \rightarrow 0 \quad (5)$$

A more stringent condition for controllability can be formulated if the airplane is required to be rotated at a certain minimum rate of pitch change $\dot{\theta}_{\min}$. Defining $C_{mCG\dot{\theta}_{\min}}$ as the coefficient for the minimum pitching moment which is necessary to overcome inertia and aerodynamic pitch-damping for sustaining $\dot{\theta}_{\min}$, the controllability condition can be written as

$$\left| \frac{\partial C_{mCG}}{\partial \delta_e} \delta_e \right| \geq C_{mCG\dot{\theta}_{\min}} \quad (6)$$

A more detailed discussion of stability criteria follows in Section 5.

2.2 The Influence of Upper Surface Blowing on LSC

Compared to conventional aircraft, the upper surface blown wing has a significant influence on the longitudinal stability and control of the vehicle.

First, the high energy jet on the upper surface of the wing substantially changes its pitching moment, partly due to the momentum of the jet, and partly due to the additional circulation, which changes the pressure distribution on the wing surfaces.

Second, the jet-flapped wing strongly influences the flow field about the lifting wing. The upwash ahead of the wing, and the downwash behind it, both have significant impact on longitudinal stability and control.

Third, due to the interaction between the jet sheet and the wing geometry, it is problematic to determine the exact location and direction of the thrust line. While a major portion of the resulting thrust force is generated by the compressor stages, other portions are contributed from duct pressures in the wing leading edge, and from the momentum of the jet sheet emerging from the slot nozzles, following the upper wing surface and the contour of the Coanda-flap, leaving the trailing edge tangentially, and curving back to the flow direction further downstream. This uncertainty about the line of action of the resulting thrust force with respect to the center of gravity of the aircraft makes it very difficult to determine pitching moments due to thrust.

All of the above mentioned problem areas require special consideration and methods that depart from the treatment of conventional aircraft.

2.3 The Need for Research

Frequent statements in the recent literature point toward a need for research in the field of stability and control for powered lift vehicles, which is the subject of this study. Some are quoted below:

It is recommended that additional studies be conducted on the Jetwing research aircraft with a modified horizontal tail. These studies should be done to confirm that such aircraft can be designed without longitudinal stability and control problems, and to further investigate the low speed capabilities of the concept [8, p. 144].

Since lift and excess thrust coefficients can be reasonably well predicted for Jetwing configurations ... future efforts should be concentrated upon predicting stability and control parameters such as pitching moment coefficients and downwash [9, p. 116].

... the change in the pressure field ahead of the [Jetwing] nozzles ... would create additional suction in the front part of the wing which, if properly accounted for, would contribute to the ... pitching moment. This would bring the theoretical pitching moment characteristic closer to the one obtained experimentally ... Another reason for the discrepancy between theoretical and experimental values of the pitching moment coefficients is the unaccounting of moment created because of turning of the free stream air as it enters the engine inlet. This can only be determined if the upwash at the inlet can be modeled [3, p. 88].

More observations and measurements on the jet wing performance with its near flow field are very much needed. Such data are essential in assisting in further development of theories [10, p. 4].

The calculation of the wing pitching moment requires the accurate determination of the surface pressure distributions with special consideration for the high energy jet sheet on the rear portion of the upper wing surface. This problem was addressed in [3]; the results showed promise, however, they were not quite satisfactory, mainly due to a lack of information about the flow angles at the inlet and at the tail of the airplane. Finding these flow angles was a major concern of this investigation and is outlined in the following sections.

3. EXPERIMENTAL DETERMINATION OF FLOW ANGLES

3.1 Preparation and Conduct of Flight Tests

Since the funds for this research project were very limited, the cost for all equipment had to be kept to a minimum. Data acquisition had to rely on existing equipment and relatively simple methods. The chosen procedure represents a compromise between low cost and required accuracy.

Flow visualization was achieved through woolen tufts that were taped to selected surfaces of the test vehicle and a rake that was mounted to the leading edge of the horizontal tail. The tufts on the vehicle surface were photographed from a camera in a chase aircraft in close formation flight, and the rake was photographed by a camera which was mounted on the test vehicle as shown in Figure 11. This camera* had an automatic film advance capability and a remote trigger which was operated by the pilot of the test aircraft. In addition to the photographic recording of flow angles, the pilot read data from panel instruments and transmitted them via radio to a ground station, where they were hand recorded.

To facilitate the correlation between the photographic pictures and the hand recorded data from panel instruments, a digital clock was installed in the view field of the tail mounted camera. Also, in order to have a reference line for the flow angles indicated by the woolen

*RICOH AUTO HALF E, focal length 25mm, f 2.8.

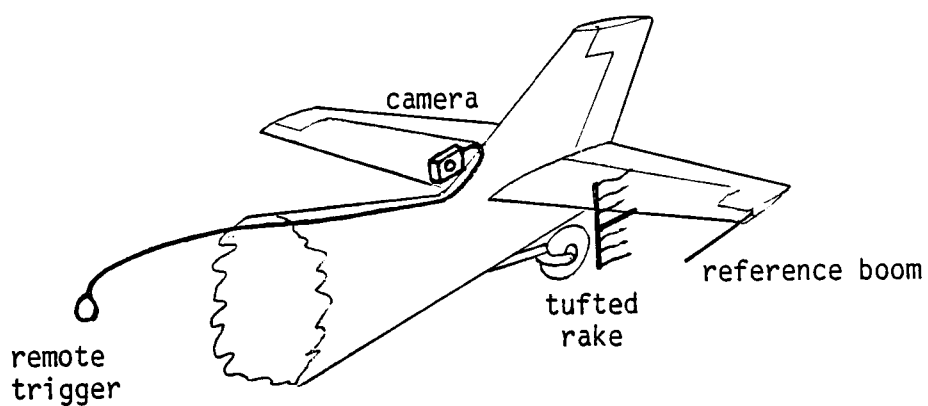


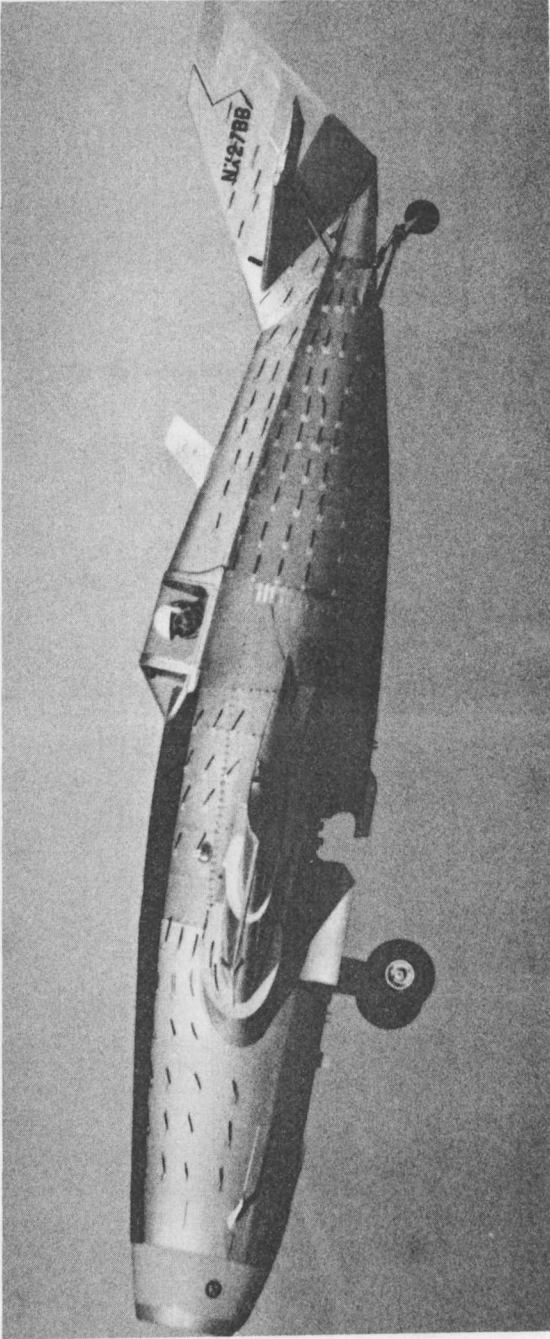
Figure 11: Experimental Set-Up for Downwash Measurements

tufts on the rake, a small boom was mounted on the tip of the stabilizer, marking the extension of the chord centerline of the horizontal tail. Typical examples for photographs taken in flight are shown in Figures 12 and 13.

Flight tests were conducted for a matrix of different airspeeds and flap settings. For each data point, the test aircraft was stabilized in level flight and the pilot triggered the tail mounted camera. At the same time the following data were recorded:

- 1) δ_e - elevator deflection, in instrument scale units
- 2) δ_s - stabilizer position, in instrument scale units
- 3) α - angle of attack, in instrument scale units
- 4) v_i - indicated airspeed, in knots
- 5) h_p - indicated pressure altitude, in feet
- 6) OAT - outside air temperature, in degrees Fahrenheit
- 7) N_1 - engine primary rotor speed, in percent of N_{1max}
- 8) Δp_{hot} - hot duct dynamic pressure, in inches of water
- 9) Δp_{cold} - cold duct dynamic pressure, in inches of water
- 10) V_{frem} - fuel remaining, in gallons

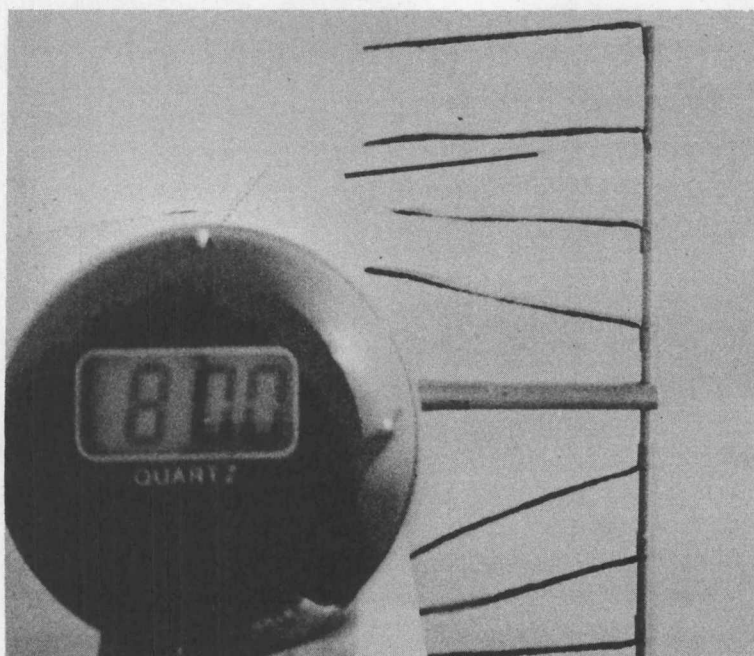
Items four through seven were displayed on standard instruments; the other data were also readily available from special instrumentation that had been installed in the aircraft for previously conducted flight test programs. A detailed description of the instrumentation set up in the test vehicle can be found in [8]. The flight test data are summarized in Table 2.



Date of Flight 9-30-83 Test No. 1 Film No. 830930-3

Flaps	15 deg	h_p	5000 ft
Gear	extended	OAT	59 °F
V_i	62 kts	N_1	83.2 %RPM
α_i	-	Δp_{hot}	67.0 in H ₂ O
δ_e	3 units	Δp_{cold}	104.5 in H ₂ O
δ_s	-10 units	V_{frem}	-

Figure 12: Example for a Photographic Picture of the Test Vehicle
Taken from a Chase Airplane



Date of Flight 6-27-84 Test No. 4 Film No. 5063-2

Flaps	15 deg	h_p	5500 ft
Gear	retracted	OAT	59 °F
v_i	60 kts	N_1	83.2 %RPM
α_i	90 units	Δp_{hot}	67.0 in H ₂ O
δ_e	2 units	Δp_{cold}	105.0 in H ₂ O
δ_s	34 units	V_{frem}	71 gal

Figure 13: Example for a Photographic Picture of the Tufted Rake
Taken by a Jetwing Tail Mounted Camera

TABLE 2. SUMMARY OF FLIGHT TEST DATA

No	Date	Test No	Flaps	Clock	V _i	δ _e	δ _s	α _i	h _{pi}	OAT	N _i	V _{frem}	Δp _{hot}	Δp _{cold}
-	m-d-y	-	deg	hrs:min	kts	units	units	units	feet	°F	%RPM	gal	in. w.	in. w.
1	9-30-83	1	15	-	62	3	-10	-	5000	47	82.0	-	65.0	104.5
2	..	2	..	-	69	3	-15	-	5000	-	81.0	-	63.8	102.0
3	..	3	..	-	61	0	5	27	5280	45	82.0	-	73.6	110.0
4	..	4	30	-	80	5	3	-	-	45	88.0	80.0	90.5	126.0
5	..	5	..	-	70	3	4	-	5340	45	86.0	78.0	81.5	118.0
6	..	6	..	-	60	8	7	-	5360	45	87.0	76.3	87.0	121.5
7	..	7	0	-	80	-5	-23	-	5200	46	75.2	68.4	48.5	85.0
8	..	8	..	-	70	-7	-23	-	5300	45	75.8	66.3	50.0	86.0
9	6-27-84	1	15	7:53	100	-2	40	10	4300	65	79.8	80.0	55.4	98.0
10	..	2	..	7:56	80	0	38	45	4700	63	78.0	75.5	52.2	91.0
11	..	3	..	7:58	70	2	34	55	5100	60	82.8	74.0	66.0	105.5
12	..	4	..	8:00	60	2	34	90	5500	59	83.2	71.0	67.0	105.0
13	..	5	..	8:04	70	-3	40	65	5400	59	81.0	65.0	59.0	98.6
14	..	6	..	8:05	80	-3	36	48	5800	58	81.2	63.5	59.4	99.0
15	..	7	30	8:08	80	3	36	20	6250	57	86.1	57.5	77.0	113.5
16	..	8	..	8:09	70	3	34	40	6500	57	86.0	56.0	77.0	110.5
17	7-11-84	1	30	(8:45) ¹	60	7	55	-	5000	61	90.0	79.0	96.2	130.5
18	..	2	..	(8:47)	70	3	72	48	5200	61	85.0	75.0	62.0	110.0
19	..	3	..	(8:48)	80	5	52	25	4750	60	88.0	-	74.6	120.2
20	..	4	0	(8:52)	100	-7	26	53	6700	58	74.2	65.2	41.4	68.5
21	..	5	..	(8:55)	90	-6	23	60	7000	56	76.2	62.7	45.0	72.8
22	..	6	..	(8:57)	80	-5	23	68	7300	55	79.0	60.5	50.0	87.5
23	..	7	..	(8:59)	100	-7	23	45	7800	55	79.0	58.5	49.5	88.0
24	..	8	..	(9:00)	120	-7	20	35	8250	55	79.5	56.0	48.5	90.0
25	7-20-84	1	0	8:52	100	-10	40	57	2400	69	66.0	86.0	31.5	65.5
26	..	2	..	9:18	90	-11	30	63	3350	65	71.5	45.6	40.0	67.5
27	9-21-84	1	30	(8:36)	60	5	50	17	3400	62	86.0	75.5	82.0	120.5
28	..	2	..	(8:39)	90	3	50	15	3700	62	86.2	71.5	80.5	121.0
29	..	3	0	(8:42)	80	-15	53	72	3700	62	68.8	66.5	35.6	69.0
30	..	4	..	(8:44)	70	-13	53	85	3720	62	78.0	60.3	54.0	92.5
31	6-19-85	1	0	8:21	80	-8	55	73	4260	52	76.0	84.0	51.5	88.0
32	..	2	..	8:24	70	-10	55	95	5400	52	75.3	81.7	50.0	84.0
33	..	3	30	8:27	60	5	58	55	4500	50	90.0	75.0	103.0	134.0
34	..	4	..	8:30	90	5	59	18	4700	50	86.0	72.4	81.0	119.0
35	..	5	45	8:33	70	7	-3	15	4400) ²	50	91.2	65.0	110.0	139.0
36	..	6	..	8:34	60	10	-15	40	4300) ³	50	91.0	62.5	-	149.0

¹ Clock times in parenthesis indicate no pictures from tail mounted camera.

² Descending flight path, @ 200 fpm.

³ Descending flight path, @ 250 fpm.

3.2 Data Evaluation

During the preliminary testing of the tail mounted camera equipment, some photographs were made with a rectangular grid held in the field of vision. This allowed checking the pictures for distortions which may have been introduced by the wide angle lens and angularities in the camera mount. The angular distortions proved to be negligible; therefore the flow angles indicated on the photographs by the tufts were true angles and could easily be measured with a protractor.

Most tufts on the rake showed flutter at some angle and airspeed. Those tufts were not considered for the flow angle determinations. Later in the program, all tufts were shortened from six to four inches, which reduced the flutter tendency substantially.

Due to the closeness to the leading edge of the stabilizer, the flow angles indicated by the tufted rake were significantly influenced by the horizontal tail. Since the main objective of this research was the air flow change due to the lifting wing, the additional changes induced by the horizontal tail had to be subtracted from the data. This was done by means of the computer program that modeled the airflow about a lifting surface.

The data reduction sequence and the flow angle correction method are outlined as follows:

1. All panel data were corrected for instrument errors. Airspeed and altimeter readings were also corrected for position errors, which stem from deviations of the static pressure, measured at the location of the static pressure probe, from

the free stream static pressure. Calibration plots for all these corrections are kept on file at UTSI's Flight Research Laboratory and were readily available for this research.

2. From the calibrated flight test data the following quantities were calculated:

- a) True airspeed

$$v_{tr} = \frac{v_c}{\sqrt{\sigma}} \quad (7)$$

The density ratio $\sigma = \rho/\rho_0$ is a function of calibrated pressure altitude h_p and outside air temperature OAT_c and is easily determined from a density altitude chart [11].

- b) Air Density at Test Altitude

$$\rho = \rho_0 \sigma \quad (8)$$

where ρ_0 as the sea level air density of a standard day [11], and σ determined according to a).

- c) Aircraft Weight at Test Time

$$W_t = W_{maxto} - \rho_f(V_{fmax} - V_{frem}) \quad (9)$$

The maximum take-off weight of the test aircraft was determined by weighing the vehicle with full fuel tank, pilot, parachute and the complete package of instruments to be used in flight test. The volume of remaining fuel at the time of the experiment was readily available from the flight test data.

d) Lift Coefficient

$$C_L = \frac{L_W}{qS} \quad (10)$$

with the dynamic pressure

$$q = \frac{\rho}{2} v_{tr}^2 = \frac{\rho_0}{2} v_c^2 \quad (11)$$

The wing lift force L_W follows from the equilibrium condition of vertical forces in steady level flight (Figure 10, page 13):

$$L_W = W_t - L_T \quad (12)$$

where the tail lift force L_T could be obtained from the balance of moments about the center of gravity:

$$L_T = \frac{M_{CGTO}}{x_T} = C_{mCGTO} \frac{\bar{c}}{x_T} qS \quad (13)$$

The tail off pitching moment coefficient C_{mCGTO} was available from wind tunnel data [7] (Figure 14).

e) Blowing Coefficient

$$C_J = \frac{F_G}{qS} \quad (14)$$

where the gross thrust F_G is a function of δ , θ , and the engine rotor speed N_1 . The ratio of static pressures

$$\delta = \frac{p}{p_0} \quad (15)$$

is a function of the test pressure altitude h_p and was obtained from atmospheric tables [12]. The ratio of

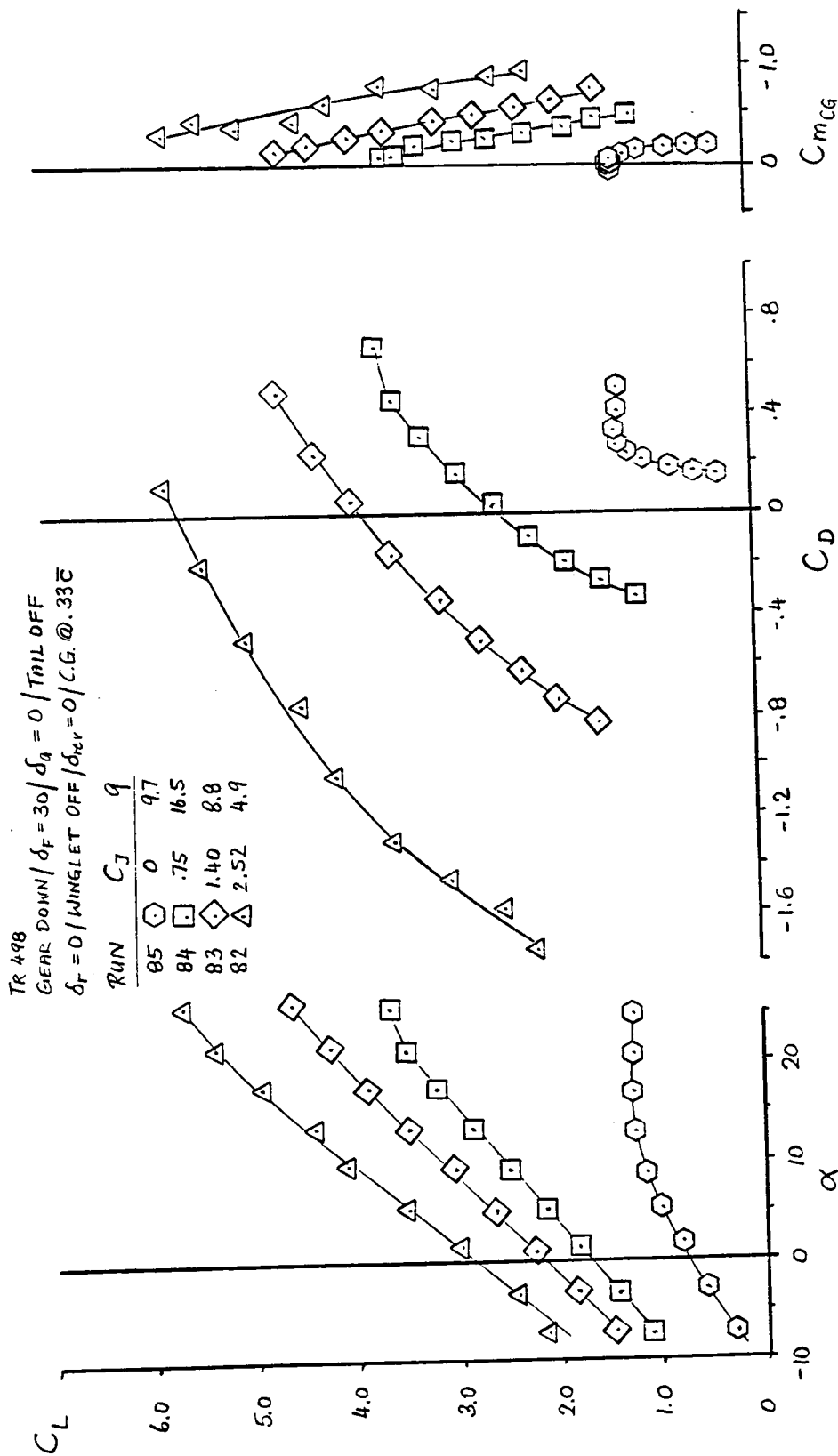


Figure 14: Example for Jetwing Wind Tunnel Data

absolute temperatures

$$\theta = \frac{T}{T_0} \quad (16)$$

could be easily calculated from flight test data with

$$\frac{T}{\text{Kelvin}} = \frac{\text{OAT}}{\text{Celsius}} + 273 \quad (17)$$

and

$$\frac{T_0}{\text{Kelvin}} = 288 \quad (18)$$

The engine rotor speed N_1 in percent of $N_{1\max}$ was directly available from flight test data. With these quantities evaluated, the gross thrust F_G could be determined from a calibration diagram which was established in [13] and is shown in Figure 15.

f) Angle of Attack

$$\alpha = f(C_L, C_J) \quad (19)$$

This angle between the free stream direction and the wing chord line is very difficult to measure in flight. The Jetwing angle of attack sensors mounted on the side of the forward fuselage and on a wingtip boom are both subject to angular distortions of the flow field about the airplane and its lifting surfaces. Therefore, the angle of attack was determined by entering the C_L and C_J values into the proper wind tunnel data diagrams [7].

