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PREDICTION OF SPAN LOADING OF STRAIGHT-WING/PROPELLER COMBINATIONS UP TO STALL

M. A. McVeigh, L. Gray, and E. Kisielowski

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UNITED TECHNOLOGY, INC.
Blue Bell, Pa. 19422
for Langley Research Center

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M. A. McVeigh
L. Gray
E. Kisielowski

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16. Abstract

A method is presented for calculating the spanwise lift distribution on straight-wing/propeller combinations. The method combines a modified form of the Prandtl wing theory with a realistic representation of the propeller slipstream distribution. The slipstream analysis permits calculations of the non-uniform axial and rotational slipstream velocity field of propeller/nacelle combinations. This non-uniform field is then used to calculate the wing lift distribution by means of the modified Prandtl wing theory.

The method utilizes non-linear aerodynamic section data for both the wing and the propeller blade airfoil sections and is applicable up to stall. The theory is developed for any number of non-overlapping propellers, on a wing with partial or full-span flaps, and is applicable throughout the aspect ratio range from 2.0 and higher.

The analysis is programmed for use on the CDC 6600 series digital computer. The computer program is used to calculate slipstream characteristics and wing span load distributions for a number of configurations for which experimental data are available. Favorable comparisons are demonstrated between the theoretical predictions and the existing data.

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SUMMARY

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The method utilizes non-linear aerodynamic section data for both the wing and the propeller blade airfoil sections and is applicable up to stall. The theory is developed for any number of non-overlapping propellers, on a wing with partial or full-span flaps, and is applicable throughout the aspect ratio range from 2.0 and higher.

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LIST OF SYMBOLS

AR	wing aspect ratio
a	lift curve slope for finite aspect ratio, per degree
a_s	speed of sound, m/sec
a_o	section lift curve slope, per degree
B	number of blades per propeller
B_n	coefficients in trigonometric series
b	wing span, m
C_D	total wing drag coefficient
C_d	section drag coefficient
C_L	total wing lift coefficient
C_l	section lift coefficient
C_Q	propeller torque coefficient, $Q/\rho n^2 D^5$
C_T	propeller thrust coefficient, $T/\rho n^2 D^4$
C_{TS}	propeller thrust coefficient, $T/q_s \pi R^2$
c	wing local chord, m
c_R	wing root chord, m
D	propeller tip diameter, m
E	edge velocity factor
F	propeller tip loss factor
i_{TL}	inclination of the propeller axis to the fuselage centerline, degrees

J	propeller advance ratio, V_o/ nD
l	wing section lift, per unit span, N
M_o	freesteam Mach number, V_o/a_s
M_v	local Mach number for propeller blade element, V/a_s
n	rotational speed, rev/sec
Q	propeller shaft torque, N.m.
q_s	average slipstream dynamic pressure, N/m^2
R	propeller tip radius, D/2, m
R_e	Reynolds number
r	local radius in propeller disk plane, m
r_s	local radius in slipstream, m
T	propeller thrust, N
u	axial component of velocity induced by a blade element in the propeller disk plane, m/sec.
V	local velocity, m/sec.
V_a	component of freestream velocity along the propeller axis, m/sec.
V_n	component of local slipstream velocity normal to the zero-lift line, m/sec.
V_o	freestream velocity, m/sec.
V_s	component of local slipstream velocity parallel to the zero-lift line, m/sec.
$\overline{V}_{s_a}$	axial component of local velocity in the fully-developed slipstream, m/sec.

V_{s_a} — momentum-weighted mean axial velocity in the fully-developed slipstream, m/sec.

V_{s_t} — tangential component of local velocity in the fully-developed slipstream, m/sec.

V_w — upwash velocity component acting in the propeller disk plane due to the presence of a lifting wing, m/sec.

v — nondimensional change in upwash due to the slipstream

X_p — distance of propeller hub forward of wing quarter chord, m

Y — spanwise co-ordinate of local wing element, m

Y^* — spanwise co-ordinate of flap end, m

Y_p — spanwise co-ordiante of propeller axis, m

α — angle of attack relative to airfoil section chord-line, degrees

α_B — angle of attack relative to fuselage centerline, degrees

α_c — corrected section angle of attack, degrees

α_c — effective angle of attack of wing section, degrees

α_g — geometric angle of attack of wing section, degrees

α_i — induced angle of attack of wing section, degrees

α_{ℓ_o} — angle of attack of airfoil section at zero lift, degrees

α_o — section angle of attack for two-dimensional air-foil, degrees

α_p — propeller axis angle of attack, degrees

α_R	geometric angle of attack of wing root, degrees
α_S	inclination of the slipstream axis to the flight path, degrees
β	pitch angle for propeller blade element, degrees
β_{mk}	multiplier for induced angle of attack, degrees
δ	magnitude of discontinuity in absolute and induced angles of attack, degrees
ϵ	geometric twist at any wing section, degrees
Γ	circulation about any wing section, m^2/sec.
λ	wing taper ratio
μ	blade speed ratio for propeller blade element
μ_T	blade tip speed ratio
υ	kinematic viscosity, m^2/sec.
ϕ	inflow angle for propeller blade element, degrees
ϕ_o	inflow angle for propeller blade element, excluding contribution from induced velocity in disk plane, degrees
ρ	ambient density, kg/m^3
σ	solidity for propeller blade element
θ	wing spanwise coordinate, $\cos^{-1}(2y/b)$
θ^*	spanwise co-ordinate for flap end, $\cos^{-1}(2y^*/b)$
Ω	rotational speed, $2\pi n$ radians/sec.
ω	angular velocity induced by a blade element behind the propeller disk plane, radians/sec.
ω'	factor to account for low aspect ratio effects

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SECTION 1

INTRODUCTION

The propeller slipstream exerts an important influence on wing load distribution, which in turn affects the aircraft stall characteristics. This effect is introduced through an increase in local velocity over the slipstream-immersed portion of the wing and a change of wing local angle of attack due to slipstream rotation. While the increased velocity tends to stabilize the flow over that wing portion, the slipstream rotation may give rise to an asymmetric stall condition due to increased local angles of attack of the wing sections behind the up-going propeller blades, and reduced angles of attack of the wing sections behind the down-going blades.

A review of the available technical literature indicates that there are no reliable theoretical or semi-empirical methods which can adequately predict the effects of a propeller slipstream on the spanwise load distribution of an entire wing. Many of the existing methods are suitable only for computing total wing forces since they are often based on gross simplifying assumptions. Thus, for example, an assumption that the slipstream-immersed portions of the wing can be treated as isolated planforms neglects the strong influence of the slipstream on adjacent wing regions. Other theoretical methods are generally classed as rigorous mathematical approaches which are usually very complex and are frequently not in sufficient agreement with experimental data to warrant their use as a design tool. Furthermore, most of the above theories use linear lift curves and as a result can not be expected to yield satisfactory agreement with test data near wing stall.

The limitation imposed by the use of linear lift curves for the wing has been successfully removed in the work reported in Reference 1. This reference presents a computerized method for predicting spanwise load distributions of straight-wing/fuselage combinations at angles of attack up to stall. This method, which is based on the Prandtl wing or "lifting line" theory as formulated by Sivells in Reference 2, provides a reliable analytical tool for predicting wing stalling characteristics of general aviation type aircraft, but is only applicable to power-off flight conditions, such as might be encountered during landing.

The current investigation extends the analysis of Reference 1 to permit calculations of span loading and stalling characteristics under power-on conditions (e.g. take-off) for wings with or without flaps and having any number of non-overlapping propellers. The present method is based on employing non-linear airfoil section characteristics for both the propeller and the wing. The basic analytical approach of this method is to retain the inherent simplicity of the Prandtl wing theory, modify the theory as required to accept non-uniform slipstream velocities, and effectively combine this modified lifting line theory with a realistic propeller theory to form a unified analytical tool.

A detailed description of this analytical method, together with the specially developed digital computer program is presented in the following pages.

SECTION 2

GENERAL REVIEW OF THE ANALYTICAL METHODS

The prime objective of the current development is to provide a practical analytical solution for determining the lift distribution and stalling characteristics of wings partially or totally immersed in a propeller slipstream. In order to depict some of the highlights of the current work relative to other approaches, this section presents a brief review of the existing analytical and experimental investigations that attempt solutions of the wing/propeller problem.

2.1 STATEMENT OF THE PROBLEM

The basic limitation in providing reliable solutions to the wing/propeller problem is related to a lack of complete understanding of the flow field generated by the wing/propeller interaction under practical operating conditions. The problem is further compounded by the difficulty of developing realistic analytical representations of this complex flow field environment so as to account for the major interaction effects acting on a wing/propeller combination. A complete solution to the problem must therefore account for all these effects, which as a minimum should include the following

(a) Local wing angle-of-attack changes due to the mean inclination and rotation of the slipstream flow.

(b) Non-uniform spanwise distribution of velocity over those portions of the wing within the slipstream.

(c) Non-uniform vertical distribution of velocity within the slipstream-immersed regions of the wing.

(d) Viscous mixing between the slipstream and freestream flow along the slipstream boundary.

In view of the real fluid flow effects involved it is unlikely that every aspect of the problem can be treated adequately, using the established analytical approaches. Historically, the approach has been to introduce a series of simplifying assumptions in order to arrive at a solution. These approaches are discussed below.

2.2 REVIEW OF EXISTING SOLUTIONS

The earliest treatment of the propeller slipstream problem is contained in the pioneering work of Koning (Reference 3) who treated the simplified case of a wing centrally immersed in a circular uninclined slipstream of uniform axial velocity without rotation. Koning applied the methods of lifting line theory to obtain a solution when the ratio of free stream velocity to slipstream velocity is close to unity. This work was extended by Glauert (Reference 4) and by Franke and Weinig (Reference 5) to a wider range of forward speeds.

Stuper (Reference 6) conducted a series of experiments to verify the predictions of Koning's theory by measuring the lift distribution on a rectangular wing with end plates under the action of a circular jet. Stuper used a specially designed fan to produce a jet without rotation and with a velocity cross-section which was approximately uniform. While the results of those experiments are somewhat impaired by the particular test arrangement used by Stuper, there is sufficient evidence to show that the Koning theory over-predicts the lift increase due to the jet.

Because of the inability of the lifting line approach, as formulated by Koning, Glauert et. al., to satisfactorily predict experimental measurements, subsequent investigators assumed that the failure of the lifting-line theory was associated with the fact that the portion of the wing immersed in the slipstream was usually of small aspect ratio. Graham, et. al.,(Reference 7) therefore approached a solution via slender body theory and the approximate lifting surface theory of Weissinger (Reference 8). Calculations made by Graham showed improved agreement with Stuper's experimental data.

Ribner and Ellis (Reference 9) generalized the Weissinger lifting surface formulation to multiple, uninclined slipstreams. Their results showed reasonable agreement with the experimental data obtained by Brenckmann (Reference 10) for the overall lift increase due to the propeller slipstream.

The test results obtained by Brenckmann represent an improvement over the experimental data of Stuper in that the former experiments utilized an infinite aspect ratio wing, thus avoiding the use of end-plates which introduce uncertainties as to the effective value of wing aspect ratio. Since Brenckmann employed a free propeller yielding a non-uniform slipstream velocity profile, the Ribner-Ellis theory, which assumes uniform velocity distributions, would not be expected to yield adequate predictions of the spanwise lift distributions as compared with Brenckmann's measurements.

Another series of tests of interest are those of Gobetz (Reference 11) and Snedeker (Reference 12) who employed a similar experimental arrangement to that of Stuper, in that a jet of air of approximately uniform velocity profile was used to simulate the propeller slipstream. These tests were designed to determine the basic effects of both wing aspect ratio and wing chord/slipstream diameter. The results were compared with theoretical calculations using the modified lifting-line theory of Rethorst (Reference 13) and it is shown that this theory is at least capable of predicting the trends of the test data.

Goland, et. al., (Reference 14) formulated a mathematical model based on potential theory approach to predict overall performance and stability characteristics of small aspect ratio wing spanning a slipstream of uniform velocity. Although this work effectively combined the R. T. Jones small aspect ratio theory with the potential flow theory to yield good correlation with test data, no attempt was made to predict and correlate the wing spanwise load distributions. This work was extended in References 15 and 16 to provide equations and charts for estimating lift and longitudinal force coefficients of STOL aircraft wings immersed in propeller slipstreams.

George and Kisielowski (Reference 17) modified the work of Reference 16 to account for non-uniformity of the propeller slipstream. In this analysis the propeller slipstream velocity was represented by a number of concentric zones of uniform velocity (staircase functions) with the wing spanning the slipstream. Although satisfactory correlations were obtained between the theoretical and experimental test data for low and moderate wing angles of attack , the theory of Reference 17 did not adequately predict lift distributions close to the wing stall.

In reviewing the above analytical attempts to solve the wing-slipstream problem, it is apparent that none of these approaches is suitable for direct application to the present problem of predicting the effects of propeller slipstream on the stall characteristics of straight wing airplanes. Either the existing theoretical models are too simplified and disregard effects which are known to be critical, (e.g. Reference 6 and 13), or the analyses are too complex and do not yield practical and reliable solutions (e.g. Reference 9). Therefore, there exists a requirement to develop an improved mathematical model capable of providing practical and reliable analytical solutions to the wing/propeller problem.

The analytical methods developed under the current program potentially represent an answer to this problem. Although this optimism is based on a few isolated correlations with the available test data, sufficient indication of the effectiveness of the developed methodology has already been obtained, as confirmed by comparative results presented later in the text. The basis for this improved mathematical model is described below.

2.3 BASIS FOR THE PRESENT ANALYSIS

A common approach of past investigations involves an idealized representation of the propeller slipstream in which the velocity is discontinuous across the slipstream boundary. This model generally requires complex solutions to the boundary conditions associated with the discontinuity.

The basis of the current analysis lies in the observation that in a real slipstream the velocity distribution remains continuous throughout the slipstream boundary.

An examination of experimental data obtained on the velocity distributions in the wakes of propellers shows that there is no sudden jump in velocity across the slipstream boundary. There is, however, a rapid increase in velocity as the boundary is crossed but the continuity of velocity is still preserved. Since the lift distribution must be continuous and the velocity distribution is continuous, then the associated circulation distribution must also be continuous. Therefore, the strength of the shed vorticity may be obtained by differentiating the spanwise distribution of circulation in the usual manner, without the complication of accounting for discontinuities in circulation.

The approach presented in the following pages utilizes a comprehensive propeller analysis to compute the slipstream flow field including swirl components of velocity. The wing-nacelle combination is then introduced into this flow field and the effects of the non-uniform propeller flow field on the wing lift distribution is computed using a modification of lifting line theory which permits the calculation of the low aspect ratio effects associated with the slipstream-immersed positions of the wing.

The validity of the simple lifting line theory utilized herein for treating wings with non-uniform spanwise velocity distributions has been verified by applying it to a problem of linearly varying spanwise velocity gradients treated in a more general and complex manner by Fejer in Reference 18. The implementation of this lifting line theory to practical wing/propeller combinations is presented in Sections 3 and 4 below.

SECTION 3

THEORETICAL ANALYSIS

This section presents a summary of the analytical methods developed for predicting the propeller slipstream effects on the spanwise load distribution of wings operating at angles of attack up to stall. The analytical approach presented herein is based upon first determining the velocity distribution in the propeller wake and then calculating its effect on the wing lift distribution. The analysis provides for the use of non-linear lift curves for both the propeller and the wing in order to realistically represent the propeller slipstream distribution and its effect on wing loading at angles of attack up to stall.

Accordingly, the first part of this section deals with the propeller slipstream calculations, and the second part presents the implementation of the slipstream parameters in the modified wing theory.

3.1 PROPELLER SLIPSTREAM ANALYSIS

The first part of the analysis deals with the propeller slipstream representation, including the required iterative solution and convergence procedures.

3.1.1 General Propeller Solution

Consider a propeller operating at an angle of attack α_p to the remote freestream of velocity V_0, as shown in Figure 1. The presence of a lifting wing behind the propeller modifies the inflow to the propeller disk through an induced upwash velocity V_w. An an approximation this upwash velocity is assumed to be uniform across the propeller disk, and to lie within the disk plane. The method used for calculating this upwash velocity is presented in Section 4.2.

For the purpose of analyzing the wing lift distribution it is assumed that the slipstream can be considered as being fully developed. With this assumption the average inclination of the contracted slipstream can be readily calculated using

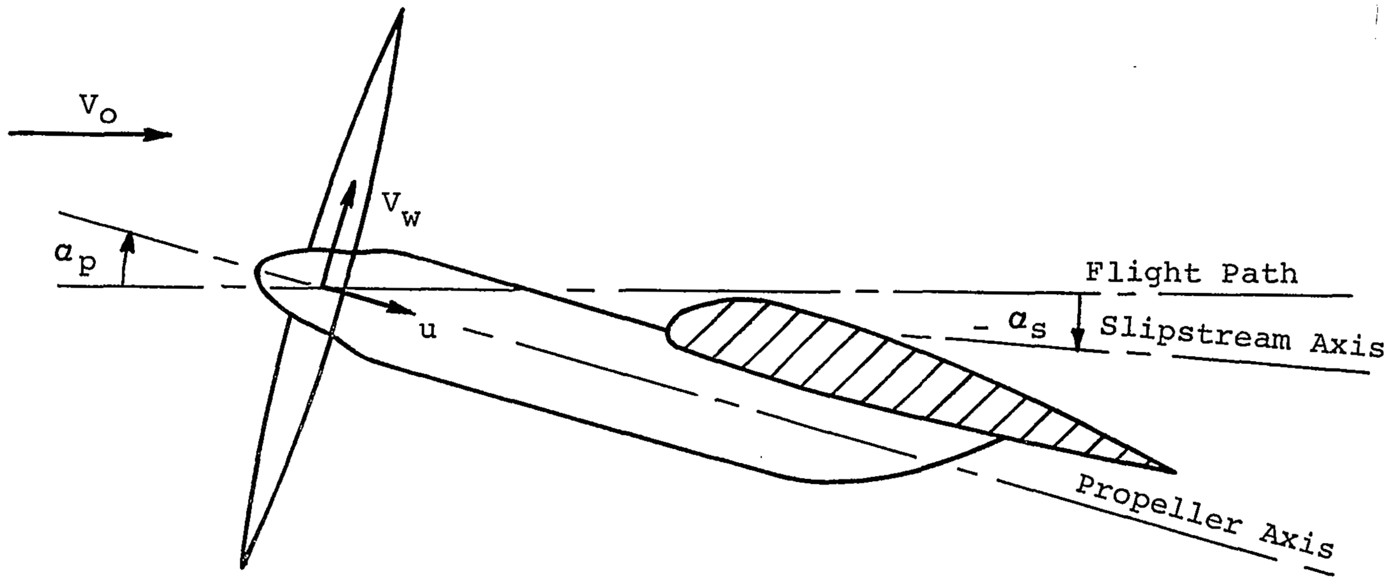


Figure 1. Notation For a Propeller Operating in the Presence of a Wing

simple actuator disk theory (e.g. Reference 19) and, in the notation of Figure 1, is obtained from

$$\tan\left(\alpha_p + \alpha_s\right) = \frac{V_0 \sin \alpha_p + V_w}{V_0 \cos \alpha_p + 2u} \tag{1}$$

where u is the axial induced velocity increment at the propeller disk. From momentum theory, u is related to propeller thrust, T, by

$$u^2 = \frac{T}{2\rho \pi R^2 V'} \tag{2}$$

where V' is the resultant velocity at the disk and is given by

$$V' = \sqrt{\left(V_0 \cos \alpha_p + u\right)^2 + \left(V_0 \sin \alpha_p + V_w\right)^2} \tag{3}$$

On combining equations (2) and (3) a quartic in u is obtained and is generally solved by iteration. However, since the present application is to conventional aircraft where α_p is small and $V_w \leq V_0$ this quartic may be reduced to a quadratic whose solution is

$$u = \frac{-V_0 \cos \alpha_p}{2} + \sqrt{\left(\frac{V_0 \cos \alpha_p}{2}\right)^2 + \frac{T}{2\rho \pi R^2}} \tag{4}$$

With the above value of u, equation (1) can be solved to yield the mean slipstream inclination α_s, relative to the freestream.

To obtain the detailed velocity distribution within the inclined slipstream it is assumed that, to a good approximation, this can be obtained directly from the solution for an isolated propeller operating in axial flow at speed

$$V_a = V_0 \cos \alpha_p \tag{5}$$

The calculation of non-uniform slipstream velocity distributions behind a propeller of arbitrary geometry is based upon established blade element-momentum theory as presented in Reference 20. While the solution to the general theory is very complex, a relatively simple and practical solution is obtained on the assumptions that the rotational energy in the slipstream is small compared to the axial energy and that the radial variation of static pressure in the slipstream can be neglected.

Standard blade element-momentum theory assumes that the flow is both incompressible and inviscid. Thus the flow in annular stream tube elements is treated in an independent manner. For any annular stream tube element, the slipstream velocities are related to both the induced velocities at the propeller disk and the radius in the fully contracted slipstream.

Following the analysis of Reference 20, the induced axial and rotational velocity components, $u, 1/2\,\omega r$, at any radius, r, in the propeller disk can be obtained by an iterative solution to the equations

$$\frac{\frac{\omega}{2}}{\left(\Omega-\frac{\omega}{2}\right)} = \frac{\sigma}{4}\left(\frac{C_l}{\cos\phi} + \frac{C_d}{\sin\phi}\right) \tag{6}$$

and

$$\frac{u}{\left(\Omega-\frac{\omega}{2}\right)r} = \frac{\sigma}{4}\left(\frac{C_l}{\sin\phi} - \frac{C_d}{\cos\phi}\right) \tag{7}$$

where, from Figure 2, the inflow angle, ϕ, is given by

$$\phi = \tan^{-1}\left[\frac{V_a + u}{\left(\Omega-\frac{\omega}{2}\right)r}\right] \tag{8}$$

and the blade section lift and drag coefficients are known in terms of angle of attack, α, given by

$$\alpha = \beta - \phi \tag{9}$$

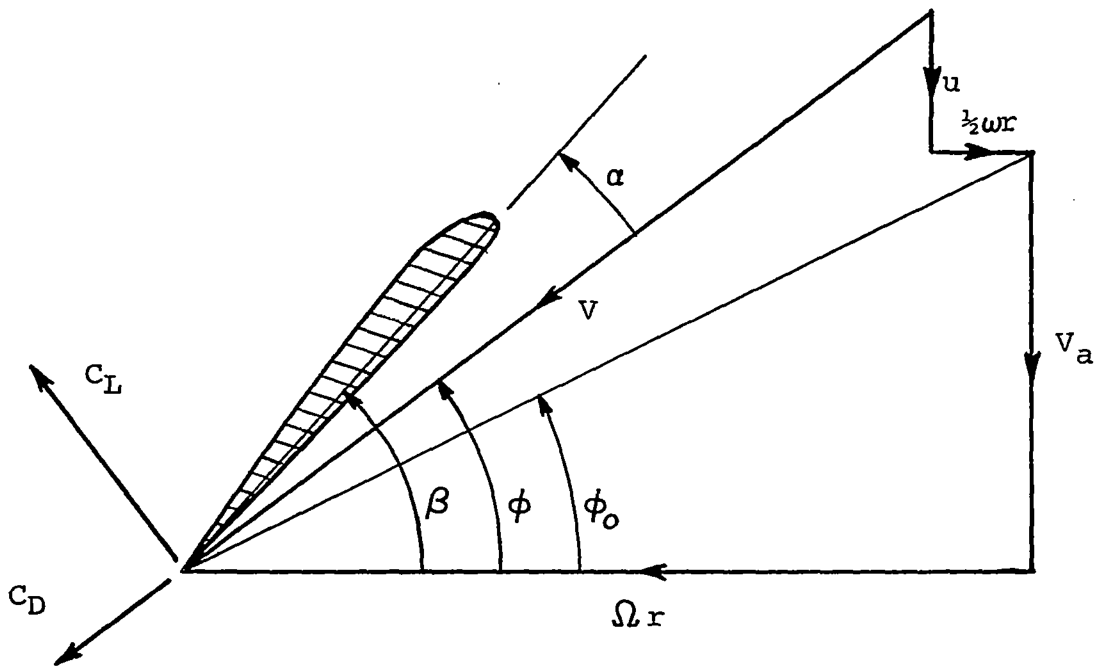


Figure 2. Blade Element Velocity Diagram

To account for the significant and well known loss of lift toward the blade tip equations (6) and (7) are rewritten in the form

$$\frac{\frac{\omega^*}{2}}{\left(\Omega-\frac{\omega}{2}\right)} = \frac{\sigma}{4F}\left(\frac{C_\ell}{\cos\phi} + \frac{C_d}{\sin\phi}\right) \tag{10}$$

and

$$\frac{u^*}{\left(\Omega-\frac{\omega}{2}\right)r} = \frac{\sigma}{4F}\left(\frac{C_\ell}{\sin\phi} - \frac{C_d}{\cos\phi}\right) \tag{11}$$

where F is the tip-loss correction factor given by Lock in Reference 21 and u^*, $1/2(\omega^* r)$ are modified values of the induced velocity components.

Equations (10) and (11) yield improved values of the inflow angle ϕ and section characteristics C_ℓ and C_d, as affected by the tip-loss correction factor F. These values are then used to obtain a better approximation for slipstream-induced velocity components u and $\omega r/2$, using equations (6) and (7).

3.1.2 Initial Calculation of Inflow Angle

The iterative solution for the system of equations (8) through (11) requires that an initial approximate value of ϕ be obtained. Equation (4) can be used if the propeller total thrust is known. However, since the propeller thrust is generally not known in advance, a method that yields a satisfactory starting value for ϕ is developed as follows:

From Figure 2 the inflow angle ϕ can be expressed as

$$\phi = \phi_o + \frac{u^*}{\Omega r} \tag{12}$$

where the resultant induced velocity increment is assumed to be normal to the local blade velocity.

On making the assumption that $C_d \leq C_\ell$ and $\frac{\omega^*}{2} \leq \Omega$ equation (11) can be written as

$$\frac{u^*}{\Omega r}\left(\frac{V_a+u^*}{\Omega r}\right)=\frac{\sigma V C_\ell}{4F\Omega r} \quad (13)$$

By solving equation (13) for $u^*/\Omega r$ and substituting this value in equation (12) an initial value for ϕ is obtained. However, the right hand side of equation (13) must first be reduced to a tractable form. This is accomplished by applying the following relationships:

(a) A linearized expression for the blade section lift curve, given by

$$C_\ell = a_o\left(\alpha-\alpha_o\right) \quad (14)$$

where a_o is a representative lift slope
α is given by equation (9), and
α_o is the angle of attack at zero lift.

(b) An expression for V, obtained from Figure 2 as,

$$V=\sqrt{V_a{}^2+(\Omega r)^2} \quad (15)$$

(c) Prandtl's expression for the tip loss factor F_p obtained from Reference 21 as

$$F_p = \frac{2}{\pi}\cos^{-1}\left[\exp\left\{-\frac{B}{2}\left(1-\frac{r}{R}\right)\sqrt{1+\left(\frac{\Omega R}{V_a}\right)^2}\right\}\right] \quad (16)$$

where B is the number of blades

Combining equations (12) through (15) and substituting for the tip loss factor, F_p, given by equation (16) leads to the following expression:

$$\frac{u^*}{\Omega r}\left(\frac{V_a+u^*}{\Omega r}\right)=\frac{\sigma a_o}{4F_p}\sqrt{1+\left(\frac{V_a}{\Omega r}\right)^2}\cdot\left(\beta-\phi_o-\alpha_o-\frac{u^*}{\Omega r}\right) \quad (17)$$

from which the solution for $u^*/\Omega r$ is obtained as

$$\frac{u^*}{\Omega r} = 1/2 \left[\sqrt{\left(\frac{V_a}{\Omega r} + X\right)^2 + 4X\left(\beta - \phi_o - \alpha_o\right)} - \left(\frac{V_a}{\Omega r} + X\right) \right] \tag{18}$$

where

$$X = \frac{\sigma a_0}{4 F_p} \sqrt{1 + \left(\frac{V_a}{\Omega r}\right)^2}$$

3.1.3 Convergence of the Iterative Propeller Solution

The iterative solution to equations (8) through (11) is naturally divergent within the normal range of the blade section lift curves. Therefore, convergence of the solution must be forced by applying a correction to each new computed value of ϕ . A correction procedure which yields rapid convergence is derived by the method presented below

Let the exact solution for inflow angle, ϕ , be expressed as

$$\phi = \phi' + \delta_1 \tag{19}$$

where ϕ' is the value used as input to the n^{th} iteration and δ_1 is a small unknown increment.

In the general iteration procedure, ϕ' is first used in equation (9) to obtain a value of α from which C_ℓ and C_d may be determined knowing the blade airfoil section characteristics. Next, equations (10) and (11) are used to solve for u^* and ω^* and these values are then substituted in equation (8) to obtain a new value of inflow angle, denoted by ϕ'' . It is this new value of inflow angle which must be corrected before proceeding to the $\overline{n+1}^{th}$ iteration.

Therefore, let the exact solution for inflow angle, ϕ , also be expressed as

$$\phi = \phi'' - \delta_2 \tag{20}$$

where δ_2 is a second small unknown increment.

Combining equations (19) and (20), to eliminate ϕ, yields

$$\delta_1 + \delta_2 = \phi'' - \phi' \tag{21}$$

Substituting equation (21) into equation (19), there follows:

$$\phi = \phi' + (\phi'' - \phi') \cdot \frac{1}{1 + (\delta_2/\delta_1)} \tag{22}$$

Equation (22) forms the basis of a method for calculating an improved value of inflow angle for input to the next iteration cycle by using the guessed and calculated values from the previous cycle. The ratio, δ_2/δ_1, remains to be determined from an approximate error analysis in the following manner:

From equation (20) the value of $\tan\phi$ is expressed to first order in δ_2 by

$$\tan\phi = \tan\phi'' - \delta_2 \sec^2\phi'' \tag{23}$$

From equations (9) and (19) the exact solution for blade lift coefficient, C_ℓ, is expressed in terms of the value C_ℓ' calculated in the n^{th} iteration cycle from

$$C_\ell = C_\ell' - a_0\,\delta_1 \tag{24}$$

where a_0 is a mean value of lift-curve slope.

Equation (10) written in terms of values for the exact solution but with the assumption that $C_d \leq C_\ell$, reduces to

$$\frac{\frac{\omega^*}{2}}{(\Omega - \frac{\omega^*}{2})} = \frac{\sigma\, C_\ell}{4F \cos\phi} = k_x \tag{25}$$

Substituting equations (19) and (24) into equation (25), and retaining only first order terms in δ_1, there follows:

$$k_x = k_x^1 \left[1 - \delta_1 \left(\frac{a_0}{C_\ell} - \tan\phi^1 \right) \right] \tag{26}$$

where

$$k_x^1 = \frac{\sigma\, C_\ell^1}{4F \cos\phi^1}$$

and where small changes in the tip loss factor are neglected.

Similarly, equation (11) reduces to

$$\frac{u^*}{(\Omega - \frac{\omega^*}{2})r} = \frac{\sigma\, C_\ell}{4F \sin\phi} = k_y \tag{27}$$

and, using equations (19) and (24), results in the expression

$$k_y = k_y^1 \left[1 - \delta_1 \left(\frac{a_0}{C_\ell} + \cot\phi \right) \right]$$

where (28)

$$k_y^1 = \frac{\sigma\, C_\ell^1}{4F \sin\phi^1}$$

Now, using equations (25) and (27), equation (8) can be expressed as

$$\tan\phi = \mu \left(1 + k_x\right) + k_y \tag{29}$$

where μ is the local forward speed ratio $(Va/\Omega r)$.

Rewriting equation (29) in terms of the values k_x', k_y' calculated in the n^{th} iteration cycle yields

$$\tan \phi'' = \mu \left(1 + k_x'\right) + k_y' \tag{30}$$

Combining equations (29) and (30) leads to the following relationship

$$\tan \phi = \tan \phi'' - \mu k_x' \left(1 - \frac{k_x}{k_x'}\right) - k_y' \left(1 - \frac{k_y}{k_y'}\right) \tag{31}$$

Finally using equations (23),(26),(28) and (31) a solution for the ratio δ_2/δ_1 is obtained as follows:

$$\frac{\delta_2}{\delta_1} = \cos^2 \phi'' \left[\mu k_x' \left(\frac{a_0}{C_\ell} - \tan \phi' \right) + k_y' \left(\frac{a_0}{C_\ell} + \cot \phi' \right) \right] \tag{32}$$

Equation (32) thus provides the essential relationship by which equation (22) is applied to obtain an improved value of inflow angle for input to the next iteration cycle. In practice, the iteration procedure is terminated when the difference $(\phi'' - \phi')$ for each successive iteration cycle has converged to within a prescribed margin of error.

3.1.4 Analysis for Slipstream Velocity Distributions

Upon reaching a converged solution for the inflow angle ϕ, the final values of ϕ, C_ℓ and C_d are then substituted in equation (6) and (7) to solve for the true induced velocity components in the propeller disk plane, u and $1/2\,\omega r$.

The local axial velocity component V_{sa} in the fully developed slipstream is obtained from Figure 1 as

$$V_{sa} = \frac{V_0 \cos a_p + 2u}{\cos\left(a_p + a_s\right)} \tag{33}$$

The local rotational velocity component, V_{st}, in the fully developed slipstream is obtained from conservation of angular momentum and is given by

$$V_{st} = \omega r \left(\frac{r}{r_s}\right) \tag{34}$$

where r_s is the local radius in the slipstream for the streamtube element which has a local radius r in the propeller disk plane.

The local radius r_s for each flow element in the slipstream is derived from a simplified application of the continuity expression to successive streamtube elements. For the n^{th} blade element station at radius r_n the corresponding radius r_{s_n} in the slipstream is given by

$$r_{s_n}^{\,2} = r_{s_1}^{\,2} + 1/2 \sum_{m=2}^{m=n} \left(r_m^2 - r_{\overline{m-1}}^2 \right) \cdot \left(\frac{2V_a + u_m + u_{\overline{m-1}}}{V_a + u_m + u_{\overline{m-1}}} \right) \tag{35}$$

where, in the notation of Figure 3, r_1 and r_{s_1} are the values of hub and nacelle radius, respectively.

The value of r_{s_n} given by equation (35) is based upon representing the slipstream by a series of concentric annular streamtubes with uniform velocity between each element station. This solution, while approximate, is found to be more than adequate for all reasonable variations between $u_{\overline{n-1}}$ and u_n .

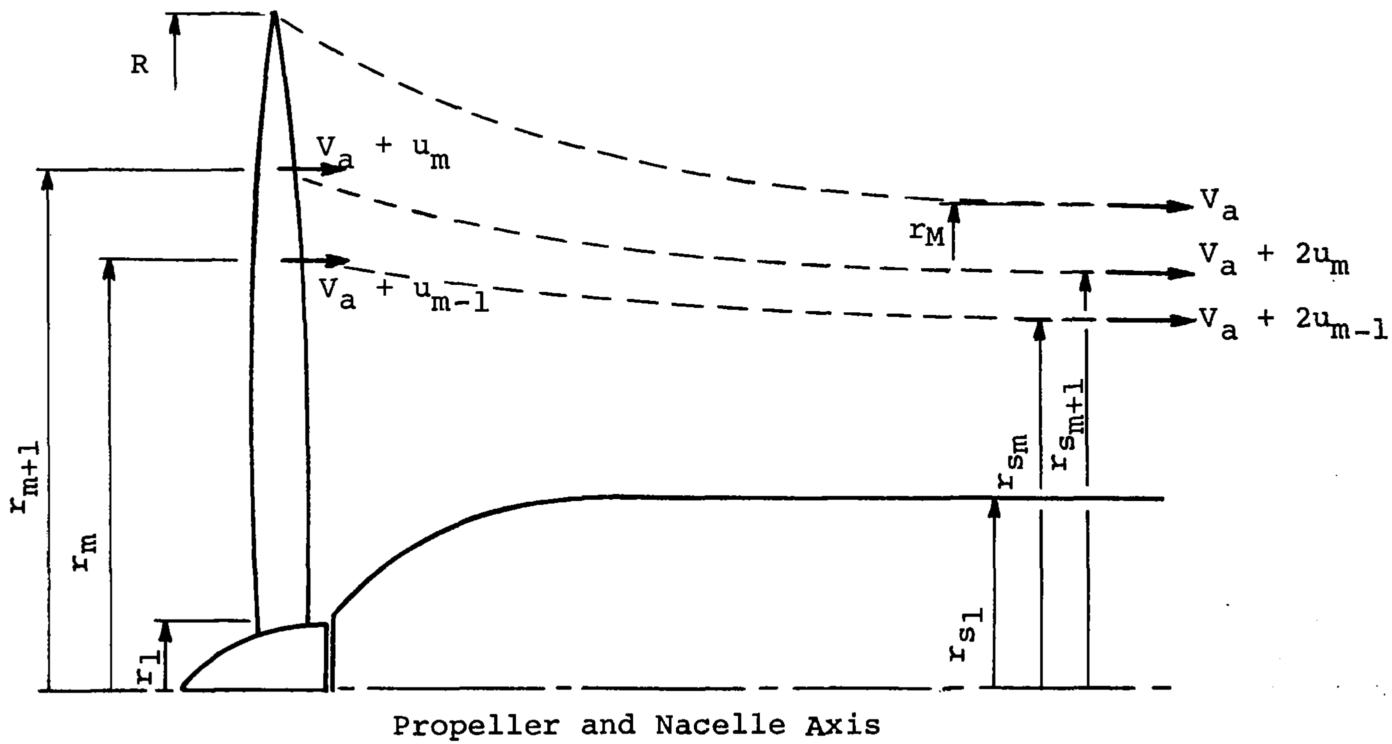


Figure 3. Analytical Model For Slipstream Contraction.

3.2 WING-IN-SLIPSTREAM ANALYSIS

In the second part of the analysis a modified form of lifting line theory is presented which uses the nonuniform slipstream velocity distribution, as determined above, to calculate the lift distribution on wings with propellers. The approach presented relies on the use of a simple physical model to obtain a solution for wing-in-slipstream loadings for a wide range of real aircraft propeller-wing combinations.

First the method is presented for an unflapped wing immersed in one or more non-overlapping slipstreams. Following this, the modifications required to include flaps are described. Finally, an extension of the analysis to include low aspect-ratio propeller-wing combinations is discussed.

3.2.1 Analysis for a Wing with no Flaps or with Full-Span Deflected Flaps

Consider the basic case of a wing in a uniform stream of velocity, V_0. If the local circulation is Γ_0, then the span load distribution at any spanwise station is given by

$$\ell_0 = \rho V_0 \Gamma_0 \tag{36}$$

The superposition of a propeller slipstream flow gives rise to an increased local velocity V_s' and an increased circulation Γ_s', which can be expressed, respectively, as

$$V_s' = V_0 + \Delta V \tag{37}$$

and

$$\Gamma_s' = \Gamma_0 + \Delta\Gamma \tag{38}$$

where ΔV and $\Delta\Gamma$ are the incremental changes in local velocity and circulation, respectively, due to propeller slipstream. Now the corresponding spanwise load distribution for the basic wing immersed in the propeller slipstream can be written as

$$\ell = \rho V_s' \Gamma_s' \tag{39}$$

Substituting equations (37) and (38) into equation (39) yields a general expression for the spanwise load distribution of a wing immersed in the propeller slipstream, as follows:

$$
\begin{aligned}
\ell &= \rho\ (V_0 + \Delta V)\ (\Gamma_0 + \Delta\Gamma) \\
&= \rho\ (V_0 + \Delta V)\ \Gamma_0 + \rho\ (V_0 + \Delta V)\ \Delta\Gamma \\
&= \rho\ V_s'\ \Gamma_0 + \rho\ V_s'\ \Delta\Gamma \\
&= \rho\ V_s'\ \Gamma_0 + \rho\ V_s'\ \Gamma_2 \\
&= \ell_1 + \ell_2
\end{aligned}
\tag{40}
$$

where $\Gamma_2 \equiv \Delta\Gamma$ is the change in wing circulation due to propeller slipstream alone.

It can be noted from equation (40) that the first component ℓ_1 of the total lift distribution is that which would be obtained if the local velocity increased while the circulation Γ_0 remained unchanged. If the circulation is unchanged, then there is no change in the trailing vorticity and therefore no change in the wing downwash field. The second term ℓ_2 represents the change in spanwise lift distribution due to the circulation Γ_2 and is therefore associated with wing downwash changes caused by the propeller slipstream.

The problem of a wing immersed in the propeller slipstream is now reduced to proper determination of local values for the resultant velocity V_s' and the circulation Γ_2 for the entire wing. This analysis is developed below.

For typical propeller/wing configurations, the resultant local velocity V_s' can be equated (within the small angle assumption) to the combined freestream and slipstream component along the wing section zero-lift line, thus

$$V_s \doteq V_s' \tag{41}$$

Using the nomenclature of Figure 4, this velocity component can be expressed as

$$V_s = V_{s_a} \cos\left(\alpha_s + \alpha_e\right) - V_{s_t} \sin\left(\alpha_s + \alpha_e\right) \tag{42}$$

Also, the corresponding component of the total flow normal to the wing section zero-lift line is given by

$$V_n = V_{s_a} \sin\left(\alpha_s + \alpha_e\right) + V_{s_t} \cos\left(\alpha_s + \alpha_e\right) \tag{43}$$

The quantitites V_{s_a} and V_{s_t} in the above equations represent the axial and swirl velocity components of the combined freestream and slipstream flow and are given by equations (33) and (34) respectively. Also, the angles α_s and α_e are known quantities which represent inclinations of the slipstream and the zero-lift line relative to the remote freestream velocity, respectively.

The extra wing circulation Γ_2, caused by the action of the propeller slipstream, is determined by equating the resulting change in wing upwash to the downwash change associated with Γ_2. This upwash change, in non-dimensional form, is defined as

$$v = \frac{V_n}{V_0} - \sin\alpha_e \tag{44}$$

Substituting equation (43) into equation (44) yields the extra upwash due to the slipstream as

$$v = \frac{V_{s_a}}{V_0} \sin\left(\alpha_s + \alpha_e\right) + \frac{V_{s_t}}{V_0} \cos\left(\alpha_s + \alpha_e\right) - \sin\alpha_e \tag{45}$$

In order to satisfy the wing boundary condition of no flow through the surface, this extra upwash or crossflow must be balanced by the combined influence of the extra bound vorticity, Γ_2, and the associated streamwise (i.e. chordwise and trailing) vorticity, $-\frac{d\Gamma_2}{dy}\,dy$.

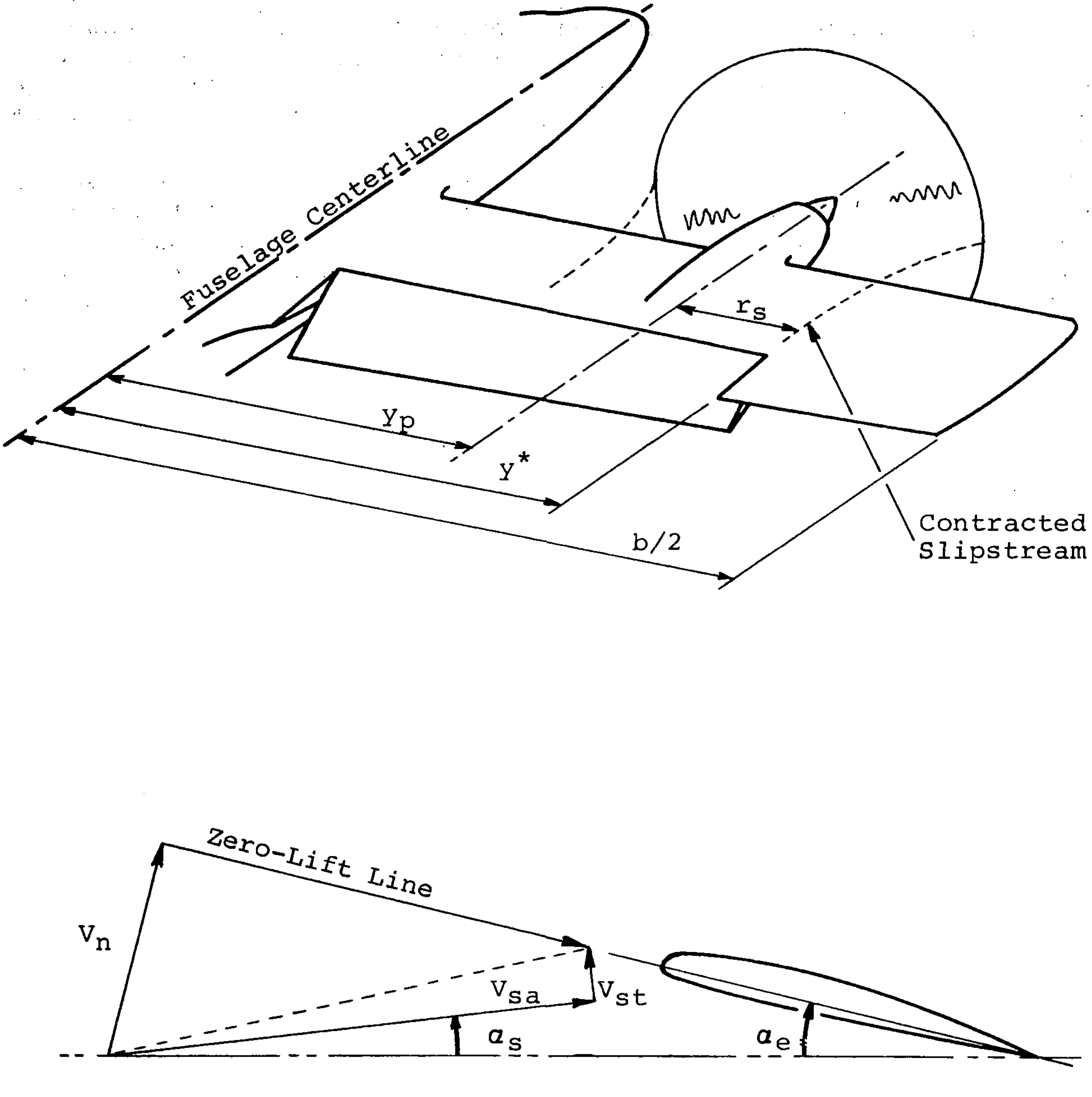


Figure 4. Notation for Wing-in-Slipstream Model

Observation of the lift on wings with slipstreams shows that the major portion of the extra loading caused by the slipstream is concentrated on and near that portion actually immersed in the slipstream and, for most configurations, the "aspect ratio" of this immersed portion is small, usually about 1.0. It is herein postulated that the distribution of extra vorticity caused by the slipstream exhibits the characteristics of that found on low aspect-ratio wings. Kuchemann, in Reference 22, shows that for low aspect-ratio wings only the chordwise and trailing vortices are required to fulfill the boundary conditions. In fact, as the aspect ratio tends to zero, the trailing vortices cancel half the upwash, and the chordwise vortices cancel the remainder.

In the present case, the net extra upwash is $V_0 v$ and the downwash change due to the trailing vortices associated with the extra circulation, Γ_2, is given by

$$W_{i_2} = \frac{1}{4\pi} \int_{-b/2}^{b/2} \frac{-\frac{d\Gamma_2}{dy_1} \cdot dy_1}{y_1 - y} \tag{46}$$

Therefore, from the above considerations it follows that

$$W_{i_2} = 1/2 \; V_0 v \tag{47}$$

and hence

$$v = \frac{1}{2\pi V_0} \int_{-b/2}^{b/2} \frac{-\frac{d\Gamma_2}{dy_1} \cdot dy_1}{y_1 - y} \tag{48}$$

Equation (48) is supported by the analysis of Reference 23, which deals with the determination of the lift on a wing passing close to a line vortex. This reference states that lifting line theory always overestimates the lift induced by rapidly changing upwash fields (such as from propellers and line vortices) by a factor of 2, due to a corresponding underestimation of the wing downwash.

Now, expressing the circulation distribution, Γ_2, as a Fourier sine series in terms of the wing spanwise angular coordinate, $\theta = \cos^{-1}\left(\frac{2y}{b}\right)$, yields

$$\Gamma_2 = 1/2 \; b \; V_o \sum_{n=1}^{\infty} B_n \sin n\theta \tag{49}$$

Combining equations (48) with (49) and performing the required mathematical operations, the Fourier coefficients B_n can be expressed as

$$B_n = \frac{4}{n\pi} \int_0^{\pi} v \sin\theta \sin n\theta \; d\theta \tag{50}$$

By limiting the series for Γ_2 to $r-1$ terms, equation (49) becomes

$$\Gamma_{2m} = 1/2 \; b \; V_o \sum_{n=1}^{r-1} Bn \sin n \frac{m\pi}{r} \tag{51}$$

where $m = 1, 2, \cdots \overline{r-1}$

and equation (50) is reduced to the summation

$$B_n = \frac{4}{nr} \sum_{m=1}^{r-1} v_m \sin \frac{m\pi}{r} \sin n \frac{m\pi}{r} \tag{52}$$

From equation (40) the lift $\mathcal{L}_2$ associated with the slipstream may be expressed as

$$\mathcal{L}_2 = \rho \; V_s \; \Gamma_2 = 1/2 \, \rho \, V_o^2 \, C_{\mathcal{L}_2} c \tag{53}$$

where the lift coefficient is based on V_o . Therefore

$$\Gamma_{2\,m} = 1/2 \; b.V_o \left(\frac{C_{\mathcal{L}_2} C}{b} \; \frac{V_o}{V_s} \right)_m \tag{54}$$

Comparing equations (51 and (54) there results

$$\left(\frac{C_{\ell_2} c}{b}\right)_k = \left(\frac{V_s}{V_o}\right)_k \sum_{n=1}^{r-1} B_n \cdot \sin n \frac{k\pi}{r} \tag{55}$$

If the relationship for the coefficients, B_n is substituted into equation (55), the lift distribution associated with the slipstream is obtained in the form

$$\left(\frac{C_{\ell_2} c}{b}\right)_k = \left(\frac{V_s}{V_o}\right)_k \cdot \frac{4}{r} \sum_{n=1}^{r-1} \frac{\sin n \frac{k\pi}{r}}{n} \sum_{m=1}^{r-1} V_m \cdot \sin \frac{m\pi}{r} \cdot \sin n \frac{m\pi}{r} \tag{56}$$

Having determined the lift associated with the slipstream upwash the overall wing lift is calculated as follows.

Let α_{i_1} and α_{i_2} be the induced angles of attached associated with the lift distributions ℓ_1 and ℓ_2 respectively, as given by equation (40). Then the total induced angle of attack at any point k is given by

$$\alpha_{i_k} = \alpha_{i_{1k}} + \alpha_{i_{2k}} \tag{57}$$

Also, re-expressing equation (40) in terms of the lift coefficients based on V_o yields

$$\left(\frac{C_\ell c}{b}\right)_k = \left(\frac{C_{\ell_1} c}{b}\right)_k + \left(\frac{C_{\ell_2} c}{b}\right)_k \tag{58}$$

Using the multipliers β_{mk} from Reference 1 in equation (58) there follows

$$\sum_{m=1}^{r-1} \left(\frac{C_\ell c}{b}\right)_m \beta_{mk} = \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1} c}{b}\right)_m \cdot \beta_{mk} + \sum_{m=1}^{r-1} \left(\frac{C_{\ell_2} c}{b}\right) \beta_{mk} \tag{59}$$

Now, by definition

$$\alpha_{i_{1k}} = \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1} c}{b}\right)_m \beta_{mk} \tag{60}$$

Adding $\alpha_{i_{2k}}$ to both sides of equation (59) and using equations (57) and (60) leads to

$$\alpha_{i_k} = \sum_{m=1}^{r-1} \left(\frac{C_\ell c}{b}\right)_m \beta mk + \alpha_{i_{2k}} - \sum_{m=1}^{r-1} \left(\frac{C_{\ell_2} c}{b}\right)_m \beta mk \quad (61)$$

Equation (61) gives the total induced angle of attack in terms of the unknown wing lift distribution, $C_\ell c/b$, the known induced angle of attack, $\alpha_{i_2} = V_0 v/V_s$, and the known slipstream lift distribution, $C_{\ell_2}\ c/b$, given by equation (56). The solution is obtained by iteration as follows.

An approximation to the overall lift distribution is calculated and equation (61) is used to obtain a first approximation to the induced angle of attack. The effective angle of attack at the wing section is obtained from

$$\alpha_{e_k} = \left[\alpha_{g_k} + \frac{V_0\, v_k}{V_{s_k}} - \alpha_{i_k} - (1-E)\ \alpha_{\ell ok}\right] \frac{1}{E} \quad (62)$$

Where α_{gk} is the section geometric angle of attack, $\alpha_{\ell ok}$, is the section zero-lift angle and E is the edge-velocity factor of Reference (1). This value of effective angle of attack is then used with the two-dimensional section lift curves at the effective section Reynolds number R_{ek} to obtain the lift coefficient C_ℓ . The value of $C_\ell c/b$ thus calculated is compared to the initial approximation and, if sufficient agreement is not obtained, a new value is computed using the method given in Reference 1. This iteration process is then repeated until guessed and calculated values agree to within prescribed tolerance.

It should be noted that the above analysis is also applicable to a wing with full-span deflected flaps, provided that appropriate airfoil characteristics are employed for wing sections with flaps.

3.2.2 Analysis for a Wing with Part-Span Deflected Flaps

The deflection of a part-span flap causes a discontinuity δ in the distribution of absolute angle of attack at the end of the flap, and produces a corresponding discontinuity in the slipstream-induced crossflow. The effect of these discontinuities on the span load distribution is treated below.

The analysis is developed for a wing having a deflected part-span flap δ_f extending from $y = -b/2$ to $y = y^*$. The most general case is that of a flap whose end lies within the slipstream, as illustrated in Figure 5.