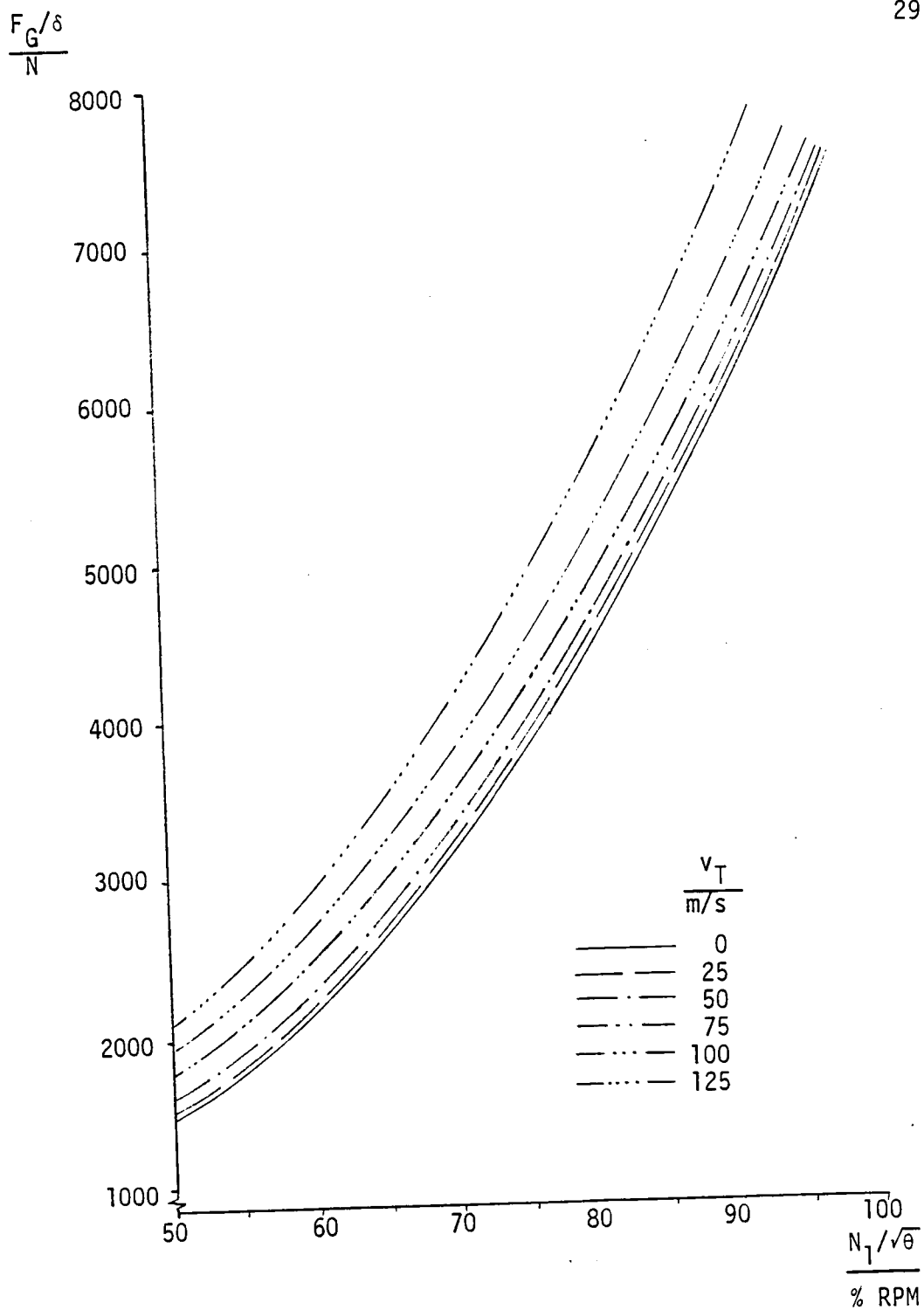


Figure 15: Jetwing Thrust Calibration Diagram

3. To correct the measured flow angles at the rake for distortions from the horizontal tail, the following procedure was used:

- a) With the wing geometry, the flap deflection δ_f , angle of attack α , air density ρ , free stream velocity u_∞ , and the blowing coefficient C_j as input data, the vortex-lattice computer program produced the flow angle ϵ_T and velocity u_T at the aerodynamic center position of the horizontal tail.
- b) With the stabilizer trim position angle i_T from the flight test data, the tail angle of attack could be determined according to Figure 16:

$$\alpha_T = \alpha + i_T - \epsilon_T \quad (20)$$

- c) With the horizontal tail geometry, the elevator deflection δ_e , tail angle of attack α_T , air density ρ , and tail free stream velocity u_T as input data, the vortex lattice computer program produced the flow angles ϵ at the rake position. These are the desired correction angles.
- d) The test data flow angles at the rake were corrected for the influence of the horizontal tail according to Figure 17:

$$\epsilon_{mc} = \alpha + i_T - (\mu - \bar{\epsilon}) \quad (21)$$

The angles ϵ_{mc} are the measured downwash angles, corrected for the distortions from the horizontal tail. They serve as a reference data base for the theoretically determined flow angles ϵ_{th} . Differences

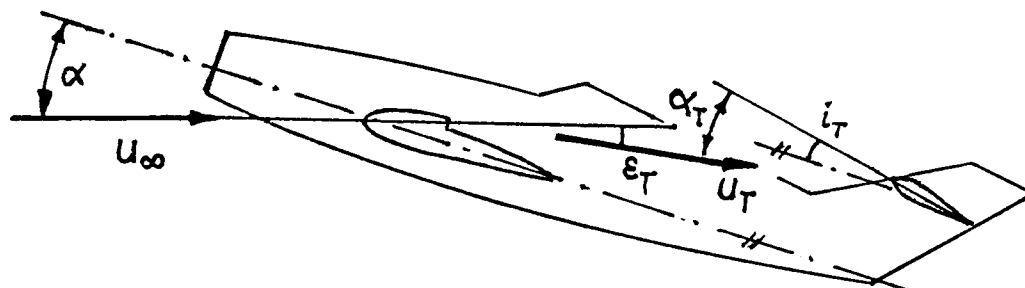


Figure 16: Flow Angles at the Horizontal Tail

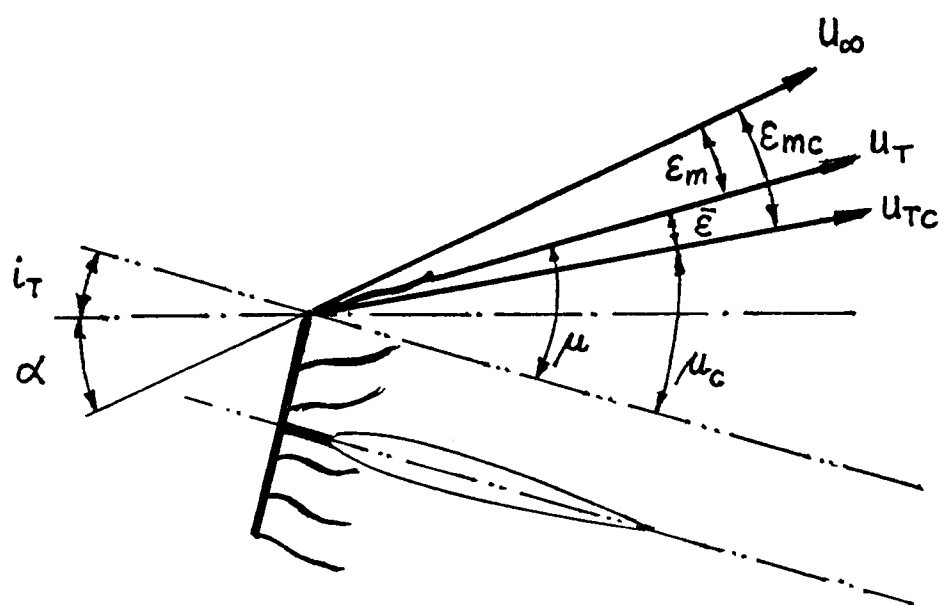


Figure 17: Flow Angle Correction for the Influence of the Horizontal Tail

reflect the influence of the fuselage, viscous effects, and inaccuracies in the experimental procedures and the vortex lattice routine. This will be discussed further in Section 6.

The results of the flight test data reduction, measured flow angles and corrections are summarized in Table 3.

TABLE 3. RESULTS OF FLIGHT TEST DATA REDUCTION

No	Date	Test No	Flaps	v_{tr}	ρ	C_J	C_L	α	δ_{el}	δ_{st}	μ	α_T	$\bar{e}$	ϵ_m	ϵ_{mc}	ϵ_{th}
-	m-d-y	-	deg	m/s	kg/m ³	-	-	deg	deg	deg	deg	deg	deg	deg	deg	deg
1	6-27-84	1	15	56.07	1.0425	.2497	.9688	4.5	-1.5	2.05	-1.5	1.29	.33	8.05	8.38	6.33
2	..	2	..	45.16	1.0311	.3569	1.4974	10.0	-1.0	1.95	0.0	3.85	1.70	11.95	13.65	9.81
3	..	3	..	39.98	1.0235	.5264	1.9194	12.0	-.5	1.75	0.0	2.30	1.06	13.75	14.81	13.37
4	..	4	..	34.52	1.0051	.7157	2.6067	18.0	-.5	1.75	2.0	2.75	1.28	17.75	19.03	19.37
5	..	5	..	40.20	1.0124	.4940	1.8869	12.5	-2.0	2.05	2.0	4.01	1.56	12.55	14.11	12.62
6	..	6	..	45.87	.9996	.3819	1.4634	9.0	-2.0	1.80	1.0	2.39	.77	9.80	10.57	9.98
7	..	7	30	47.06	.9853	.4226	1.3942	2.0	.0	1.80	-11.0	-5.39	-3.14	14.80	11.66	9.77
8	..	8	..	40.30	.9853	.5763	1.8970	5.0	.0	1.70	-9.2	-5.31	-3.09	15.90	12.81	12.70
9	7-20-84	1	0	53.88	1.1100	.1692	.9994	9.5	-4.0	2.05	6.0	9.33	3.42	5.55	8.97	3.20
10	..	2	..	48.24	1.0800	.1961	1.1823	11.5	-4.2	1.50	5.5	10.26	3.72	7.50	11.22	3.94
11	6-19-85	1	0	43.81	1.0740	.3542	1.5519	14.0	-3.2	2.80	1.5	11.82	4.49	15.30	19.79	6.55
12	..	2	..	39.35	1.0273	.4282	2.0035	18.2	-4.0	2.80	1.0	11.09	4.06	20.00	24.06	11.22
13	..	3	30	33.21	1.0680	.9406	2.6703	12.0	.7	2.98	-9.0	1.41	.89	23.98	24.87	15.51
14	..	4	..	50.31	1.0620	.3687	1.1649	1.0	.7	3.02	-8.5	-4.04	-2.12	12.52	10.40	8.64
15	..	5	45	39.45	1.0660	.6882	1.8605	-1.0	1.2	-.15	-22.0	-14.24	-9.11	20.85	11.74	12.75
16	..	6	..	33.12	1.0740	.9595	2.6073	4.2	2.0	-.90	-22.0	-24.63	-16.35	25.30	8.95	26.09

4. THEORETICAL DETERMINATION OF FLOW ANGLES

4.1 Literature Findings

Aerodynamicists realized the powerful lift augmenting effects of a jet-flap as early as 1938 in Germany [2], but the phenomenon remained dormant for many years since the engine technology had not advanced enough to supply a jet-flapped wing with a high energy jet sheet of sufficient mass flow.

When the development of turbojet engines made rapid progress in the 1950's, particularly when two and three spool engines with high bypass ratios came into existence, the supply of high energy air mass flow in flight was no longer a problem, and the interest in the jet-flapped wing was newly sparked.

A large number of analytical methods for predicting the aerodynamic characteristics of jet-flapped wings have since been developed in various countries. They include simple handbook type methods relying largely on empirical formulae [14], as well as very sophisticated methods that employ complex analytic modeling of the real aerodynamics [15].

The basic theoretical properties of a thin, two-dimensional jet in inviscid flow were formulated by Maskell and Gates [16] in 1955. Spence's classic paper solved the two-dimensional jet-flapped wing problem in 1956 for the jet ejection from the trailing edge [17], and over a plain hinged flap [18], using a Lifting Line analysis. Together with Maskell, he extended the analysis to the three-dimensional case of a finite wing with full-span blowing and constant jet angle. Wing

loading, chord, and blowing, are assumed to have elliptical distribution along the span.

With these restrictive assumptions, the section lift coefficient, C_ℓ , section blowing coefficient, C_j , and the downwash angle, α_i , are constant along the span. The total lift coefficient, C_L , is equal to C_ℓ , and the overall blowing coefficient C_j equals C_j .

To allow for part-span blowing and non-elliptical spanwise distribution, Williams added some empirical corrections to Spence's basic approach [14].

Jacobs published a semi-empirical method in 1972, which replaces the jet-flap by an equivalent mechanical flap, both, for two and three dimensions [19,20].

Both methods predict overall performance parameters of a jet-flapped wing very well, as could be shown during a UTSI research effort by Kimberlin, Solies, and Sinha [9]. Since these methods are relatively simple, they are considered valuable tools for the preliminary design of jet-flapped aircraft. They cannot, however, serve for the analysis of the flow field about the blown wing. For this, a different analytical method has to be used.

The classic paper for predicting the downwash angles and the wake characteristics for the power off conditions behind conventional plain and flapped wings was published by Silverstein, Katzoff, and Bullivant in 1939 [21]. The downwash calculations are based on a linear Lifting Line Theory, with bound horseshoe vortices at the wing quarter chord, and semi-infinitely long straight line vortices, forming the planar

trailing vortex sheet (Figure 18). No curvature of the vortex sheet is introduced, but as an approximation for the purpose of calculating the downwash at the tail, the entire sheet is considered uniformly displaced downward by the amount equal to the displacement of the center of the sheet near the tail. In spite of these relatively crude simplifications, the resulting downwash calculations were accurate enough to warrant the publication of design charts with parametric variations of wing geometry and tail positions [22].

A theoretical evaluation of the downwash behind jet-flapped wings was given by Ross in 1961 [23]. It is based on Maskell and Spence's solutions for the jet sheet. The wing incidence and jet deflection are assumed to be small, and the displacement of the jet is taken into account by displacing the downwash field so that the relative distance between the jet and the tailplane is correct. It is also assumed that the spanwise loading of the finite wing is elliptic and that the trailing vorticity is generated at one particular chordwise position. The wake is considered to be a flat sheet, and so the effect of the roll-up of the wake has been neglected.

A jet-flapped wing vortex lattice theory with nonlinear wake and tip flows was developed by Chen and Wu in 1983 [10]. This method allows for large angles of attack and jet deflection and arbitrary wing and flap geometry; the mutual interaction of all vortices is accounted for and the shape of wing tip and wake vortices is determined with great accuracy within the limitations of the potential flow assumptions. The method does not consider a roll-up of the jet sheet, but its shape is

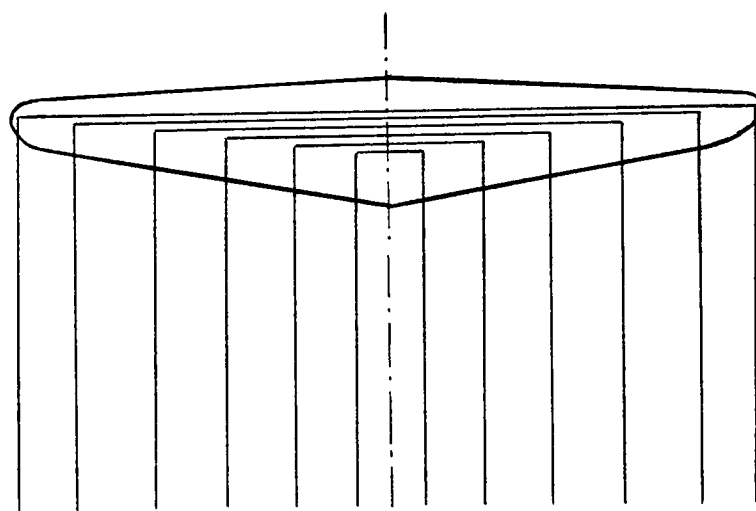


Figure 18: Horseshoe Vortex Distribution for Downwash Calculations
Used in NACA TR 651

streamlined through iteration, such that its position is very close to the actual one.

Even though this method models a quite complicated flow field very accurately, it can still be considered relatively simple since its basic principle is the discrete singularity distribution of elementary horse-shoe vortices, based on the Lifting Line Theory, to which the Biot-Savart Law can be applied.

Therefore, this method was considered a good compromise between required accuracy and economy of computer time and was chosen as a tool for the flow field studies, which are subject to this dissertation. Additional information on these methods can be found in [3] and [24].

4.2 Vortex-Lattice Method and Computer Program

4.2.1 General

A three-dimensional nonlinear vortex-lattice method, applied to the wing, jet sheet, wake and tip flows has been developed and programmed in a FORTRAN computer code.

The program determines the aerodynamic performance of a jet-flapped wing as well as flow directions and velocities at arbitrary points in the flow field about it. The basic approach to the problem is similar to that one outlined by Chen and Wu in [10]. Elementary vortex elements represent the wing, wake, tip vortices and jet sheet. The wing thickness is not considered, at this point. The camber of the airfoil centerline, however, is reflected in the geometry of the lifting surface representing the wing. The mutual interaction of all vortices is accounted for. Since the shapes of the jet sheet, wake, and tip

vortices are unknown a priori, they are determined simultaneously as part of the solution. The posed problem is subject to two boundary conditions:

1. The kinematic boundary condition, which requires that the flow is tangent to all solid surfaces:

$$w_n = 0 \quad (22)$$

where w_n is the normal velocity at a surface element.

2. The dynamic boundary condition, which states that an element of the jet sheet with the momentum J' and the radius of curvature R can sustain the pressure differential

$$\Delta p = - \frac{J'}{R} \quad (23)$$

The mixed boundary conditions as well as the unknown positions of the jet sheet and free vortices make the problem very difficult and require iterative techniques for its solution. Details of the program are presented in the following paragraphs.

4.2.2 Potential Flow

The basic underlying principle of this and other vortex-lattice methods is the assumption of an incompressible, non-viscous fluid. Even though this idealized fluid does not exist in nature, this potential flow model greatly simplifies the mathematical treatment of certain problems in fluid dynamics. Lift forces and large scale flow patterns can be determined with very good accuracy. The model fails to predict

phenomena like viscous drag, turbulence, dissipation, etc. In potential flow all vorticity is restricted to singularities and the velocities in the flow field result from direct superposition of all contributing elements [25].

4.2.3 Biot-Savart Law

The fundamental law, which is employed for all velocity computations in the potential flow field, is the Law of Biot and Savart [26], which was originally formulated in electrodynamics to describe the magnetic field strength around a wire conducting a current. In its aerodynamic analogy the law formulates the induced velocity $d\vec{w}$ at a point P (Figure 19a) due to a vortex filament $d\vec{s}$ of strength Γ :

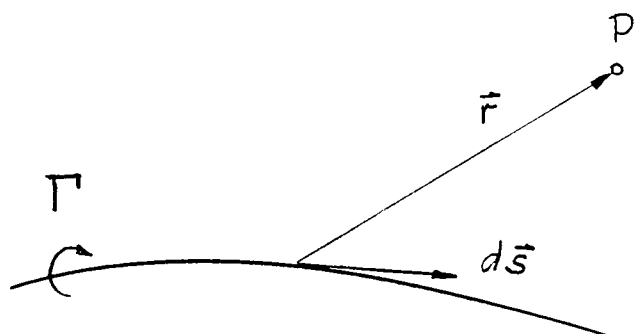
$$d\vec{w} = \frac{\Gamma}{4\pi} \frac{\vec{r} \times d\vec{s}}{r^3} \quad (24)$$

For a straight vortex filament of length AB and strength Γ (Figure 19b) this becomes

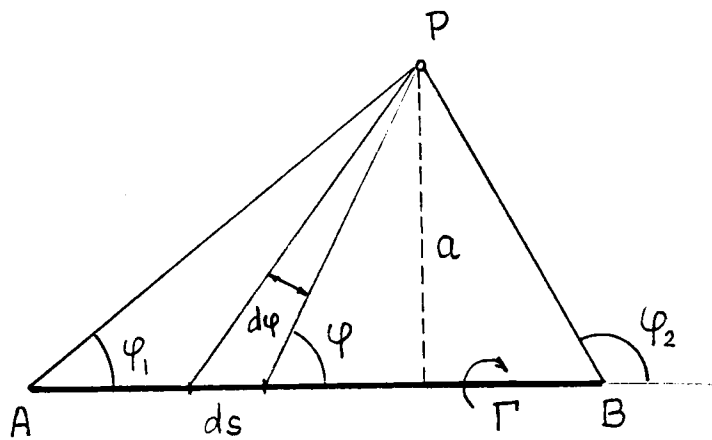
$$w = \frac{\Gamma}{4\pi a} (\cos \phi_1 - \cos \phi_2). \quad (25a)$$

4.2.4 Vector Analysis

All points of interest on the wing geometry or in the flow field are referenced in Cartesian coordinates as sketched in Figure 20. This makes the use of vectors for the solution of geometrical problems particularly convenient; the consistent use of vector analysis eliminates the need for determining angles and for using trigonometric functions, which consume valuable computer time and would be necessary if conventional geometric analysis were used.



a)



b)

Figure 19: Parameters for the Application of the Biot-Savart Law

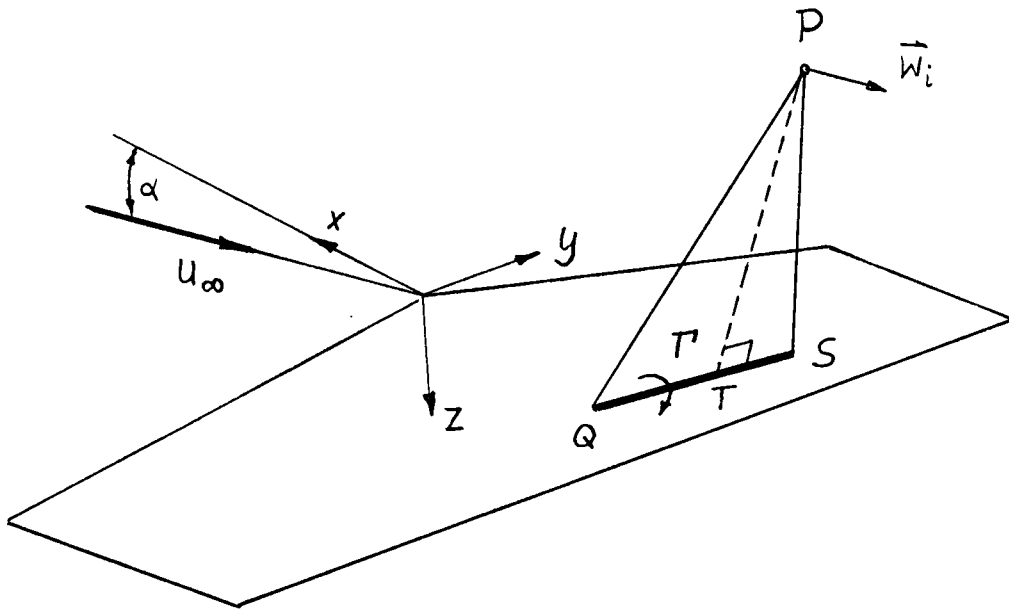


Figure 20: Coordinate System and Sample Vortex Filament with Induced Velocity at a Point in the Flow Field

4.2.4.1 Finding Point T (Figure 21a,b)

A frequent problem for the application of the Biot-Savart Law is the finding of the shortest distance of the point P from the vortex filament. Since the vectors $\overline{PT}$ and $\overline{QS}$ are perpendicular to each other, their dot product is zero:

$$\overline{PT} \cdot \overline{QS} = 0 \quad (26)$$

where

$$\overline{PT} = \overline{OT} - \overline{OP} \quad (27)$$

and

$$\overline{QS} = \overline{OS} - \overline{OQ} \quad (28)$$

Writing the vectors in component representation, yields

$$\overline{OP} \equiv (XP, YP, ZP) \quad (29)$$

$$\overline{OQ} \equiv (XQ, YQ, ZQ) \quad (30)$$

$$\overline{OS} \equiv (XS, YS, ZS) \quad (31)$$

These vector components are identical with the Cartesian coordinates of the points P, Q, and S and are readily available in the computer program. The components for the unknown point T write similarly

$$\overline{OT} \equiv (XT, YT, ZT) \quad (32)$$

With these components, the dot product Equation (26) can be written

$$(XT-XP)(XS-XQ) + (YT-YP)(YS-YQ) + (ZT-ZP)(ZS-ZQ) = 0 \quad (33)$$

Another vector equation incorporating point T can be written as

$$\overline{OT} = \overline{OQ} + \lambda \overline{QS} \quad (34)$$

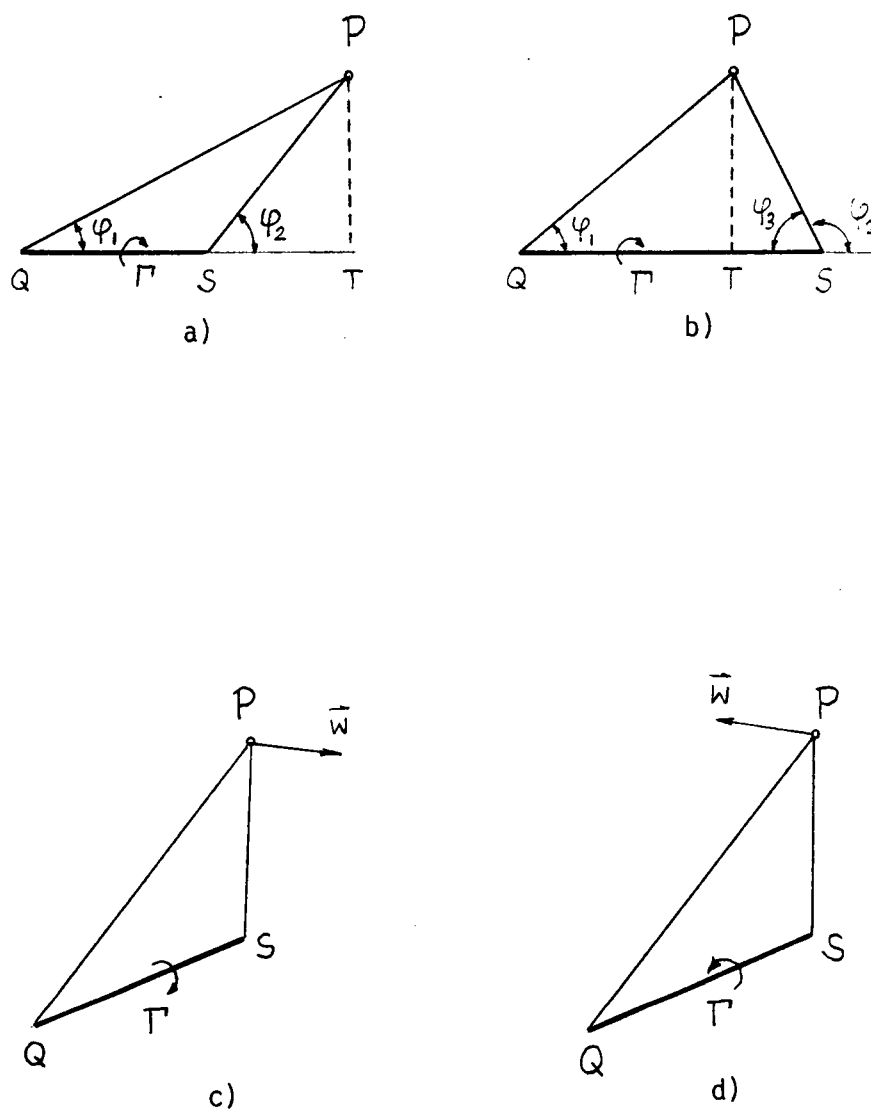


Figure 21: Different Geometries and Vortex Spins for the Application of the Biot-Savart Law

In component form, this yields three equations:

$$X_T = X_Q + \lambda(X_S - X_Q) \quad (35)$$

$$Y_T = Y_Q + \lambda(Y_S - Y_Q) \quad (36)$$

$$Z_T = Z_Q + \lambda(Z_S - Z_Q) \quad (37)$$

Equations (33), (35), (36), and (37) are a system of linear equations for the unknowns X_T , Y_T , Z_T , and λ and can easily be solved. Cramer's rule was used in this case, since the system is small and has many zero entries.