Following the preceding treatment of a wing with no flaps or with full-span deflected flaps, the total wing lift distribution given by equation (40) can be divided into two portions and can be expressed in non-dimensional form as

$$\left(\frac{C_{\ell} c}{b}\right) = \left(\frac{C_{\ell_1} c}{b}\right) + \left(\frac{C_{\ell_2} c}{b}\right) \tag{63}$$

where $C_{\ell_2} \cdot c/b$ is the lift distribution associated with slipstream-induced upwash and $C_{\ell_1} \cdot c/b$ is the remainder of the distribution.

In the present case, however, the slipstream-induced upwash $V_0 v$, given by equation (48), is discontinuous at the end of the flap as shown in Figure 5. The net discontinuity in crossflow at the edge of the flap, $y = y^*$, can be obtained from equation (45) by considering the upwash on both sides of the flap end. Thus, using equation (45) for the flapped side of the wing, at $y = y^* - o$, this extra upwash can be expressed as follows

$$V_0 \cdot v_{(y^*-o)} = V_{s\alpha}^* \sin\left(\alpha_s^* + \alpha_e^* + \delta\right) + V_{st}^* \cdot \cos\left(\alpha_s^* + \alpha_e^* + \delta\right) - V_0 \sin\left(\alpha_e^* + \delta\right) \tag{64}$$

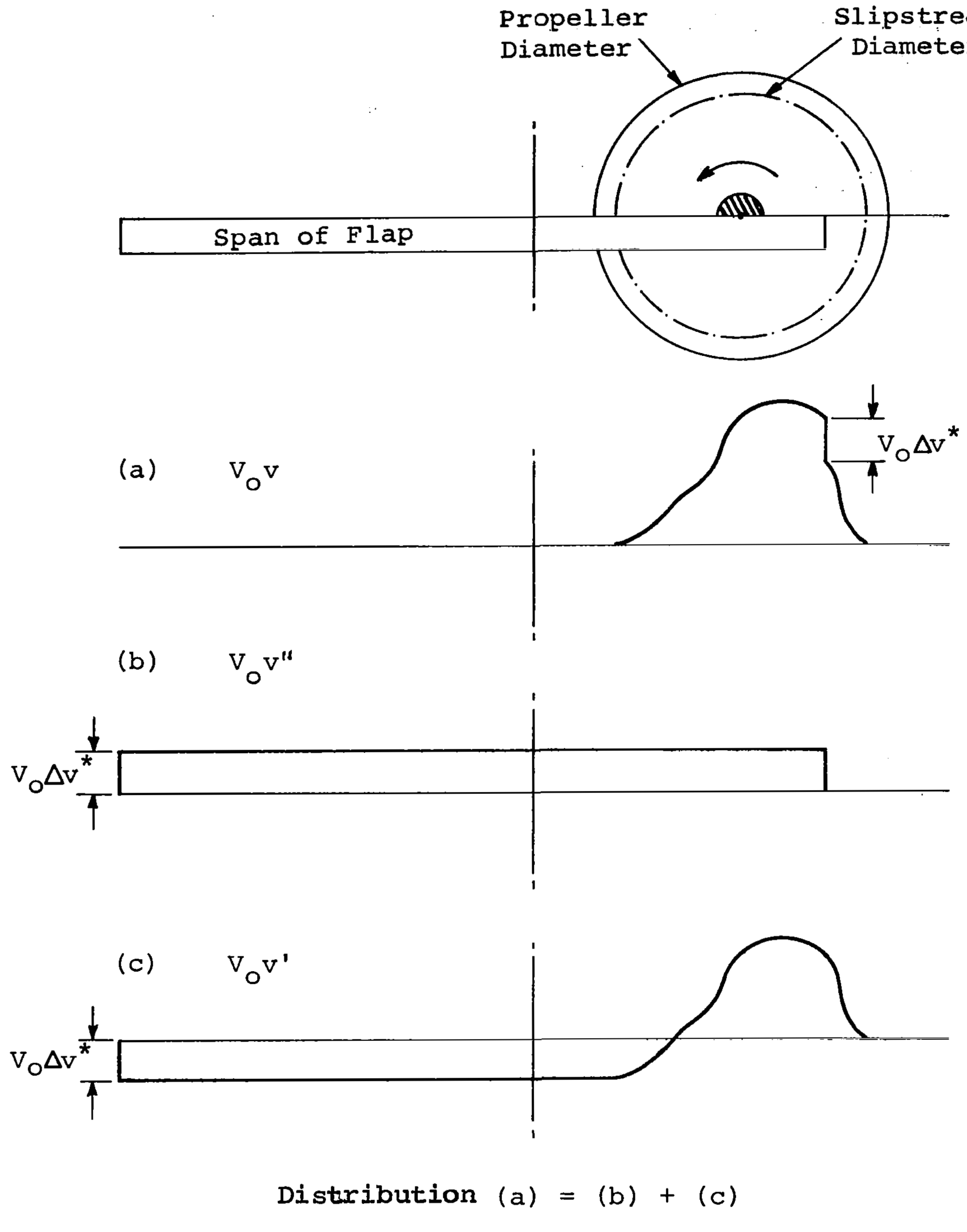


Figure 5. Mathematical Representation of Flap Discontinuity

Similarly, for the unflapped side at $y = y^* + o$, the crossflow is given by

$$V_o\, v_{(y^*+o)} = V_{sa}^* \sin\left(\alpha_s^* + \alpha_e^*\right) + V_{st}^* \cos\left(\alpha_s^* + \alpha_e^*\right) - V_o \sin\left(\alpha_e^*\right) \tag{65}$$

The net discontinuity in crossflow is obtained as the difference between equations (64) and (65), thus

$$V_o\, v_{(y^*-o)} - V_o\, v_{(y^*+o)} = V_o\, \Delta v^* \tag{66}$$

Because of the discontinuity in crossflow, $V_o\, \Delta v^*$, given by equation (66), the solution for the lift distribution C_{ℓ_2} can not be obtained from a simple Fourier series for Γ_2, as was possible in equation (48). Therefore, the distribution $V_o\, v$ is split into two portions, one a continuous distribution $V_o . v'$ and the other a step function distribution $V_o . v''$, where

$$V_o v = V_o\, v' + V_o\, v''$$

and

$$\begin{aligned} V_o v'' &= V_o\, \Delta v^* \qquad \text{for} \quad -b/2 \le y < y^* \\ &= 0 \qquad\qquad \text{for} \quad y^* \le y \le b/2 \end{aligned} \tag{67}$$

Now, it is necessary to relate the velocity distributions given by equation (67) to their corresponding circulation distributions. Since Γ_2 is the total circulation corresponding to $V_o\, v$, as given by equation (48), it can also be split into two distributions, i.e., Γ_2', corresponding to $V_o\, v'$ and Γ_2'' corresponding to $V_o\, v''$, where

$$\Gamma_2 = \Gamma_2' + \Gamma_2'' \tag{68}$$

Thus, using equations (48) and (68), the velocity distributions given by equation (67) can now be re-expressed in terms of the corresponding circulation distributions Γ_2' and Γ_2'' as follows:

$$V_0 v = V_0 v' + V_0 v''$$

$$= \frac{1}{2\pi} \int_{-b/2}^{b/2} \frac{-\frac{d\Gamma_2'}{dy_1} \cdot dy_1}{y_1 - y} + \frac{1}{2\pi} \int_{-b/2}^{b/2} \frac{-\frac{d\Gamma_2''}{dy_1} \cdot dy_1}{y_1 - y} \tag{69}$$

where

$$V_0 v' = \frac{1}{2\pi} \int_{-b/2}^{b/2} \frac{-\frac{d\Gamma_2'}{dy_1} \cdot dy_1}{y_1 - y} \tag{70}$$

and

$$V_0 v'' = \frac{1}{2\pi} \int_{-b/2}^{b/2} \frac{-\frac{d\Gamma_2''}{dy_1} \cdot dy_1}{y_1 - y} \tag{71}$$

Now the problem reduces to determining Γ_2' and Γ_2'' and the corresponding lift distribution $C_{\ell_2} \cdot c/b$. This is accomplished as outlined below.

Since $Vo.v'$ is continuous then, following the analysis of subsection 3.2.1, Γ_2' can be expressed as the simple Fourier series

$$\Gamma_2' = \frac{1}{2} b \, Vo \sum_{n=1}^{r-1} B_n \sin n\theta \tag{72}$$

where the coefficients B_n are given by:

$$B_n = \frac{4}{n.r} \sum_{m=1}^{r-1} \left(v - v'' \right)_m \sin \frac{m\pi}{r} \sin n \frac{m\pi}{r} \tag{73}$$

The relationship for Γ_2'' is obtained from the analysis of Reference 24 in the form

$$\frac{2\,\Gamma_2''}{b\,Vo} = \frac{\Delta v^*}{\pi} \left[2\left(\pi - \theta^*\right) \sin\theta - \left(\cos\theta - \cos\theta^*\right) \log\left\{ \frac{1 - \cos\left(\theta + \theta^*\right)}{1 - \cos\left(\theta - \theta^*\right)} \right\} \right] \tag{74}$$

The distribution Γ_2'' given by equation (74) satisfies the required discontinuity in the crossflow, $Vo.v'' = Vo.\Delta v^*$ in equation (71).

The circulation distributions Γ_2' and Γ_2'', determined in equations (72) and (74) respectively, can now be used to obtain the corresponding lift distribution $C\ell_2 \; c/b$ associated with the slipstream-induced upwash. This is accomplished by rearranging equation (54) and using equation (68), thus

$$\frac{C\ell_2 \, c}{b} = \left(\frac{Vs}{Vo}\right) \left(\frac{2\,\Gamma_2}{b\,Vo}\right)$$

$$= \left(\frac{Vs}{Vo}\right) \left(\frac{2\,\Gamma_2'}{b\,Vo} + \frac{2\,\Gamma_2''}{b\,Vo}\right) \tag{75}$$

Finally, substituting equations (72), (73) and (74) into equation (75) yields the lift distribution C_{ℓ_2} c/b at any point k on the wing, in the form

$$\left(\frac{C_{\ell_2} c}{b}\right)_k = \left(\frac{V_s}{V_o}\right)_k \left[\frac{4}{r} \sum_{n=1}^{r-1} \frac{\sin n \frac{k\pi}{r}}{n} \sum_{m=1}^{r-1} \left(v_m - v_m''\right) \sin \frac{m\pi}{r} \sin n \frac{m\pi}{r} \right.$$

$$\left. + \frac{\Delta v^*}{\pi} \left\{ 2\left(\pi - \theta^*\right) \sin \frac{k\pi}{r} - \left(\cos \frac{k\pi}{r} - \cos \theta^*\right) \log \frac{1 - \cos\left(\frac{k\pi}{r} + \theta^*\right)}{1 - \cos\left(\frac{k\pi}{r} - \theta^*\right)} \right\}\right] \tag{76}$$

The above analysis gives the solution for the case where the flap extends from the left wing tip to a point $y = y^*$ on the right wing. The solution for a flap extending between $-y^* \leq y \leq y^*$, or any other combination of flap positions, is obtained by superposition of solutions as shown in Figure 6.

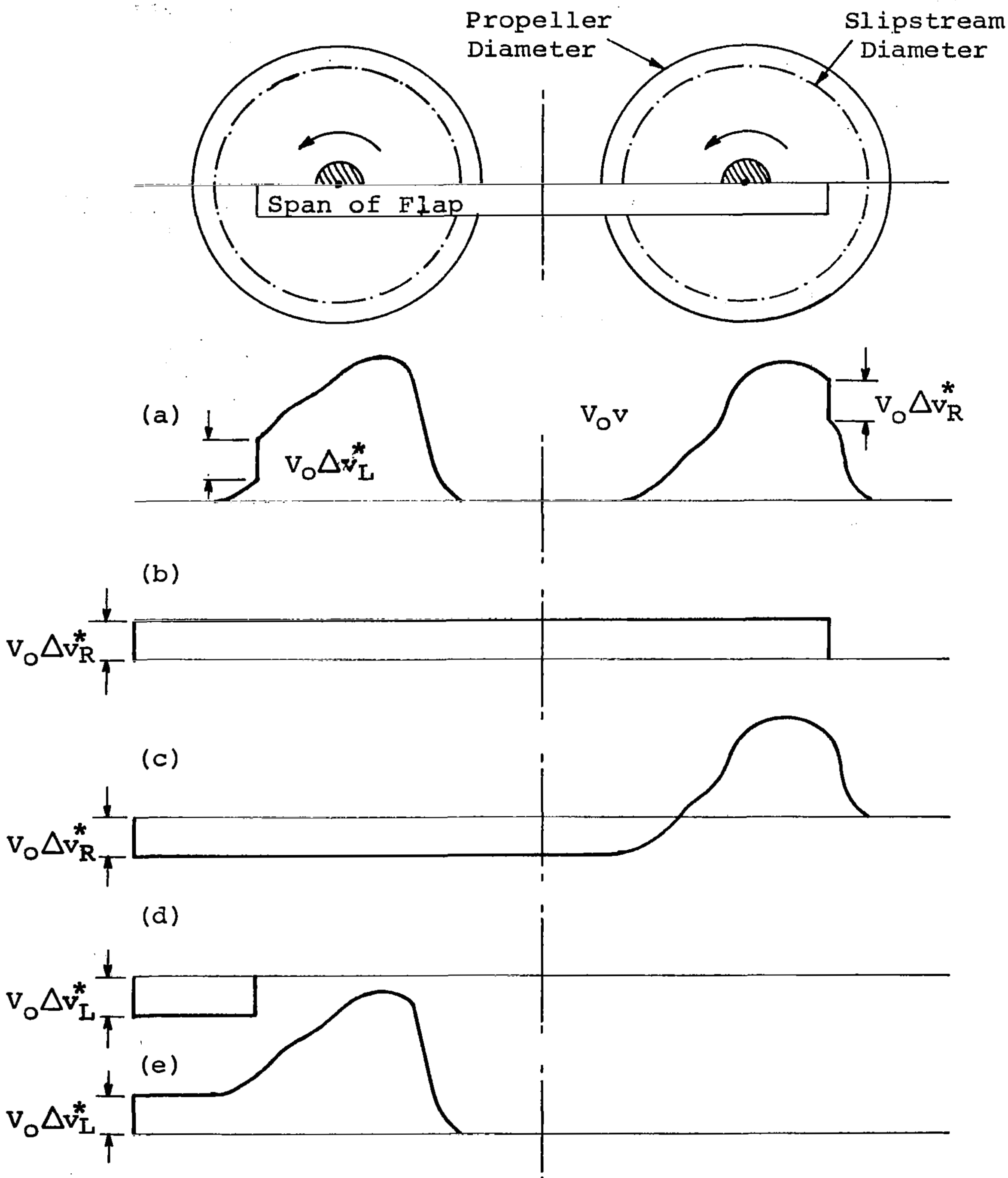
It should be noted that equation (76) represents only one part of the solution for the total lift distribution C_ℓ c/b as given in equation (63). It is now necessary to obtain an appropriate solution for the distribution C_{ℓ_1} c/b . This is accomplished as outlined below.

The discontinuity δ in absolute angle of attack caused by the flap deflection also affects the distribution C_{ℓ_1} c/b. This distribution, although continuous, possesses an infinite derivative at $y = y^*$. Therefore, the multipliers β_{mk} developed in Reference 1 can not be used directly to obtain the induced angle of attack due to this distribution. This restriction is removed by the following analytical approach.

The distribution C_{ℓ_1} c/b can also be divided into two portions, thus

$$\left(\frac{C_{\ell_1} c}{b}\right) = \left(\frac{C_{\ell_1}' c}{b}\right) + \delta \left(\frac{C_{\ell_1}'' c}{b\,\delta}\right) \tag{77}$$

where C_{ℓ_1}'' c/bδ is the lift distribution due to a unit



Distribution (a) = (b) + (c) + (d) + (e)

Figure 6. Method for Superposition of Solutions

discontinuity δ, and $C_{\ell_1}' c/b$, is the remainder.

Applying the multipliers β_{mk} to equation (77) yields the following

$$\sum_{m=1}^{r-1} \left(\frac{C_{\ell_1} c}{b}\right)_m \beta_{mk} = \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1}' c}{b}\right)_m \beta_{mk} + \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1}'' c}{b\,\delta}\right)_m \delta\, \beta_{mk} \quad (78)$$

Also, applying the multiplers β_{mk} to equation (63) yields the total lift distribution $C_\ell c/b$ as

$$\sum_{m=1}^{r-1} \left(\frac{C_\ell c}{b}\right)_m \beta_{mk} = \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1} c}{b}\right)_m \beta_{mk} + \sum_{m=1}^{r-1} \left(\frac{C_{\ell_2} c}{b}\right)_m \beta_{mk} \quad (79)$$

Substituting equation (78) into equation (79), yields

$$\sum_{m=1}^{r-1} \left(\frac{C_\ell c}{b}\right)_m \beta_{mk} = \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1}' c}{b}\right)_m \beta_{mk} + \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1}'' c}{b\,\delta}\right)_m \delta\, \beta_{mk} + \sum_{m=1}^{r-1} \left(\frac{C_{\ell_2} c}{b}\right)_m \beta_{mk} \quad (80)$$

In order to obtain the iterative solution to equation (80), it is now necessary to relate the lift distributions given in equations (63) and (77) to their corresponding induced angle of attack distributions. If α_i is the induced angle of attack distribution corrresponding to the total lift distribution $C_\ell c/b$, and α_{i_1} and α_{i_2} are the induced angle of attack distributions corresponding to the lift components $C_{\ell_1} c/b$ and $C_{\ell_2} c/b$, then from equation (63) there follows

$$\alpha_i = \alpha_{i_1} + \alpha_{i_2} \quad (81)$$

Also, applying similar considerations to equation (77) there results

$$\alpha_{i_1} = \alpha_{i_1}' + \alpha_{i_1}'' \tag{82}$$

Substituting equation (82) into equation (81) yields the relationship for the induced total angle of attack as follows:

$$\alpha_i = \alpha_{i_1}' + \alpha_{i_1}'' + \alpha_{i_2} \tag{83}$$

where $\alpha_{i_1}'' = \delta$ over the flap span
$\quad o$ outside the flap span

It can be noted in equation (83) that the induced angle of attack distribution α_{i_1}' must be continuous, since its corresponding lift distribution $C_{\ell_1}' \, c/b$ as given in equation (77) is continuous. Therefore, this induced angle of attack distribution is obtained directly using the multipliers, thus

$$\alpha_{i_{1_k}}' = \sum_{m=1}^{r-1} \left(\frac{C_{\ell_1} \, c}{b} \right)_m \beta_{mk} \tag{84}$$

The above relationship can be re-expressed in terms of the total induced angle of attack distribution α_i by using equation (83), thus

$$\sum_{m=1}^{r-1} \left(\frac{C_{\ell_1}' c}{b} \right)_m \beta_{mk} = \alpha_{i_k} - \alpha_{i_{1_k}}'' - \alpha_{i_{2_k}} \tag{85}$$

Now, equation (85) can be substituted into equation (80) to eliminate the $C_{\ell_1}' \, c/b$ distribution and to yield the total lift distribution $C_{\ell} \, c/b$ in the desired multiplier form as follows

$$\sum_{m=1}^{r-1}\left(\frac{C_\ell c}{b}\right)_m \beta mk = \alpha_{ik} - \alpha_{i_1k}'' - \alpha_{i2k} + \sum_{m=1}^{r-1}\left(\frac{C_{\ell_1}'' c}{b\,\delta}\right)_m \delta\, \beta mk + \sum_{m=1}^{r-1}\left(\frac{C_{\ell_2} c}{b}\right)_m \beta mk \qquad (86)$$

Finally, rearranging the equation (86), the total induced angle of attack at any point k on the wing can be related to the corresponding total lift distribution and the know distributions of induced angle of attack α_{i_1}'' and α_{i_2} and their corresponding lift distributions $C_{\ell_1}'' c/b$ and $C_{\ell_2}\, c/b$, respectively. The resulting relationship is

$$\alpha_{i_k} = \sum_{m=1}^{r-1}\left(\frac{C_\ell c}{b}\right)_m \beta mk + \delta\left[\frac{\alpha_{i_k}''}{\delta} - \sum_{m=1}^{r-1}\left(\frac{C_{\ell_1}'' c}{b\,\delta}\right)_m \beta mk\right] + \alpha_{i_{2_k}} - \sum_{m=1}^{r-1}\left(\frac{C_{\ell_2} c}{b}\right)_m \beta mk \qquad (87)$$

where from the analysis of Reference 1

$$\left(\frac{C_{\ell_1}'' c}{b\ \delta}\right)_k = \frac{2}{\pi}\left[\left(\cos\theta^* - \cos\theta_k\right)\log\left\{\frac{1-\cos\left(\theta_k+\theta^*\right)}{1-\cos\left(\theta_k-\theta^*\right)}\right\} + 2\left(\pi-\theta^*\right)\sin\theta_k\right] \qquad (88)$$

and $C_{\ell_2} c/b$ has been already determined in equation (76).

Equation (87) is analogous to equation (61) developed in subsection 3.2.1 for no flap deflection. This equation is also solved by an iteration procedure, similar to that used for solving equation (61). Thus, upon obtaining the required convergence of the iterative solution, equation (87) yields the total lift distribution $C_\ell\, c/b$ for a wing with a deflected flap within the propeller slipstream.

3.2.3 Extension of the Wing Analysis to Small Aspect Ratios

To provide added flexibility to the methodology developed herein, the wing analysis treated in Sections 3.2.1 and 3.2.2 is extended to include wings of small aspect ratio. This analysis is particularly useful for the current application, since much of the available test data on spanwise loadings for wings in slipstream falls within the low aspect ratio range. The correlations of this extended analysis with the corresponding test data where appropriate is shown in Section 5.0.

The modification of the present analysis to small aspect ratio wings is based on the wing theory of Kuchemann (Reference 22), as outlined below.

In equations (61) and (87) a set of multipliers was used to obtain the induced angle of attack distributions for a wing with no flaps and with part-span deflected flaps, respectively. These multipliers were obtained from the fundamental equation of the high-aspect-ratio, lifting-line theory which expresses the induced angle of attack in terms of the span loading,

$$\alpha_i = \frac{b}{8\pi} \int_{-b/2}^{b/2} \frac{\frac{d\left(C_\ell\, c/b\right)}{dy_1}\, dy_1}{y_1 - y} \tag{89}$$

Kuchemann, Reference 22, has shown that this equation may be generalized to wings of any aspect ratio by writing

$$\alpha_i = \frac{\omega' b}{8\pi} \int_{-b/2}^{b/2} \frac{\frac{d\left(C_\ell\, c/b\right)}{dy_1}\, dy_1}{y_1 - y} \tag{90}$$

where ω' is a factor which varies between 1 for high aspect ratio ($AR \rightarrow \infty$), and 2 for low aspect ratio ($AR \rightarrow 0$) .

Kuchemann obtained the following equation for ω'

$$\omega' = 2 - \left[1 + \frac{4}{AR^2} \right]^{-1/4} \tag{91}$$

If the multipliers β_{mk} are rederived using equation (91), a new set of multipliers, β'_{mk}, is obtained related to the old set by

$$\beta'_{mk} = \omega' \beta_{mk} \tag{92}$$

This new set of multipliers β'_{mk} may then be used to calculate induced angle of attack throughout the entire aspect ratio range.

The second equation that must be modified is that which defines the edge-velocity factor E. In Reference 1, this quantity is given by

$$\begin{aligned} E &= \frac{a_e - a_o}{a_o - a_{\ell o}} \\ &= \sqrt{1 + \frac{4}{AR^2}} \\ &= \frac{a_o}{a} \end{aligned} \tag{93}$$

where a_o is the two-dimensional lift curve slope $(AR \rightarrow \infty)$ and a is the corresponding value for finite aspect ratio.

Reference 22 presents an expression for the ratio of the lift curve slopes as

$$\frac{a_o}{a} = \frac{2 - \pi \omega' \cot\left(\frac{\pi \omega'}{2}\right)}{2 \omega'} \tag{94}$$

where ω' is given by equation (91).

Thus, substituting equation (94) into equation (93) yields the edge-velocity factor E. applicable to all values of aspect ratio as

$$E = \frac{2 - \pi\omega' \cot\left(\frac{\pi\omega'}{2}\right)}{2\omega'} \tag{95}$$

In the extended analysis equation (95) is used in place of equation (93).

Finally, the expression for the lift distribution associated with a discontinuity in induced angle of attack, as given by equation (88), must be modified in the following form

$$\left(\frac{C_{\ell_1}'' c}{b\,\delta}\right) = \frac{2}{\pi\omega'}\left[\left(\cos\theta^* - \cos\theta\right)\log\left\{\frac{1-\cos(\theta+\theta^*)}{1-\cos(\theta-\theta^*)}\right\} + 2\left(\pi-\theta^*\right)\sin\theta\right] \tag{96}$$

Equation (96) is now applicable to any value of aspect ratio. This equation is implemented in the computer program and extends the program capabilities to wings with low and high aspect ratios ranging from about 2.0 to infinity.

SECTION 4

DIGITAL COMPUTER PROGRAM

The theoretical analysis presented in Section 3 was programmed for use on the CDC 6600 series digital computer. This was accomplished by extensively modifying the computer program of Reference 1 to include the propeller slipstream and the wing in-slipstream analysis.

This section presents a description of the combined computer program logic, the selection and assembly of the pertinent airfoil section characteristics, and a sample computer output. Wherever appropriate, the discussion is directed towards those features of the modified program that are directly relevant to the treatment of the propeller slipstream and its effect on the wing spanwise loading. Additional information pertaining to computations of the wing loading for a basic wing/fuselage combination can be obtained from Reference 1.

4.1 PROPELLER SLIPSTREAM COMPUTATIONS

This subsection presents the methodology and the associated airfoil section data used in computations of the propeller slipstream velocity distributions, which are later implemented in the overall solution for the wing spanwise loading of a general wing/propeller combination. The basic computational steps for implementing the slipstream velocity distributions into the wing analysis are summarized in subsection 4.2

The essential steps in the propeller slipstream solution are given below.

4.1.1 Computational Procedures for Propeller Slipstream Velocity Distributions

(a) Calculate the propeller angle of attack and tip speed ratio from

$$\alpha_p = \alpha_B + i_{TL} \qquad (97)$$

$$\mu_T = \frac{J \cos \alpha_p}{\pi} \tag{98}$$

(b) At each selected station on the propeller blade obtain local values of the solidity, blade speed ratio and inflow angle ϕ_0 from

$$\sigma = \frac{B \ \bar{c}_p}{2 \ \pi \ \bar{r}} \tag{99}$$

$$\mu = \frac{J \ \cos \alpha_p}{\pi \ \ \bar{r}} \tag{100}$$

$$\phi_0 = \tan^{-1} \mu \tag{101}$$

(c) obtain an approximate solution for the tip loss factor using

$$F_p = \frac{2}{\pi} \cos^{-1} \left[\exp \left\{ - \frac{B}{2} \left(1 - \bar{r} \right) \sqrt{1 + \left(\frac{1}{\mu_T} \right)^2} \right\} \right] \tag{102}$$

(d) Calculate an initial value for the quantity $(u^*/\Omega r)$ from

$$\frac{u^*}{\Omega r} = \frac{1}{2} \left[\sqrt{(\mu + k)^2 + 4k \ \left(\beta - \phi_0 \ - \alpha_{\ell o} \right)} - (\mu + k) \right] \tag{103}$$

where, by definition

$$k = \frac{\sigma \ a_0}{4 \ F_p} \sqrt{1 + \mu^2} \tag{104}$$

and a_0, $\alpha_{\ell o}$ are the lift-curve slope and angle of attack at zero lift, respectively, for a linearized approximation to the tabulated airfoil section characteristics.