4.2.4.2 Computing Induced Velocity w (Figure 21c,d)

Recalling the Biot-Savart law, Equation (25a),

$$w = \frac{\Gamma}{4\pi a} (\cos \phi_1 - \cos \phi_2) \quad (25a)$$

Introducing the angle $\phi_3 = 180^\circ$, this equation writes

$$w = \frac{\Gamma}{4\pi a} (\cos \phi_1 + \cos \phi_3) \quad (25b)$$

With the coordinates of the points P, Q, S, and T known, these equations can be written as follows:

$$w = \frac{\Gamma}{4\pi PT} \left(\frac{QT}{PQ} - \frac{ST}{PS} \right) \quad (38a)$$

and

$$w = \frac{\Gamma}{4\pi PT} \left(\frac{QT}{PQ} + \frac{ST}{PS} \right) \quad (38b)$$

where the distances PT, etc., are easily computed by using Pythagoras'

theorem:

$$PT = \sqrt{XPT^2 + YPT^2 + ZPT^2} \quad (39)$$

where

$$XPT = XT - XP \quad (40)$$

$$YPT = YT - YP \quad (41)$$

$$ZPT = ZT - ZP \quad (42)$$

4.2.4.3 Finding Velocity Vector $\vec{w}$

While the Biot-Savart Law (38) yields the length of the induced velocity vector, its direction in Cartesian coordinates is yet unknown. Here the vector cross product can help (Figure 21b):

$$\vec{c} = \overline{PQ} \times \overline{PT} \quad (43a)$$

In coordinate form, this writes

$$XC = YPQ \times ZPT - ZPQ \times YPT \quad (44)$$

$$YC = ZPQ \times XPT - XPT \times ZPT \quad (45)$$

$$ZC = XPQ \times YPT - YPQ \times XPT \quad (46)$$

If the vortex spin is reversed (Figure 21d), Equation (43a) becomes

$$\vec{c} = - \overline{PQ} \times \overline{PT} = \overline{PT} \times \overline{PQ} \quad (43b)$$

The vector $\vec{c}$ is normal to $\overline{PQ}$ and $\overline{PT}$ and therefore has the proper direction of the induced velocity vector, but it has yet to be scaled

$$\vec{w} = \lambda \vec{c} \quad (47)$$

where

$$\lambda = \sqrt{\frac{w^2}{XC^2 + YC^2 + ZC^2}} \quad (48)$$

The components of the induced velocity vector are finally

$$XW = \lambda XC \quad (49)$$

$$YW = \lambda YC \quad (50)$$

$$ZW = \lambda ZC \quad (51)$$

Careful attention must be given to the orientation of the vortex and the relative position of the point P to the end points of the vortex filament, for this determines whether Equations (38a) or (38b), and (43a) or (43b), are applicable.

4.2.5 Vortex Representation of Jet Sheet

An analysis of a curved jet sheet in two-dimensional flow was given by Preston [27] and extended by Spence [17]. An element of the jet, as sketched in Figure 22, with thickness δ , mean velocity v_J , and radius of curvature R , can sustain the pressure difference

$$p_1 - p_2 = - \frac{\rho v_J^2}{R} \quad (52)$$

The linear strength of a vortex distributed along the centerline of the jet is shown to be

$$\gamma_J = \frac{u}{R} \left(\frac{J'}{\rho u^2} - \delta \right) \quad (53)$$

With u as the average of the velocities above and below the jet, and

$$J' = \rho v_J^2 \delta \quad (54)$$

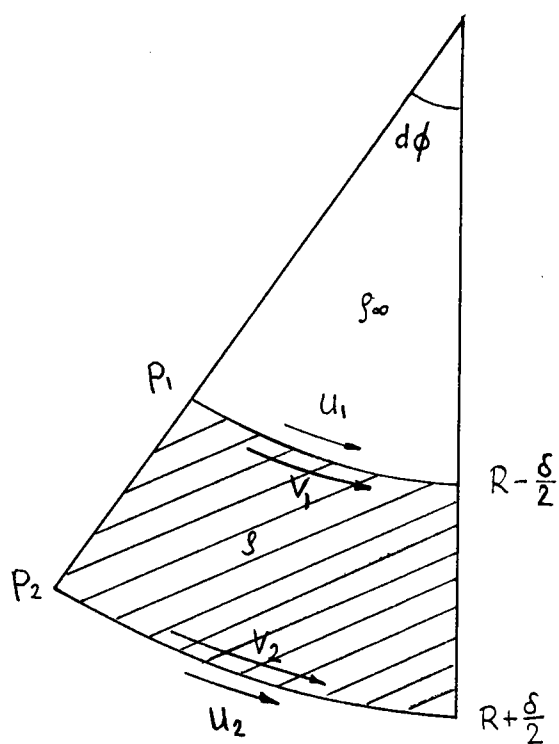


Figure 22: Element of a Curved Jet Sheet in Two-Dimensional Nonviscous Flow

the local momentum flux of the jet per unit span. Keeping the momentum flux constant and letting the jet thickness go to zero, with the jet velocity approaching infinity, yields

$$p_1 - p_2 = \frac{J'}{R} \quad (55)$$

and

$$\gamma_J = \frac{J'}{\rho u R} \quad (56)$$

The total strength of a jet sheet of unit span is

$$\Gamma_J = \int_{te}^{\infty} \gamma_J ds \approx \frac{J'}{\rho_{\infty} u_{\infty}} (\delta_J + \alpha) \quad (57)$$

Details of these derivations are shown in Spence's paper [17].

Going to three dimensions requires special considerations of partial span blowing and downwash effects due to the trailing vortices. This is outlined in the following:

A key parameter in the theoretical and experimental treatment of powered lift is the blowing coefficient

$$C_J = \frac{\dot{m}_J v_J}{qS} \quad (58)$$

where $\dot{m}_J v_J = J$ is the jet momentum, which can be measured experimentally and represents the product of actual values of mass flow $\dot{m}_J$ and the velocity v_J . For design purposes these two quantities are usually determined theoretically by assuming nonviscous isentropic flow. The ideal values $\dot{m}_{\mu}$ and v_{μ} can be used for the definition of the blowing coefficient

$$C_{\mu} = \frac{\dot{m}_{\mu} v_{\mu}}{qS} \quad (59)$$

This theoretical blowing coefficient C_{μ} is usually higher than the corresponding actual blowing coefficient C_J , since viscous effects reduce nozzle exit areas, result in scrubbing losses, etc. Therefore, for accurate predictions, appropriate corrections have to be made.

In wind tunnel and flight tests the Jetwing blowing coefficient was determined as

$$C_J = \frac{F_G}{qS} \quad (60)$$

The measured gross thrust F_G reflects the losses in the jet momentum J . Another problem, however, occurs through the use of the wing reference area S for the partially blown wing. Theoretical analyses like Spence's [17] are based on a lifting surface with full span blowing and do not consider fuselage cut-out or part-span blowing. The overall three-dimensional blowing coefficient C_J of a part-span blown wing does not represent the same blowing per unit span as the blowing coefficient C_J of the full-span blown wing.

Comparing, for example, two wings with the same reference areas, one full-span blown and the other half-span blown, then for the same blowing coefficient C_J the sectional blowing intensity per unit span, J/b , would be twice as high for the half-span blown wing.

In using the theoretical formulae that were derived from full-span blown wings, it is important to use similar blowing intensities for the calculation of derivatives, etc. Therefore, the blowing coefficient has to be modified for part-span blown wings by defining a "mean

sectional blowing coefficient"

$$C_j = \frac{\dot{m}_j v_j}{q S_B} \quad (61)$$

where S_B is the wing reference area corresponding to the spanwise extent of blowing. In the chosen example the blown reference area becomes

$$S_B = \frac{1}{2} S \quad (62)$$

and

$$C_j = 2C_j \quad (63)$$

With the blowing coefficient defined this way, the blowing intensity $\dot{m}_j v_j / b$ is the same for both wings, if their sectional blowing coefficients are equal.

This correction method is used, e.g., by Williams in [14]. Others also consider the "carry over effect" that the blown portion of a part-span blown wing has on the unblown area, by using a reference area

$$S' = k_{C0} S_B \quad (64)$$

with a carryover factor $k_{C0} > 1$ [28].

In the computer program that was developed during this research effort, k_{C0} could be freely chosen and was generally assumed to be one. For selected conditions, it could be determined iteratively by matching the computed lift coefficient with wind tunnel data. The sectional blowing coefficient C_j relates to the overall blowing coefficient C_j as follows:

$$C_j = \frac{C_j S}{k_{CO} S_B} = \frac{F_G}{q k_{CO} S_B} \quad (65)$$

For the Jetwing test aircraft the ratio of the blown reference area to the wing area is

$$\frac{S_B}{S} = .573 \quad (66)$$

The three-dimensional jet momentum is

$$J = \dot{m}_J v_J = J' b_J \quad (67)$$

where b_J as the spanwise extent of the blowing. The sectional blowing coefficient can be written

$$C_j = \frac{J}{q S_B} = \frac{J'}{q \bar{c}_B} \quad (68)$$

where $\bar{c}_B$ as the mean geometric wing chord of the blown section. From Equations (56) and (68) follows the linear strength of a vortex sheet replacing the jet:

$$\gamma_J = \frac{u C_j \bar{c}_B}{2R} \quad (69)$$

This equation is the "dynamic boundary condition", which relates the jet sheet vorticity to its shape (R) and to the flow velocity in its immediate vicinity (u). The strength of a discrete vortex on a jet sheet panel i becomes

$$\Gamma_{Ji} = \int_{x_1}^{x_2} \gamma_J ds = \frac{u C_j \bar{c}_B}{2R_i} \Delta s_i \quad (70)$$

with Δs_i as the length of the panel. With Equations (57) and (68) the total strength of a streamwise strip of the jet sheet writes

$$\Gamma_{Js} = \int_{te}^{\infty} \gamma_J ds = \frac{1}{2} u_{\infty} C_J \bar{c}_B (\delta + \alpha - \alpha_{i\infty}) \quad (71)$$

where $\alpha_{i\infty}$ is the change of the free stream flow angle due to the induced velocities by the trailing vortices far behind the wing. These angles are determined in the computer program through iterative changes of the jet sheet position.

4.2.6 Vortex Representation of Wing and Free Vortices

The distribution of discrete vortices on wing panels, jet panels, and free tip and wake vortices is schematically shown in Figure 23. The wing panels are positioned on a surface that extends from the wing leading edge to its trailing edge, following the cambered centerline of the Jetwing NACA 23015 airfoil. Each panel hosts a "horseshoe" vortex, with its front part across the panel quarter chordline, and its tails along the sides of the panel. The tails of each horseshoe vortex follow the edges of other panels backward to the trailing edge of the wing and from there either the free wake vortices or the contour of the jet sheet.

According to Helmholtz's second vortex theorem a vortex line in a potential flow field has to either be closed in itself, or end only at the boundary of the flow field [17]. Therefore, it is assumed that the vorticity of the bound wing vortices is not discontinued at the trailing edge, as some authors do, but carried away with the jet sheet to infinity.

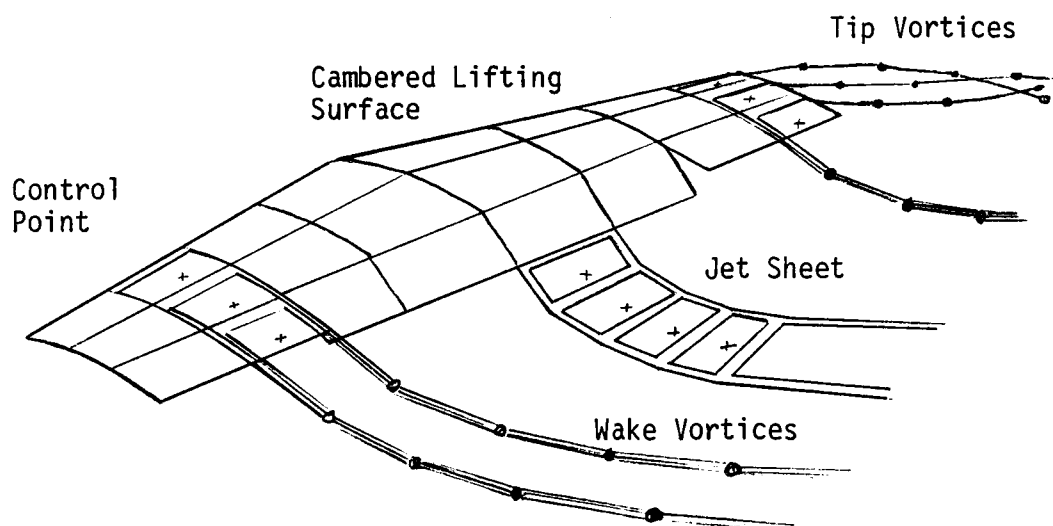


Figure 23: Schematic Illustration of the Singularity Distribution for Modelling of the Wing and Various Flow Components

At the wing tips the outside tail of each horseshoe vortex becomes instantly a free tip vortex. All free vortices are constructed of straight line segments which are interconnected at nodal points. Starting at the wing tip or trailing edge, the vortex is lined up with the direction of the airflow consecutively at each nodal point. At some chosen distance behind the wing, the segmentation of the free vortices and the jet sheet is discontinued and replaced by semi-infinite vortex lines, that follow the free stream direction.

The rationale behind the line-up of free vortices with streamlines is the similarity of the potential flow model with the real flow field. In the real viscous fluid all vorticity is generated at the wing surface and carried away downstream along the wing wake and tip vortices and is finally dissipated far downstream. The vorticity is therefore confined to the boundary layer on the wing surface and the wing wake and tip vortices. Also, it can be shown that a vortex filament which is skewed relative to curved streamlines in a potential flow field causes a pressure differential across those streamlines [17]. Therefore, vortex elements that are not aligned with the streamlines should be confined to the wing surface and jet sheet, which both cause pressure differences across the surface.

The center of the three-quarter chord line of each panel marks the control point. At this point the induced velocity by the bound and free vortices has to be such that the resulting normal velocity on the panel vanishes. This condition is equivalent to the requirement that the flow has to be parallel to a solid surface and is referred to as the "kinematic boundary condition."

The determination of the strength of each bound vortex is outlined as follows:

At first, the required induced normal velocity w_{nr} has to be determined for each wing panel control point. Due to the angle of attack of the free stream velocity and the slope of each panel, caused by the camber of the airfoil centerline, there will be a normal component w_n on each panel. In addition, the jet vortices will induce a normal velocity w_{nj} on each wing panel. The kinematic boundary condition requires

$$w_{nr} = -(w_n + w_{nj}) \quad (72)$$

The induced normal velocity, Δw_{ni} , at panel i due to a horseshoe vortex Γ_{wj} at wing panel j can be written

$$\Delta w_{ni} = a_{ij} \Gamma_{wj} \quad (73)$$

where a_{ij} is the influence coefficient for the two panels. It can be shown that

$$a_{ij} = a_{ji} \quad (74)$$

which means that a vortex at panel i causes the same effect at panel j , as a vortex of same strength at panel j would cause at panel i .

The influence coefficient a_{ij} is equivalent to that normal velocity Δw_{ni} at panel i which is induced by a vortex Γ_j of strength "1" at panel j . Therefore, the influence coefficients can be determined by using the Biot-Savart Law (38):

$$a_{ij} = \sum_k \frac{1}{4\pi P_i T_k} \left(\frac{Q_k T_k}{P_i Q_k} + \frac{S_k T_k}{P_i S_k} \right) \quad (75)$$

Here P_i indicates the control point at panel i ; Q , S , and T are points according to Figure 20, page 42, and the dummy variable k indicates the summation over all the segments of the horseshoe vortex "1" at panel j , including all the tail parts like tip or wake vortices and the semi-infinite ends.

If the number of wing panels is n , then the n^2 influence coefficients a_{ij} form an $n \times n$ matrix $[A]$. The n required normal velocities w_{nr} and the n unknown vortex strengths Γ_w can be written as column matrices $\{w_{nr}\}$ and $\{\Gamma_w\}$, respectively. This permits to write Equation (73) for all n wing panels in matrix form:

$$[A]\{\Gamma_w\} = \{w_{nr}\} \quad (76)$$

Expression (76) represents a system of n linear equations for the n unknown vortex strengths Γ_w which are solved by a Gauss-Seidel Iteration subroutine [29].

4.2.7 Shape of Jet Sheet

At this point, a restriction is imposed on the model by assuming that no transverse transport of momentum takes place within the jet sheet, so that the momentum flux is constant along each jet vortex line. Laser velocimeter measurements on the Jetwing research aircraft [8] have shown that the spanwise variation of C_j is not very large, justifying the above simplification for this case (Figure 24).

Another simplifying assumption is that the nonviscous jet sheet extends from the wing trailing edge to infinity without breaking or rolling up. Such "far field" considerations go beyond the scope of this dissertation and require more analytical effort.

This dissertation focuses on "near field" effects in relative proximity to the lifting wing. Therefore the assumption above appears to be a reasonable simplification. The shape of the jet sheet is described by the following exponential function:

$$z(x) = z_{te} + a_0(1 - e^{b_0(x-x_{te})}) + c_0(x - x_{te}) \quad (77)$$

with the coordinates as shown in Figure 25, and a_0 , b_0 , c_0 as constants. The slope of the jet sheet is expressed by

$$\frac{dz}{dx}(x) = -a_0 b_0 e^{b_0(x-x_{te})} + c_0 \quad (78)$$

With the boundary conditions

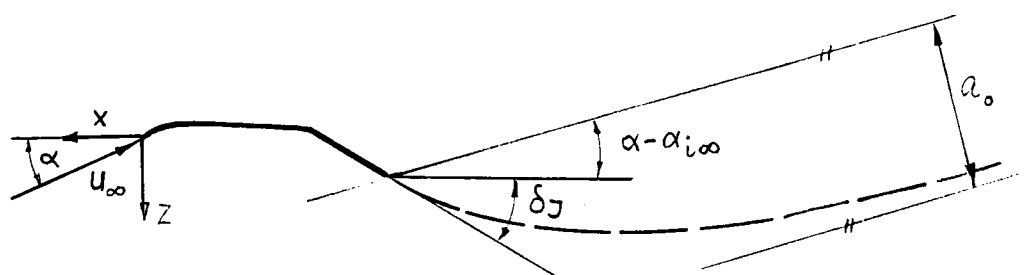
$$\left. \frac{dz}{dx} \right|_{x \rightarrow -\infty} = \tan(\alpha - \alpha_{i\infty}) \quad (79)$$

and

$$\left. \frac{dz}{dx} \right|_{x=x_{te}} = -\tan \delta_J \quad (80)$$

the constants become

$$c_0 = \tan(\alpha - \alpha_{i\infty}) \quad (81)$$



a)



b)

Figure 25: Parameters for Jet Sheet Description and Change of Jet Sheet Position and Panel Velocities after Iterations

and

$$b_0 = \frac{c_0 + \tan \delta_J}{a_0} \quad (82)$$

The values for α , δ_J , x_{te} , and z_{te} are fixed input parameters for each run of the program and the constants a_0 and $\alpha_{i\infty}$ can be arbitrarily chosen, at first. The program then varies the values of a_0 and $\alpha_{i\infty}$ during the iteration of the jet sheet position, until a shape is found, that makes the jet sheet streamlined, i.e., the normal velocities at the control points of the jet panels vanish, or become less than a small tolerance (Figure 25b). To make the process convergent, the variation of a_0 and $\alpha_{i\infty}$ follows a Newton-Raphson iteration scheme [29], with quotients of finite differences replacing derivatives, as follows:

$$a_0^i = a_0^{i-1} - \frac{\Delta a_0}{\Delta \bar{w}_{n1}} \bar{w}_{n1}^{i-1} \quad (83)$$

and

$$\alpha_{i\infty}^i = \alpha_{i\infty}^{i-1} - \frac{\Delta \alpha_{i\infty}}{\Delta \bar{w}_{n2}} \bar{w}_{n2}^{i-1} \quad (84)$$

where the superscript i denotes the iteration level, and

$$\Delta a_0 = a_0^{i-1} - a_0^{i-2} \quad (85)$$

and

$$\Delta \alpha_{i\infty} = \alpha_{i\infty}^{i-1} - \alpha_{i\infty}^{i-2} \quad (86)$$

The term $\bar{w}_{n1}$ denotes the average normal velocity of some jet panels close to the wing trailing edge, while $\bar{w}_{n2}$ is the average normal

velocity of some jet panels further downstream. The finite differences of these average velocities are then

$$\Delta \bar{w}_{n1} = \bar{w}_{n1}^{i-1} - \bar{w}_{n1}^{i-2} \quad (87)$$

and

$$\Delta \bar{w}_{n2} = \bar{w}_{n2}^{i-1} - \bar{w}_{n2}^{i-2} \quad (88)$$

4.2.8 Lift Distribution

The spanwise distribution of circulation can be obtained by a chordwise summation of the wing panel vortices at each strip according to Equation (76) plus, if applicable, the addition of the total vortex strength of a jet strip according to Equation (71):

$$\Gamma_j = \sum_i \Gamma_{wij} + \Gamma_{JS_j} \quad (89)$$

where the indices i and j mark the chordwise and spanwise station, respectively. The total circulation becomes

$$\Gamma = \frac{1}{b} \sum_j \Gamma_j \Delta y_j \quad (90)$$

with the wing span b and the width of a strip Δy_j .

The total lift L relates to Γ according to the celebrated Kutta-Joukowski Law:

$$L = \rho_\infty u_\infty b \Gamma \quad (91)$$

with ρ_∞ and u_∞ as free stream density and velocity values. From this follows the overall lift coefficient:

$$C_L = \frac{L}{qS} \quad (92)$$

with the dynamic pressure

$$q = \frac{\rho_\infty}{2} u_\infty^2 \quad (93)$$

The spanwise lift distribution is marked by the local lift coefficient at each strip:

$$C_{Lj} = \frac{2\Gamma_j}{u_\infty \bar{c}_j} \quad (94)$$

with $\bar{c}_j$ as the wing chord at station j .

The induced drag can be determined as follows. Local induced drag coefficients for each strip write

$$C_{DIj} = C_{Lj} \alpha_{Ij} \quad (95)$$

Here the α_{Ij} are downwash angles at the wing quarter chord, induced by only the trailing parts of all elementary horseshoe vortices [25]. From this, the overall induced drag coefficient writes

$$C_{DI} = \frac{1}{S} \sum_j C_{DIj} \Delta S_j \quad (96)$$

where

$$\Delta S_j = \bar{c}_j \Delta y_j \quad (97)$$

The induced drag factor k results as the ratio of the actual induced drag obtained in Equation (96) over the theoretical minimum for induced drag for the given wing geometry:

$$k = \frac{C_L^2 / (\pi A + 2C_J)}{C_{DI}} \quad (98)$$

with the aspect ratio

$$A = \frac{b^2}{S} . \quad (99)$$

4.2.9 Computational Process

The program runs through the following steps:

1. Start up by reading data defining the wing and panel geometry. Also, angle of attack α , free stream velocity u_∞ , air density ρ_∞ , flap deflection δ_F , the blowing coefficients C_J and C_{j_s} , and initial values for the jet sheet coefficients a_0 and α_{i_∞} , are input parameters.
2. Horseshoe vortices and control points are positioned on all wing panels and the shapes of wake and tip vortices are defined. As a good first approximation according to [30], these free vortices are straight lines, extending from their origin at the wing tip or trailing edge to infinity with a slope

$$\frac{dz}{dx} = \tan \frac{\alpha}{2} . \quad (100)$$

3. A jet sheet shape is computed, using Equation (77). Jet panels are defined, jet vortices positioned, and their strength calculated according to Equation (70).
4. Knowing the shape of bound vortices, free vortices and jet sheet, the influence coefficients α_{ij} can be determined with Equation (75).

5. The normal velocities on each wing panel are calculated.
While the free stream components w_n follow directly from geometrical considerations, the jet induced normal velocities w_{nJ} are determined as a summation of all Δw_{nJ} 's found with the Biot-Savart Law (38).
6. With the required induced normal velocities w_{nr} from Equation (72) and the influence coefficients α_{ij} known, the wing panel vortex strengths can be determined according to Equation (76) and a Gauss-Seidel Iteration technique.
7. The lift coefficient C_L is determined according to Equations (89) through (92).
8. For the origin of each free vortex and subsequently each nodal point, the resulting velocity from the free stream and all vortex induced velocities is computed, and each vortex filament is aligned with that velocity's direction.
9. Steps four through eight are repeated until the variation of the lift coefficients is less than a small tolerance ΔC_L .
This process is the "inner iteration", defining the shape of the free vortices. The process shows rapid convergence; typically after four iterations the C_L variation is less than 0.003.
10. The normal velocities on all jet panels are computed, considering the free stream contributions and the summation of induced velocities from all bound, free and jet panel vortices.
11. The shape of the jet sheet is modified by changing the values of $\alpha_{i\infty}$. Then steps three through ten are repeated, until the

average velocity $\bar{w}_{n2}$ is less than a given tolerance or an iteration count reaches a maximum. The program then switches to modify the constants a_0 in a similar fashion. This alternating variation of jet sheet parameters continues until all jet panel normal velocities are less than a small tolerance or a maximum iteration count is reached. This "outer iteration" converges slowly; normally more than thirty iterations are required to reduce the jet panel normal velocities to a small percentage (3%) of the free stream velocity. Here the proper choice of the initial values for a_0 and $\alpha_{i\infty}$ can be very helpful in saving computer time.

12. The program reads as input data the total number and the Cartesian coordinates of all points in the flow field, where resulting flow velocities and flow angles are desired. Those are computed subsequently by multiple use of Equation (38).
13. As a final step the program prints the resulting geometry of the jet sheet and free vortices, circulation and lift distributions, and the flow angles and velocities for the chosen points in the flow field. Optionally, it generates a data file containing this information for a plotting program. This permits to present the results in graphical form.

Examples of resulting outputs are shown on the next pages. More results are presented in Section 6.

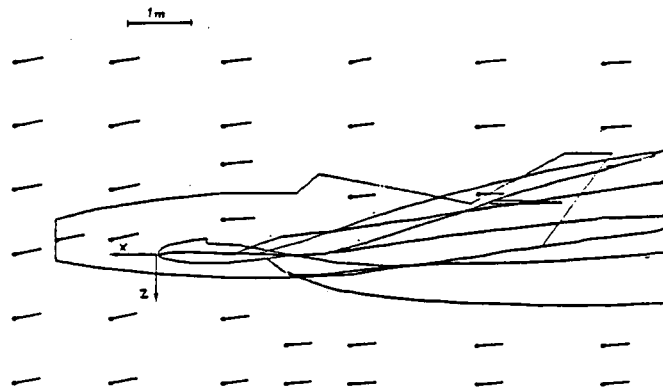
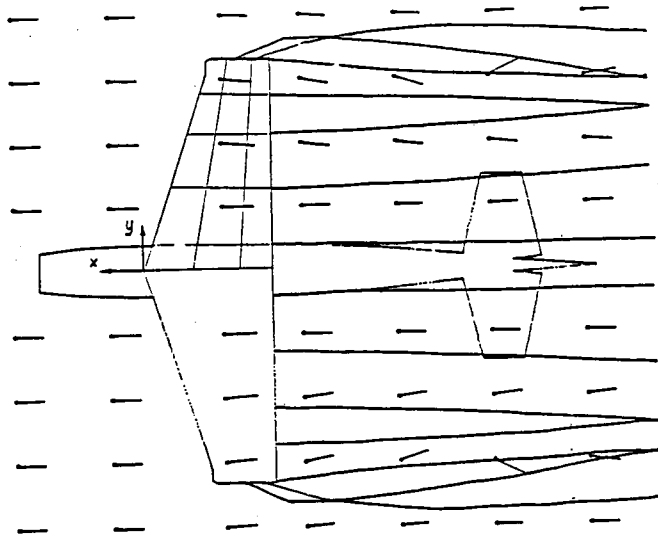
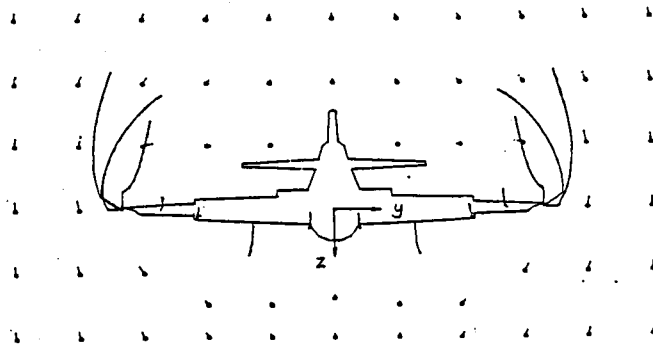
a) Plane $y=0$ b) Plane $z=-0.7$ c) Plane $x=-5.0$

Figure 26: Plotting Sample of Jetwing Flow Field, Jet Off

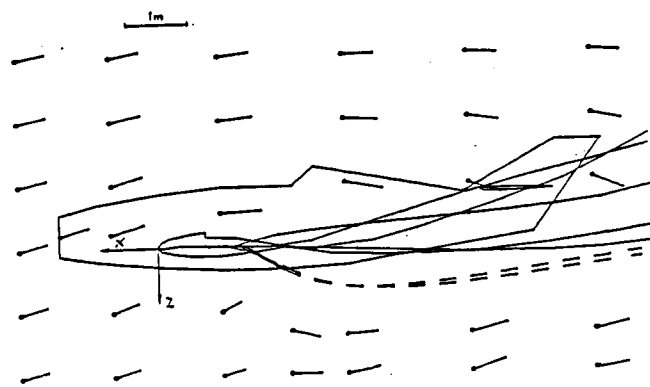
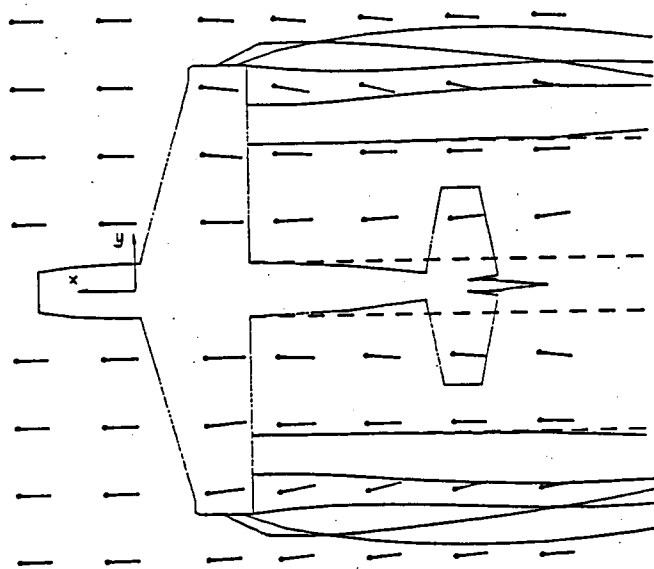
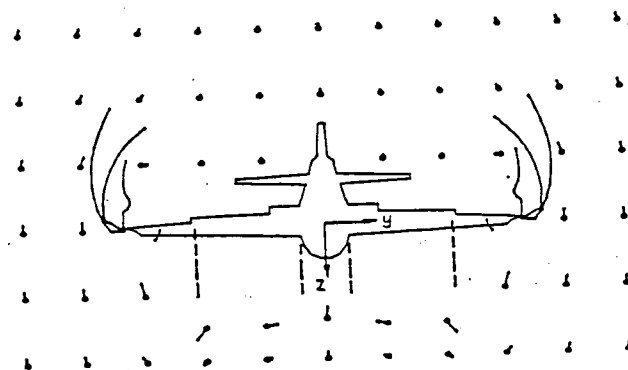
a) Plane $y=0$ b) Plane $z=-0.7$ c) Plane $x=-5.0$

Figure 27: Plotting Sample of Jetwing Flow Field, Jet On

5. DETERMINATION OF LONGITUDINAL STABILITY PARAMETERS

5.1 General Derivations

As outlined in Section 2.1, longitudinal stability and control theory has to consider pitching moments about the center of gravity of the aircraft in flight. With pitch-up moments defined positive (Figure 28), moments become stabilizing, if their derivatives with respect to angle of attack are negative.

A classic approach [31] in stability analysis has been the determination of a neutral point, defined as that center of gravity position at which the derivative

$$\frac{dC_m}{dC_L} = 0 \quad (101)$$

A C.G. position ahead of the neutral point makes this derivative negative, resulting in positive longitudinal static stability. Conversely, a C.G. position behind the neutral point makes this derivative positive, indicating longitudinal static instability.

Etkins [32] points out, however, that this derivative is a trim slope criterion and does not lead to clear-cut conclusions about stability with respect to angle of attack disturbances at constant speed.

In general, C_m and C_L are functions of angle of attack α , control surface deflections δ_e , Mach number M , Reynolds number Re , thrust coefficient C_T , and dynamic pressure q . Neglecting Reynolds number effects, which are usually very small, and assuming fixed controls, this writes

$$C_L = f(\alpha, M, C_T, q) \quad (102)$$

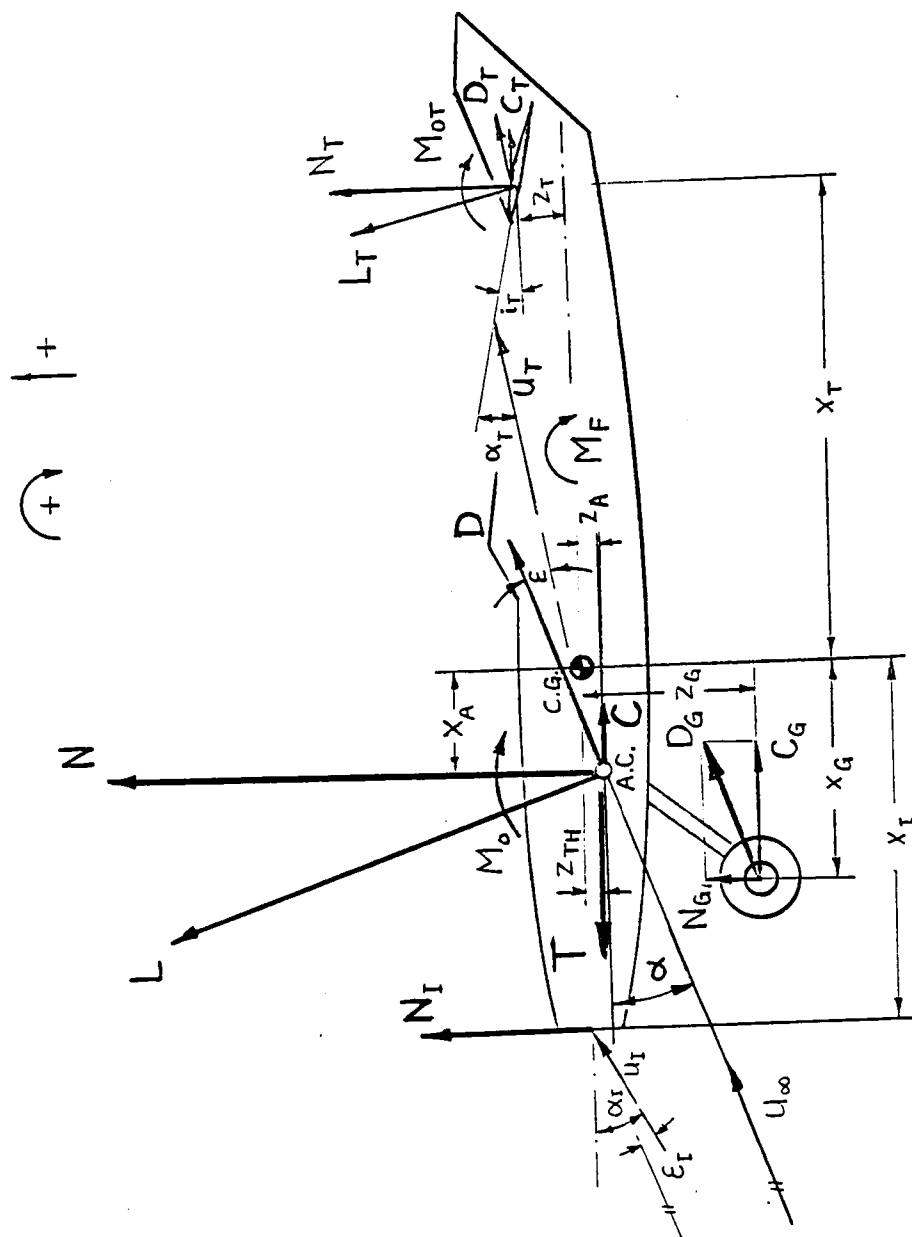


Figure 28: Major Forces and Moments for Jetwing Longitudinal Stability Considerations

$$C_m = g(\alpha, M, C_T, q) \quad (103)$$

The derivative dC_m/dC_L does not exist, unless M , C_T , and q are functions of C_L . When that is the case, then

$$\frac{dC_m}{dC_L} = \frac{\partial C_m}{\partial \alpha} \frac{\partial \alpha}{\partial C_L} + \frac{\partial C_m}{\partial M} \frac{\partial M}{\partial C_L} + \frac{\partial C_m}{\partial C_T} \frac{\partial C_T}{\partial C_L} + \frac{\partial C_m}{\partial q} \frac{\partial q}{\partial C_L} \quad (104)$$

This equation has meaning only when a specific kind of flight is prescribed, e.g. horizontal unaccelerated flight, or rectilinear climbing flight at full throttle. With this kind of condition imposed, M , C_T , and q are definite functions of C_L , and the derivative dC_m/dC_L exists.

It can be shown that it relates to the trim curves of the airplane (32). A plot of the elevator angle to trim versus air speed will have a zero slope when dC_m/dC_L is zero, and a negative slope when the C.G. lies aft of the neutral point so defined. This reversal of slopes indicates a tendency toward instability with respect to speed but can be misleading as to pitch-stability.

As stated in Section 2.1, in flight testing the lift coefficient C_L is much easier to determine than the angle of attack and therefore a preferred parameter, but extreme care has to be exercised, when the derivative $\partial C_m/\partial \alpha$ is substituted by dC_m/dC_L . Only if all but the first term on the right hand side of Equation (104) become zero, then the derivative dC_m/dC_L becomes a true stability parameter, equivalent to $\partial C_m/\partial \alpha$. This is the case, e.g., for an ideally rigid aircraft in gliding flight at low Mach numbers. Neglecting compressibility,

$$\frac{\partial C_m}{\partial M} = 0 \quad (105)$$

and aeroelastic effects,

$$\frac{\partial C_m}{\partial q} = 0 \quad (106)$$

Equation (104) becomes for the Jetwing:

$$\frac{dC_m}{dC_L} = \frac{\partial C_m}{\partial \alpha} \frac{\partial \alpha}{\partial C_L} + \frac{\partial C_m}{\partial C_J} \frac{\partial C_J}{\partial C_L} . \quad (107)$$

The derivative $\partial C_m / \partial C_J$ indicates trim moment changes due to throttle adjustments or speed changes, e.g. in unaccelerated level flight. It can be further expanded to

$$\frac{\partial C_m}{\partial C_J} = \frac{\partial C_{M_0}}{\partial C_J} + \frac{\partial C_m}{\partial \epsilon} \frac{\partial \epsilon}{C_J} \quad (108)$$

In this equation, the first term on the right side represents pitching moment changes due to the change of blowing over the wing itself, while the second term indicates pitching moment changes due to downwash angle changes in the flow field, caused by the change of the blowing coefficient. These terms have significance for control considerations and if "stick fixed" and "stick free" neutral points are determined in unaccelerated level flight at various airspeeds. For the investigation of the pitch-stability of the aircraft the term $\partial C_m / \partial \alpha$ has to be investigated. It was shown in [33] that this is a valid measure for stability with respect to angle of attack disturbances at constant speed.

A further analysis requires a detailed study of all airplane parts which contribute significant pitching moments. The basic derivations on the next pages follow Perkins and Hage [31], but differentiate with

respect to angle of attack α , rather than the lift coefficient C_L . In addition, some Jetwing specifics are incorporated.

The lift and drag forces are by definition always perpendicular and parallel to the wind, respectively. It is therefore inconvenient to use these forces to obtain moments, for their arms to the center of gravity vary with angle of attack. For this reason, all forces are resolved into normal and chordwise forces whose axes remain fixed with the airplane and whose arms are therefore constant. Resolving the tail force along the same axes as shown in Figure 28, the following summation of moments about the airplane's center of gravity is obtained:

$$M_{CG} = N x_A - C z_A + M_o + N_I x_I + T z_{TH} + N_G x_G - C_G z_G + M_F \\ - N_T x_T + C_T z_T + M_{oT} \quad (109)$$

Normalizing all terms by $qS\bar{c}$ and differentiating with respect to alpha yields

$$\frac{\partial C_{m_{CG}}}{\partial \alpha} = \frac{\partial C_N}{\partial \alpha} \frac{x_A}{\bar{c}} - \frac{\partial C_C}{\partial \alpha} \frac{z_A}{\bar{c}} + \frac{\partial C_{m_o}}{\partial \alpha} + \frac{\partial C_{NI}}{\partial \alpha_I} \frac{\partial \alpha_I}{\partial \alpha} \frac{x_I}{\bar{c}} \\ + \frac{\partial C_J}{\partial \alpha} \frac{z_{TH}}{\bar{c}} + \frac{\partial C_{NG}}{\partial \alpha} \frac{x_G}{\bar{c}} - \frac{\partial C_{CG}}{\partial \alpha} \frac{z_G}{\bar{c}} + \frac{\partial C_{m_F}}{\partial \alpha} \\ + \eta_T \frac{S_T}{S} \frac{\partial \alpha_T}{\partial \alpha} \left[- \frac{\partial C_{NT}}{\partial \alpha_T} \frac{x_T}{\bar{c}} + \frac{\partial C_{CT}}{\partial \alpha_T} \frac{z_T}{\bar{c}} + \frac{\partial C_{m_{oT}}}{\partial \alpha_T} \right] \quad (110)$$

where $\eta_T = \rho_T/\rho_\infty$ is the tail efficiency, accounting for a reduction of dynamic pressure due to interaction with parts of the wing wake and

fuselage boundary layer. The contributions of the components are discussed briefly in the following.

5.1.1 Wing

The coefficients for the normal and tangential force (Figure 28) write

$$C_N = C_L \cos \alpha + C_D \sin \alpha \quad (111)$$

and

$$C_C = C_D \cos \alpha - C_L \sin \alpha \quad (112)$$

The derivatives with respect to α yield

$$\frac{\partial C_N}{\partial \alpha} = \frac{\partial C_L}{\partial \alpha} \cos \alpha - C_L \sin \alpha + \frac{\partial C_D}{\partial \alpha} \sin \alpha + C_D \cos \alpha \quad (113)$$

and

$$\frac{\partial C_C}{\partial \alpha} = \frac{\partial C_D}{\partial \alpha} \cos \alpha - C_D \sin \alpha + \frac{\partial C_L}{\partial \alpha} \sin \alpha + C_L \cos \alpha \quad (114)$$

Reasoning that α is normally a small angle and the drag coefficient C_D is a small number with respect to unity, the above equations can be approximated by

$$\frac{\partial C_N}{\partial \alpha} = \frac{\partial C_L}{\partial \alpha} \quad (115)$$

and

$$\frac{\partial C_C}{\partial \alpha} = \frac{\partial C_D}{\partial \alpha} - C_L = \frac{\partial C_D}{\partial C_L} \frac{\partial C_L}{\partial \alpha} - C_L \quad (116)$$

For constant speed, only the lift induced drag component changes significantly with angle of attack. Maskell and Spence [34] give an expression for the induced drag of a jet wing:

$$C_{Di} = \frac{kC_L^2}{\pi A + 2C_j} \quad (117)$$

where the factor $k \geq 1$ accounts for deviations from an elliptical lift distribution. Differentiating Equation (117) with respect to C_L yields

$$\frac{\partial C_{Di}}{\partial C_L} = \frac{2kC_L}{\pi A + 2C_j} \quad (118)$$

Now Equation (116) writes

$$\frac{\partial C_C}{\partial \alpha} = \left[\frac{2kC_L}{\pi A + 2C_j} - \alpha_a \right] \frac{\partial C_L}{\partial \alpha} \quad (119)$$

where α_a is the aerodynamic angle of attack in radians, differing from the geometric angle of attack α_g by the angle of attack for zero lift α_o (Figure 29):

$$\alpha_a = \alpha_g + \alpha_o \quad (120)$$

The derivative $\partial C_{m0}/\partial \alpha$ equals zero, by definition of the aerodynamic center of the wing. Substituting this and Equations (115) and (119), the wing term in Equation (110) becomes

$$\left. \frac{\partial C_{mCG}}{\partial \alpha} \right|_w = \frac{\partial C_L}{\partial \alpha} \frac{x_A}{\bar{c}} + \left[\frac{-2kC_L}{\pi A + 2C_j} + \alpha_a \right] \frac{\partial C_L}{\partial \alpha} \frac{z_A}{\bar{c}} \quad (121)$$

This wing stability equation contains a lift term (first) and a drag term (second). Contrary to conventional aircraft, the Jetwing lift curve slope $\partial C_L/\partial \alpha$ is a function of the blowing coefficient C_j , but

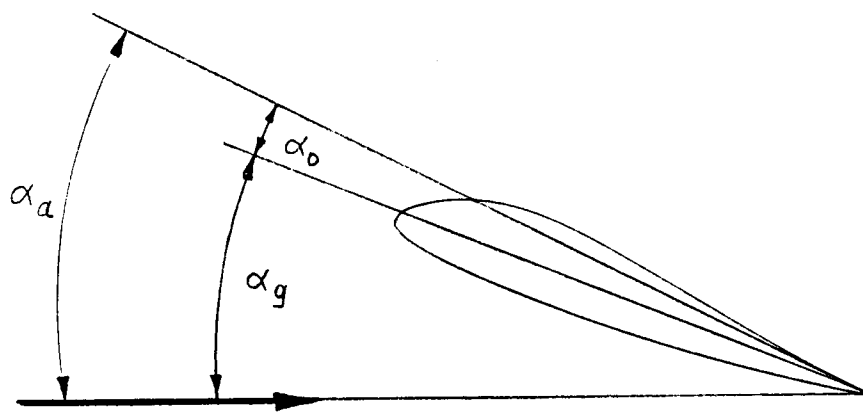


Figure 29: Relationship between Geometric, Aerodynamic, and Zero Lift Angles of Attack

similar to conventional aircraft, it is always positive. Obviously, for positive x_A (C.G. behind the wing A.C., Figure 28, page 70), the first term is destabilizing, while for negative x_A (C.G. ahead of A.C.) this wing term gives a stabilizing contribution. Entering some typical values for the bracket term yields

$$\left[\frac{2kC_L}{\pi A + 2C_J} - \alpha_a \right] \equiv \left[\frac{-2.0 \times 1.4 \times 1.16}{3.14 \times 4.48 + 2 \times .37} + 18.5 \times \frac{\pi}{180} \right] \quad (122)$$

$$= [-.22 + .32] > 0$$

If the C.G. is above the wing aerodynamic center, as is the case for the Jetwing, z_A will be positive according to Figure 28 and this term will be destabilizing. For

$$\alpha_a < \left| \frac{2kC_L}{A + 2C_J} \right| \quad (123)$$

the drag term would become negative and therefore give a stabilizing contribution to the aircraft in flight. This, however, is purely theoretical, for Jetwing wind tunnel data show that inequality (123) is never true.

The bracket term in Equation (121) is negligible for small values of C_L , C_J , and α_a , but becomes significant at higher values of these parameters. Comparing, for example, Jetwing level flight conditions at airspeeds of 200 knots and 64 knots, yields values of .003 and .124 for the bracket term, respectively. This means that for an equal vertical and horizontal displacement of the aircraft center of gravity from the wing aerodynamic center ($x_A = z_A$), at high C_L values the drag term

contributes 12% to the destabilizing effects of the wing. For the Jetwing, $z_A > x_A$, which amplifies this fact further.

The lift curve slope $\partial C_L / \partial \alpha$ varies with changing blowing coefficient C_J , e.g., $\partial C_L / \partial \alpha = 3.58$ for $C_J = 0.0$, and $\partial C_L / \partial \alpha = 6.74$ for $C_J = 2.52$. This amount of blowing almost doubles the destabilizing wing contribution in Equation (121). In summary, both the lift and the drag of the main wing are destabilizing for the Jetwing, due to the aircraft's center of gravity location behind and above the wing aerodynamic center. While the drag term may be neglected at low C_L values, it grows to approximately 15% of the lift term at high C_L values. The Upper Surface Blowing changes the lift curve slope of the wing significantly and therefore amplifies the destabilizing effect of the wing.

5.1.2 Inlet

The normal force at the inlet duct entrance results from the turning of the incoming mass flow (Figure 30) and can be determined from momentum considerations:

$$N_I = \dot{m}_I u_I \sin \alpha_I \quad (124)$$

Engine mass flow $\dot{m}_I$ and inlet velocity u_I can be determined from engine performance data, while the angle between the duct axis and the local flow at the duct entrance for the Jetwing writes

$$\alpha_I = \alpha + \varepsilon_I \quad (125)$$

where ε_I denotes the upwash angle induced by the lifting wing. Differentiating with respect to α yields

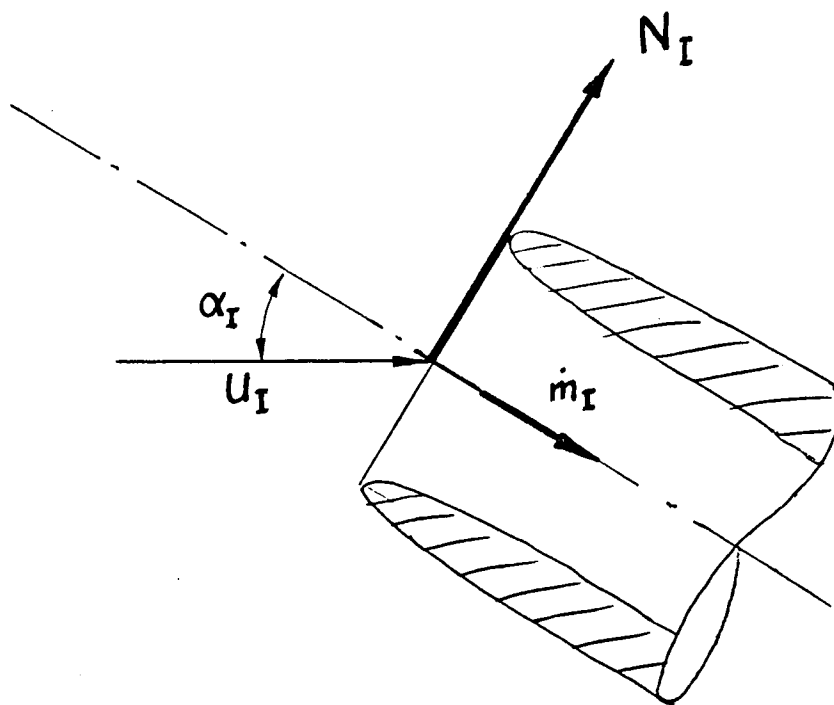


Figure 30: Normal Force at the Inlet

$$\frac{\partial \alpha_I}{\partial \alpha} = 1 + \frac{\partial \epsilon_I}{\partial \alpha} \quad (126)$$

The inlet term in the stability Equation (110) becomes

$$\left. \frac{\partial C_{m_{CG}}}{\partial \alpha} \right|_I = \frac{\partial C_{NI}}{\partial \alpha_I} \frac{\partial \alpha_I}{\partial \alpha} \frac{x_I}{\bar{c}} = \frac{\dot{m}_I u_I}{q S \bar{c}} \cos \alpha_I \left(1 + \frac{\partial \epsilon_I}{\partial \alpha} \right) \frac{x_I}{\bar{c}} > 0 \quad (127)$$

This proves the inlet normal force to be destabilizing.