(e) Compute an initial inflow angle at each blade element station from

$$\phi' = \phi_0 + \frac{u^*}{\Omega r} \tag{105}$$

(f) Obtain an initial value for the quantity defined as

$$k_x = \frac{\left(\frac{u^*}{\Omega r}\right) \tan \phi'}{1 - \left(\frac{u^*}{\Omega r}\right) \tan \phi'} \tag{106}$$

(g) As the first step in the basic iteration routine, calculate a better approximation for the tip loss factor from

$$F = \left(\frac{F}{F_p}\right) \frac{2}{\pi} \cos^{-1}\left[\exp\left\{-\frac{B}{2}\left(1-\overline{r}\right)\sqrt{1+\left(\frac{1}{\overline{r}\tan\phi'}\right)^2}\right\}\right] \tag{107}$$

where F/F_p is obtained by interpolating the results from the tip loss correction tables for specified values of $B, \overline{r}$ and $\sin\phi'$. A listing of the tip loss correction tables stored and utilized by the computer program is presented in Appendix A.

(h) Calculate the blade section angle of attack and the blade section Mach number from

$$\alpha_b = \beta - \phi'$$
$$M_v = \frac{M_0 \; \pi \; \overline{r}}{J\;(1+k_x)\;\cos\alpha_p} \tag{108}$$

Then obtain the section characteristics C_ℓ and C_d by interpolation and/or extrapolation of the data presented in the propeller airfoil tables, for the specified airfoil section geometry and values of α_b and M_v .

(i) Compute the following quantities defined as

$$k_x = \frac{\sigma}{4F}\left[\frac{C_\ell}{\cos\phi'} + \frac{C_d}{\sin\phi'}\right] \tag{109}$$

$$k_y = \frac{\sigma}{4F}\left[\frac{C_\ell}{\sin\phi'} - \frac{C_d}{\cos\phi'}\right] \tag{110}$$

$$k_z = \mu\left(1 + k_x\right) + k_y \tag{111}$$

and then calculate a new value of ϕ from

$$\phi'' = \tan^{-1} k_z \tag{112}$$

(j) If the absolute magnitude of $(\phi''-\phi')>0.1$ degrees, then the solution for ϕ requires reiteration. In this case the value of ϕ to be substituted for ϕ' in steps (g) through (i) is obtained from

$$\phi = \phi' + (\phi''-\phi')\ \frac{1}{k_c} \tag{113}$$

where k_c is given by

$$k_c = 1 + \cos^2 \phi'' \left[\mu \, k_x \left\{ \frac{a_0}{C_\ell} - \tan \phi' \right\} + k_y \left\{ \frac{a_0}{C_\ell} + \cot \phi' \right\} \right] \tag{114}$$

(k) If the absolute magnitude of $(\phi'' - \phi') \leq 0.1$ degrees then the final slipstream velocity components for the streamtube element passing through the specified blade element station are determined as follows. First, calculate the true induced axial velocity ratio in the propeller disk plane using

$$\frac{u}{\Omega r} = \frac{1}{2} \left[\sqrt{\mu^2 + \frac{4 \, F \, k_y \, k_z}{(1 + k_x)^2}} - \mu \right] \tag{115}$$

and then obtain the axial velocity ratio in the fully contracted slipstream from

$$\frac{V_{s_a}}{V} = 1 + \frac{2}{\mu} \left(\frac{u}{\Omega r} \right) \tag{116}$$

(l) Obtain the local radius in the fully contracted slipstream which corresponds to the specified blade element station from

$$\bar{r}_s = \left[\bar{r}_{s_p}^2 + \left(\bar{r}^2 - \bar{r}_p^2 \right) \left(\frac{1}{2} + \frac{V_a}{V_{s_a} + V_{s_{a_p}}} \right) \right] \tag{117}$$

where $\bar{r}_{s_p}$, $\bar{r}_p$ and $V_{s_{a_p}}/V_a$ are the values corresponding to the immediately preceding inboard blade element station. The velocity ratio at the outer slipstream boundary is taken as unity, as is that at the hub/nacelle boundary unless a blade element station is specified at the hub.

(m) Compute the tangential velocity ratio in the fully contracted slipstream as

$$\frac{V_{st}}{V_a} = \frac{2}{\mu} \left(\frac{u}{\Omega r}\right) \frac{k_x}{k_y} \frac{\bar{r}}{\bar{r}_s} \tag{118}$$

(n) Having obtained solutions for the flow corresponding to all propeller blade element stations $m = 1$ (at the hub) through $m = M$ (at the blade tip), calculate the value of the integrated propeller thrust coefficient from

$$C_T = \sum_{m=2}^{M} \frac{1}{2} \left[\left(\frac{d\,C_T}{d\,\bar{r}}\right)_m + \left(\frac{d\,C_T}{d\,\bar{r}}\right)_{\overline{m-1}} \right] \left[r_m - r_{\overline{m-1}} \right] \tag{119}$$

where
$$\frac{d\,C_T}{d\,\bar{r}} = \left(\pi \quad \bar{r}\right)^3 \frac{F\ k_y\ k_z}{\left(1 + k_x\right)^2} \tag{120}$$

(o) Obtain the momentum value of propeller thrust coefficient from

$$C_{T_S} = \left[\frac{1}{1 + \dfrac{\pi^3\ \mu_r{}^2}{\left(8\ C_T\right)}} \right] \tag{121}$$

(p) Compute the integrated propeller torque coefficient using

$$C_Q = \sum_{m=2}^{M} \frac{1}{2} \left[\left(\frac{d\,C_Q}{d\,\bar{r}}\right)_m + \left(\frac{d\,C_Q}{d\,\bar{r}}\right)_{\overline{m-1}} \right] \left[r_m - r_{\overline{m-1}} \right] \tag{122}$$

where $$\frac{dC_Q}{d\bar{r}} = \left(\pi \bar{r}\right)^3 \frac{r \; F \; k_x \; k_z}{2\left(1+k_x\right)^2} \tag{123}$$

(q) Calculate the value of the momentum-weighted average axial velocity ratio in the fully contracted slip-stream from

$$\frac{\overline{V}_{sa}}{V_a} = \sqrt{1 + \frac{8 \; C_T}{\pi^3 \; \mu_T{}^2}} \tag{124}$$

4.1.2 Propeller Blade Section Characteristics

The analytical methods developed herein require that suitable aerodynamic characteristics be employed for the blade sections of propellers used on general aviation-type aircraft. The information on typical blade sections was obtained from the available technical literature and is summarized in Table I.

As can be noted from this table, early blade sections used in typical propellers are of the USNPS and Clark Y airfoil series. These sections have very similar profiles and members of each series are uniquely identified by the value of thickness/chord ratio alone.

Later blade sections are of the NACA 16-series family, which have a wider application in modern propeller design because of their superior low-drag characteristics (see Reference 25). These considerations also apply to the use of NACA 64 and 65 airfoil series. All of the latter air-foils are specified in terms of both a design lift coefficient and a thickness/chord ratio.

Based on a review of published experimental measurements of propeller airfoil section characteristics, it is evident that the most reliable data for the current application can be obtained from tests conducted in three wind tunnel

Table I. Typical Propeller Blade Sections.

Airfoil Series	Design Lift Coefficients	Thickness/Chord Ratios	References
USNPS	-	0.05 to 0.35	29
Clark Y	-	0.07 to 0.50	17,30,31
NACA 16XXX	0.2 to 0.7	0.04 to 0.40	17,30,32,33,34
NACA 64-XXX	0 to 0.2	0.07 to 0.26	35
NACA 65-XXX	0 to 0.2	0.04 to 0.40	35,36

facilities only. These are the Langley Low Turbulence Pressure Tunnel (Reference 26), for section data at low speed conditions ($M \simeq 0.15$), and both the Langley and Ames High Speed Wind Tunnels (References 27 and 28, respectively) for section data at high speeds ($0.3 \leq M \leq 0.85$). Experimental data available from tests in these facilities were therefore used as the basis for preparation of the required section characteristics for all selected airfoils with the exception of the USNPS and Clark Y series. The section data for the latter two airfoils was generated from the measurements obtained in the Langley Variable-Density Tunnel.

Application of the present analytical methods requires information on the two-dimensional behavior of both lift and drag for the specified blade airfoils. However, an important simplification in preparing these airfoil characteristics is realized through the use of a constant value for drag coefficient on the basis of the following approximation.

From the propeller analysis it can be noted that the contributions of the blade section drag coefficient C_d to the axial and swirl velocity components in the slipstream are given, approximately, by $(C_d/C_\ell)\tan\phi$ and $(C_d/C_\ell)\cot\phi$ respectively, where ϕ is the inflow angle. For low speed flight conditions appropriate to general aviation type aircraft, the contributions of blade section drag to the local axial velocity component in the slipstream are found to be negligible, whereas the contributions to the local swirl velocity are typically not more than a few percent. Thus it is considered a justifiable simplification in the computer program to substitute a representative constant value for C_d in place of the actual variations as a function of angle of attack and Mach number.

It is thus evident that realistic application of the propeller-slipstream analysis demands that selected data on blade section lift characteristics be accurately defined as a function of local angle-of-attack and Mach number for those typical airfoil sections identified above.

Table II summarizes the airfoil sections for which aerodynamic characteristics have been obtained and identifies the source references. In general it is apparent that insufficient data exist to enable a thorough coverage of all the possible variations in section geometry, angle-of-attack and

Table II. Summary of Propeller Airfoil Sections Tabulated for Use in the Computer Program.

Airfoil Series	Thickness/Chord Ratios in Percent									Mach No Range	Source References
USNPS	4	6	8	10	12	14	16	18	20	0.07	37
Clark Y	6	8	10	11.7	14	18	22			0.07	38
NACA 161XX	6	9	-	15	-	30				0.3 to 0.8	39
163XX	6	9	12	15	21	-					
165XX	6	9	12	15	21	30					
167XX	-	9	12	15	-	-					
NACA 64-0XX	6	9	12	15	18	21				0.15	40,41
64-2XX	6	9	12	15	18	21					
64-4XX	-	9	12	15	18	21					
NACA 65-0XX	6	9	12	15	18	21				0.15	40
65-2XX	6	9	12	15	18	21					
65-4XX	-	10	12	15	18	21					

Mach number range. Accordingly a number of simple empirical techniques have been developed to permit a reasonable extrapolation of the available data, as will be discussed later in the text.

Furthermore, in preparation of the final section characteristics, faired curves of the experimental C_ℓ versus α were utilized. The data was carefully selected so as to best define the non-linearities in the faired curves. In general the data represents the full range of the experimental measurements extending from the zero lift condition to a point close to stall and in most cases through the stall.

A complete computer listing of the tabulated section characteristics for the propeller airfoils listed in Table II, is presented in Appendix B. The airfoil tables are arranged so as to provide the maximum flexibility in their use in the computer program. These tables can be easily extended or deleted to include other airfoil families or specially modified aerodynamic characteristics of the selected sections.

These tables form the basis for look-up procedures which through interpolation and extrapolation of the stored data provide the required values of C_ℓ for specified blade sections. These table look-up procedures are described in detail in the next subsection.

4.1.3 Table Look-Up Procedures for Propeller Airfoil Characteristics

The propeller airfoil data tables are read in and stored by the computer immediately prior to execution of the propeller-slipstream calculations. The computer program provides data tables for up to 9 airfoil families, identified by an airfoil series code between 1 and 9 inclusive, but is capable of accepting a maximum of 150 tables. This storage capacity is considered more than adequate under most circumstances but could be extended, if required, by an internal program change. As a rule the only tables read in will be those sets corresponding to the blade sections of the propeller-wing configuration being evaluated.

Each table, as it is read in by the computer, is indexed consecutively in order to permit efficient operation of the look-up procedure. For proper utilization of these data tables, it is essential that they be assembled in a special order. The assembly of all tables for each given airfoil family must be in ascending order of Mach number, thickness/chord ratio and design lift coefficient. However, the sets of tables for any airfoil family may be assembled in any order.

As an initial step in the table look-up procedure, the computer program first searches through the tables to locate and index those particular tables required for interpolation as each propeller blade element station is specified. The actual look-up procedure utilizes linear interpolation throughout and is performed first for the required value of α , secondly for the value of Mach number, thirdly for the section thickness/chord ratio and finally for the design lift coefficient of the airfoil family specified.

To permit satisfactory operation of the computer program for conditions outside the range of the data tables a series of simple extrapolation procedures have been developed empirically from the available experimental data. These procedures are outlined below.

For angles of attack outside the tabulated range in each table it is assumed that the value of C_ℓ remains constant, and for a Mach number outside the given range the extrapolation procedure determines a correction to the required value of α , defined as α_c , thus

$$\alpha_c = \alpha_{0_T} + \left(\alpha - \alpha_{0_T}\right)\sqrt{\frac{1 - M_T^2}{1 - M^2}} \qquad (125)$$

where the subscript T denotes values for the table to be extrapolated.

This method is based on an application of the standard Prandtl-Glauert rule for the change in lift-curve slope with Mach number and assumes that the extrapolated family of lift curves can be represented by a simple adjustment of the angle of attack scale about α_0 point.

For section thickness/chord ratios outside the given range of tables at each value of design lift coefficient it is assumed that the airfoil characteristics will be invariant. While this assumption does not satisfactorily represent the general reduction in lift-curve slope for thick sections ($t/c \geq 0.2$) the existing data does not provide a base for a better approximation.

For a section design lift coefficient c_{ℓ_i} outside the tabulated range, an extrapolation procedure is used to obtain a corrected value of c_ℓ defined as c_{ℓ_c}, thus

$$c_{\ell_c} = c_{\ell_T} + \left(c_{\ell_i} - c_{\ell_{i_T}}\right) \frac{k_{c_{\ell_i}}}{\sqrt{1-M^2}} \tag{126}$$

where the subscript T denotes values for the table to be extrapolated and $k_{c_{\ell_i}}$ is an empirical constant which generally varies for each airfoil family and thickness/chord ratio. This constant has been determined for each airfoil family used herein, and constitutes an inherent part of the computer program table look-up subroutine.

4.2 WING IN-SLIPSTREAM COMPUTATIONS

This subsection presents the method of implementation of the propeller slipstream distributions obtained above into the spanwise load calculations of a propeller/wing combination. The essential computational steps are described below.

4.2.1 Computational Procedures for Spanwise Loading on a Wing with no Flaps, or with Full-Span Deflected Flaps

(a) Obtain the wing basic geometric parameters namely, section chord ratio c/c_R, twist distribution ϵ, thickness-chord ratio t/c, and camber distribution. Then calculate the wing section Reynolds number R_e based on the local chord c and the local resultant velocity V, thus

$$R_e = \frac{V\,c}{\nu} \tag{127}$$

where V is the combined freestream and slipstream velocity given in equation (3) and ν is the kinematic viscosity. Also, obtain the section zero-lift angle α_{ℓ_0}.

(b) Compute the wing-induced upwash function, f, from the following equation which is based on a simple horse-shoe model of the wake (Reference 19)

$$f = \frac{\left(\pi/4-\bar{y}_p\right) \quad 1/\bar{x}_p}{\sqrt{\left(\pi/4-\bar{y}_p\right)^2+\bar{x}_p^{\,2}}} + \frac{\left(\pi/4+\bar{y}_p\right) \quad 1/\bar{x}_p}{\sqrt{\left(\pi/4+\bar{y}_p\right)^2+\bar{x}_p^{\,2}}}$$
$$- \frac{\sqrt{\left(\pi/4+\bar{y}_p\right)^2+\bar{x}_p^2}-\bar{x}_p}{\left(\pi/4+\bar{y}_p\right)\sqrt{\left(\pi/4+\bar{y}_p\right)^2+\bar{x}_p^{\,2}}}$$
$$- \frac{\sqrt{\left(\pi/4-\bar{y}_p\right)^2+\bar{x}_p^{\,2}}-\bar{x}_p}{\left(\pi/4-\bar{y}_p\right)\sqrt{\left(\pi/4-\bar{y}_p\right)^2+\bar{x}_p^{\,2}}} \qquad (128)$$

where $\bar{y}_p = 2y_{p/b}$ and $\bar{x}_p = 2\,\bar{x}_p/b$

are the non-dimensional spanwise and chordwise locations of the right-hand propeller hub.

(c) Calculate the geometric angle of attack at each wing station from

$$\alpha_g = \alpha_B + \alpha_R + \epsilon + \Delta\epsilon_N + \alpha_B T\left[\frac{R}{-}\frac{d\bar{u}}{du}-1\right] \qquad (129)$$

where α_B is the fuselage angle of attack
α_R is the wing/fuselage root setting
ϵ is the local geometric twist

$T\left[\frac{R}{-}\frac{d\bar{u}}{du}-1\right]$ is the correction factor for fuselage upwash given in Reference (1) and $\Delta\epsilon_n$ is the setting of the equivalent chord line of the nacelle above the wing chord line at the nacelle station. The quantity $\Delta\epsilon_n$ is only to be included when a computation station coincides with the nacelle location.

(d) Calculate the following initial approximation to the overall wing lift coefficient

$$C_{L_{APPROX}} = \frac{1}{\left(1+\frac{1.82}{AR}\right)}\left(\alpha_B+\alpha_R-0.4\,\alpha_{\ell o_{TIP}}-0.6\,\alpha_{\ell o_{ROOT}}\right) \qquad (130)$$

(e) Compute the wing-induced upwash at the propeller disc using equation (128) as follows:

$$V_W = \frac{C_{L\,APPROX.}\, f}{\pi^2 \quad AR} \tag{131}$$

(f) Calculate the propeller thrust-line angle of attack and average inclination of the propeller slipstream to the freestream from

$$\alpha_p = \alpha_B + i_{TL} \tag{132}$$

$$\alpha_s = \tan^{-1}\left\{\frac{V_o \sin \alpha_p + V_W}{\overline{V}_{sa}}\right\} \tag{133}$$

where i_{TL} is the propeller thrust-line angle relative to the fuselage centerline.

(g) At each wing station calculate the effective angles of attack, the resultant local slipstream velocity, V , and the non-dimensional slipstream upwash, v , from the following equations:

$$\alpha_e = \alpha_g - \alpha_{\ell_o} \tag{134}$$

$$V = V_{sa} \;/\cos\left(\alpha_p + \alpha_s\right) \tag{135}$$

$$v = \left(\frac{V_{sa}}{V_o}\right) \sin\left(\alpha_s + \alpha_e\right) + \left(\frac{V_{st}}{V_o}\right)\cos\left(\alpha_s + \alpha_e\right) - \sin \alpha_e \tag{136}$$

(h) Calculate the distribution of lift due to slipstream upwash $C_{\ell_2}\, c/b$ using equation (56).

(i) Using the effective angles of attack, α_e , computed from equation (134) find the values of section lift coefficient, C_ℓ , from the two-dimensional section data at the proper values of Reynolds number, thickness-chord ratio and camber level.

(j) Calculate an initial approximation to the spanwise loading distribution using

$$\frac{C_{\ell}\,c}{b} = C_{\ell}\left(\frac{AR}{AR+1.8}\right)\left(\frac{c}{c_R}\right)\left(\frac{c_R}{b}\right)\left[\frac{1}{2}+(1+\lambda)\sqrt{1-\left(\frac{2y}{b}\right)^2}\right] \tag{137}$$

where λ is the wing taper ratio.

(k) Compute the values of induced angle of attack for this load distribution using equation (61) and determine the resultant section angles of attack from equation (62).

(l) From the section data obtain the values of lift coefficient corresponding to the resultant angles of attack from step (k) and calculate the new values of the span loading, $C_{\ell}\,c/b$.

(m) Compare the approximate values of span loading with the calculated values. If these are not in sufficiently close agreement, compute a new set of approximate values of $C_{\ell}\,c/b$ using the procedures presented in subsection 3.2.2 of Reference 1. Repeat the iteration process until the required convergence is achieved.

(n) Integrate the new span load distribution to obtain the overall wing lift coefficient C_L and calculate a new value of wing-induced upwash at the propeller disc using equations (128) and (131).

(o) Repeat steps (f), (g), (h), (i), (k), (l), (m), (n) until the approximate and calculated values of span loading are in satisfactory agreement.

(p) Having determined the lift distribution obtain the section profile drag and pitching moment values from the section data and calculate the overall wing lift, drag, and pitching moment coefficients.

4.2.2 Computational Procedures for Spanwise Loading on a Wing with Part-Span Deflected Flaps

(a) Calculate an initial approximation to the flapped wing lift distribution from the following equations

$$\frac{c_{\ell} c}{b} = \frac{1}{2} c_{\ell_R} \left(\frac{c}{c_R}\right)\left(\frac{c_R}{b}\right)\left\{1 + \sqrt{1 - \left(\frac{y}{y^*}\right)^2}\right\} \quad 0 \le y \le y^* \tag{138}$$

$$= \frac{1}{2} c_{\ell_R} \left(\frac{c}{c_R}\right)\left(\frac{c_R}{b}\right)\left\{1 - \sqrt{1 - \left(\frac{1-y}{1-y^*}\right)^2}\right\} \quad y^* \le y \le \frac{b}{2}$$

where c_{ℓ_R} is the value of the lift coefficient at the root obtained from the flapped section data at the angle of attack

$$\alpha = \alpha_B + \alpha_R \tag{139}$$

(b) Determine the uncorrected values of lift coefficient $c^*_{\ell_u}$ at each flap end as follows

$$c^*_{\ell_u} = \frac{c_{\ell} c}{b}\left(\frac{b}{c}\right)\frac{1}{FF} \tag{140}$$

where FF is the correction factor which accounts for the change in the two-dimensional section data at the flap end. The calculation procedure for obtaining these correction factors is described in detail in subsection 4.1.3 of Reference 1, and will not be duplicated here.

(c) For the values of $c^*_{\ell_u}$ obtained in step (b) above obtain the corresponding angles of attack α_0 from the data for flapped sections. Calculate the corresponding corrected angles of attack $\alpha_{c\delta}$ at each end of the flap from

$$\alpha_{c\delta} = E \cdot FF\left(\alpha_0 - \alpha_{\ell_0}\right) + \alpha_{\ell_0} \tag{141}$$

(d) Using the same procedure as in step (c) above, calculate the values of angle of attack $\alpha_{c\delta=0}$ on the unflapped sides of the wing. Then obtain the first approximation for the values of the discontinuities in angle of attack δ , thus

$$\delta = \alpha_{c\delta=0} - \alpha_{c\delta} \tag{142}$$

(e) Integrate the lift distribution given by equation (138) to obtain an approximate value of the overall flapped lift coefficient, C_L , and using equation (3), determine the wing-induced upwash at the propeller disc. Then calculate the value of slipstream inclination, α_s , using equation (5).

(f) Using the values of δ and α_s from steps (d) and (e), respectively, calculate the distribution of slipstream crossflow from the following equation:

$$v = \left(\frac{V_{sa}}{V_0}\right)\sin\left(\alpha_s + \alpha_e\right) + \left(\frac{V_{st}}{V_0}\right)\cos\left(\alpha_s + \alpha_e\right) - \sin\alpha_e \tag{143}$$

and use equations (64) and (65) to determine the discontinuities in crossflow $\Delta v^* = v''$.

NOTE: In the most general case of a wing having two propellers, (one mounted on each wing panel), rotating in the same direction, the slipstream-induced crossflow distribution will be different at the same spanwise station y on each side of the fuselage centerline. This difference is caused by upward slipstream swirl velocities on one wing panel and downward on the other, occurring at the same spanwise stations on each side of the fuselage, i.e. $v(y) \neq v(-y)$. In the case of two propellers rotating in opposite directions, each slipstream-induced crossflow is symmetrical about the fuselage centerline and equation (143) need only be applied once, since $v(y) = v(-y)$.

(g) Using the appropriate values of the discontinuities δ and $\Delta v^* = v''$ from steps (d) and (f), respectively, compute the lift distribution $C_{\ell_2} \cdot c/b$ using equation (76).

NOTE: For the most general case, as discussed in step (f) above, this lift distribution must be calculated separately for each wing panel.

(h) Determine the lift distribution $C_{\ell_1}'' \cdot c/b$, corresponding to the left and right spanwise discontinuities from equation (88).

(i) Calculate the overall induced angle-of-attack distribution α_i from equation (87), using the approximate span load distribution computed in step (a) above.

(j) Compute the effective resultant section angle of distribution from the following equation

$$\alpha_e = \frac{\alpha_g - \alpha_i + \alpha_{i_g} - \alpha_{\ell_0} \left(1 - E \dfrac{c_{\ell_{max}}}{(c_{\ell_{max}})_0}\right)}{E \dfrac{c_{\ell_{max}}}{(c_{\ell_{max}})_0}} \qquad (144)$$

where α_g is the geometric angle of attack, $c_{\ell_{max}}$ is the value of $c_{\ell_{max}}$ obtained from the corrected section data and $(c_{\ell_{max}})_0$ is the uncorrected value of $c_{\ell_{max}}$.

(k) Using the values of α_e from step (j) above, obtain the corresponding values of lift coefficient c_{ℓ_0} from the uncorrected two-dimensional section lift data. Then determine the correct values of lift coefficient c_ℓ by scaling, as follows:

$$c_\ell = c_{\ell_0} \frac{c_{\ell_{max}}}{(c_{\ell_{max}})_0} \qquad (145)$$

(l) Calculate the distribution $c_\ell \cdot c/b$ from (145) and compare this calculated distribution with the approximate distribution. If agreement between the distributions is not sufficiently close, calculate a new and better approximation using the procedures presented in subsection 3.2.2 of Reference 1.

(m) Repeat steps (b) through (l) above, until agreement is reached between the approximate and calculated values of the span load distribution.

(n) Having determined the lift distribution in step (m), calculate the corresponding value of the overall integrated wing lift coefficient C_L.

4.2.3 Wing Section Characteristics

The wing airfoil section characteristics for typical general aviation aircraft are presented in Section 4.2 of Reference 1, and will not be duplicated in this report. These characteristics are used directly in the current computer program and constitute a part of the overall tool for prediction of stalling characteristics of general wing/propeller combinations.

4.2.4 Table Look-Up Procedures for Wing Section Characteristics

The table look-up subroutine for wing section characteristics used in the current program is identical to that described in Section 4.2 of Reference 1.

4.3 DESCRIPTION OF THE COMPUTER PROGRAM LOGIC

The computational procedures described in Section 3.0 have been programmed for use on a CDC 6600 series digital computer. The program user instructions are given in Appendix C. The flow diagram for the program is shown in Figure 7 and a listing of the program is presented in Appendix D. The program was accomplished by an extensive restructuring and enlargement of the basic power-off wing stall analysis program contained in Reference 1.