5.1.3 Thrust

Some authors [31] use the derivative dC_{TH}/dC_L together with the assumption of unaccelerated flight in their analysis of the thrust contribution to static longitudinal stability. With the arm z_{TH} (Figure 28, page 69), the pitching moment coefficient due to thrust becomes

$$C_m = \frac{T z_{TH}}{q S \bar{c}} \quad (128)$$

Considering unaccelerated level flight, the dynamic pressure is a function of the lift coefficient:

$$q = \frac{\rho}{2} v^2 = \frac{W}{S} \frac{1}{C_L} \quad (129)$$

With this, Equation (128) becomes

$$C_{m_{TH}} = \frac{T}{W} \frac{z_{TH}}{\bar{c}} C_L \quad (130)$$

Assuming that the net-thrust for a jet aircraft is constant with air-speed, the following expression is derived:

$$\left. \frac{dC_m}{dC_L} \right|_{TH} = \frac{T}{W} \frac{z_{TH}}{\bar{c}} \quad (131)$$

From this the authors proceed to conclude, that

"jet propelled airplanes with thrust lines below the C.G. (z_{TH} positive) will be destabilized by the thrust contribution, whereas thrust lines above the C.G. will be stabilizing."
[31, p. 244]

This, however, is an erroneous conclusion with respect to pitch-stability. While Equation (131) makes a statement about trim moment alterations due to changes in airspeed, it does not describe the airplane's stability behavior, as was shown at the beginning of this section. The valid thrust term in the stability equation (110) is $\partial C_J / \partial \alpha$. This raises the question, whether the thrust changes with angle of attack. There are two areas where the angle of attack enters the thrust analysis.

First, the engine inlet and diffuser may show less efficiency at high angles of attack which are associated with large turning angles of the incoming air mass flow. A reduction in diffuser efficiency would in turn reduce the engine thrust [39]. Assuming the position of the resulting thrust vector as depicted in Figure 28, page 70, the reduction of thrust with increasing angle of attack has a stabilizing influence on the aircraft.

Second, the ambient static pressure near the nozzle exit on the upper wing surface changes with angle of attack. This has a direct impact on the jet velocity, particularly since the Jetwing nozzles operate under subsonic, i.e. "unchoked" conditions. The suction on the upper surface of airfoils increases with higher angles of attack; for the Jetwing this translates to an increase of thrust. With the geometry

as shown in Figure 28, page 70 , this contributes a destabilizing component to the aircraft.

Assuming that the effects of inlet angle and nozzle pressure changes on the thrust are relatively small and cancel each other, the stability term for the thrust in Equation (110) writes

$$\frac{\partial C_J}{\partial \alpha} = 0 \quad (132)$$

for angle of attack changes at constant airspeed. The thrust gives therefore no direct contribution to the aircraft's longitudinal static stability.

Indirectly, however, the thrust destabilizes the aircraft in flight, since it increases the lift curve slope and therefore amplifies the wing contribution, which is destabilizing for the case of the Jetwing, as was shown in Section 5.1.1. Also, at high thrust settings, the destabilizing inlet normal force is larger due to the increased engine mass flow, as shown in Section 5.1.2.

5.1.4 Gear

From Figure 31 it is obvious that

$$\frac{\partial C_{NG}}{\partial \alpha} > 0 \quad (133)$$

and

$$\frac{\partial C_{CG}}{\partial \alpha} < 0 \quad (134)$$

Entering this in the stability equation (110), yields

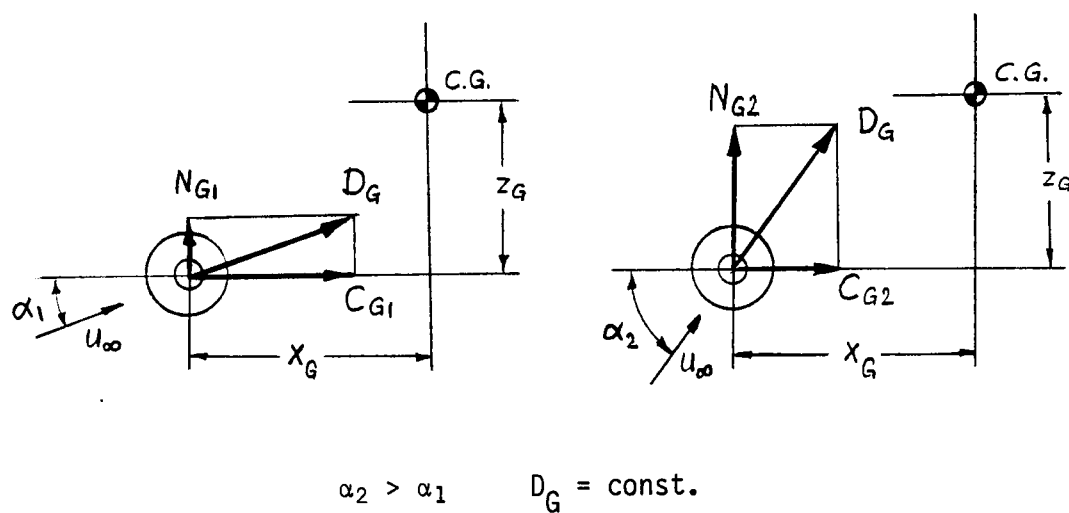


Figure 31: Gear Forces at Different Angles of Attack

$$\left. \frac{\partial C_{mCG}}{\partial \alpha} \right|_G = \frac{\partial C_{NG}}{\partial \alpha} \frac{x_G}{c} - \frac{\partial C_{CG}}{\partial \alpha} \frac{z_G}{c} > 0 \quad (135)$$

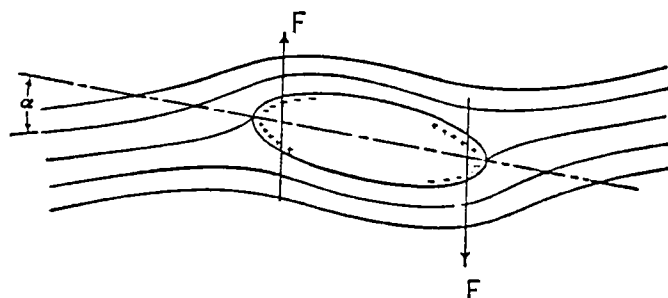
With Equation (133), all terms in Equation (135) are positive; therefore, the extended gear destabilizes the Jetwing aircraft in flight. It can be shown that it would be stabilizing if it were positioned behind the C.G. Since the drag of the gear is constant for a given air-speed, the stability contribution depends solely on the relative change of the drag components with angle of attack. For the small angle of attack changes considered in stability analysis, the aerodynamic contribution of the gear can be neglected.

Indirectly, the gear stabilizes the Jetwing, since the extension of the undercarriage moves the aircraft's C.G. closer to the wing A.C. Numerical estimates show that the C.G. is lowered by 1.3 cm, and moved forward by 1.0 cm. This reduces the wing drag term by 6.5% and the wing lift term by 6.0%.

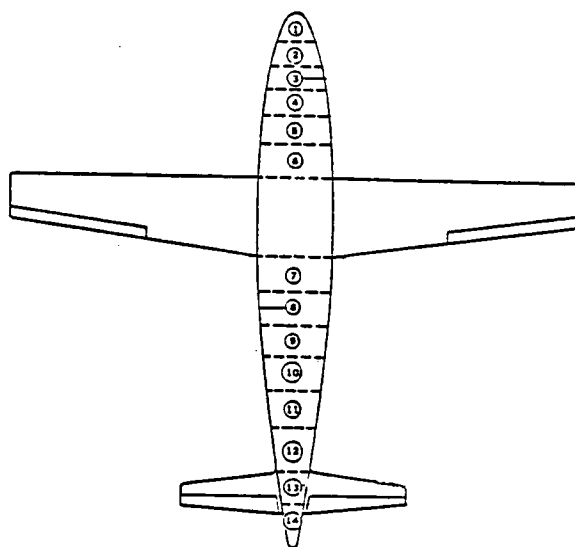
5.1.5 Fuselage

For longitudinal stability considerations, the frictional lift and drag force contributions of the fuselage are usually neglected [32]. A body of revolution with angle of attack, however, generates a significant pitching moment (Figure 32a), which increases with angle of attack and is therefore destabilizing.

The variation of the fuselage longitudinal pitching moment with angle of attack is greatly affected by the upwash in front of the wing and the downwash behind the wing. Munk linked the variation of fuselage pitching moment with angle of attack to the body volume and



a)



b)

Figure 32: Potential Flow About a Body of Revolution and a Typical Layout for Computing Fuselage Moments

dynamic pressure [35], and Multhopp developed a method to include the wing interference in Munk's formula [36]. The equation for the approximate pitching moment variation is then

$$\frac{\partial M_F}{\partial \alpha} = \frac{q}{k_F} \int_0^x w_F^2 \frac{\partial \beta}{\partial \alpha} dx \quad (136)$$

with

$$k_F = 1748 \frac{N}{m^2} \quad (137)$$

and w_F as the local width of the fuselage (Figure 32b). The local flow angle β is

$$\beta = \alpha - \epsilon \quad (138)$$

with ϵ as the lift induced downwash angle. Ahead of the wing the induced upwash adds to the angle of the free stream, making $\partial \beta / \partial \alpha$ greater than unity and having a destabilizing influence on the fuselage, while behind the wing the induced downwash subtracts from the free stream angle, making

$$\frac{\partial \beta}{\partial \alpha} = 1 - \frac{\partial \epsilon}{\partial \alpha} \quad (139)$$

and therefore reducing the unstable contribution of the fuselage behind the wing.

5.1.6 Horizontal Tail

The last two terms in Equation (110) have been found to be negligible when compared to other terms. This leaves

$$\left. \frac{\partial C_{m_{CG}}}{\partial \alpha} \right|_T = - \frac{\partial C_{NT}}{\partial \alpha_T} \frac{\partial \alpha_T}{\partial \alpha} \frac{x_T}{\bar{c}} \frac{S_T}{S} \eta_T \quad (140)$$

Since from Figure 16, page 31,

$$\alpha_T = \alpha - \epsilon_T + i_T \quad (141)$$

the derivative $\partial \alpha_T / \partial \alpha$ can be written

$$\frac{\partial \alpha_T}{\partial \alpha} = 1 - \frac{\partial \epsilon_T}{\partial \alpha} \quad (142)$$

and Equation (140) becomes

$$\left. \frac{\partial C_{m_{CG}}}{\partial \alpha} \right|_T = - \frac{\partial C_{NT}}{\partial \alpha_T} \frac{x_T}{\bar{c}} \frac{S_T}{S} \eta_T \left(1 - \frac{\partial \epsilon_T}{\partial \alpha} \right) \quad (143)$$

This relationship marks the horizontal tail as having a stabilizing influence, as long as

$$\frac{\partial \epsilon_T}{\partial \alpha} < 1 \quad (144)$$

This parameter is therefore of prime importance in this investigation.

5.2 Determination of Derivatives from Experimental Data

5.2.1 Flight Test

Flight test data were collected during a previous research program [8], measuring control stick forces and position for stabilized level flight conditions (Figure 33). The "so found" stick-fixed and stick-free stabilities give valuable information about the handling qualities of the airplane.

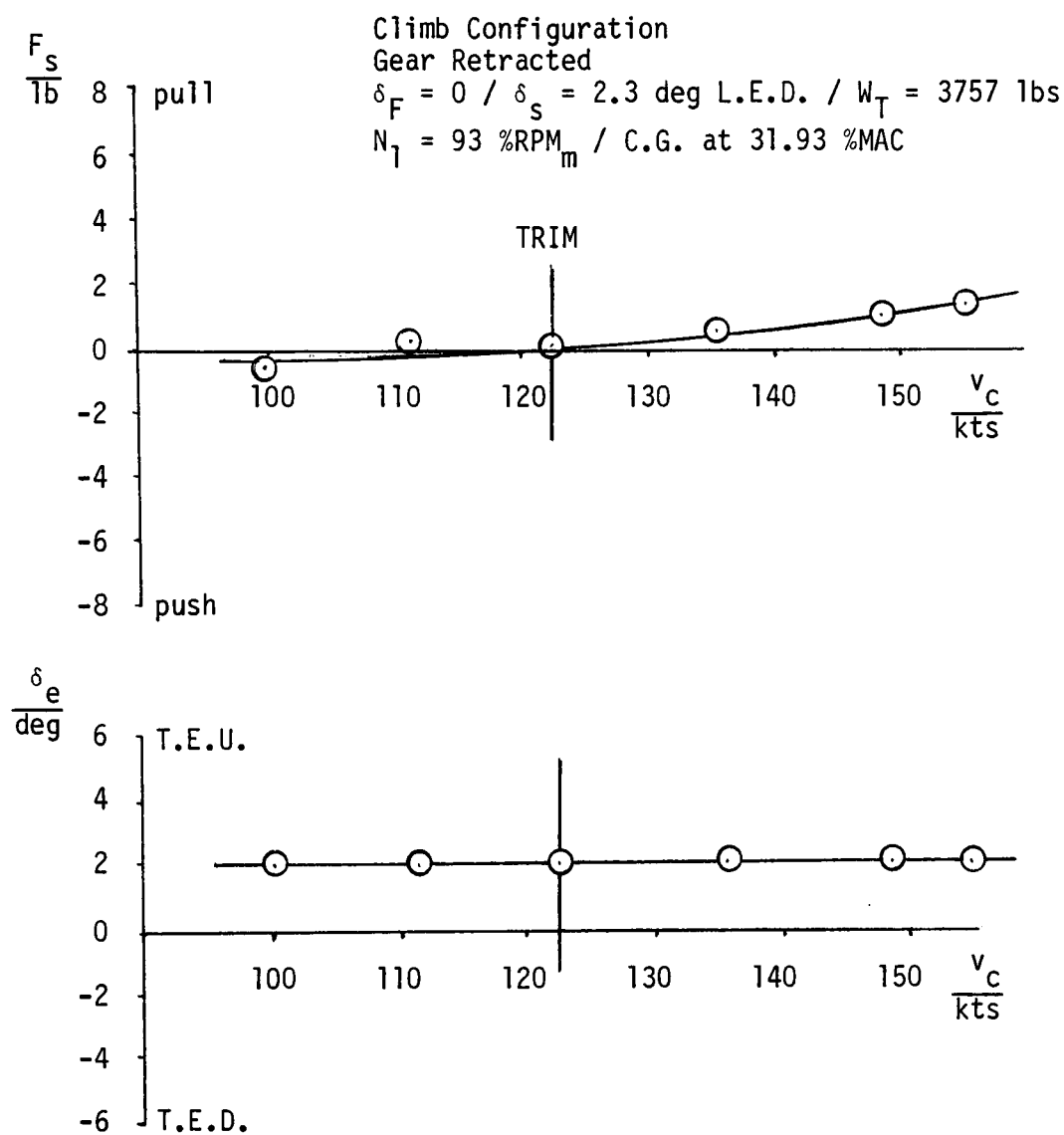


Figure 33: Example for Jetwing Elevator Positions and Stick Forces versus Airspeed

For the pilot, an airplane shows stable flight characteristics if it requires forward positioning and forward push force on the control stick in order to make it fly faster. Conversely, pull forces and rearward positioning are desirable characteristics for slower airspeeds.

For the Jetwing, the flight test data show unfavorable longitudinal stability characteristics. As the plots show, very small or zero changes in stick position and force are required for airspeed variations. For some configurations the stick forces are even reversed. This makes it difficult for the pilot to stabilize the airplane with respect to airspeed. He also reports poor pitch stability of the aircraft [8].

With regard to downwash changes at the tail, the flight test data can be analyzed further to give insight with respect to this parameter. Since stick forces are related to hinge moments at the elevator, the flow angles at the tail could be determined from the data. Including other flight test data, the following expressions can be derived:

$$\frac{\epsilon_2 - \epsilon_1}{\alpha_2 - \alpha_1} = \frac{\Delta \epsilon}{\Delta \alpha} \approx \frac{d\epsilon}{d\alpha} \quad (145)$$

Since the angles α_1 and α_2 were determined at different airspeeds, the derivative would have to be written

$$\frac{d\epsilon}{d\alpha} = \frac{\partial \epsilon}{\partial \alpha} + \frac{\partial \epsilon}{\partial C_L} \frac{\partial C_L}{\partial \alpha} + \frac{\partial \epsilon}{\partial C_J} \frac{\partial C_J}{\partial \alpha} + \dots \quad (146)$$

This term does not represent a true indication of angle of attack stability, as was shown before. Therefore, no more effort is made here to further examine these data.

5.2.2 Wind Tunnel

Another approach was taken by Moore [37] in examining the Jetwing wind tunnel data with respect to longitudinal stability parameters and horizontal tail sizing. Since in wind tunnel testing parameters like α , u_∞ , and C_j can be varied independently, the data offer the extraction of partial derivatives.

Moore's approach for finding $\partial\epsilon/\partial\alpha$ and the minimum required tail area for neutral longitudinal static stability is outlined in the following.

A simplified longitudinal stability equation [31] writes

$$\left. \frac{dC_m}{dC_L} \right|_{TOT} = \left. \frac{dC_m}{dC_L} \right|_{T0} - \frac{\partial C_{LT}/\partial\alpha_T}{\partial C_{LW}/\partial\alpha_W} V_T \eta_T \left(1 - \frac{\partial\epsilon}{\partial\alpha} \right) \quad (147)$$

where the indices TOT, T0, T, and W stand for total, tail off, tail, and wing, respectively. The horizontal tail volume is

$$V_T = \frac{S_T}{S} \frac{x_T}{\bar{c}} \quad (148)$$

and the lift curve slopes of wing and tail are

$$a_T = \frac{\partial C_{LT}}{\partial\alpha_T} \quad (149)$$

$$a_W = \frac{\partial C_{LW}}{\partial\alpha_W} \quad (150)$$

Solving for the downwash derivative, yields

$$\frac{\partial\epsilon}{\partial\alpha} = \frac{a_W}{a_T V_T \eta_T} \left[\left. \frac{dC_m}{dC_L} \right|_{TOT} - \left. \frac{dC_m}{dC_L} \right|_{T0} \right] + 1 \quad (151)$$

The terms in this equation can be determined from Jetwing geometry and wind tunnel data. Extractions from these are shown in Figures 34 and 35.

For comparison, empirical equations from [31] and [38] give

$$\frac{\partial \epsilon}{\partial \alpha} = \frac{114.6 a_w}{\pi A} \quad (152)$$

and

$$\frac{\partial \epsilon}{\partial \alpha} = 20 a_w \frac{\lambda^{0.3}}{A^{0.725}} \left(\frac{3 \bar{c}}{x_T} \right)^{0.25} \quad (153)$$

The minimum required horizontal tail area for neutral static longitudinal stability follows from Equation (147)

$$\left. \frac{dC_m}{dC_L} \right|_{TOT} = 0 = \left. \frac{dC_m}{dC_L} \right|_{T0} - \frac{a_T}{a_w} \frac{S_T}{S} \frac{x_T}{\bar{c}} \eta_T \left(1 - \frac{\partial \epsilon}{\partial \alpha} \right) \quad (154)$$

Solving for the tail area yields

$$S_{Tmin} = \left. \frac{dC_m}{dC_L} \right|_{T0} \frac{a_w}{a_T} \frac{\bar{c}}{x_T} \frac{S}{\eta_T} \frac{1}{(1 - \partial \epsilon / \partial \alpha)} \quad (155)$$

Results are shown in Section 6.

5.3 Determination of Derivatives from Theoretical Data

The vortex-lattice routine for jet-flapped wings, which was developed during this study, is particularly suited for the calculation of partial derivative stability parameters. By slightly varying one input parameter while keeping the others constant, the program output shows the response of the flow field to this perturbation. The desired

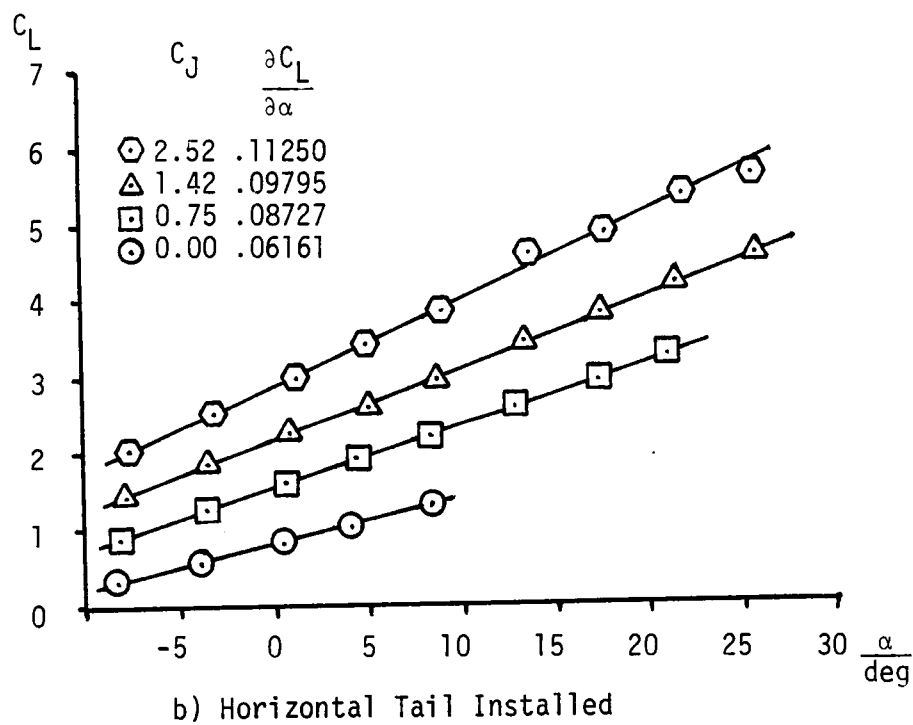
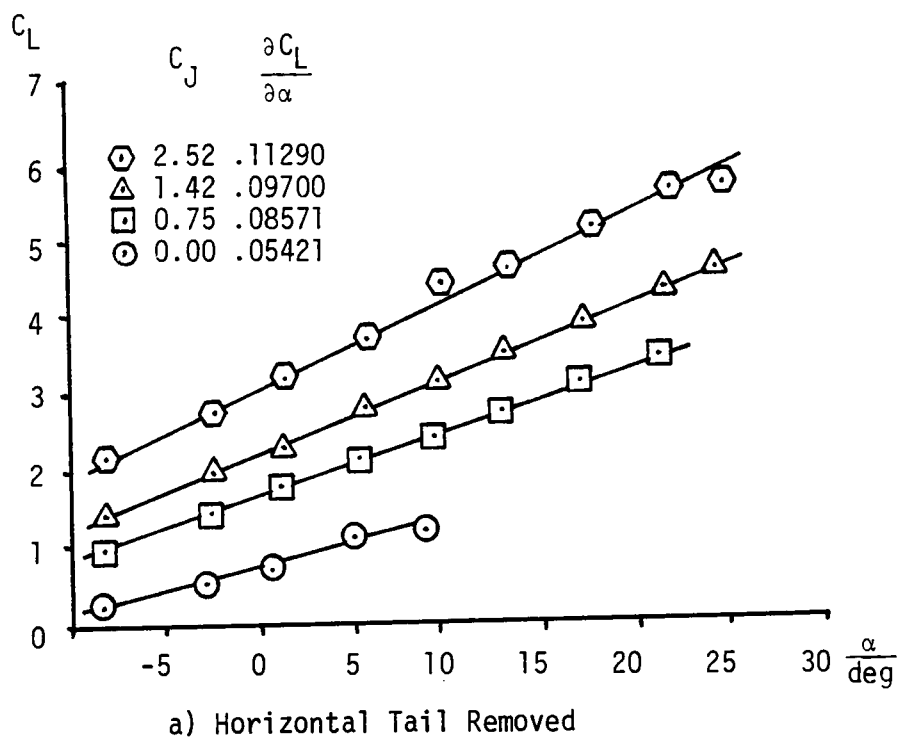
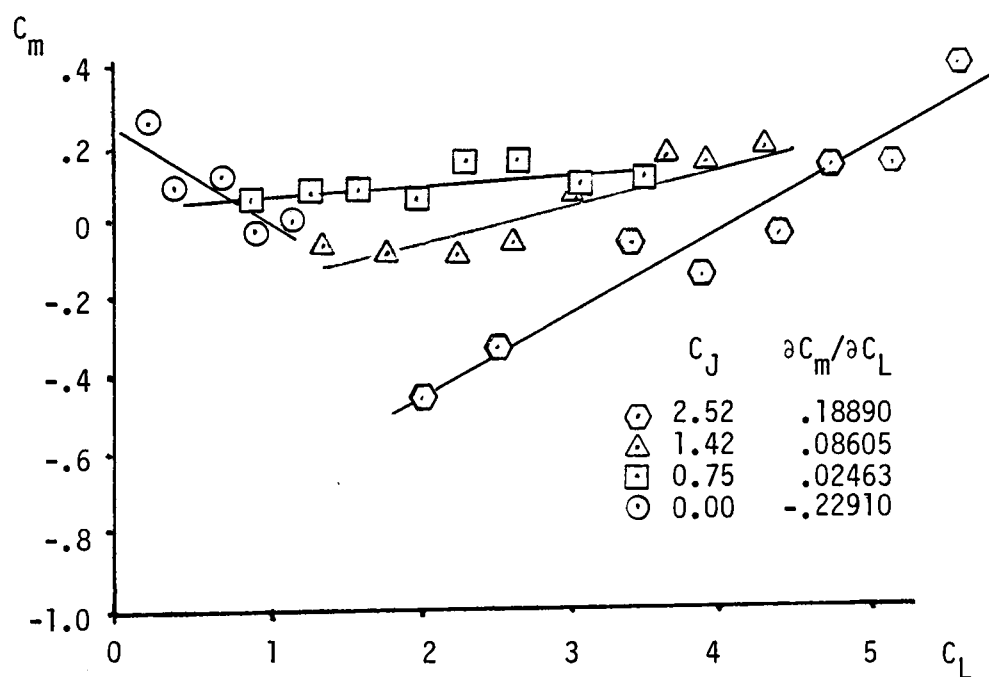
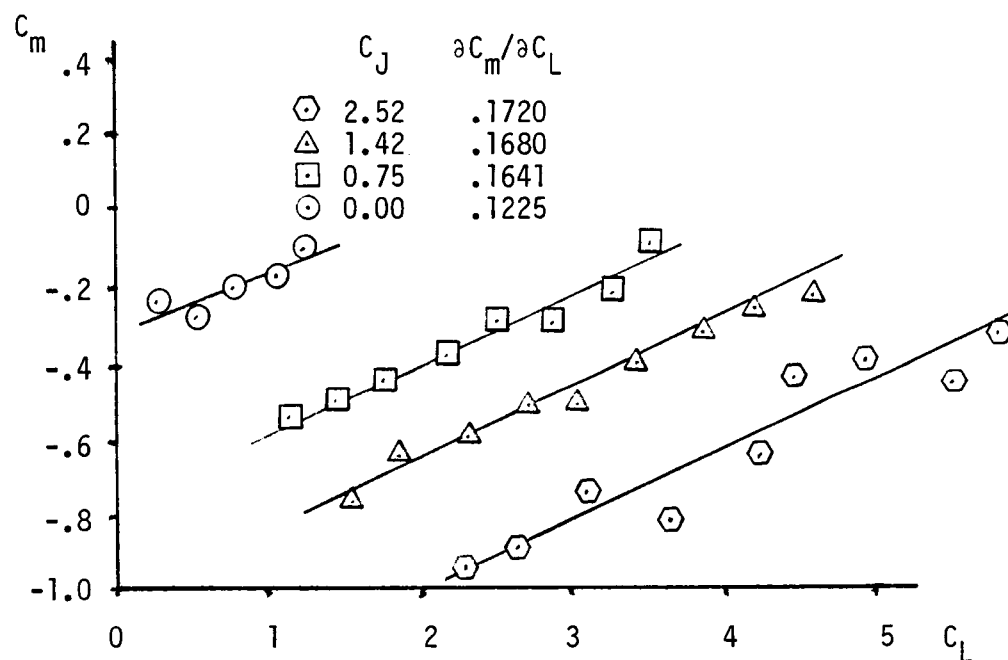


Figure 34: Determination of $\frac{\partial C_L}{\partial \alpha}$ Derivatives from Wind Tunnel Data



a) Horizontal Tail Installed



b) Horizontal Tail Removed

Figure 35: Determination of $\partial C_m / \partial \alpha$ Derivatives from Wind Tunnel Data

partial derivatives can then be obtained as direct approximations through finite differences. If the program, e.g., produces a tail downwash angle ϵ_{T1} in response to the input parameter α_1 , and a second angle ϵ_{T2} in response to α_2 , then the desired partial derivative can be written

$$\frac{\epsilon_{T2} - \epsilon_{T1}}{\alpha_2 - \alpha_1} = \frac{\Delta\epsilon_T}{\Delta\alpha} \approx \frac{\partial\epsilon_T}{\partial\alpha} \quad (156)$$

The chosen parameters and results are discussed in the next section.

6. PRESENTATION AND DISCUSSION OF RESULTS, CONCLUSIONS AND RECOMMENDATIONS

6.1 Chosen Geometry of Panels and Vortex Filaments

For cases that required high resolution, the wing was segmented into eight strips along the semi-span, and three chordwise sections each, totaling 48 panels (Figure 36a). The length of jet sheet panels, wing tip and wake vortex filaments was chosen to be 0.5 meters, extending downstream to Trefftz plane at $x=10.0$ meters. The total number of elements for this model amounted to 128 jet panels, 224 wake, and 102 tip vortex filaments. A complete run for one data point, i.e. a particular set of input parameters for velocity, air density, angle of attack, flap deflection, and blowing coefficient required approximately two hours of Central Processing Unit (CPU) time on a VAX 780/11 digital computer. A simplified model used 18 wing panels (Figure 36b), a Trefftz plane at $z=8.0$ meters and a length of .75 meters for the jet panels and vortex filaments. These simplifications and a relaxation of error margins for the iterations reduced the required CPU time for one run to approximately 15 minutes.

For comparison with sample data from the literature, plain rectangular and trapezoidal wing planforms were used.

6.2 Comparison of Lift Coefficients

To check the nonlinear vortex-lattice program, the spanwise distribution of local circulations and lift coefficients was computed for a rectangular wing and a trapezoidal wing with a taper ratio of 0.5. The results were compared with those from Multhopp's lifting line

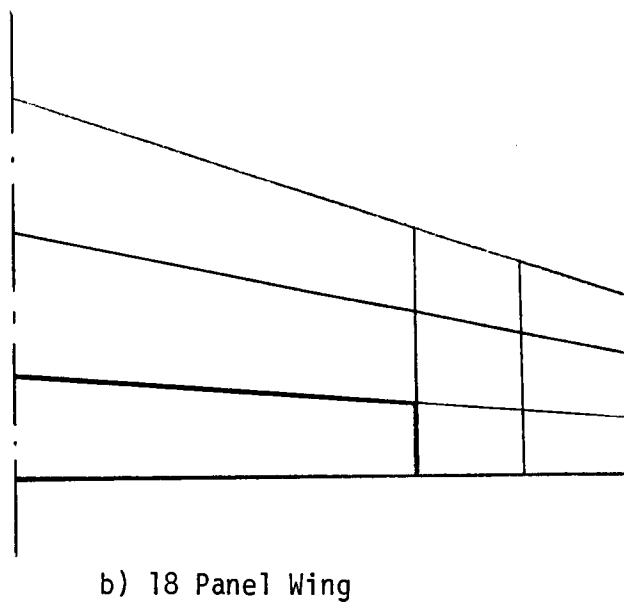
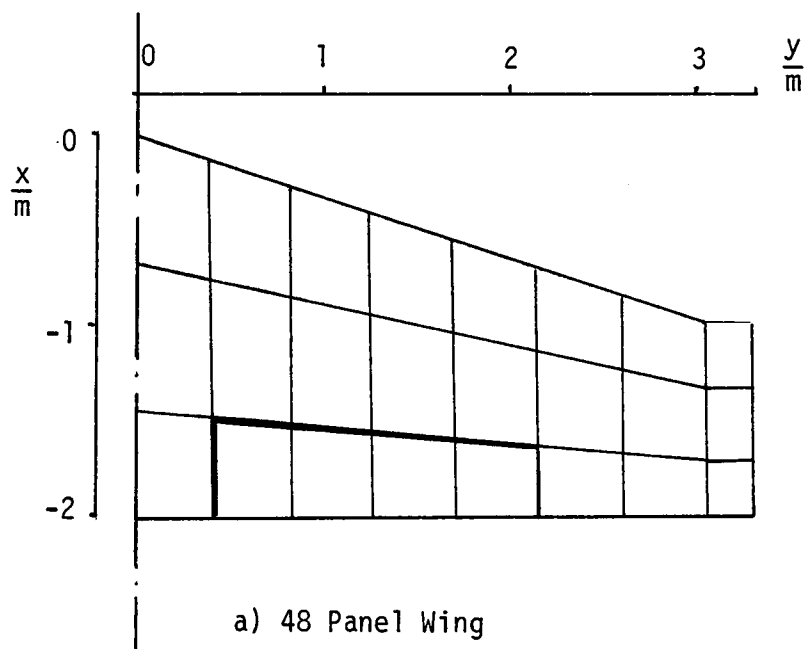


Figure 36: Chosen Geometry for Jetwing Panels

theory and the extended lifting line theory by Weissinger [25] (Figure 37). Results from more sophisticated methods like lifting surface theories and experimental data generally show values in between those lifting line methods. The results from the non linear vortex lattice routine show excellent agreement with the literature findings, giving confidence in the method.

Overall lift coefficients generated by the program for the simplified wing were compared with Jetwing tail off wind tunnel data (Figure 38). As expected, for the no blowing case, this model overpredicts the lift, mainly due to the fact that this simplified model does not consider the fuselage cut out for the flaps. With the sectional blowing coefficient adjusted to this model, the lift prediction for the jet flapped wing agrees fairly well with wind tunnel data. As expected, the decline in wind tunnel lift curve slopes at high angles of attack, which is caused mainly by viscous effects, cannot be predicted by the nonviscous vortex lattice theory.

Also, the program does not model the actual blowing of the jet exhaust over portions of the upper wing surface, which may explain why it underpredicts the lift at high values of the blowing coefficient.

6.3 Comparison of Downwash Data

Again, for the purpose of checking the non linear vortex-lattice program, the downwash angles in a Trefftz plane located one semi-span behind a rectangular and a tapered wing were computed and compared to literature data (Figure 39). There is good agreement in the center portion of the wing up to 60% of the semi-span, with the non linear

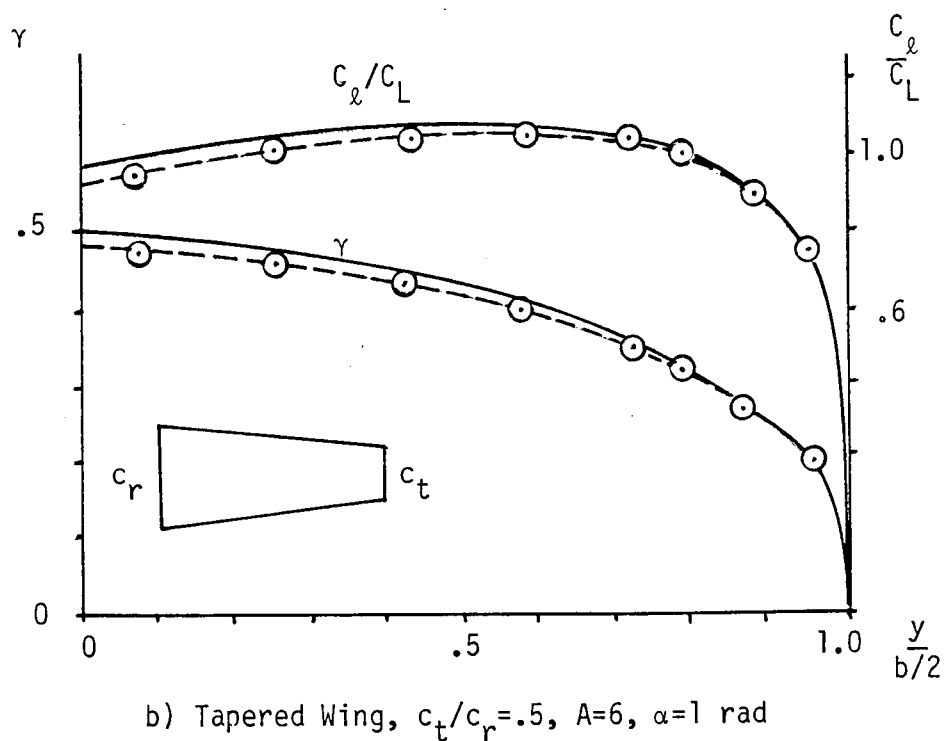
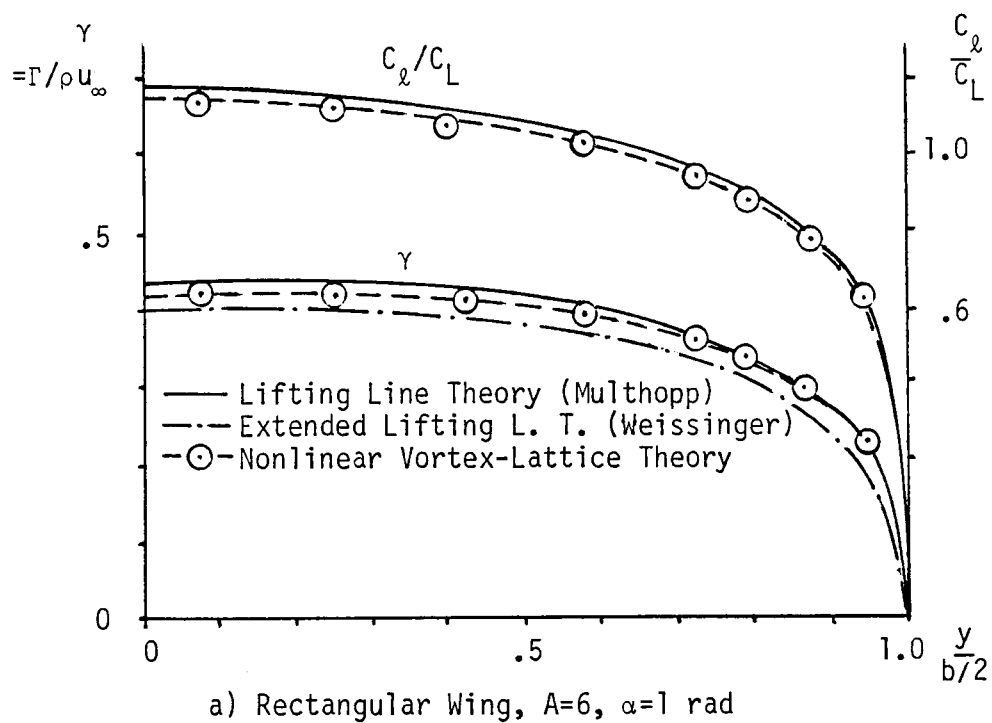


Figure 37: Comparison of Lift Distributions Resulting from Different Theories

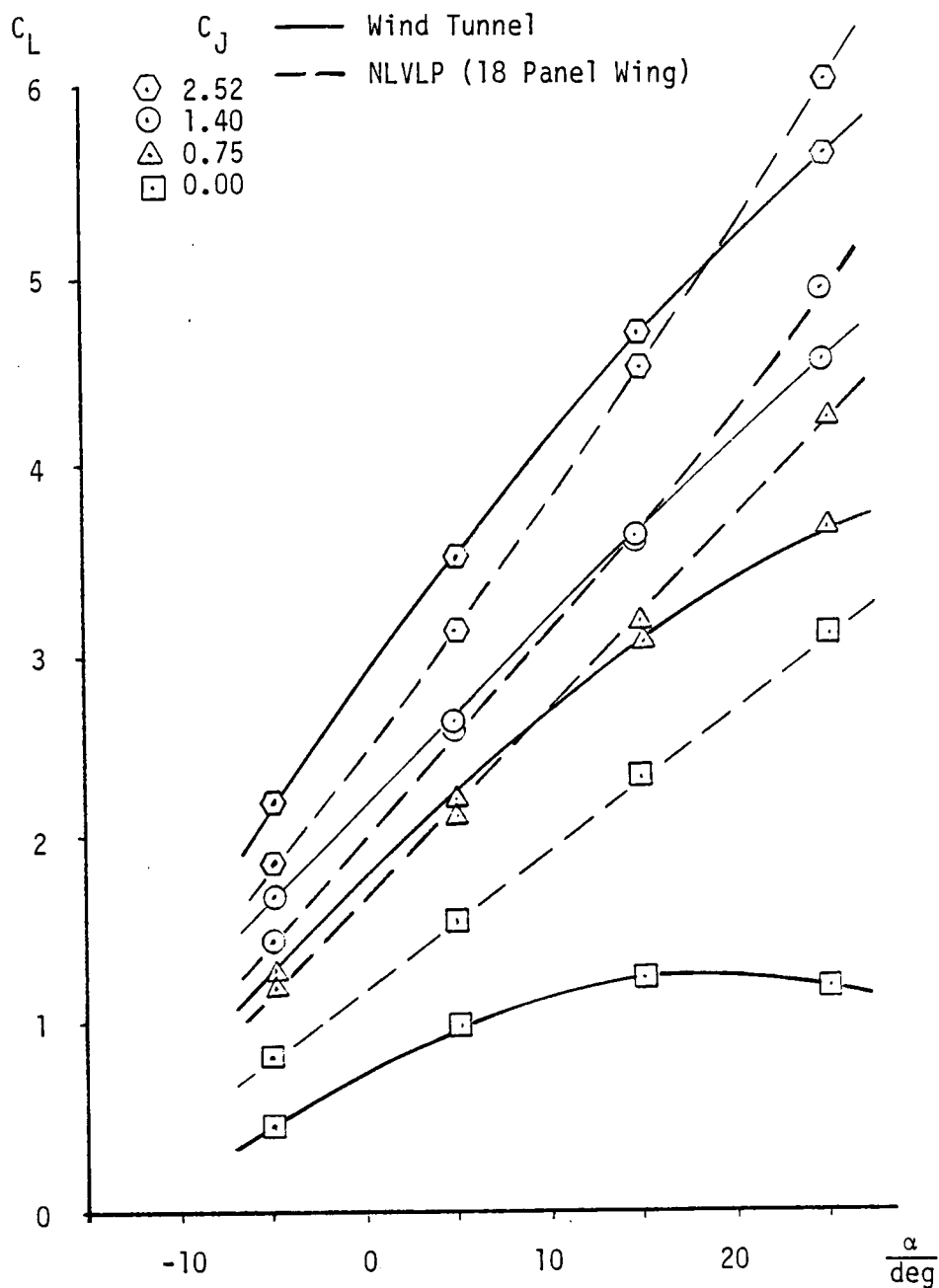


Figure 38: Comparison of Wind Tunnel Data with Results from Non-linear Vortex-Lattice Theory for a Simplified Jetwing Model

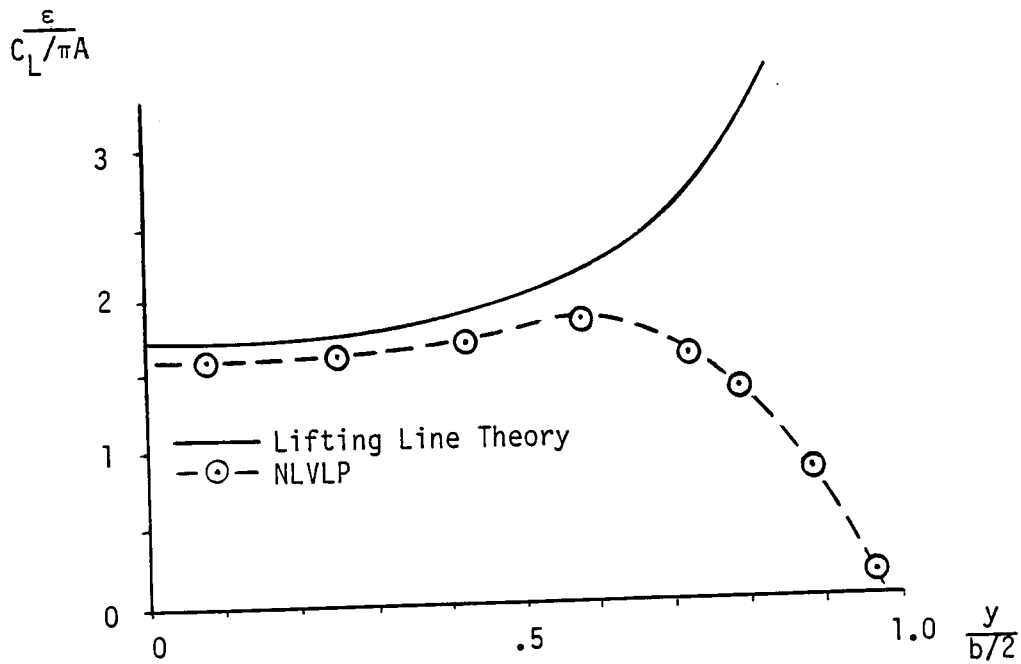
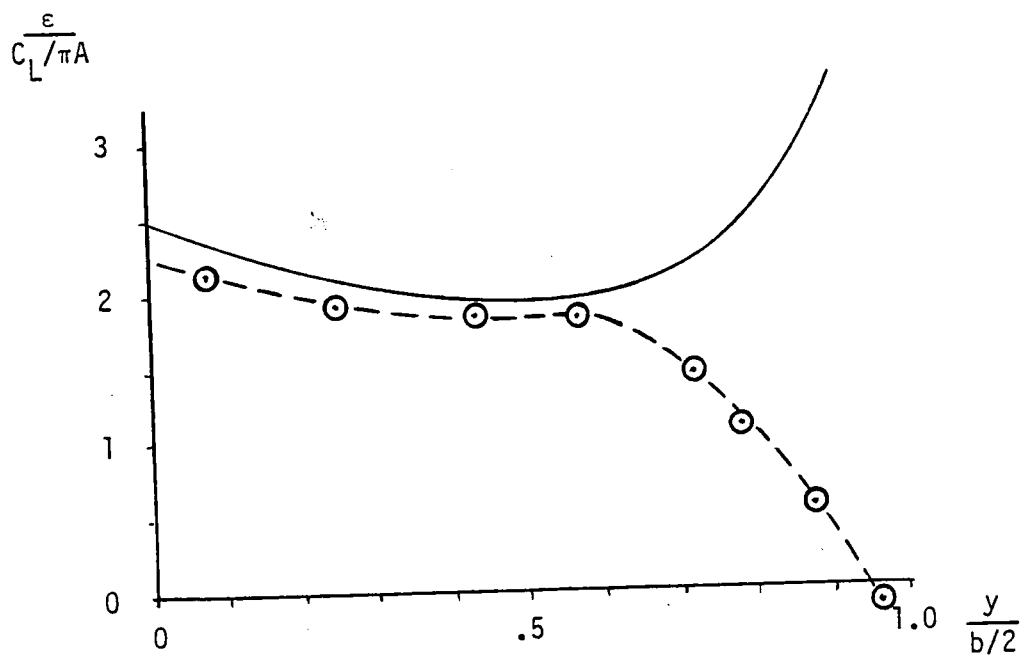
a) Rectangular Wing, $A=6$ b) Tapered Wing, $c_t/c_r = 0.5$, $A=6$

Figure 39: Comparison of Results from Simple Lifting Line Theory and Nonlinear Vortex-Lattice Theory for the Downwash Angle Distribution Behind Untwisted Wings ($x=-b/2$, $z=0$)

vortex-lattice predictions being slightly less than the values of the pure lifting line theory. This is coherent with the previous findings on the lift coefficients, and it can be concluded that the results from the non linear theory are more accurate.

The discrepancy between the curves in the outboard portions of the wing finds its explanation in the fact that the trailing wake vortex sheet in the nonlinear theory is curved upward, while in the linear lifting line theory it is not. In this theory the location of the downwash angles under consideration is in the plane of the trailing vortices resulting in a downwash distribution with a singularity at the wingtip, i.e., in the immediate vicinity of $y=b/2$ the vertical velocities jump from very large downwash values inside to very large upwash values outside of the wing span (Figure 40a).

In the program based on the nonlinear vortex-lattice theory, the trailing free vortices are aligned with the local flow angles and therefore curved upward from the plane $z=0$ (Figure 40b). This and the roll up of the free tip vortices results in the different vertical velocity distribution with a smooth transition from downwash to upwash across $y=b/2$. It is obvious that these results give a better representation of the actual flow field and further increase the confidence in the method.