The program is initiated by reading in the basic wing-fuselage configuration parameters. In this input format, provision has been made to include an increment representing the drag coefficient of the nacelles. If the calculations are to be performed for the power-on case this is indicated to the program by setting the parameter NSLIP equal to 1. If NSLIP=0, the slipstream calculation loops are bypassed and the program only computes the power-off characteristics.

The computer program arrays are dimensioned to enable calculations of the span loading to be made using 10 control points per semispan.

For twin propeller aircraft computations where the propellers are situated near the center of each wing panel or

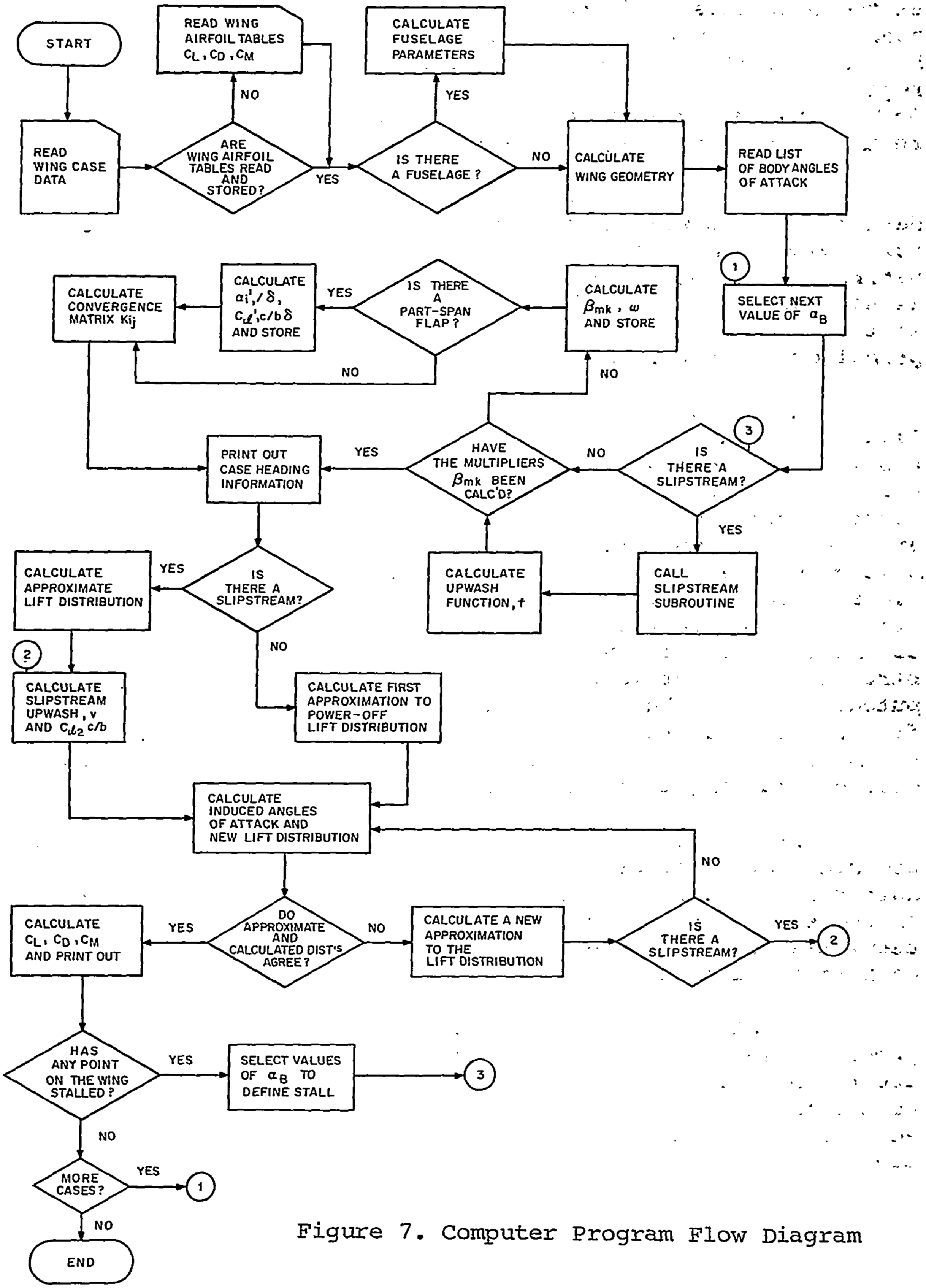


Figure 7. Computer Program Flow Diagram

at the wing tips, this number of wing control stations is adequate. However, for single propeller configurations, a better definition of the span loading in the slipstream region is obtained if the number of control stations is doubled to 20 per semispan. This is readily achieved by redimensioning the required arrays.

Having input the basic data, the required wing section data tables are read in and stored on tape. If the case is for a wing with fuselage the required transformation parameters are computed. The list of fuselage angles of attack is now read in and the first value in the list is selected. If the computation is to be performed for a power-on case the propeller slipstream subroutine is then called.

Execution of the slipstream subroutine shown in Figure 8 is initiated with input and storage at the propeller tip loss correction factor tables. This is followed by reading and storing the required blade section data tables, together with the data specifying the basic propeller geometry and operating condition. The program then proceeds with the main computations as the parameters for each successive blade element are read in. For each blade station, the solution for blade section angle of attack and lift is iterated to convergence. The velocity components for the corresponding streamtube element in the contracted slipstream are then computed. Finally, having obtained the complete velocity distribution for the slipstream, the slipstream velocities at the wing control stations are determined by interpolation before returning to the main program logic.

Having calculated the slipstream velocity distributions the wing upwash function and the induced angle-of-attack multipliers are now computed. If a part span deflected flap is present the parameters associated with the spanwise discontinuities are calculated together with the factors used to correct the two-dimensional section data.

The matrix of coefficients K_{ij} used in the iteration procedure is now computed and stored. If the calculations are to include slipstream effects, the slipstream inclination to the freestream, the slipstream upwash function, v , and the loading associated with this upwash function, C_{ℓ_2} c/b, are computed.

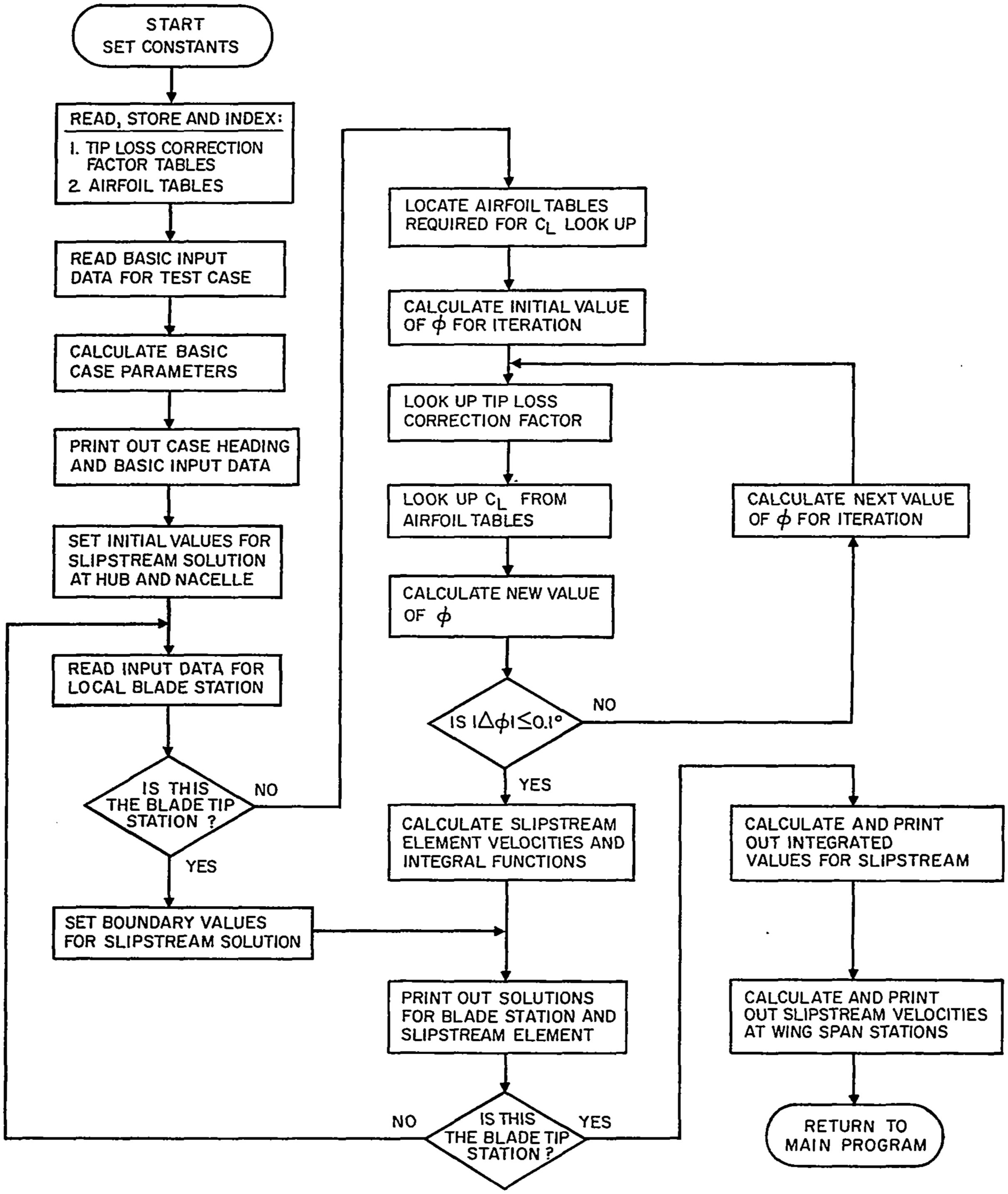


Figure 8. Logic Diagram for Propeller Slipstream Subroutine

The central program iteration loop is now entered. A new lift distribution is computed and compared to an approximate input value. If convergence is not achieved a new and better approximate value is computed. If slipstream effects are being considered, this new lift distribution is used to recalculate the upwash at the propeller discs and a modified slipstream inclination is obtained. A new upwash distribution is calculated and the basic iteration loop reentered.

Once convergence is obtained, the program computes and prints out the overall wing integrated values of C_L, C_D, etc. together with the distributions. If stall is detected at any wing station, the program enters a routine to select values of fuselage angle of attack that will define the exact stall angle more closely.

4.4 SAMPLE OUTPUT

A typical output obtained from the computer program, as described above, is presented in Tables III and IV. Table III shows sample computations for the spanwise lift distribution on a wing-in-slipstream, whereas Table IV presents a sample output for the slipstream velocity distribution used in the wing computations.

Table III - Sample Output for Lift Distribution on a Wing - In - Slipstream

```
                          LR 284 AR=3.04 WITH SS CT''=0.64

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        BODY ANGLE OF ATTACK, DEG. . =       4.25      VALUE OF DISCRIMINANT. . . . =      0.001
        BODY HEIGHT / SPAN . . . . . =       0.00      BODY WIDTH / SPAN. . . . . . =      0.00
        ASPECT RATIO . . . . . . . . =       3.04      WING HEIGHT / SPAN . . . . . =      0.00
        WING BODY INCIDENCE, DEG . . =       0.00      TIP THICKNESS CHORD. . . . . =      0.15
        ROOT THICKNESS CHORD . . . . =       0.15      GEOMETRIC TWIST, DEG . . . . =      0.00
        NUMBER OF SPANWISE STATIONS. =      20.00      AERODYNAMIC TWIST, DEG . . . =      0.00
        FLAP SPAN / WING SPAN. . . . =       0.00      TAPER RATIO. . . . . . . . . =      1.00
        FLAP SETTING, DEG. . . . . . =       0.00      REYNOLDS NUMBER. . . . . . . =      0.63

             COORDINATES OF MOMENT REFERENCE POINT          X=      0.00          Z=      0.00

./../../../../../../../../../../../../../../../../../../../../../../../../../../../../../../../../../.
       3  ITERATIONS REQUIRED TO CONVERGE FOR ANGLE OF ATTACK EQUAL TO      4.25

        SPANWISE STATIONS-2Y/B
( 1)    0.987688E 00 ( 2)     0.951056E 00 ( 3)     0.891006E 00 ( 4)     0.809017E 00 ( 5)     0.707107E 00
( 6)    0.567786E 00 ( 7)     0.453991E 00 ( 8)     0.309018E 00 ( 9)     0.156436E 00 (10)     0.183157E-05
(11)   -0.156432E 00 (12)    -0.309014E 00 (13)    -0.453988E 00 (14)    -0.587783E 00 (15)    -0.707104E 00
(16)   -0.809015E 00 (17)    -0.891005E 00 (18)    -0.951055E 00 (19)    -0.987687E 00 (

        SECTION PITCHING MOMENT COEFFICIENT
( 1)    0.000000E 00 ( 2)     0.000000E 00 ( 3)     0.000000E 00 ( 4)     0.000000E 00 ( 5)     0.000000E 00
( 6)    0.000000E 00 ( 7)     0.000000E 00 ( 8)     0.000000E 00 ( 9)     0.000000E 00 (10)     0.000000E 00
(11)    0.000000E 00 (12)     0.000000E 00 (13)     0.000000E 00 (14)     0.000000E 00 (15)     0.000000E 00
(16)    0.000000E 00 (17)     0.000000E 00 (18)     0.000000E 00 (19)     0.000000E 00 (

        EFFECTIVE SECTION ANGLE OF ATTACK
( 1)    0.216154E 01 ( 2)     0.427324E 01 ( 3)     0.589261E 01 ( 4)     0.615528E 01 ( 5)     0.413006E 01
( 6)    0.114550E 01 ( 7)    -0.328316E 01 ( 8)    -0.463838E 01 ( 9)     0.189371E 01 (10)     0.208239E 01
(11)    0.189371E 01 (12)    -0.463839E 01 (13)    -0.328315E 01 (14)     0.114549E 01 (15)     0.413008E 01
(16)    0.615531E 01 (17)     0.589264E 01 (18)     0.427328E 01 (19)     0.216160E 01 (

        SECTION PROFILE DRAG COEFFICIENT
( 1)    0.627489E-02 ( 2)     0.689538E-02 ( 3)     0.757830E-02 ( 4)     0.801825E-02 ( 5)     0.723848E-02
( 6)    0.604193E-02 ( 7)     0.666712E-02 ( 8)     0.703933E-02 ( 9)     0.619995E-02 (10)     0.628537E-02
(11)    0.619995E-02 (12)     0.703933E-02 (13)     0.666712E-02 (14)     0.604192E-02 (15)     0.723850E-02
(16)    0.801827E-02 (17)     0.757832E-02 (18)     0.689540E-02 (19)     0.627490E-02 (

        SECTION INDUCED DRAG COEFFICIENT
( 1)    0.716377E-02 ( 2)    -0.470249E-02 ( 3)    -0.264332E-01 ( 4)    -0.309915E-01 ( 5)    -0.330864E-02
( 6)    0.622957E-02 ( 7)    -0.482516E-01 ( 8)    -0.813102E-01 ( 9)     0.733644E-02 (10)     0.724601E-02
(11)    0.733644E-02 (12)    -0.813105E-01 (13)    -0.482515E-01 (14)     0.622955E-02 (15)    -0.330885E-02
(16)   -0.309920E-01 (17)    -0.264338E-01 (18)    -0.470285E-02 (19)     0.716370E-02 (

        DISTRIBUTION OF SECTION LIFT COEFFICIENT-CL
( 1)    0.228259E 00 ( 2)     0.451254E 00 ( 3)     0.622260E 00 ( 4)     0.649997E 00 ( 5)     0.436134E 00
( 6)    0.120965E 00 ( 7)    -0.346702E 00 ( 8)    -0.489813E 00 ( 9)     0.199976E 00 (10)     0.219901E 00
(11)    0.199976E 00 (12)    -0.489814E 00 (13)    -0.346701E 00 (14)     0.120964E 00 (15)     0.436137E 00
(16)    0.650000E 00 (17)     0.622263E 00 (18)     0.451256E 00 (19)     0.228265E 00 (

        DISTRIBUTION OF SECTION LIFT COEFFICIENT-CLS
( 1)    0.838396E-01 ( 2)     0.165745E 00 ( 3)     0.228556E 00 ( 4)     0.238744E 00 ( 5)     0.160192E 00
( 6)    0.444305E-01 ( 7)    -0.127343E 00 ( 8)    -0.179908E 00 ( 9)     0.734512E-01 (10)     0.807697E-01
(11)    0.734512E-01 (12)    -0.179909E 00 (13)    -0.127343E 00 (14)     0.444303E-01 (15)     0.160193E 00
(16)    0.238745E 00 (17)     0.228557E 00 (18)     0.165747E 00 (19)     0.838419E-01 (
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        FUSELAGE ANGLE OF ATTACK,DEGREES. .=  4.25000     INDUCED DRAG COEFFICIENT,CDI  =  -0.01950
        LIFT COEFFICIENT,CL . . . . . . . .=  0.13549     PROFILE DRAG COEFFICIENT,CD   =   0.00678
        LIFT COEFFICIENT,CLS. . . . . . . .=  0.04976     NACELLE DRAG COEFFICIENT,CDN  =   0.00000
        PITCHING MOMENT COEFFICIENT,CM. . .=  0.00000     TOTAL   DRAG COEFFICIENT,CD   =  -0.01272
        PITCHING MOMENT COEFFICIENT,CMS . .=  0.00000     TOTAL   DRAG COEFFICIENT,CDS  =  -0.00467
```

Table IV - Sample Output for Propeller Velocity Distribution

PROPELLER SLIPSTREAM ANALYSIS - NRC LR-284 FIG 26 CT''NOM=0.64, BETA75=25, J=0.605

PROPELLER - WING GEOMETRY		PROPELLER - NACELLE GEOMETRY		PROPELLER OPERATING CONDITION	
NUMBER OF PROPS	= 2	NO OF BLADES PER PROP	= 4	LEFT / RIGHT PROP ROTN	= RH, LH
PROP FWD COORD 2.XP/B	= 0.5667	HUB DIA / PROP DIA	= 0.1673	PROP ADVANCE RATIO	= 0.6050
PROP SPAN COORD 2.YP/B	= 0.6179	NACELLE DIA / PROP DIA	= 0.1673	FLIGHT MACH NUMBER	= 0.0581
PROP DIA / WING SPAN	= 0.4753	PROP AXIS REL BODY AXIS	= 0.000 DEG	PROP ANGLE OF ATTACK	= 4.250 DEG

BLADE ELEMENT GEOMETRY						BLADE ELEMENT SOLUTION					SLIPSTREAM ELEMENT SOLUTION			
RB/RP	CB/RP	PITCH	A/F SER	CLI	T/C	F	MACH	ALPHA	CL	CD	RS/RP	USA/UA	UST/UA	PHIS
0.2000	0.2500	55.350	NACA 16	0.500	0.153	1.040	0.082	3.461	0.582	0.010	0.1983	1.2525	0.3364	15.033
0.3000	0.2500	49.200	NACA 16	0.500	0.106	0.994	0.105	6.981	0.848	0.010	0.2878	1.4486	0.4344	16.694
0.4000	0.2500	43.350	NACA 16	0.500	0.090	0.972	0.131	7.968	0.970	0.010	0.3750	1.6170	0.4776	16.455
0.5000	0.2500	37.700	NACA 16	0.500	0.085	0.956	0.159	7.645	0.957	0.010	0.4610	1.7270	0.4672	15.139
0.6000	0.2500	32.350	NACA 16	0.500	0.080	0.933	0.188	6.403	0.878	0.010	0.5467	1.7843	0.4309	13.579
0.7000	0.2500	27.300	NACA 16	0.500	0.075	0.893	0.217	4.722	0.764	0.010	0.6326	1.7986	0.3810	11.962
0.8000	0.2500	22.850	NACA 16	0.500	0.070	0.811	0.246	2.739	0.655	0.010	0.7190	1.7895	0.3369	10.663
0.9000	0.2500	18.900	NACA 16	0.500	0.065	0.638	0.275	0.699	0.487	0.010	0.8067	1.6853	0.2688	9.063
0.9500	0.2500	17.100	NACA 16	0.500	0.062	0.473	0.289	-0.458	0.367	0.010	0.8518	1.5679	0.2195	7.970
1.0000	0.2500	15.350	NACA 16	0.500	0.060	0.000	0.307	0.000	0.000	0.000	0.9012	1.0000	0.0000	0.000

PROPELLER THRUST COEFFICIENT, CT'' = 0.6327
PROPELLER THRUST COEFFICIENT, CT = 0.2462
PROPELLER TORQUE COEFFICIENT, CQ = 0.0374
MOMENTUM WGTD SLIPSTREAM VEL RATIO = 1.6455

SLIPSTREAM VALUES AT WING CONTROL STATIONS

K	2Y/B	RS/RP	USA/UO	UST/UO	UST/USA
1	0.9876	0.7780	1.7148	0.2903	0.1693
2	0.9510	0.7009	1.7865	0.3452	0.1932
3	0.8910	0.5745	1.7840	0.4136	0.2318
4	0.8090	0.4020	1.6470	0.4730	0.2872
5	0.7071	0.1876	1.2490	0.3175	0.2541
6	0.5877	0.0633	1.2490	-0.1071	-0.0858
7	0.4539	0.3448	1.5544	-0.4613	-0.2968
8	0.3090	0.6498	1.7918	-0.3712	-0.2071
12	-0.3090	0.6498	1.7918	-0.3712	-0.2071
13	-0.4539	0.3448	1.5544	-0.4613	-0.2968
14	-0.5877	0.0633	1.2490	-0.1071	-0.0858
15	-0.7071	0.1876	1.2490	0.3175	0.2541
16	-0.8090	0.4020	1.6470	0.4730	0.2872
17	-0.8910	0.5745	1.7840	0.4136	0.2318
18	-0.9510	0.7009	1.7865	0.3452	0.1932
19	-0.9876	0.7780	1.7148	0.2903	0.1693

SECTION 5

VERIFICATION OF THE DEVELOPED THEORY

This section presents a series of correlations between the predicted results, obtained from the computer program described in Section 4 and the available experimental data. A discussion of these correlations has been separated into two natural categories. The first part deals with a verification of the solution for an isolated propeller-nacelle configuration, while the second part considers the combined wing-in-slipstream case.

5.1 CORRELATIONS FOR AN ISOLATED PROPELLER

A majority of the available experimental data on isolated propellers is limited to measurements of total thrust and torque. Even in the few reported studies where the propeller slipstream velocities were measured, the data presented is generally incomplete and insufficient to permit a comprehensive evaluation of the propeller analysis. It was therefore necessary to establish the overall adequacy of the analytical predictions by presenting a series of partial correlations with the applicable data from each experimental source.

Correlations of the elemental loading on a propeller blade are limited to the experimental data reported in Reference 30. This data is presented for two 2.8-foot diameter model propellers of similar design, but different twist distributions. The experimental loadings were obtained directly from measurements of the slipstream velocity and swirl angle in a plane immediately behind the propeller disc. This test information forms the basis for the correlations shown in Figures 9, 10, and 11.

Figure 9 presents comparisons between the predicted and measured elemental thrust and torque loadings, expressed as ratios of predicted over measured values, versus predicted local lift coefficient at a blade radius of 75.2 percent. As can be noted from this figure, the thrust loading predictions, employing the tabulated airfoil characteristics, are in satisfactory to good agreement with the test data throughout the range of the lift curve. Also, the

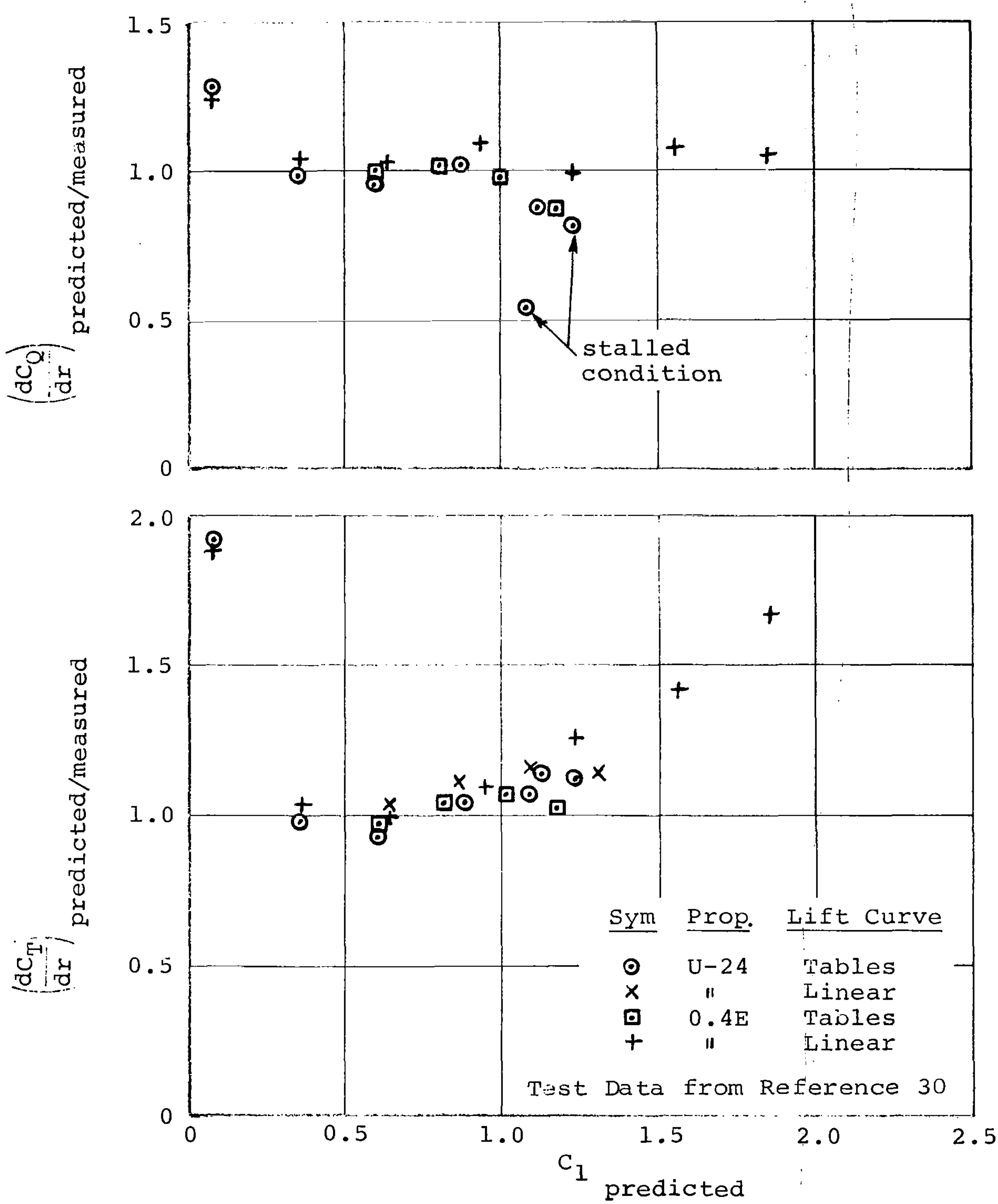


Figure 9. Correlation between Predicted and Measured Elemental Thrust and Torque Loadings at 75 Percent Radius.

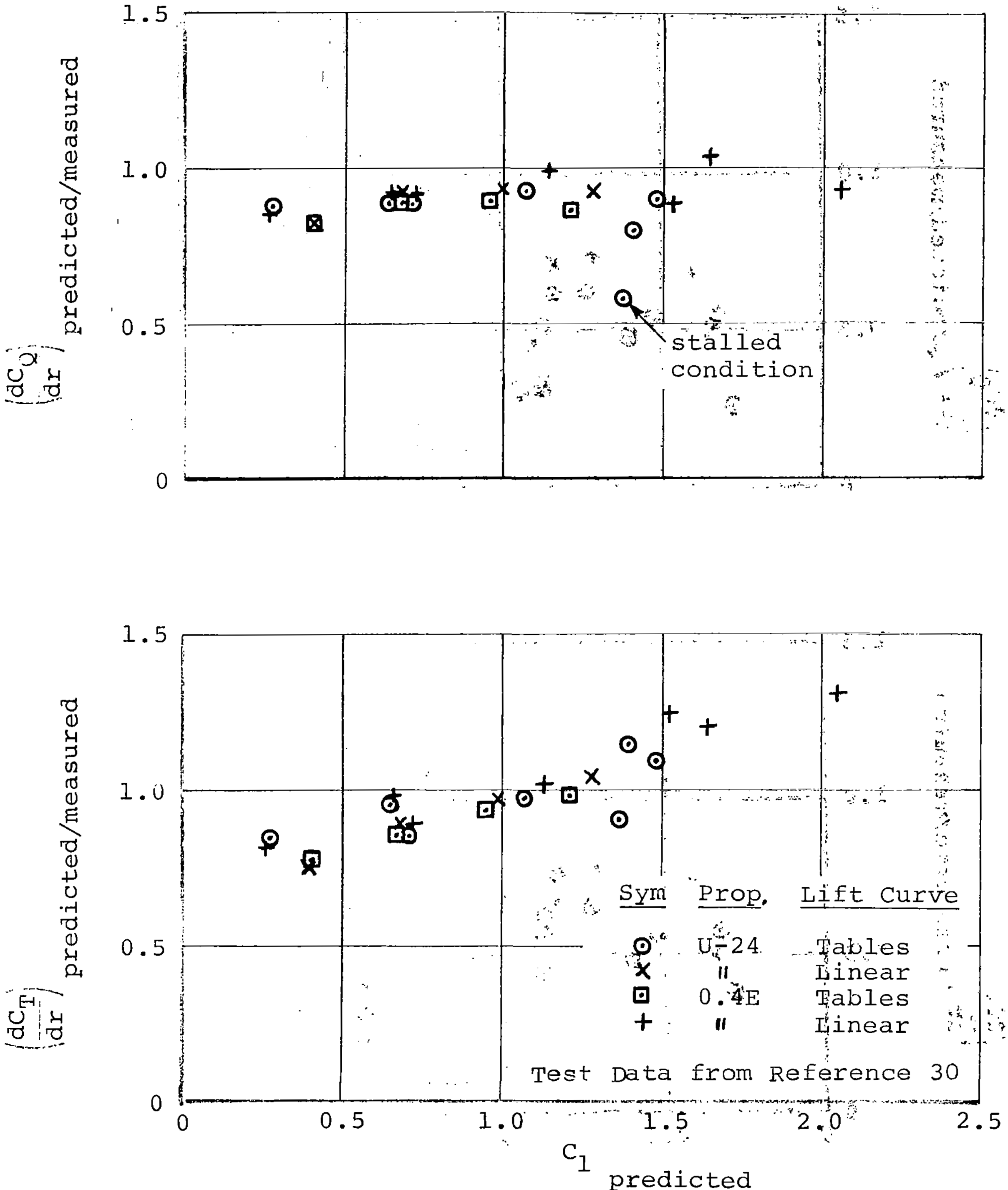


Figure 10. Correlation between Predicted and Measured Elemental Thrust and Torque Loadings at 52 Percent Radius.

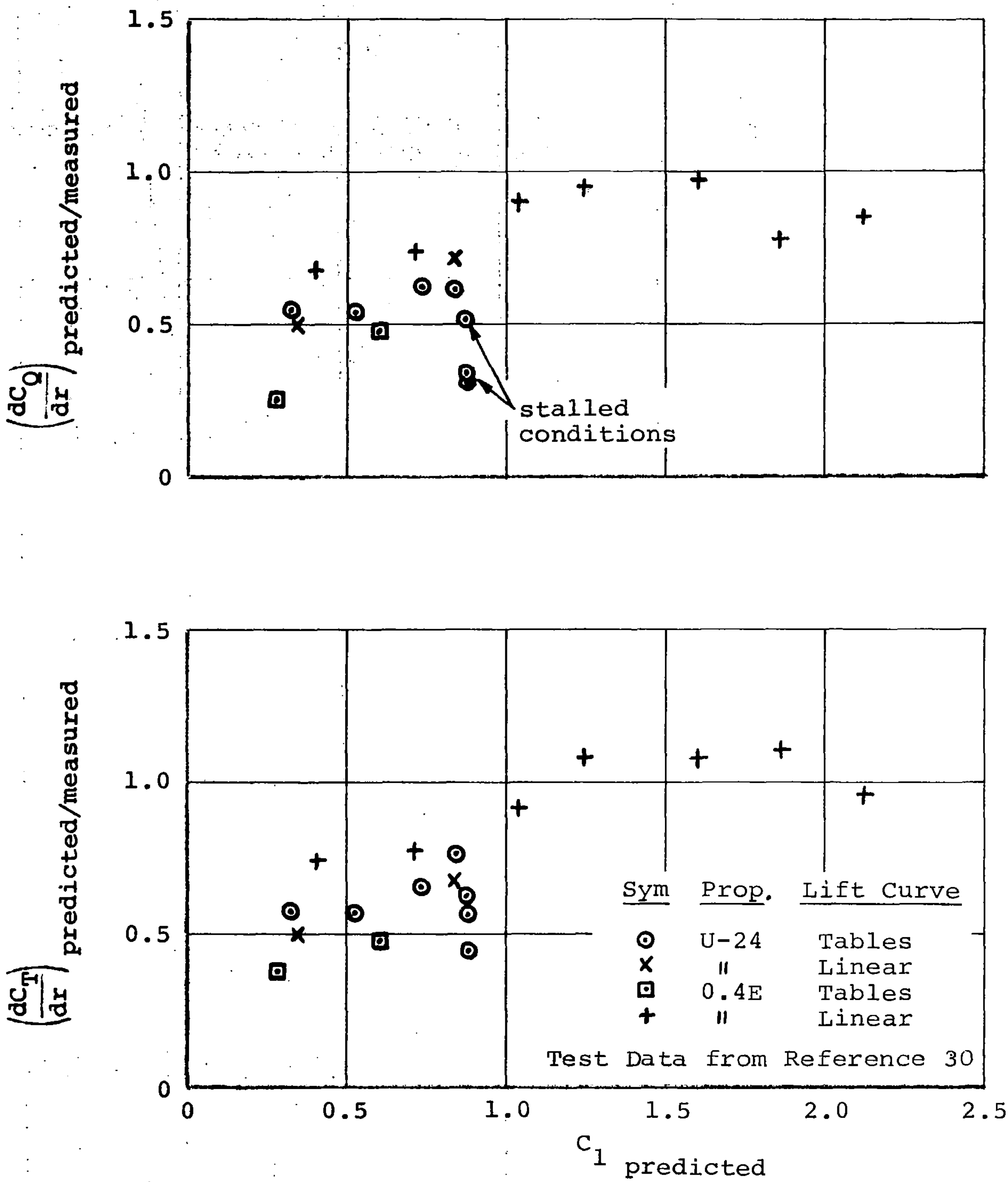


Figure 11. Correlation between Predicted and Measured Elemental Thrust and Torque Loadings at 25 Percent Radius.

corresponding torque loadings are in reasonable agreement, except at conditions near the stall, where the discrepancies may be attributed to an underestimation of the drag coefficient, assumed constant at a nominal value of 0.01. However, it should be noted that the experimental torque loading is particularly sensitive to the measurement of slipstream swirl angle, and therefore may be subject to appreciable experimental error. Figure 9 also includes a comparison of the predicted results obtained by using an unstallable linear lift curve to approximate the airfoil characteristics. The limitation of the linearized representation is reflected by an inferior prediction of thrust loadings near and above the stall point.

Figure 10 shows similar correlations, to those presented in Figure 9 but for a blade station further inboard at 52 percent radius. In this case, satisfactory to good correlations are also indicated.

Figure 11 shows similar comparisons to those shown in Figures 9 and 10, but at a blade radius near the hub, at 25.3 percent. While an increased scatter in the correlations may be partly attributed to the smaller magnitude of the measured quantities, it is evident that the assumption of an unstallable linear lift curve offers a better correlation for both the thrust and torque loadings. A suggestion that the stall point for this airfoil should be extended to a higher angle-of-attack, is consistent with the probable existence of a favorable boundary layer development caused by centrifugal pumping near the hub region.

In reviewing the correlations shown in Figures 9 through 11, it is apparent that there may be a restricted region of the blade close to the hub where stall delay effects are present. However, there is clearly an insufficient substantiation of this phenomenon to permit any rational empirical treatment.

Figures 12 and 13 present correlations between the predicted and measured values of the axial and swirl velocity distributions within the slipstream of propellers operating at relatively low advance ratios. The experimental data shown was obtained from Reference 17, which presents slipstream velocity measurements for two 39-inch diameter propeller-nacelle models, in a plane approximately 0.44 diameters

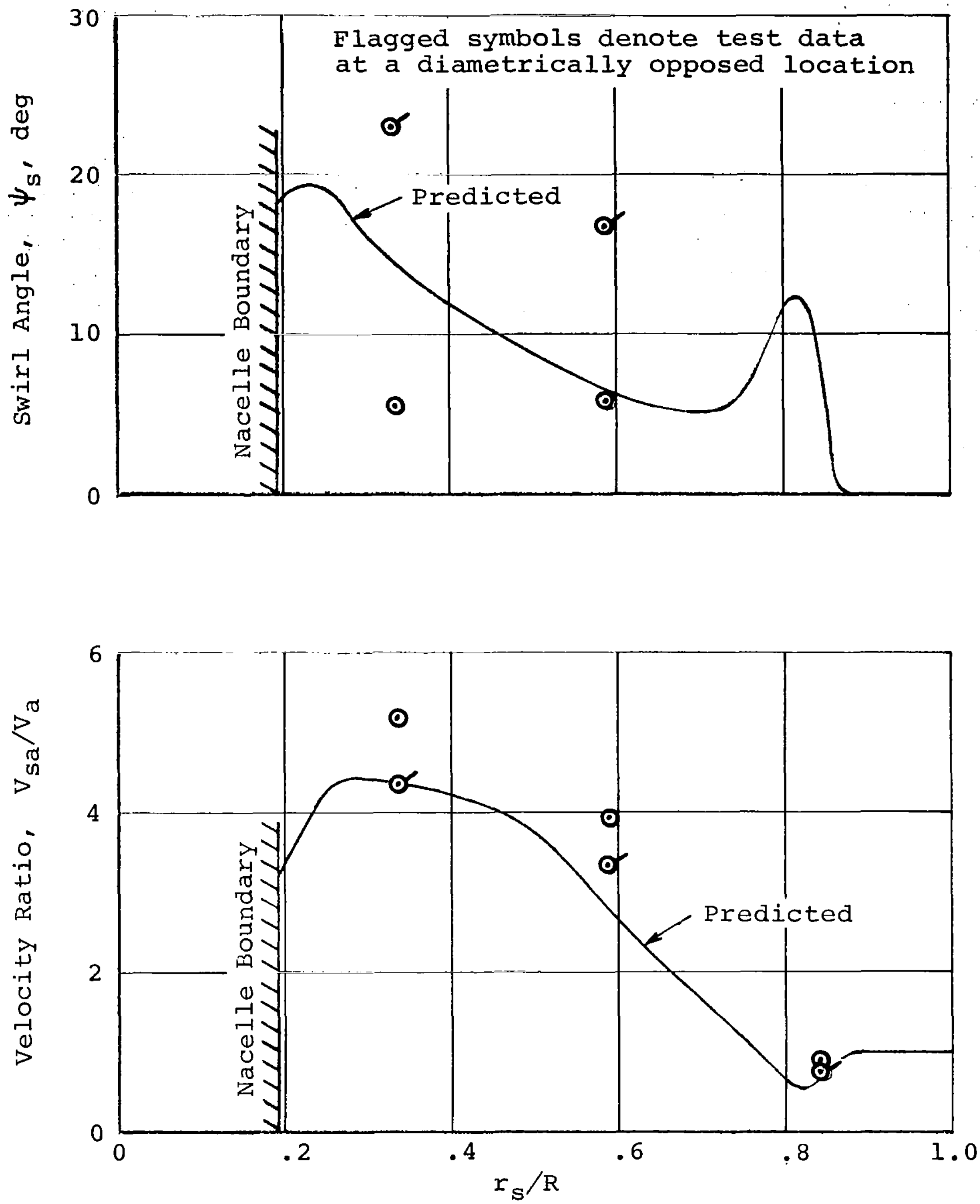


Figure 12. Comparison Between Predicted and Measured Distributions of Slipstream Axial Velocity and Swirl Angle for the P-2 Propeller of Reference 17 at J = 0.12.

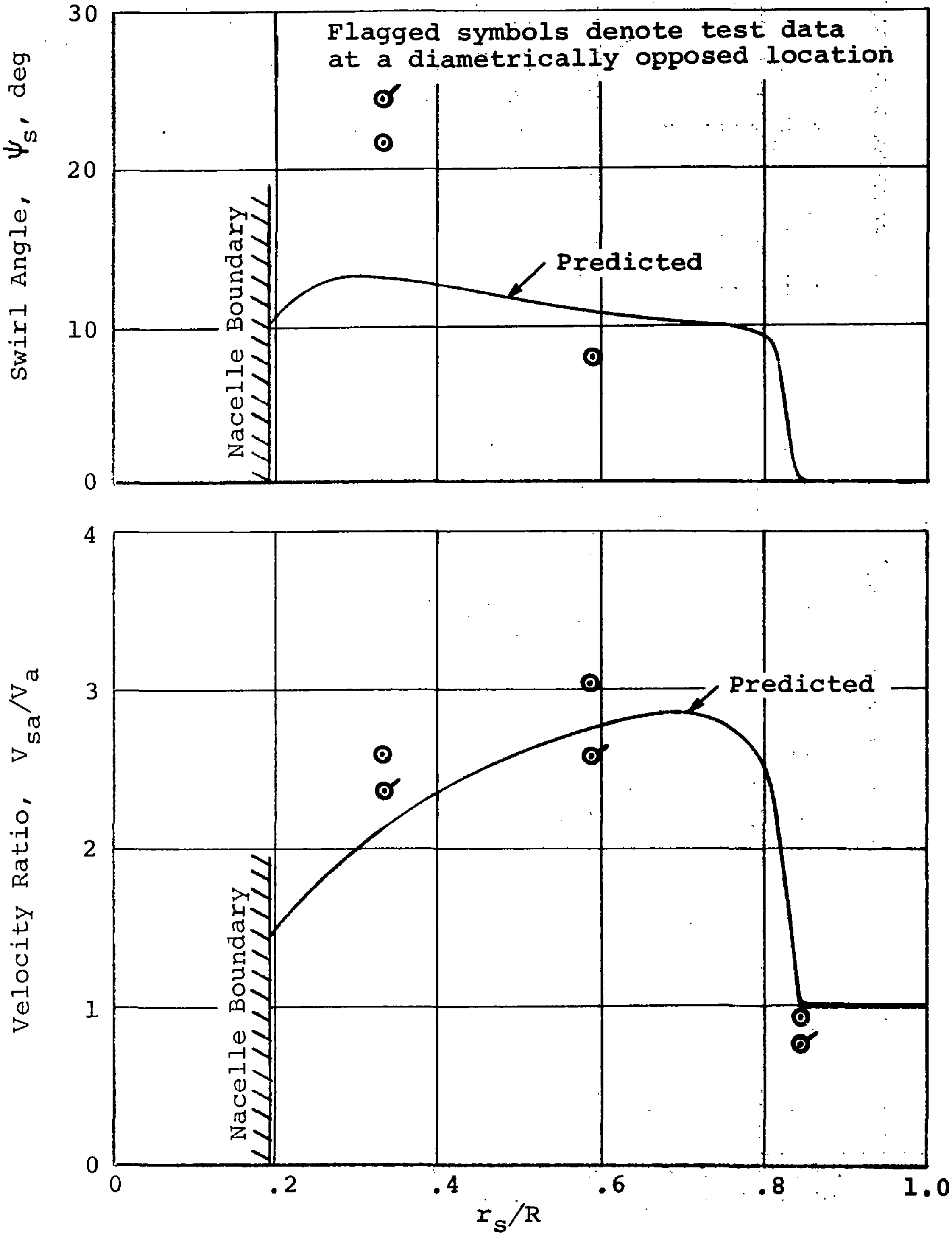


Figure 13. Comparison Between Predicted and Measured Distributions of Slipstream Axial Velocity and Swirl Angle for the P-1 Propeller of Reference 17 at J = 0.26.

downstream of the propeller disc. The slipstream velocity measurements were obtained using an eight-probe rake mounted symmetrically about the propeller axis.

Figure 12 shows a comparison between the predicted and measured slipstream velocities for a propeller designed with high taper and twist so as to produce an axial velocity peak well inboard. As can be seen from this figure, the predicted axial velocity distribution within the slipstream is in good agreement with the corresponding test data. However, the swirl angle prediction can not be properly assessed because of the excessive scatter of the experimental data points. It should be noted that flagged and unflagged test points shown in Figure 12 represent image positions on each side of the propeller.

Figure 13 shows a similar degree of correlation between the predicted and measured slipstream velocities for a propeller having a more conventional plan form and twist distribution. In this case, the predicted results are presented for a propeller speed reduced to 80 percent of the reported value. This correction was introduced to overcome an apparent discrepancy in the test measurements, as suggested by the authors of Reference 17.

Figure 14 presents two additional correlations for slipstream swirl angle distributions based on the test data of Reference 42. The experimental measurements were obtained at a distance of 2 diameters downstream of an isolated propeller. It can be seen from Figure 14 that the analysis generally predicts profiles of the experimental distributions, although an incremental shift in the swirl angle of up to 4 degrees is evident. However, the absence of the corresponding test data on the axial velocity distributions precludes a proper explanation of this shift in the slipstream swirl angle.

One aspect of the analysis not considered in the above correlations is an assumption that the slipstream may be considered as fully developed or fully contracted, for the purpose of predicting the wing span loading. While the effect of slipstream contraction on wing loading can only be properly assessed by comprehensive measurements, some observations on the rate of slipstream contraction can be made from the available test data.

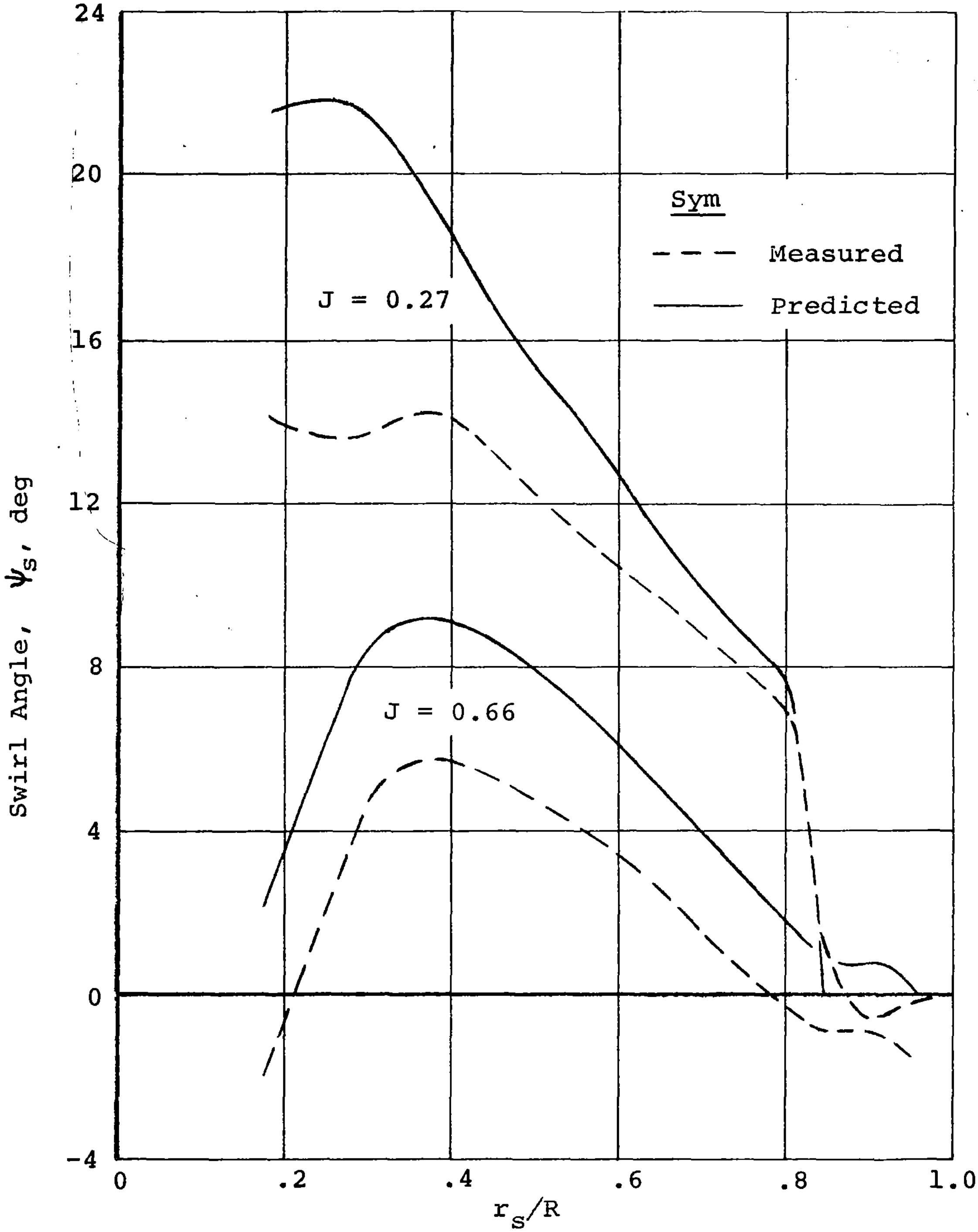


Figure 14. Comparison Between Predicted and Measured Distributions of Slipstream Swirl Angle for Typical Test Conditions Reported in Reference 42.

In practical aircraft configurations the propeller disc plane is generally located between one-half and one diameter forward of the wing quarter-chord line. From the correlations shown in Figures 12 and 13, it may be inferred that slipstream contraction could be fully developed at distances within 0.44 (D) behind the propeller. If this is the case, then the rate of slipstream contraction is significantly higher than that predicted by potential theory.

From the foregoing discussion and the correlations presented above, it can be concluded that the computerized analytical method developed herein yields more than adequate solution for the non-uniform propeller slipstream velocity distributions, which can be confidently used for prediction of wing spanwise loadings.

5.2 CORRELATIONS FOR WING-IN-SLIPSTREAM

This subsection presents the correlations of the theory with experimental data on wing spanwise loadings with slipstream effects. In selecting experimental data to thoroughly test the theoretical model the following criteria were used:

- Complete information on the geometric parameters of wings, nacelles and propellers.
- Adequate definition of the wing and propeller airfoil sections used in the tests.
- A thorough description of the propeller operating conditions in terms of blade angle, advance ratio, and rotational speed.
- An accurate determination of the spanwise lift distribution obtained by chordwise pressure surveys.

It was found that the amount of available test data that meets all of the above criteria is extremely limited. However, sufficient data was obtained from the technical literature to provide a fairly adequate basis for verification of the theoretical model. Some test data was obtained on wings immersed in jets, wings having centrally mounted propellers, and wings with propellers placed at different spanwise stations up to and including the wing tips.

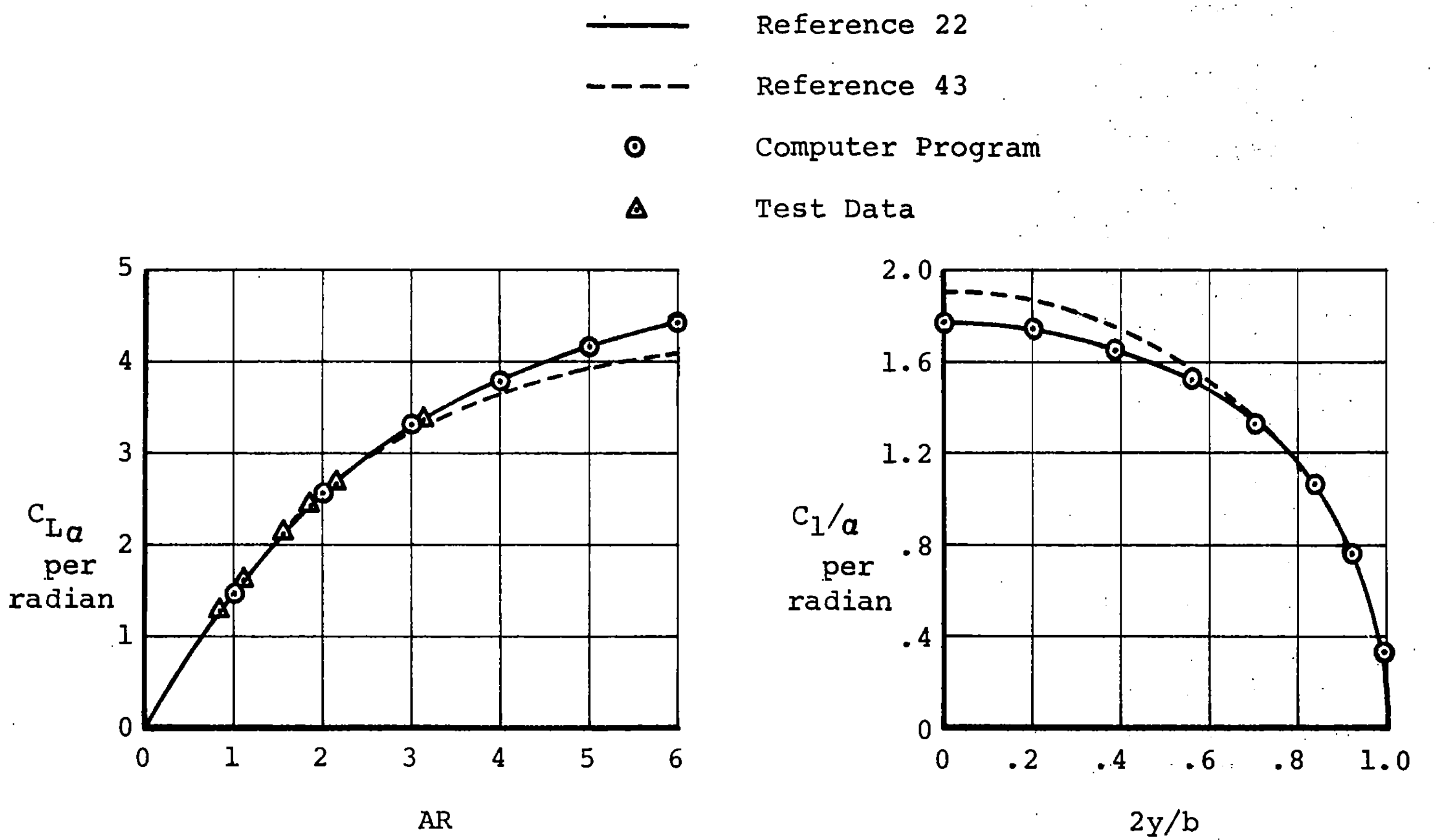
With few exceptions, the majority of the available test data was obtained at wing angles of attack below stall. Therefore, the adequacy of the developed methods to predict the span load distribution at the onset of stall could not be thoroughly verified. However, based on the correlations presented herein, at angles of attack close to stall, it can be inferred that the span loading at stall can be reasonably well predicted using the present analysis. Unfortunately, no test data is available on spanwise lift distributions for wings with part-span deflected flaps. Therefore, correlations for this case can not be presented at the present time.