To compare the program outputs with experimental data, some jet-wing flight test angle of attack measurements were used [8]. These data had been taken by means of a flow angle sensor, mounted on a boom extending 1.32 meters upstream of the right wing tip leading edge. The measurements were corrected for instrument errors according to

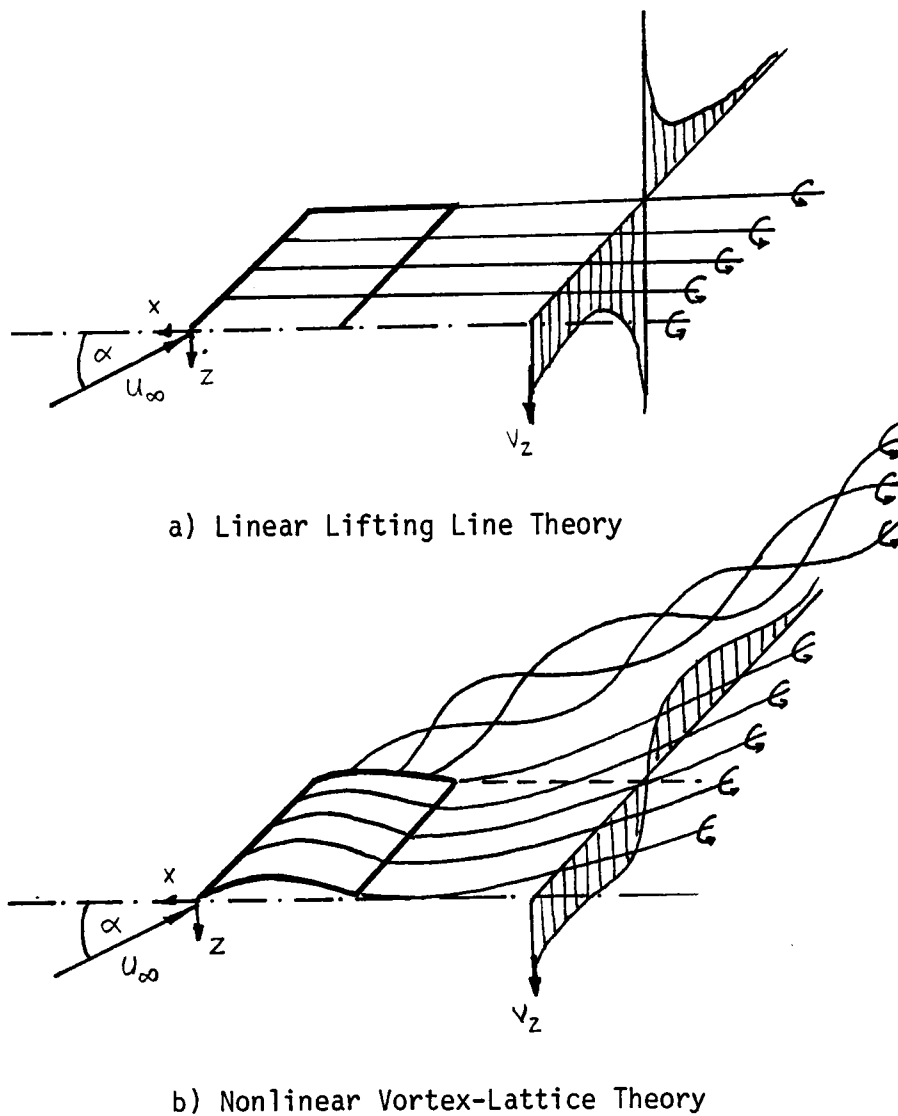


Figure 40: Comparative Sketches of Vortex Arrangement and Downwash Distribution for Two Theories

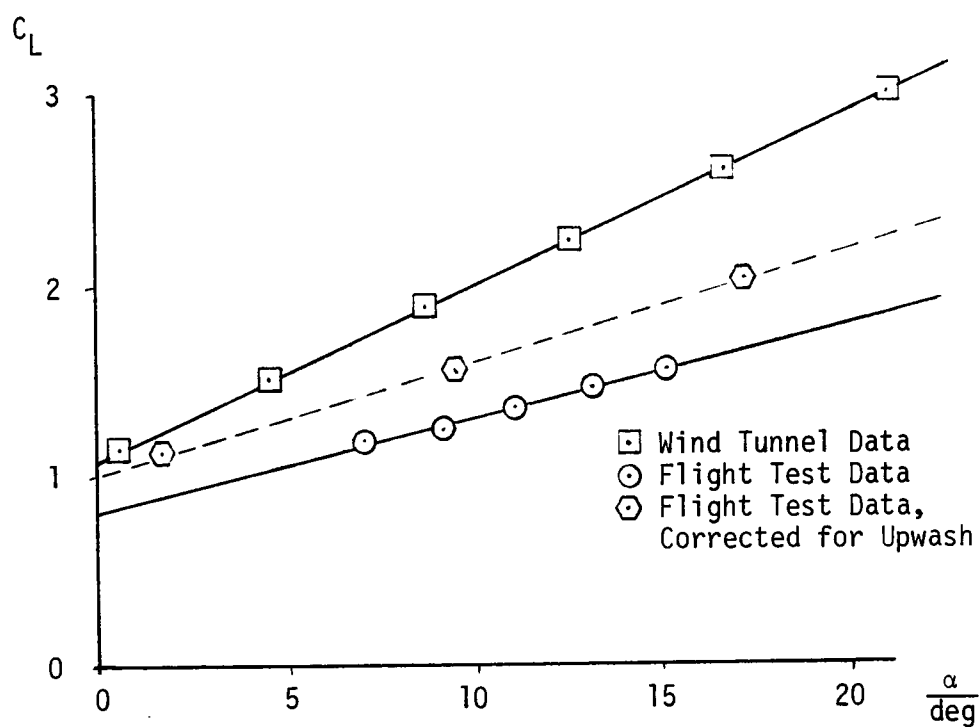
ground test calibrations, but no upwash compensation was applied to the data, assuming that the flow at the location of the sensor was relatively free of disturbances from the wing.

To clarify this point, the nonlinear vortex-lattice program was used to generate upwash data at the location of the α -sensor for the same flow conditions as existed in the flight tests. These upwash angles were subtracted from the flight test angle of attack measurements and compared with wind tunnel data (Figure 41). The results show substantial upwash, averaging approximately five degrees. The corrected flight test data are much closer to the wind tunnel data; the remaining discrepancy represents a summation of rigging errors of the α -sensor, inaccuracies resulting from the simplified wing geometry in the vortex-lattice program, flow disturbances associated with wind tunnel testing, and measurement errors in general.

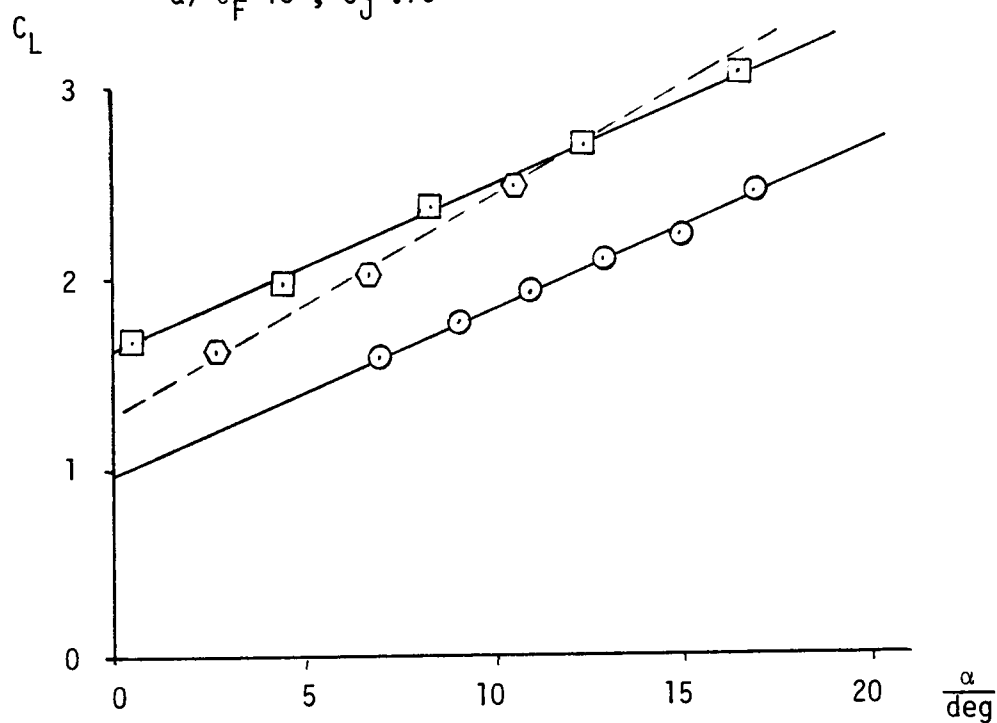
Finally, program generated flow angles were compared with flow angle measurements near the horizontal tail as described in Section 3 of this study. The vortex-lattice program was not only used to compute the theoretical downwash angle ϵ_{th} at the top tuft position on the rake for each flight test data point, but also, in a second step, to compute the perturbation $\bar{\epsilon}$ of the flow due to the horizontal tail surface itself. The measured flow angles ϵ_m were then corrected, yielding ϵ_{mc} .

The three downwash angles resulting from each flight test data point and the reduction methods outlined in Section 3 are listed in Table 3 and are shown in Figures 42 and 43.

Only for the 15 degree flap position the theoretical and experimental data compare fairly well; the average difference between the



a) $\delta_F = 15^\circ$, $C_J = 0.75$



b) $\delta_F = 30^\circ$, $C_J = 0.75$

Figure 41: Correction of Flight Test Angle of Attack Measurements for Upwash at the Boom Mounted Sensor

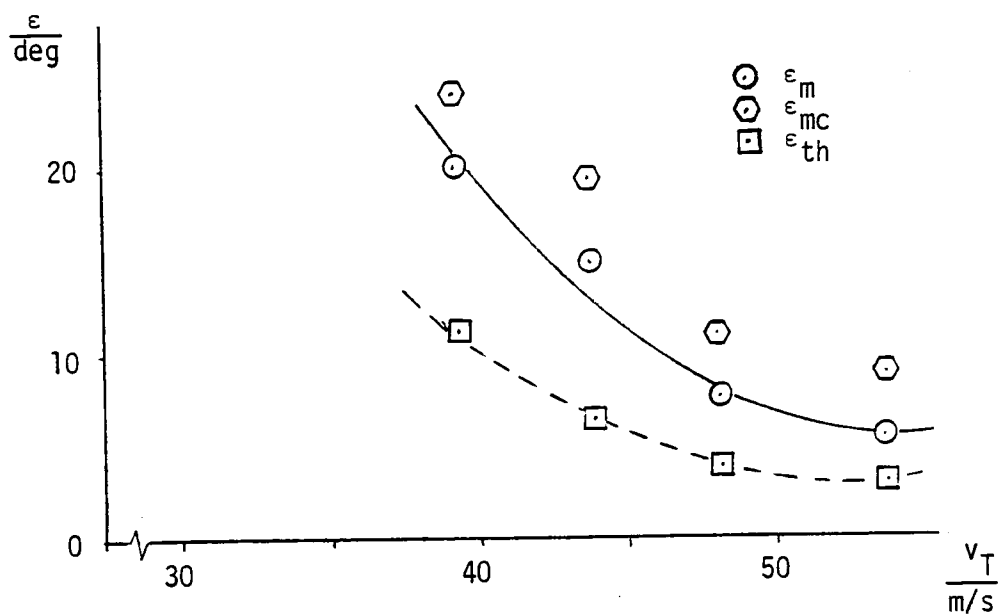
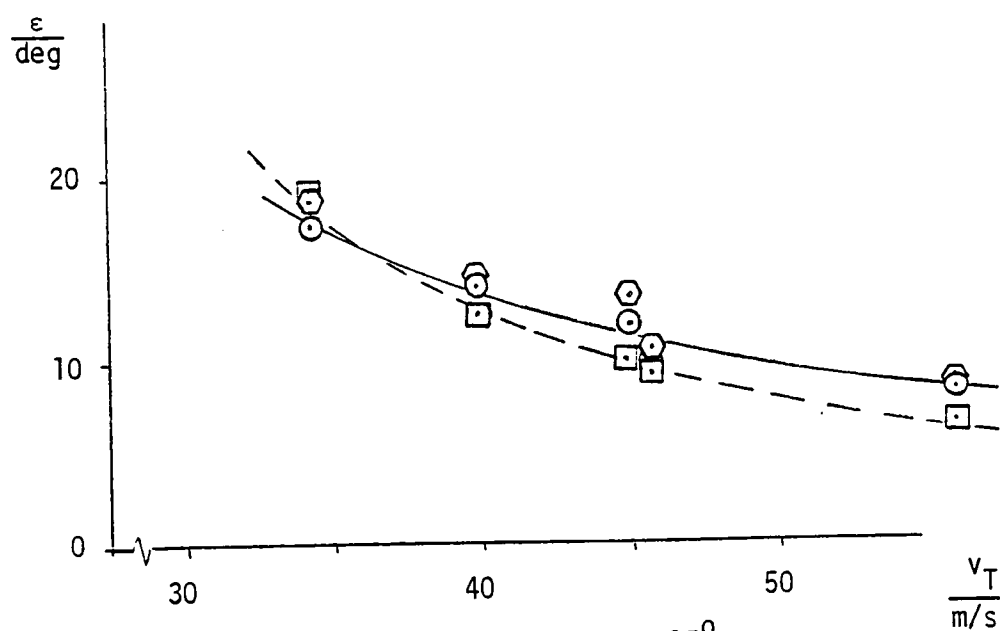
a) Level Flight Conditions, $\delta_F = 0$ b) Level Flight Conditions, $\delta_F = 15^\circ$

Figure 42: Jetwing Downwash Angles versus Airspeed at the Top Tuft Rake Position for Zero and 15 Degrees Flap Deflection

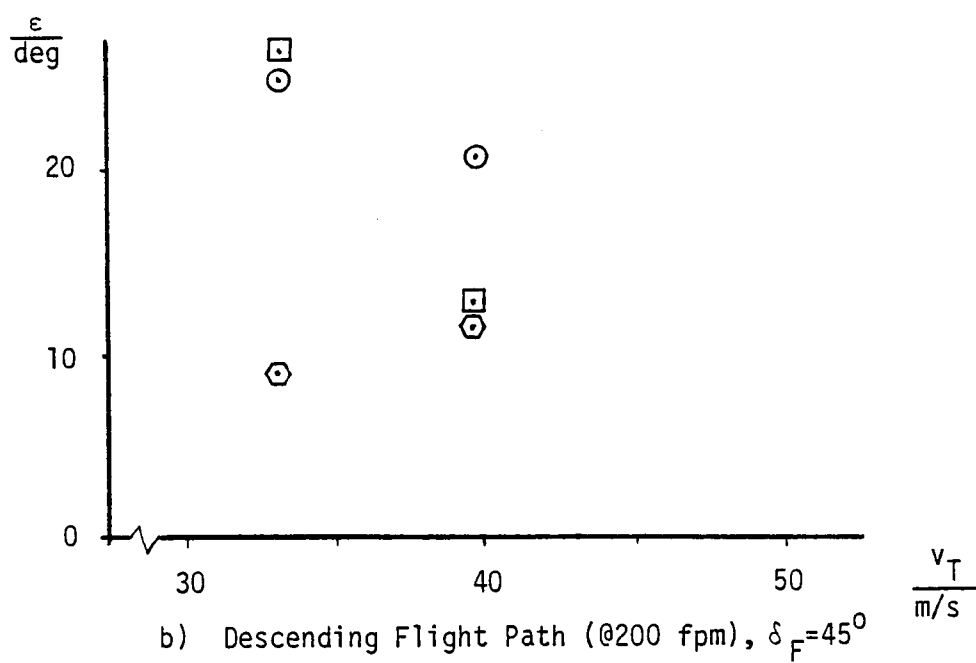
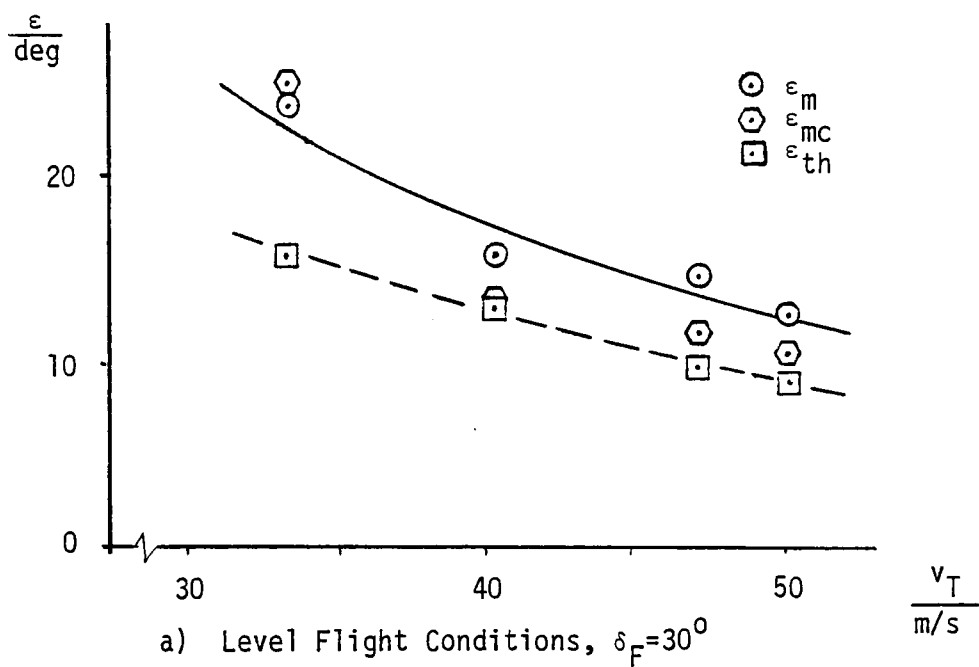


Figure 43: Jetwing Downwash Angles versus Airspeed at the Top Tuft Rake Position for 30 and 45 Degrees Flap Deflection

two curves representing the experimental and theoretical data is approximately five degrees for the zero degree flap position, and approximately four degrees for the 30 degree flap position. No curve fits were attempted for the 45 degree flap position, since only two data points were available and for the slow speed point the pilot reported a tail stall close to the time when the data were recorded.

The corrected measured downwash angles are closer to the theoretical values in some cases, and further away from them in others. There appears to be no general consistency, except for the zero degree flap case, where the corrected angles are all approximately four degrees larger than the measured ones. Some of the possible causes for the remaining discrepancy between the data are outlined in the following.

On the experimental side, one weak point of the data sampling method is the correlation of panel instrument data with the tuft angles on the photographic picture. All data were taken in stabilized level flight conditions which assured a reasonable steadiness of parameters like airspeed, altitude, power setting, stabilizer, trim angle, etc. The application of proper calibration procedures for these data reduces the errors to the approximate 2% margin common for flight testing.

A larger uncertainty remains with more transient data like angle of attack, elevator deflection angle, and the tuft angles. It has to be remembered that the Jetwing research aircraft showed static longitudinal instability for the tested configurations, which required

frequent changes of the elevator position by the pilot to keep the airplane's pitch attitude reasonably constant.

The airplane's inertia will make it resist high frequency changes in the flow conditions, but the tufts themselves will easily follow flow disturbances caused by small gusts of the turbulent wake of the wing. This may contribute to the data scatter particularly in case of zero flap deflection, where the angles of attack are high and the turbulent wake is close to the horizontal tail. Besides the possible errors from the changing angles, there remains some uncertainty about the actual angle of attack in flight. The windtunnel wall effects may have caused the aircraft to generate a slightly different lift coefficient than it would for the same angle of attack in free flight, or, in different words, for the same lift coefficient as in free flight, the aircraft may have been at a slightly different angle of attack in the windtunnel.

On the theoretical side the major error sources come from simplifying assumptions which were necessary to limit the amount of theoretical effort and computer time. The simplifications include the neglect of viscosity, thicknesses of wing and horizontal tail, and the presence of the fuselage and vertical tail. Also, the reduction of wing panels to a minimum number was done to stay within budget constraints.

Naturally, the computer program is sensitive to errors in the input data. If the α -input into the program, for example, differs from the actual angle of attack, then the computer code cannot possibly produce the proper downwash angles.

This may be the main cause of error for the chosen procedure of

correcting the measured downwash for influences from the horizontal tail. The production run for the perturbation angles uses the theoretical tail angle of attack as input value; this term contains an output downwash angle from a previous run of the program.

6.4 Generation of Downwash Data and Derivatives for Longitudinal Static Stability

To produce data for the facilitation of a longitudinal stability analysis, the nonlinear vortex-lattice routine was used to generate upwash and downwash angles as well as derivatives with respect to α at ten selected points along the fuselage centerline and three points along a horizontal line through the aerodynamic center of the horizontal tail. The runs were repeated for a matrix of three selected blowing coefficients and four flap settings. The airspeed was chosen to be 50 m/s (97 kts) and the angle of attack 10 degrees. For the determination of derivatives, α was increased to 11 degrees for each run. Resulting program outputs are represented in Figures 44 through 47.

The plots of upwash angles ahead of the wing and downwash behind the wing show an increase of the angles for both, blowing coefficient and flap deflection, as could be expected. For higher blowing values and flap deflections the downwash angles are quite substantial. This complies with tuft photos taken in flight. Fairly independent of flap setting and blowing coefficient, there is always strong downwash and change of downwash in the area right above the rear portion of the wing. This also complies with photos taken of tufts on the fuselage side. In the area near the wing there are some crossovers of curves with different blowing coefficients for both, the angles and the

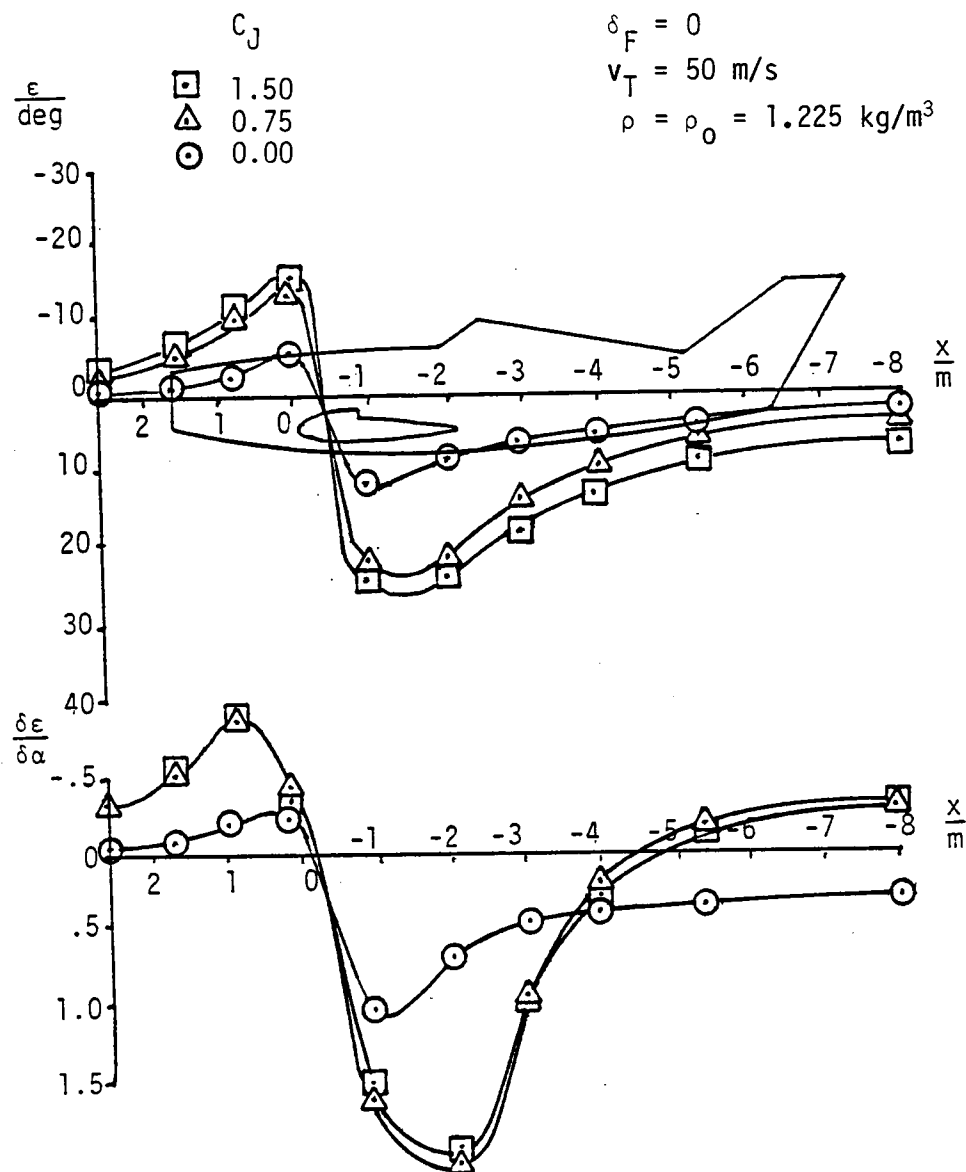


Figure 44: Jetwing Downwash Angle and $\partial \epsilon / \partial \alpha$ Distribution Along the Fuselage Centerline for Zero Degrees Flap Deflection

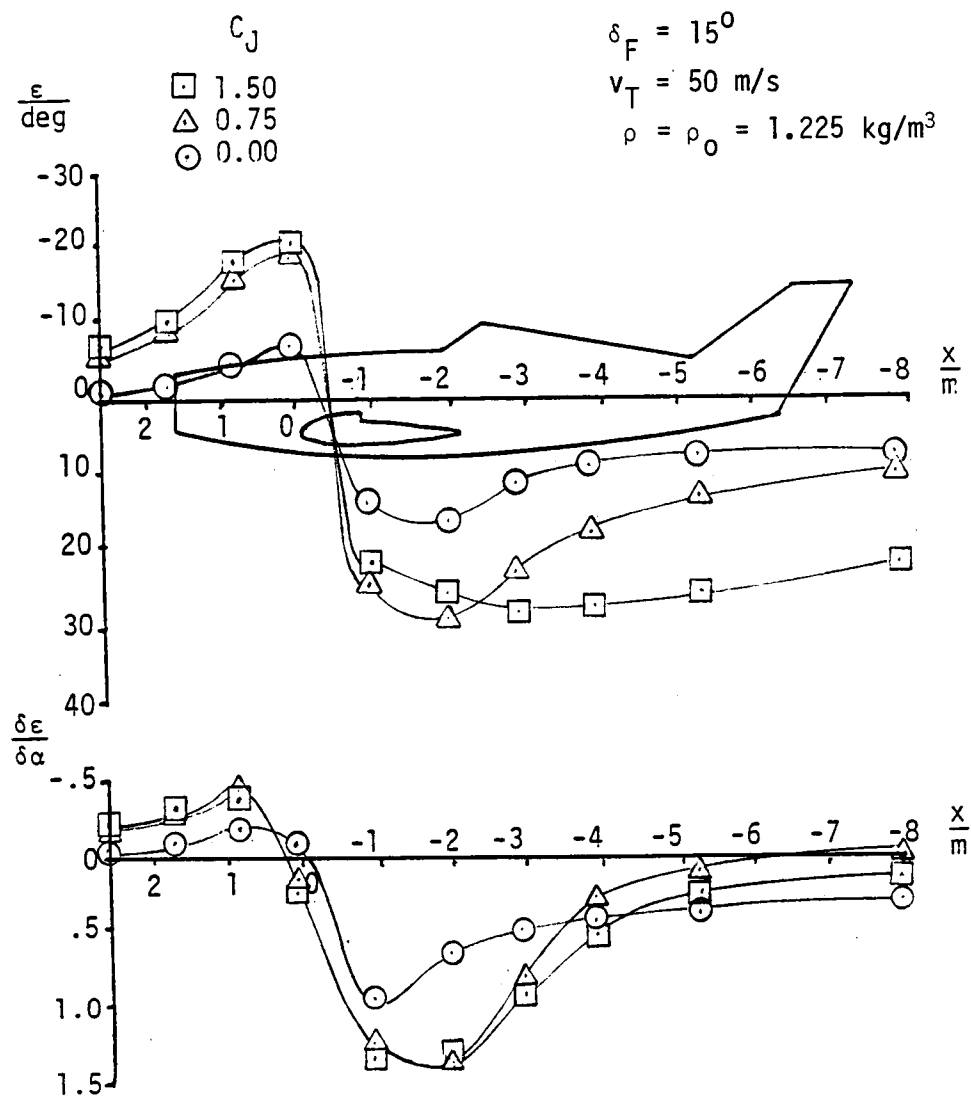


Figure 45: Jetwing Downwash Angle and $\partial \epsilon / \partial \alpha$ Distribution Along the Fuselage Centerline for 15 Degrees Flap Deflection