The correlations that are presented below show the applicability of the analysis to low aspect-ratio wings, the capability to predict wing-in-jet effects, the prediction of span loading for single propeller configurations and, finally, the ability to predict the lift distributions on twin engine aircraft including those having tip-mounted propellers.

5.2.1 Correlations for Low Aspect Ratio Wings

The applicability of the present method to low aspect ratio wings, (see subsection 3.2.3), was verified by performing correlations of the span loading on a rectangular wing of aspect ratio equal to 1.0. These correlations which are shown in Figure 15, are based on the analytical results of Reference 22 and the available test data obtained from a number of sources.

Figure 15 (a) shows a comparison of the predicted span loading (expressed as C_ℓ/α), with the analytical data of the two selected References. As can be noted from this figure, the predicted results are in satisfactory agreement with the results of References 22 and 43. Also Figure 15 (b) shows a comparison between the predicted and measured variations of lift-curve slope for a rectangular wing versus aspect ratio. Again, the computed results match those of References 22 and 43, and agree with the corresponding experimental values.



(b) Variation of Lift Curve Slope with Aspect Ratio

(a) Spanwise Lift Distribution for Rectangular Wing, AR = 1

Figure 15. Verification of Low Aspect Ratio Analysis

5.2.2 Correlation for Centrally-Mounted Propellers and Jets

In Reference 6 Stuper measured the lift distribution on a rectangular wing having a uniform circular jet of air blowing over the center span. The jet was produced by a specially designed fan generating a uniform jet flow without rotation. This test data was chosen for comparison because it provides a check of the wing-in-slipstream theory, without reference to the propeller analysis.

Figure 16 shows a comparison between the predicted and measured spanwise lift distributions from Stuper's test; and again satisfactory agreement between the theory and the experimental data is obtained.

Measurements of lift distributions on wings with centrally mounted propellers are presented in Reference 29. In this series of tests, data was obtained for a full-scale wing/propeller combination in the large 30' X 60' wind tunnel at NASA, Langley. The propeller had a diameter of 4 feet and the wing was rectangular with a 5 foot chord. Aspect ratio was varied by changing the wing span.

Figure 17 presents a comparison between the theoretical predictions and the experimental data obtained for aspect ratio of 6.0 for wing alone, wing and nacelle, and wing, nacelle and propeller. Similar comparisons for a wing aspect ratio of 3.0 are shown in Figure 18. It can be noted from these figures that the combined wing/propeller theory predicts the span loading very well, except near the tips where the experimental data shows the characteristic square-tip loading which cannot be predicted using lifting line theory.

It should be noted that for both the Stuper jet case (Figure 16) and the central propeller cases (Figures 17 and 18), twenty stations per semispan were used in the computations in order to obtain adequate definition of the load distribution within the propeller slipstream region. If the propeller slipstream is not present, sufficient definition is generally achieved with the standard 10 points per semispan.

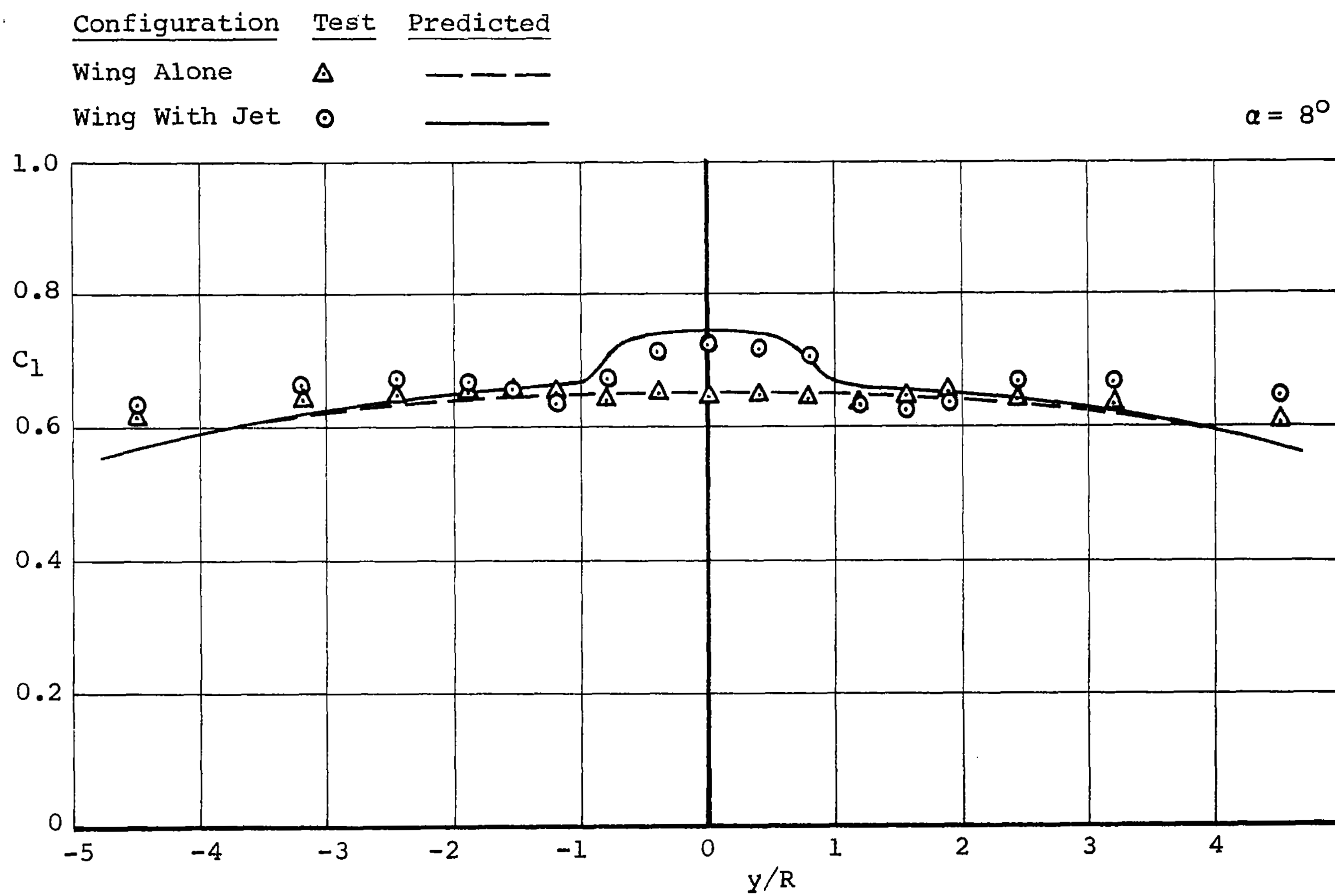


Figure 16. Comparison Between Predicted Spanwise Loading and Measurements of Reference 6 for a Rectangular Wing With End Plates Subjected to a Uniform Jet; $V_s/V_o = 1.36$.

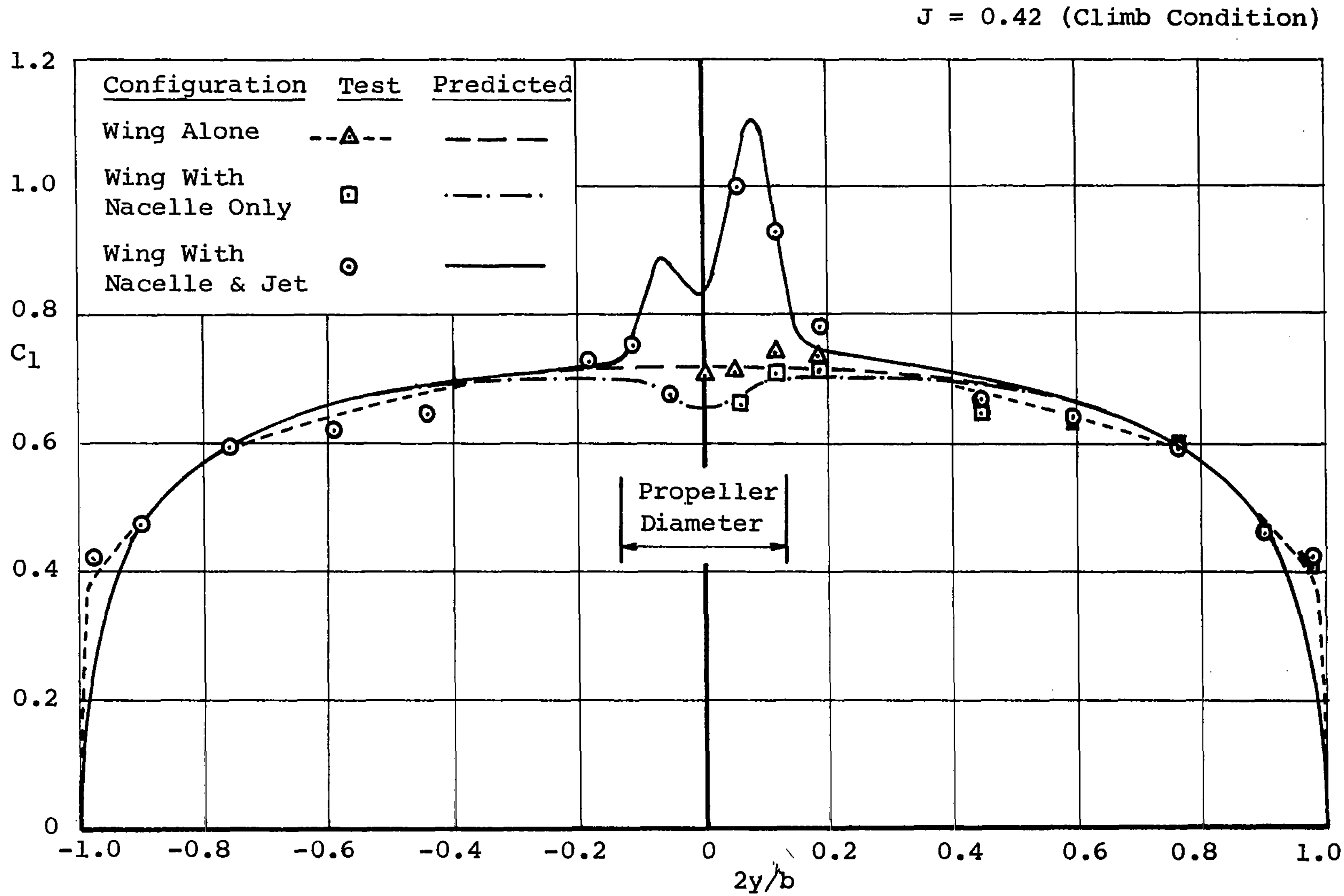


Figure 17. Predicted Versus Measured Spanwise Loadings for the Rectangular Wing of Reference 29 With a Centrally-Mounted Propeller; AR = 6.

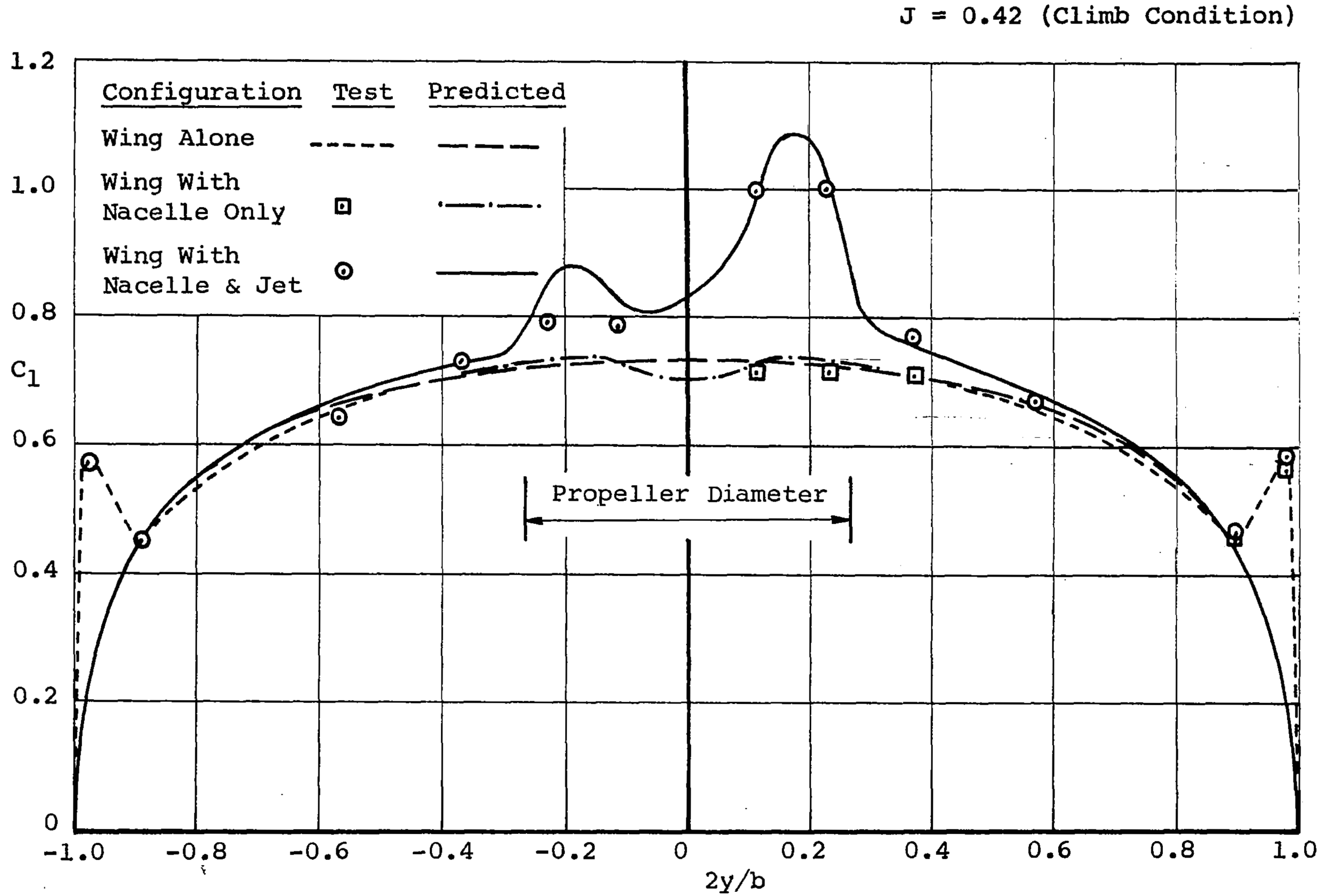


Figure 18. Predicted Versus Measured Spanwise Loadings for the Rectangular Wing of Reference 29 With a Centrally-Mounted Propeller; AR = 3.

5.2.3 Correlation for Twin Propeller Configurations

Reference 42 presents the results of wind tunnel tests on a reflection-plane model of a twin-engined tilt-wing VTOL configuration. The model tested consisted of a low aspect ratio rectangular (18" X 26") unswept wing with a nacelle and propeller situated at 62 percent of the semispan. The wing airfoil section was a NACA 0015, the propeller blade sections were of the NACA 16 series and the propeller diameter was 26". The test report presents measured spanwise load distributions at various wing angles of attack for a limited range of propeller thrust coefficients.

Figures 19, 20, and 21 show comparisons of the predicted and measured span loadings for power-off and power-on conditions, for propeller thrust coefficients of $C_{T_S}=0$, 0.36 and 0.64, respectively. It was found that in order to match the measured lift distributions power-off, the theoretical calculations had to be performed at angles of attack slightly below those values quoted in Reference 42. For example, in order to match the C_l distribution for 5^o angle of attack, the calculations had to be made at 4.25^o. The reason for this discrepancy is not clear since, as is shown elsewhere, predictions for other wings, power-off, agree with the experimental data. The discrepancy could be attributable to tunnel flow inclination effects. In the comparisons shown for power-on conditions, the angle of attack values used are those that match the power-off loading.

Despite the differences noted above, the theoretical predictions of the spanwise lift distribution agree well with the experimental data of Reference 42 except near the wing root. This discrepancy is attributed to the presence of tunnel wall boundary layer effects, as mentioned in Reference 42.

In Reference 44, a series of tests are reported that were made on rectangular wings or aspect ratio 2.28, 3.26, and 4.7 with an underslung nacelle and propeller placed at 83 percent, 58 percent and 40 percent of the semispan, respectively. The propeller was the same propeller used in the tests of Reference 42. The wing airfoil section was a NACA 4415 series. The tests were conducted for wing angles of attack of 0^o through 120^o at various values of propeller thrust coefficient.

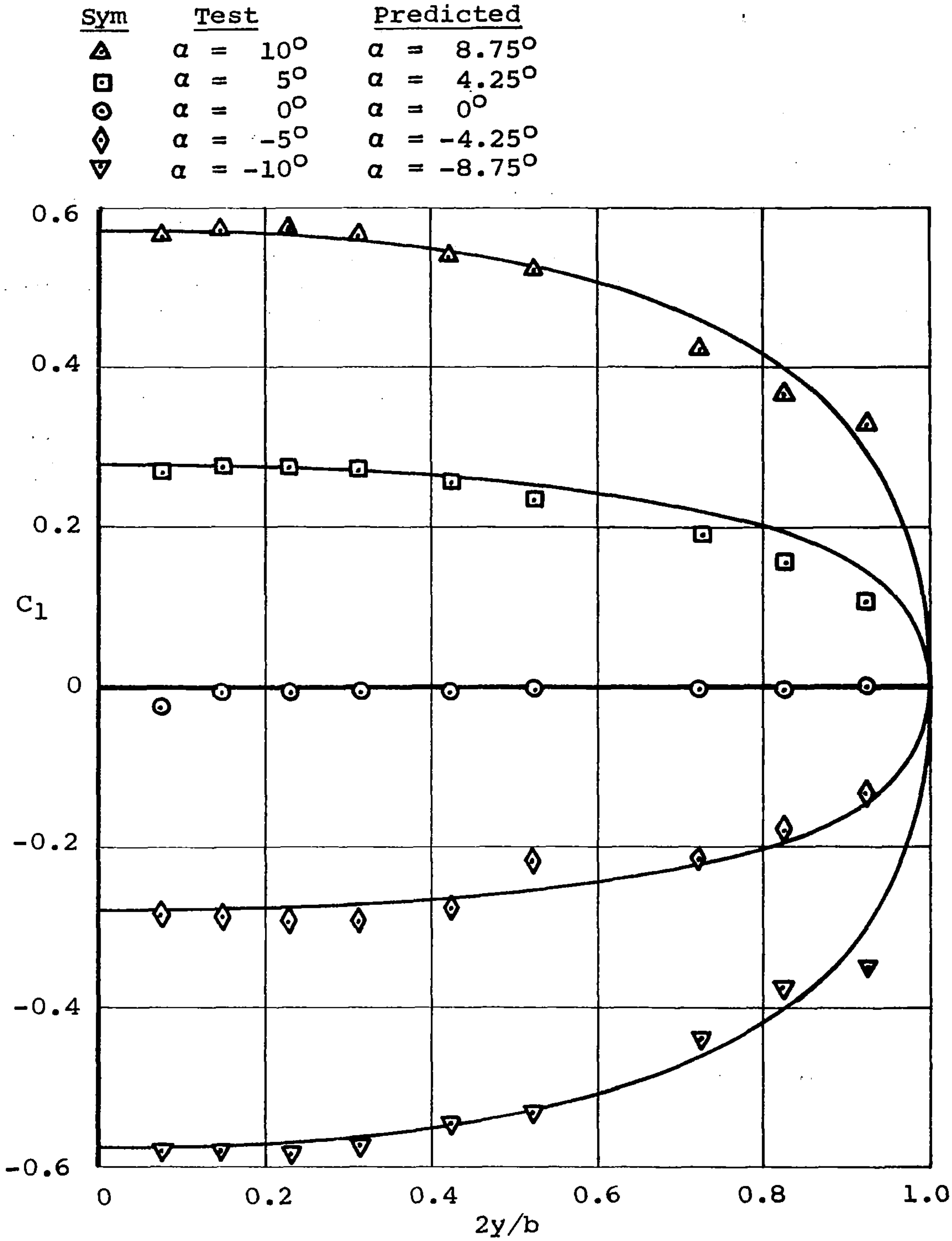


Figure 19. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 42; AR = 3.0, C_{T_S} = 0.

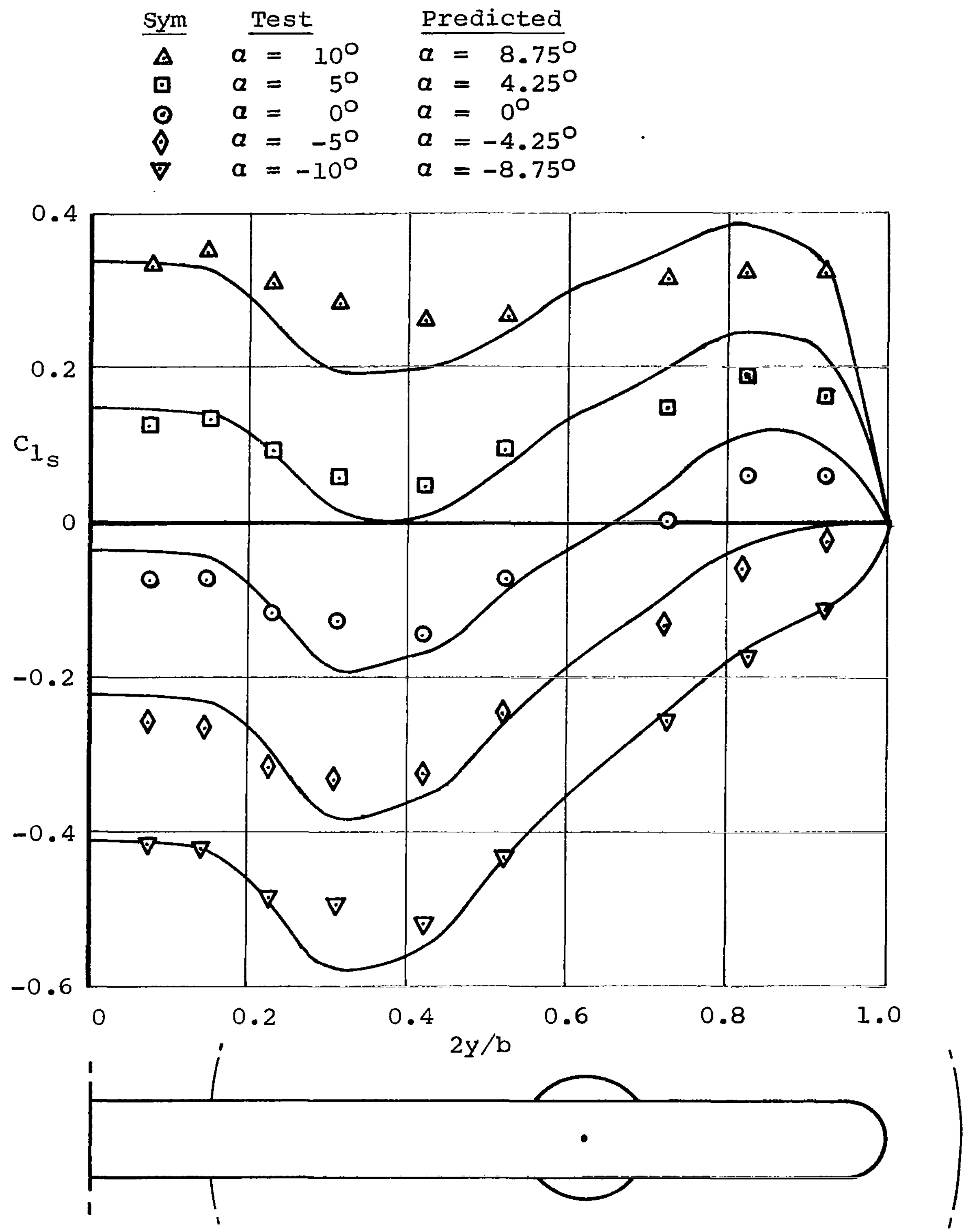


Figure 20. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 42; AR = 3.0, C_{T_S} = 0.36, β_{75} = 25°.