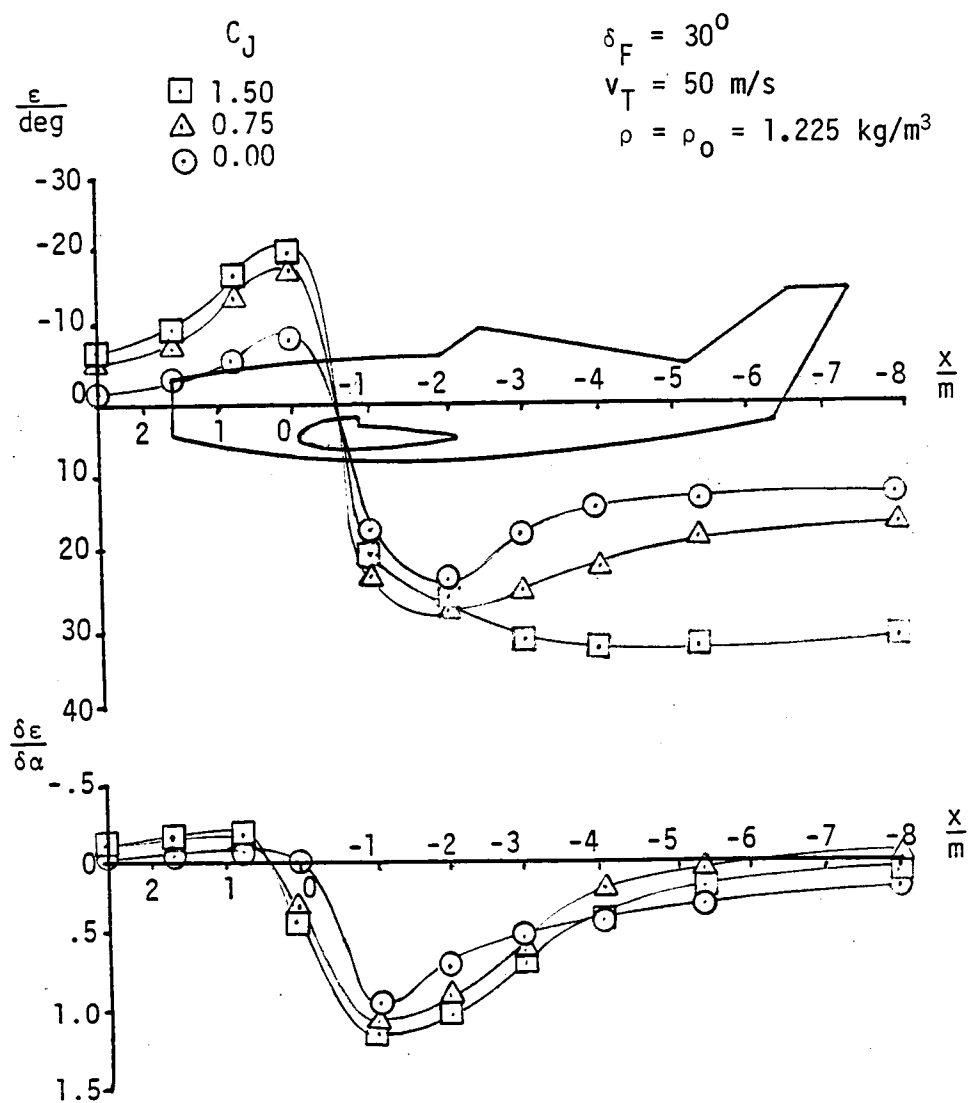


Figure 46: Jetwing Downwash Angle and $\partial \epsilon / \partial \alpha$ Distribution Along the Fuselage Centerline for 30 Degrees Flap Deflection

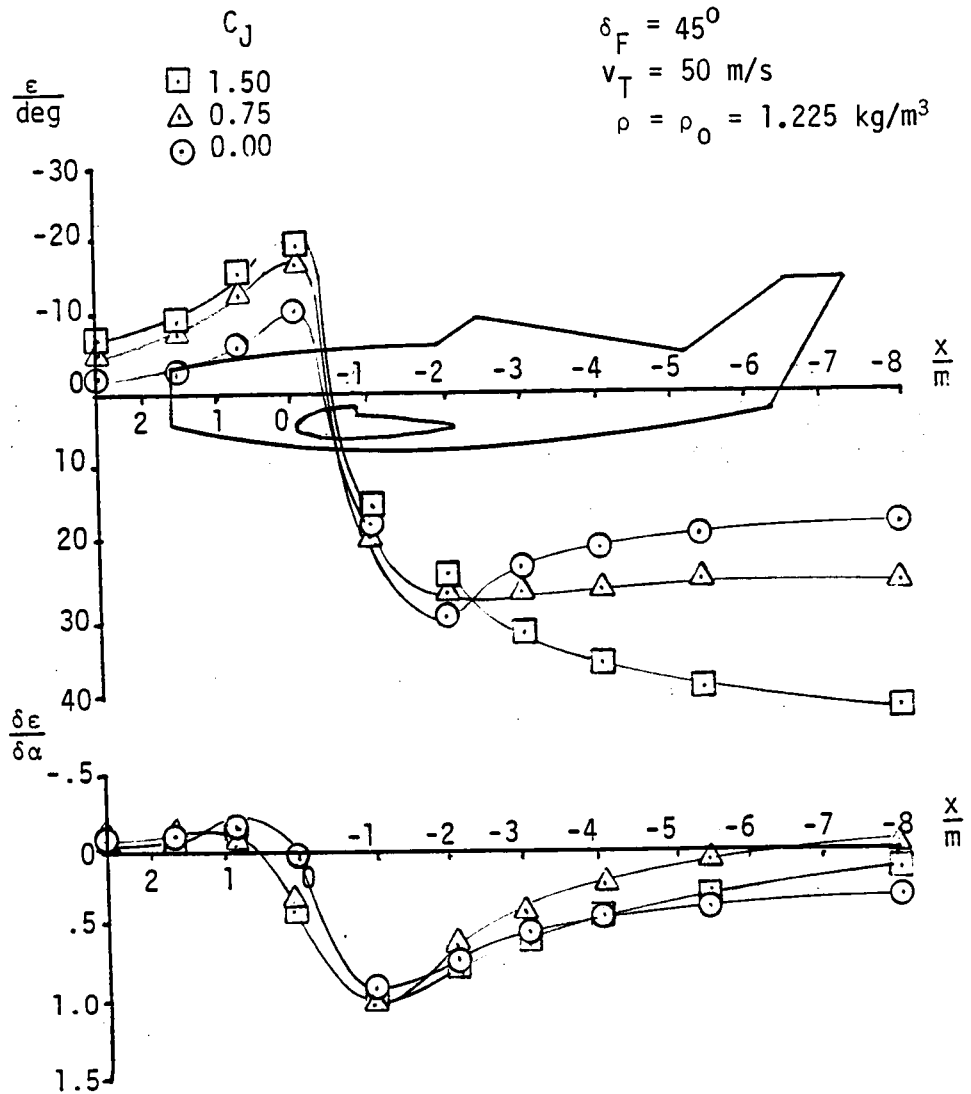


Figure 47: Jetwing Downwash Angle and $\partial \epsilon / \partial \alpha$ Distribution Along the Fuselage Centerline for 45 Degrees Flap Deflection

derivatives. This is not fully understood, but may be explained by the entrainment effect that the stronger blowing jet has on the upstream flow field.

A surprising result is the fact that the derivative curves for the higher blowing coefficients cross over the line for no blowing around $x=-4.0$ meters behind the wing. This would indicate an increase of the stability contribution of the horizontal tail due to blowing. The phenomenon shows for all flap settings. If the results are accurate, then the only explanation for this behavior of the flow lies in the particular geometry of the trailing tip and wake vortices and the influence of the jet sheet. It appears that the presence of the jet sheet reduces the downwash at the tail, induced by the trailing vortices.

Once blowing, however, an increase in the blowing coefficient also slightly increases the downwash derivative, meaning less stability contribution of the horizontal tail. A study of the downwash distribution along the semi-span of the horizontal tail shows an increase of the downwash toward the tip by an approximate factor of 1.5.

These data points are included in Figure 48, which show the result of Moore's analysis [37], derived from generalized terms in the longitudinal stability equations and slopes of windtunnel data curves. The diagram also shows points obtained from empirical formulae in Domash [38] and Perkins & Hage [31].

The data agree with the results from the nonlinear vortex-lattice program only for the case of no blowing. It has to be remembered that these formulae and Moore's analysis give general terms and

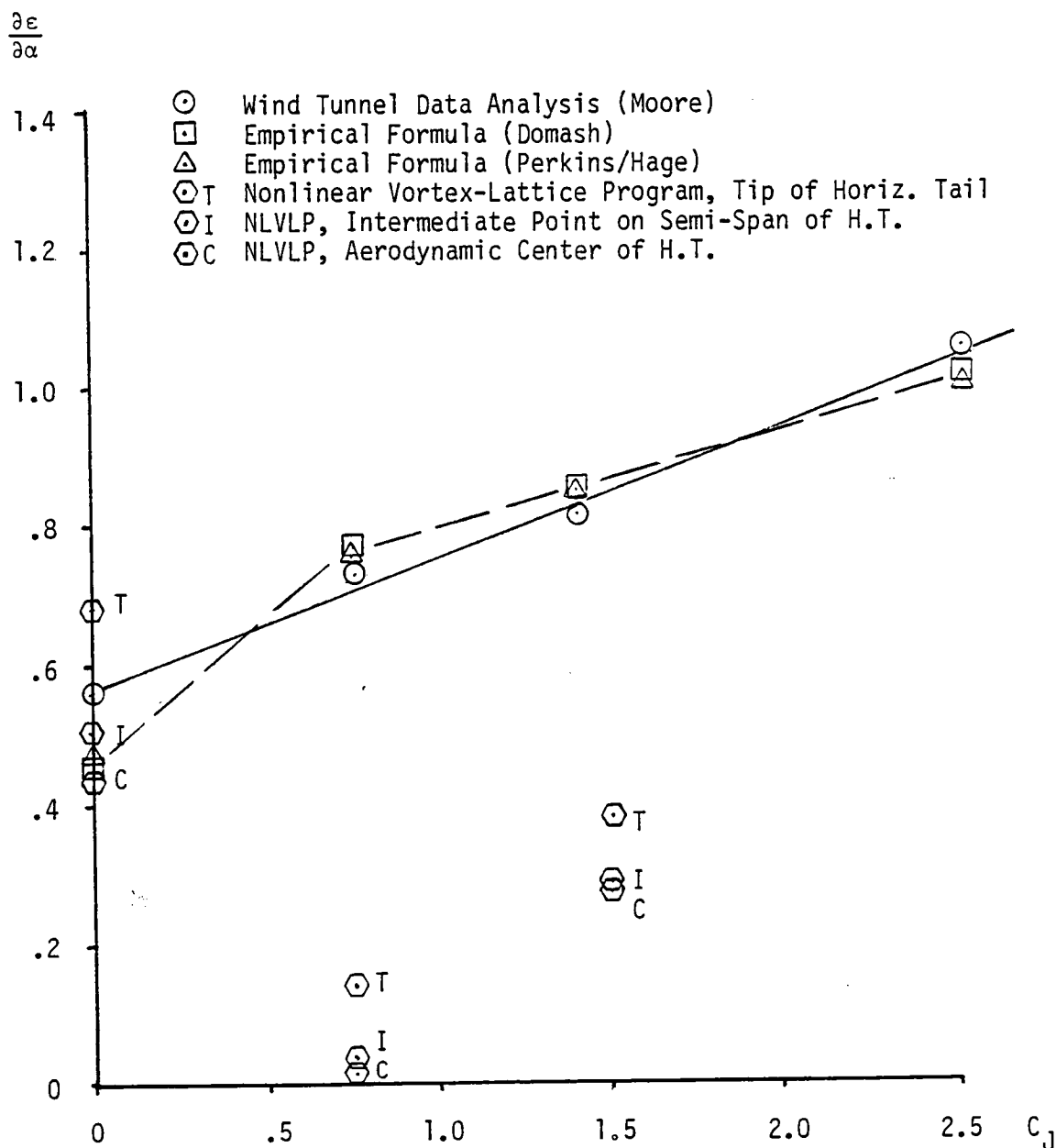


Figure 48: Comparison of Downwash Derivatives at the Jetwing Tail Resulting from Different Methods

average values designed for linearized theory, while the nonlinear vortex-lattice routine produces downwash angles and derivatives at specified points and for specific input conditions. These data points show the influence of special wake geometrics, etc., which are subject to change with angle of attack and other input data.

The minimum required horizontal tail areas for neutral static longitudinal stability were determined by Moore [37] for four values of the blowing coefficient:

C_J	0.00	0.75	1.42	2.00
S_T/ft^2	9.59	32.38	56.40	199.00
S_T/m^2	0.89	3.01	5.24	18.49

6.5 Conclusions and Recommendations

The overall results of this study are summarized as follows:

1. The Upper Surface Blowing strongly influences the flow field around the aircraft. The upwash ahead of the wing and the downwash behind it are substantially increased. This is of importance for the pitch control of the airplane. The vortex-lattice program generated flow angles can be used to quantify the pitching moment contributions of selected parts of the airplane. This may be used, for example, to correct the tail off C_m wind tunnel data for the influence of inlet and fuselage. The remainder represents the wing pitching moment which can be used as an experimental data base for the comparison with theoretical methods [3].

2. For each increase in the blowing coefficient, the flow field about the blown wing affects the longitudinal static stability in a destabilizing manner at all major parts of the airframe.
 - a) The inlet normal force becomes more destabilizing due to a substantial increase of the upwash variation at the inlet entrance.
 - b) The forward fuselage becomes more destabilizing for the same reasons.
 - c) The rear fuselage is subject to increased downwash variation which would have a stabilizing influence if the flow were nonviscous [31]; for a drag generating aft body in viscous flow, however, this effect is reduced or even reversed [25].
 - d) The horizontal tail becomes less stabilizing due to the fact that the downwash derivative grows with increasing blowing coefficient.
3. There is some surprising indication that for certain cases, the presence of the jet sheet may actually improve the stability contribution of the tail, however, more research is needed before final conclusions can be made.
4. The flight test method used in this study for the sampling of downwash angles in flight produced some reasonable data, however, for accurate comparisons of theoretical results with flight tests, some improvement of the equipment and procedures is recommended.
5. The FORTRAN code which was developed as part of this study,

based on a nonlinear vortex-lattice theory, proved to be an excellent tool for the flow field analysis about a jet-flapped wing. Improvement efforts should aim at making it faster by refining the numerical iteration schemes. The flow modeling can be further improved by incorporating viscous effects and the downstream break-up of the jet sheet and wake.

BIBLIOGRAPHY

BIBLIOGRAPHY

1. Schubauer, G. B., "Jet Propulsion with Special Reference to Thrust Augmentations," NACA TN 442, 1933.
2. Hagedorn, H., and Ruden, P., "Windkanal Untersuchungen an einem Junkers-Doppelfluegel mit Ausblaseschlitz am Knick des Hauptfluegels," Technischer Bericht des Institutes fuer Aeromechanik und Flugtechnik der Technischen Hochschule Hannover, Lilienthal Gesellschaft, Bericht A64, 1938.
3. Sinha, A. K., "Study of Lift Characteristics of Upper Surface Blown Jetwing with Rectangular Nozzle," Thesis (Ph.D.), The University of Tennessee, Knoxville, December 1983.
4. "Jetwing from Colorado," Air International, Vol. 14, No. 2, pp. 69-71, February 1978.
5. "Jetwing, a Very Quiet Machine," Professional Pilot, pp. 16-19, April 1978.
6. Fink, D. E., "Propulsive Lift Concepts Flight Tested," Aviation Week and Space Technology, September 1977.
7. Jetwing JW-1 Windtunnel Data, UTSI Flight Research Laboratory, Tullahoma, Tennessee, December 1978.
8. Kimberlin, R. D., "A Flight Test Evaluation of the Ball-Bartoe Jetwing Propulsive Lift Concept," UTSI Report 81-1, The University of Tennessee Space Institute, Tullahoma, Tennessee, July 1, 1981.
9. Kimberlin, R. D., Solies, U. P., and Sinha, A. K., "A Flight Test Evaluation and Analytical Study of the Ball-Bartoe Jetwing Propulsive Lift Concept without Ejector," UTSI Report 82/17, The University of Tennessee Space Institute, Tullahoma, Tennessee, October 1, 1982.
10. Chen, Z. L. and Wu, J. M., "Jet Wing Vortex Lattice Theory with Nonlinear Wake and Tip Flows," AIAA-83-0263, Reno, Nevada, January 1983.
11. Herrington, R. M., et al. USAF Flight Test Engineers Handbook, AF-TR No. 6273, Air Force Flight Test Center, Edwards Air Force Base, 1961, Rev. 1966.
12. "United States Standard Atmosphere," NACA TR 1235, 1962.

13. Trimbach, J. M., "The V- γ Performance Method as Applied to the Ball-Bartoe Jetwing STOL Aircraft," Thesis (M.S.), The University of Tennessee, Knoxville, June 1981.
14. Williams, J., Butler, S. F. J., and Wood, M. N., "The Aerodynamics of Jet Flaps," A.R.C. R&M No. 3304, London, 1961.
15. Shen, C. C., Lopez, M. L., and Wasson, N. F., "Jet-Wing Lifting-Surface Theory Using Elementary Vortex Distributions," Journal of Aircraft, Vol. 12, No. 5, pp. 448-456, May 1975.
16. Maskell, E. C. and Gates, S. B., "Preliminary Analysis for a Jet-Flap System in Two-Dimensional Inviscid Flow," A.R.C. C.P. 359, London, June 1955.
17. Spence, D. A., "The Lift Coefficient of a Thin, Jet-Flapped Wing," A.R.C. Proc. Series A, Vol. 238, No. 1212, pp. 46-48, London, January 1961.
18. Spence, D. A., "The Lift on an Aerofoil with a Jet-Augmented Flap," The Aeronautical Quarterly, Vol. IX, Part 3, pp. 287-299, August 1958.
19. Jacobs, W. F., "Theory of the Jet-Augmented Flap in Two Dimensions," Report No. LG 72 E R0023, Lockheed-Georgia Company, November 1, 1972.
20. Jacobs, W. F., "Theory of the Jet-Augmented Flap in Three Dimensions," Report No. LG 73 E R0023, Lockheed-Georgia Company, December 1, 1972.
21. Silverstein, A., Katzoff, S., and Bullivant, W. K., "Downwash and Wake Behind Plain and Flapped Airfoils," NACA TR 651, 1939.
22. Silverstein, A. and Katzoff, S., "Design Charts for Predicting Downwash Angles and Wake Characteristics Behind Plain and Flapped Wings," NACA TR 648, 1939.
23. Ross, A. J., "The Theoretical Evaluation of the Downwash Behind Jet-Flapped Wings," A.R.C. R&M No. 3119, London, 1961.
24. Woolard, H. W., "A Review of Some Air Force STOL Aircraft Aerodynamic Prediction Methods," AIAA-74-992, Los Angeles, California, August 1974.
25. Schlichting, H. und Truckenbrodt, E., Aerodynamik des Flugzeuges, Band I and II, Springer Verlag, Berlin, 1960.
26. Abbot, H. T. and von Doenhoff, A. E., Theory of Wing Sections, Dover Publications, New York, 1959.

27. Preston, J. H., "Note on the Circulation in Circuits Which Cut the Streamlines in the Wake of an Aerofoil at Right Angles," A.R.C. R&M No. 2957, London, 1954.
28. Wertz, R. D., "Application of Spence's Methods and Data to Prediction of Lift Coefficients of the Jetwing," Unpublished notes, UTSI Flight Research Laboratory, 1981.
29. Burden, R. L., Faires, J. D., and Reynolds, A. C., Numerical Analysis, Prindle, Weber & Schmidt, Boston, 1978.
30. Gersten, K., "Tragfluegeltheorie bei Unterschallgeschwindigkeit," WGL, S. 25-40, Jb. 1958.
31. Perkins, C. D. and Hage, R. E., Airplane Performance Stability and Control, John Wiley and Sons, New York, 1963.
32. Etkin, B., Dynamics of Flight - Stability and Control, Second Edition, John Wiley and Sons, New York, 1982.
33. Schade, R. O., "Free-Flight Tunnel Investigation of Dynamic Longitudinal Stability as Influenced by the Static Stability Measured in Wind Tunnel Force Tests Under Condition of Constant Thrust and Constant Power," NACA TN 2075, 1950.
34. Maskell, E. C. and Spence, D. A., "A Theory of the Jet Flap in Three Dimensions," A.R.C. Proc. Series A, Vol. 251, pp. 407-425, London, 1959.
35. Munk, M. M., "The Aerodynamic Forces on Airship Hulls," NACA TR 184, 1923.
36. Multhopp, H., "Zur Aerodynamik des Flugzeugrumpfes," Luftfahrt Forschung Band T8, S. 52-66, 1941; Translation: NACA TM 1036, 1942.
37. Moore, D. A., "Jetwing Longitudinal Stability and Tail Sizing Calculations," Unpublished notes, UTSI Flight Research Laboratory, Tullahoma, Tennessee, 1985.
38. Domash, D. O., Sherby, S. S., and Conolly, T. F., Airplane Aerodynamics, Fourth Edition, Pitman Publishing Corporation, New York, 1967.
39. Kerebrock, J. L., Aircraft Engines and Gas Turbines, The MIT Press, Cambridge, Massachusetts, 1977.
40. Rooke, K. A., "An Evaluation of Nozzle Pressure Ratio as a Means of Determining In-Flight Gross Thrust of the Ball-Bartoe Jetwing with the Pratt & Whitney JT 15 D-1 Turbofan Engine Installed," Thesis (M.S.), The University of Tennessee, August 1980.

41. McCormick, B. W., Aerodynamics, Aeronautics, and Flight Mechanics, John Wiley and Sons, New York, 1979.
42. McCormick, B. W., Aerodynamics of V/STOL Flight, Academic Press, New York, 1967.
43. Martynov, A. K., Practical Aerodynamics, The MacMillan Company, New York, 1965.
44. Keuthe, A. M. and Chow, C. Y., Foundations of Aerodynamics, Third Edition, John Wiley and Sons, New York, 1976.
45. Betz, A., "Eine Anschauliche Ableitung des Biot-Savartschen Gesetzes," ZAMM Bd. S, S. 149-151, 1928.
46. Uchiyama, N., "Studies of Various Rocket Fin-Body Configurations with Angle of Attack in Linearized Subsonic, Transonic and Supersonic Flow," Thesis (Ph.D), The University of Tennessee, Knoxville, December 1977.
47. Mendenhall, M. R. and Spangler, S. B., "Calculation of the Longitudinal Aerodynamic Characteristics of Upper-Surface-Blown Wing-Flap Configurations," AIAA-79-0120, New Orleans, January 1979.
48. Taylor, A. S., "A Theoretical Investigation of the Longitudinal Stability, Control and Response Characteristics of Jet-Flap Aircraft," Parts I and II, A.R.C. R&M No. 3272, London, 1962.
49. Turabian, K. L., A Manual for Writers of Term Papers, Theses, and Dissertations, The University of Chicago Press, Chicago, 1973.
50. Anonymous, Guide to the Preparation of Theses and Dissertations, Fifth Edition, The Graduate School of The University of Tennessee, Knoxville, 1983.

VITA

Uwe Peter Solies was born on November 23, 1949 in Wetzlar, West Germany. He received his general education attending elementary and high schools in West German cities and graduated with the "Abitur" degree in 1968. Then he joined the German Luftwaffe and served for two years in a fighter unit.

After graduating as Reserve Lieutenant, he started his engineering education at the Ruhr University (RUB) at Bochum, West Germany, in Fall 1970. While he was a student in Bochum, he also completed a total of six months practical training in general manufacturing processes in industry, including two months of flight simulator design and human engineering in the Netherland's National Aerospace Laboratory.

After graduating from RUB with a prediploma degree in Mechanical Engineering in 1972, he started a program in Aerospace Engineering at the Rhine Westphalian Technical University (RWTH) at Aachen, West Germany. There he joined the "FVA", an aeronautical research association, which was founded by Theodore von Karman, when he was a professor for aerodynamics in Aachen in 1920. Uwe Peter Solies became actively involved in several research, design and building projects, such as a fiberglass sailplane, a hydraulic flap control system, a small rocket for atmospheric research, noise reduction on tow airplanes, etc. During his time in Aachen he also became an enthusiastic glider pilot and accumulated 300 hours of flight experience with 21 different types of gliders and motorgliders.

He graduated from RWTH in 1978 with a "Diplom-Ingenieur" degree and came to the University of Tennessee Space Institute at Tullahoma, Tennessee, for a Ph.D program in Aerospace Engineering. Here he worked with the UTSI Flight Research Division on problems of aircraft performance, stability and control in theory and flight test. His major research activity focused on powered lift technology. He also became the founding president of the Soaring Club of UTSI, which has been a very active group since then.

He accepted an offer for a faculty position at Mississippi State University and its renowned Rapsheet Flight Research Laboratory, where he will work as Assistant Professor in Aerospace Engineering after his graduation as Doctor of Philosophy from the University of Tennessee.