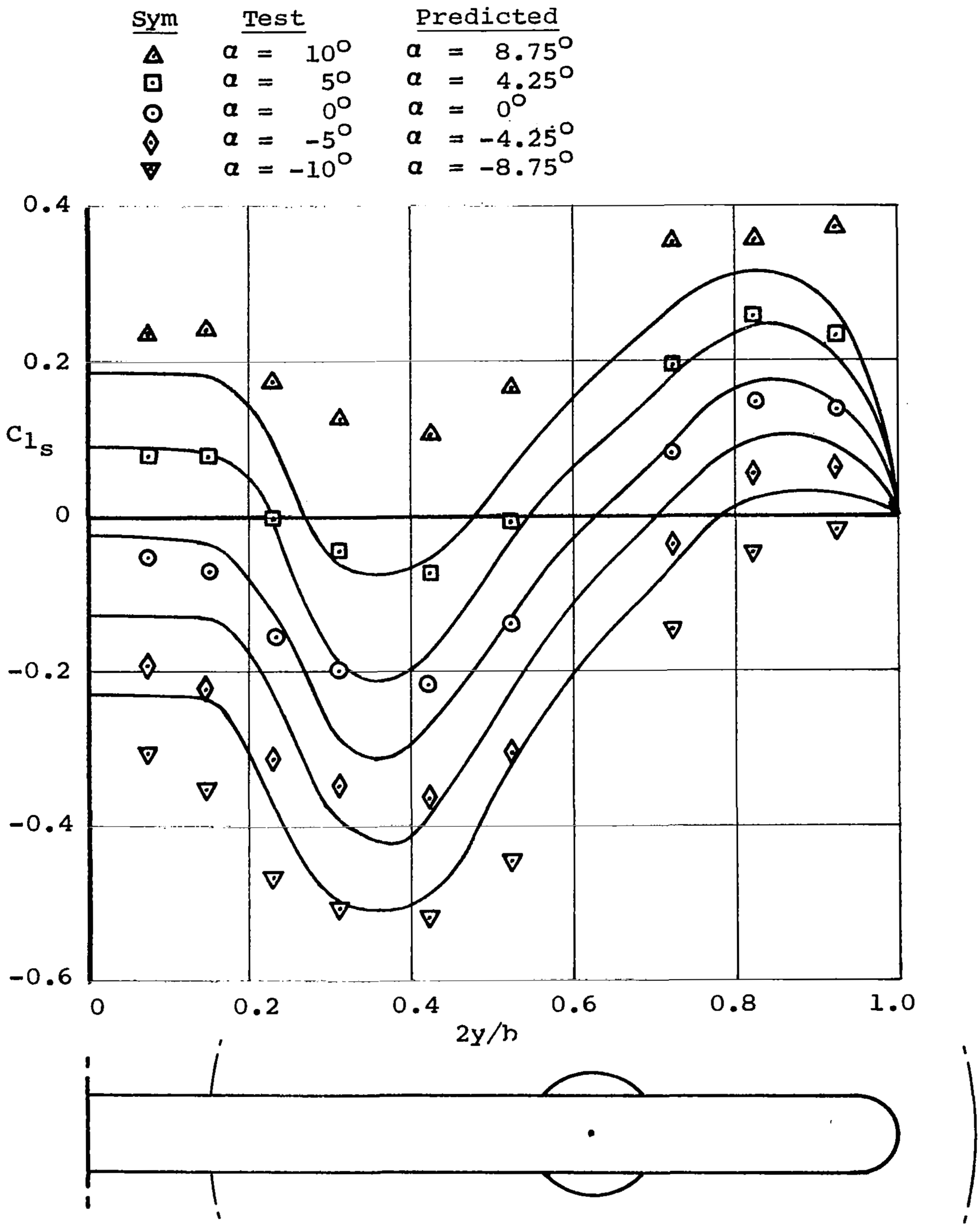


Figure 21. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 42; AR = 3.0, C_{T_S} = 0.64, β_{75} = 25°.

Figures 22, 23, and 24 show power-off correlations of the spanwise lift distribution obtained using the present theoretical analysis for the three wing aspect ratios at wing angles of attack below stall. The agreement between the theory and test is good throughout all values of wing aspect ratio except near the wing root where substantial wall effects are evident. Figures 25 through 27 show the theoretical span loading versus the measured loading for a propeller thrust coefficient $C_{T_S} = 0.4$. In all cases, excellent agreement is obtained between the predictions and the test distributions.

Although test data was obtained at angles of attack up to 120°, the angles of attack were either below stall or well above stall. Thus no data was obtained at the point of initial stall onset. No check of the theory close to the stall point is, therefore, available from this test series.

5.2.4 Effect of Propeller Rotation

The direction of rotation of propellers of multi-propeller configurations may introduce appreciable changes in the wing span loading. For example, rotation of propellers in the same direction of a twin-propeller configuration causes asymmetry in the span loading, which in turn gives rise to the aircraft rolling moment.

Although no test data exists to verify this aspect of the present theory, computations were performed for the configuration of Reference 42 to demonstrate the ability of the computer program to handle different propeller rotations. The predicted results showing the effect of propeller rotations on the wing span loading are presented in Figure 28. The results are applicable to wing aspect ratio of 3.0, wing angle of attack of 10° and propeller thrust coefficient of $C_{T_S} = 0.64$.

As can be noted from Figure 28 counterclockwise rotation of both propellers (as viewed from the rear) results in the asymmetric span loading, which could be integrated to yield the aircraft rolling moment due to power effects.

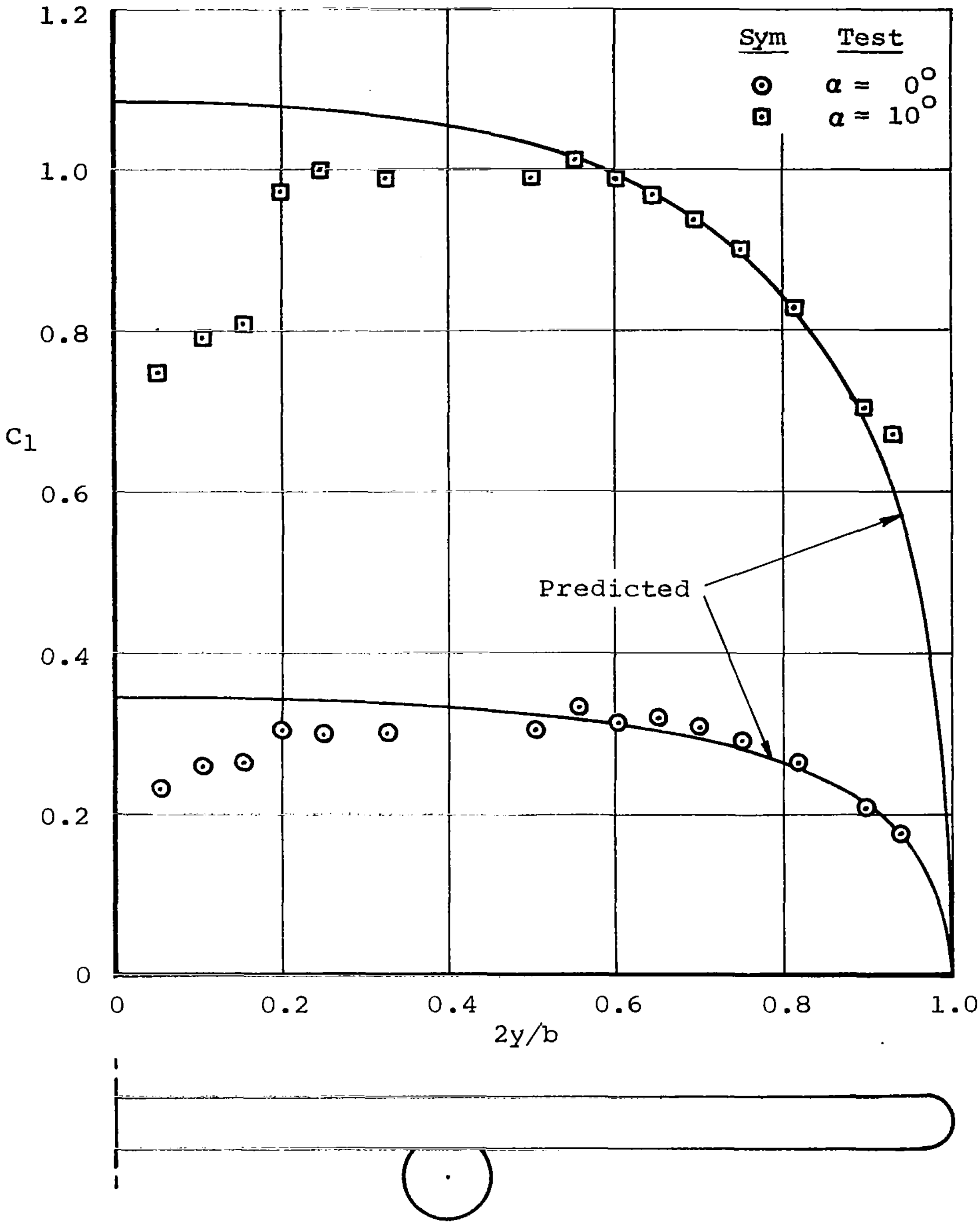


Figure 22. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 44; AR = 4.7, $C_{T_S} = 0$.

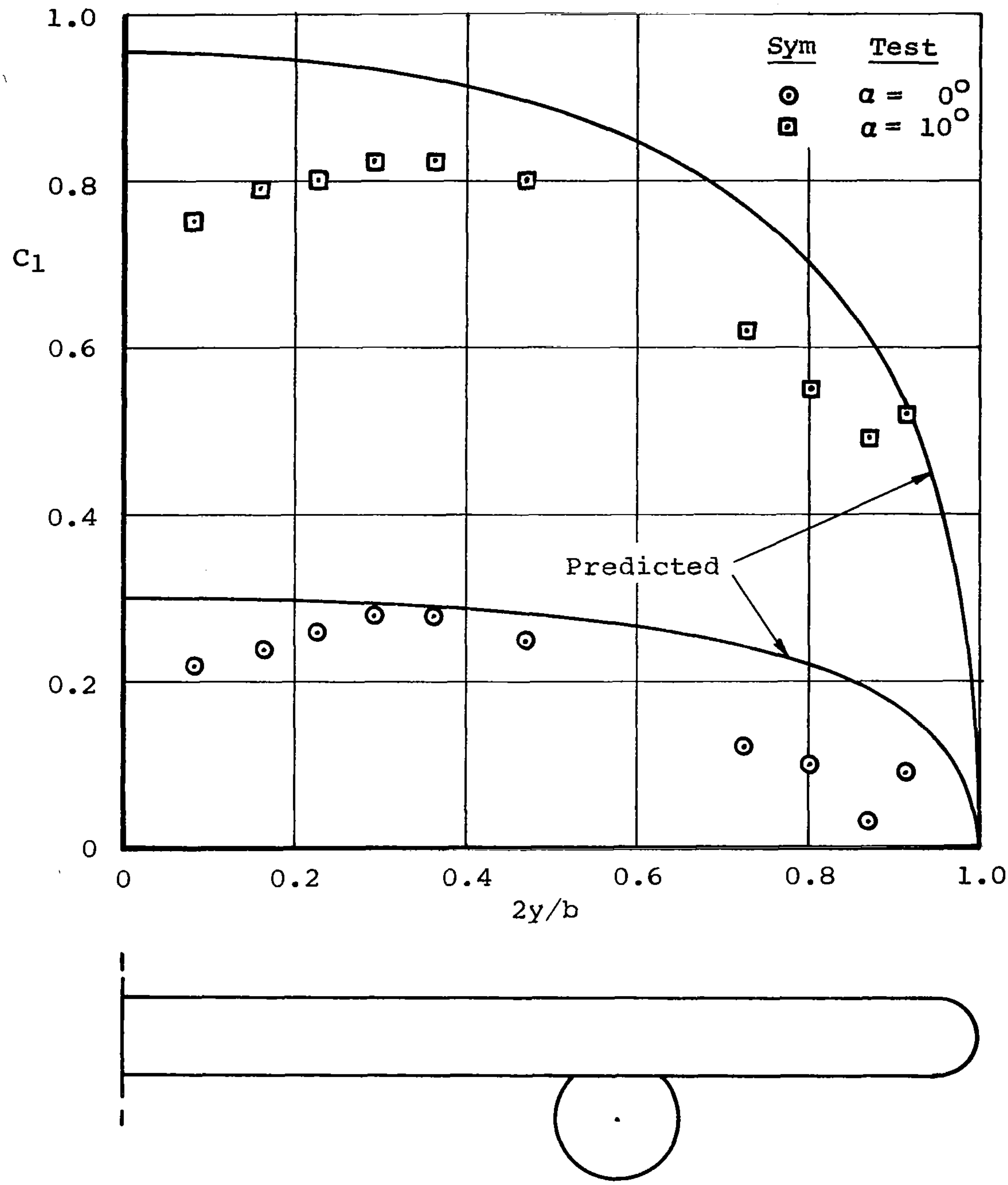


Figure 23. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 44; AR = 3.26, $C_{T_S} = 0$.

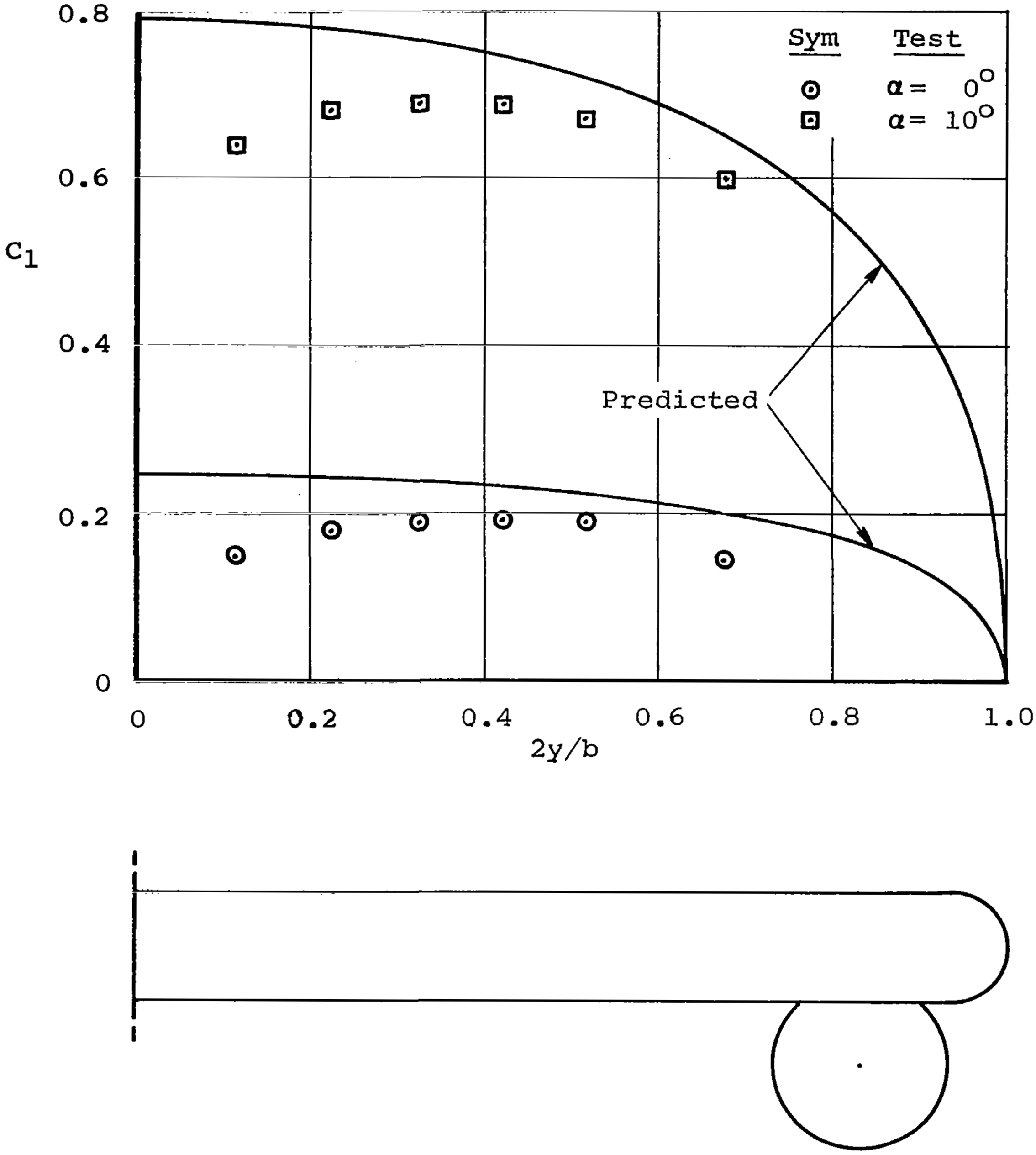


Figure 24. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 44; AR = 2.28, $C_{T_S} = 0$.

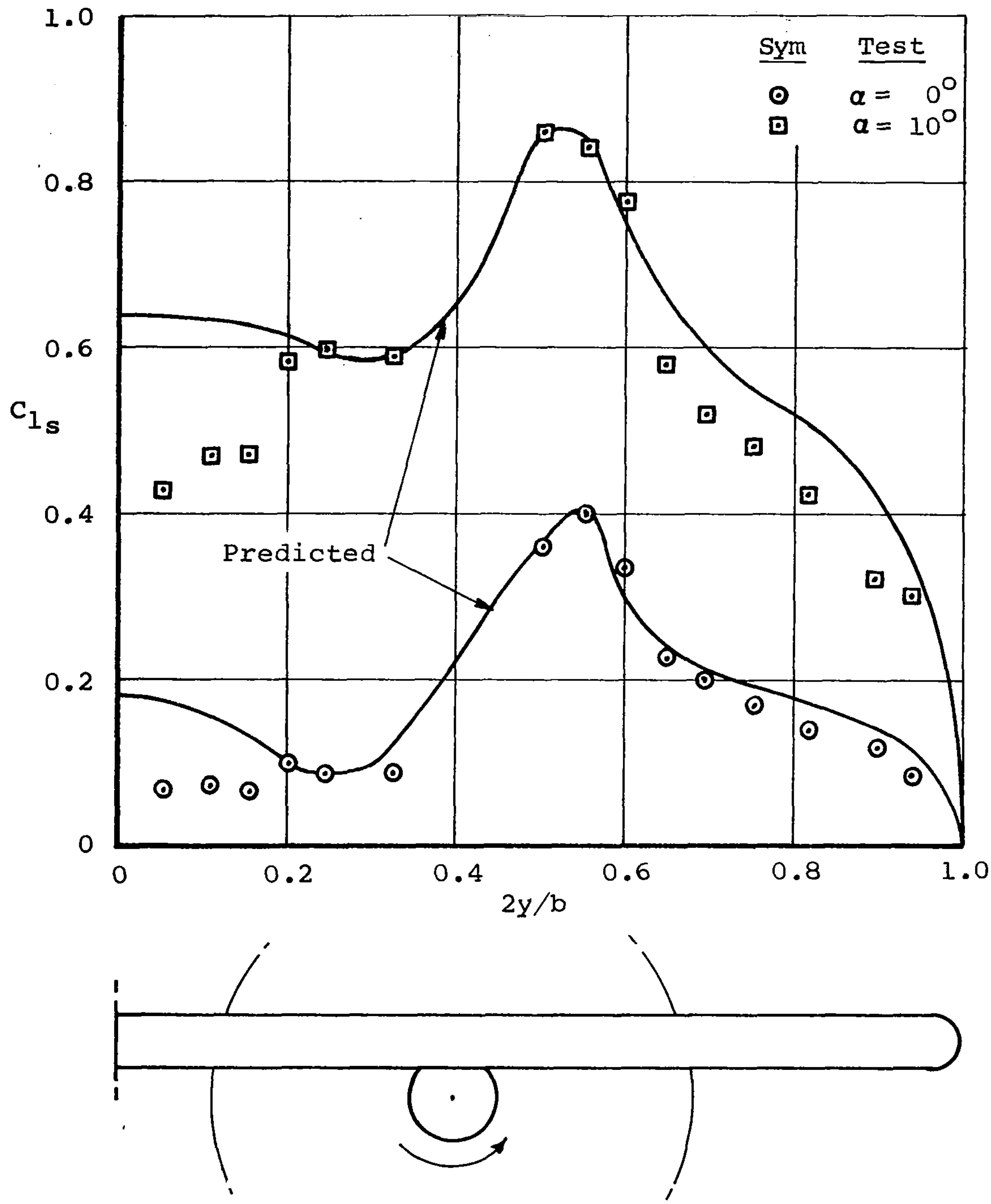


Figure 25. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 44; AR = 4.7, C_{T_S} = 0.4.

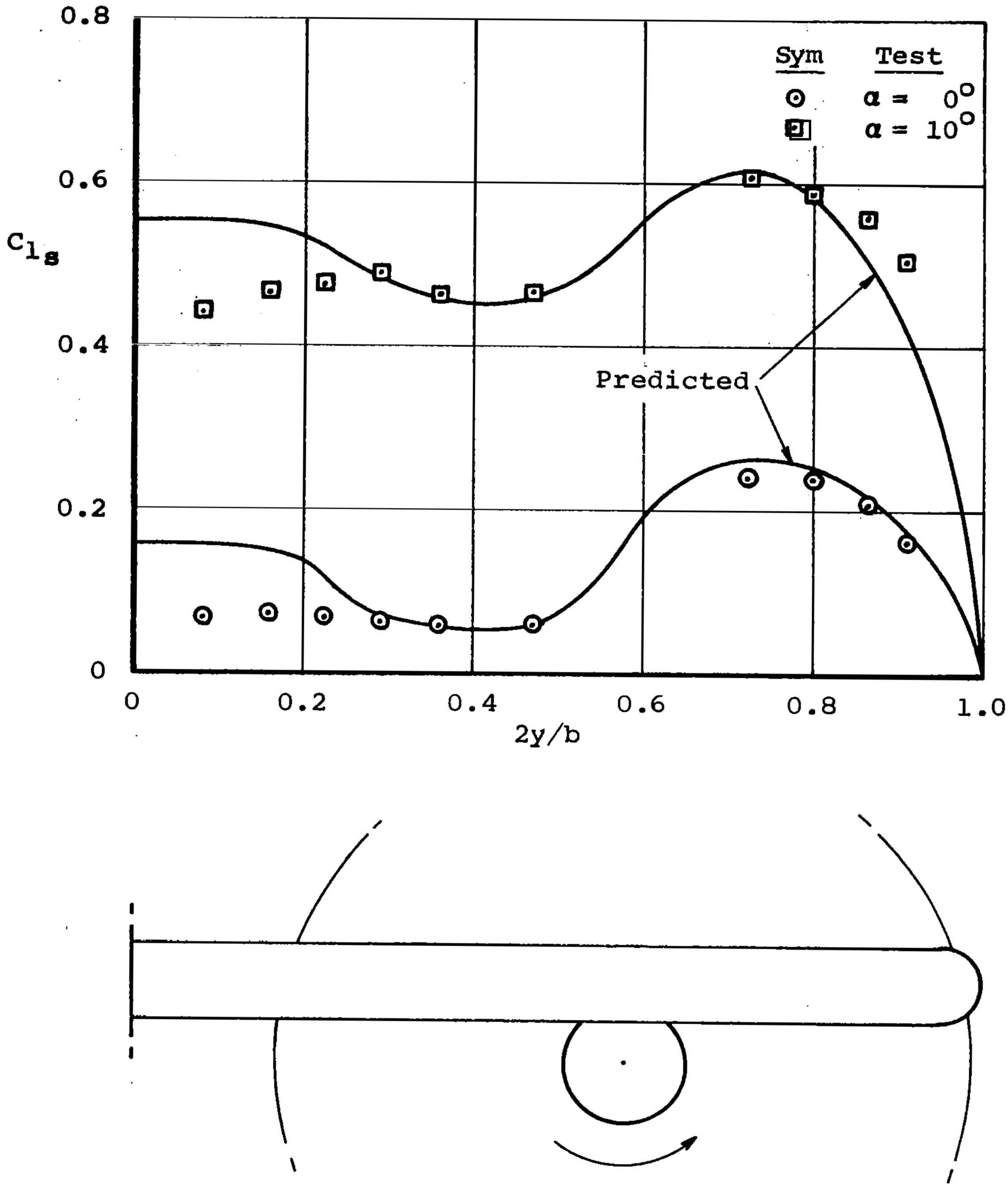


Figure 26. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 44; AR = 3.26, C_{T_S} = 0.4.

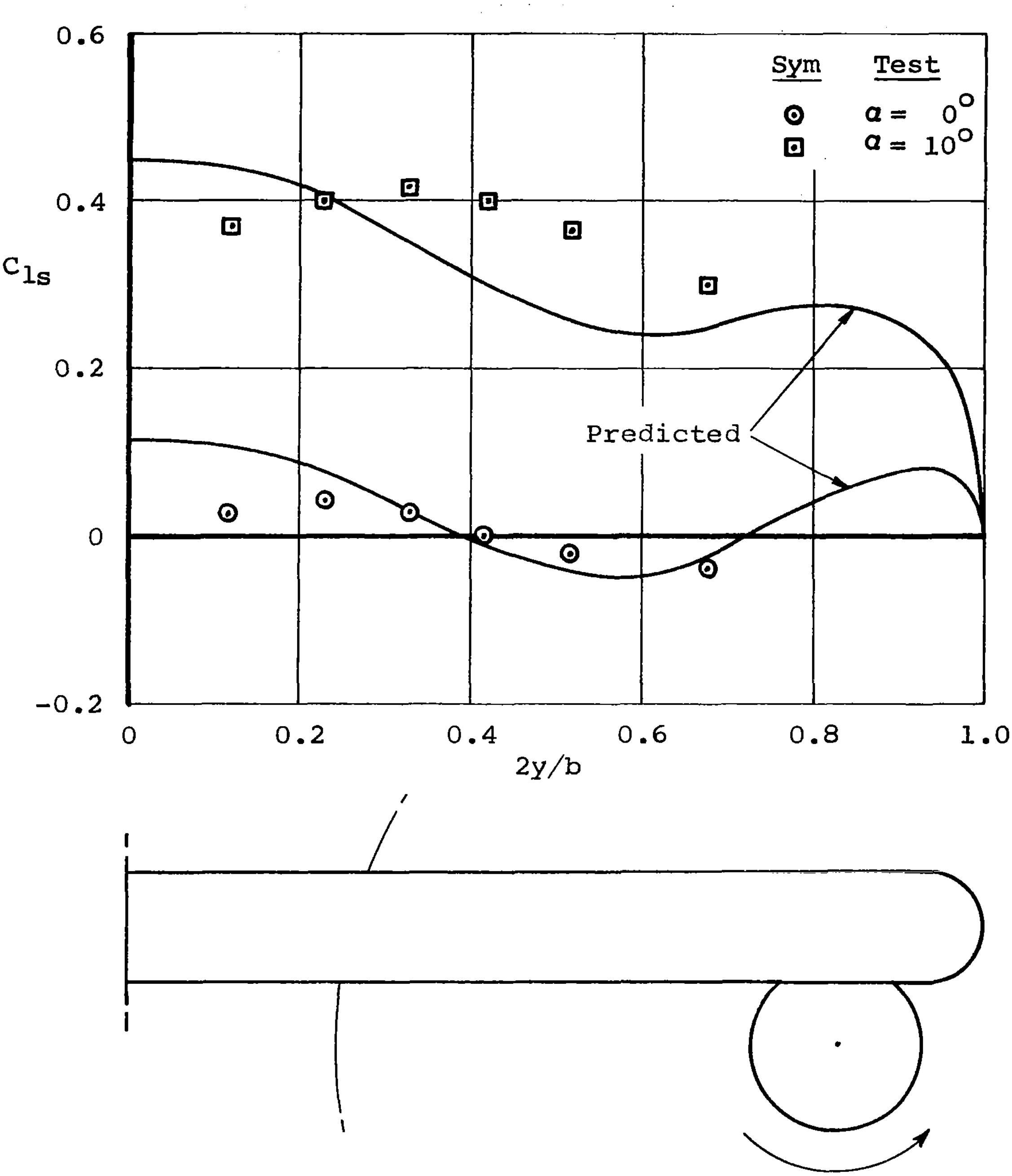


Figure 27. Predicted Versus Measured Spanwise Loadings for the Twin-Propeller Configuration of Reference 44; AR = 2.28, C_{T_S} = 0.4.

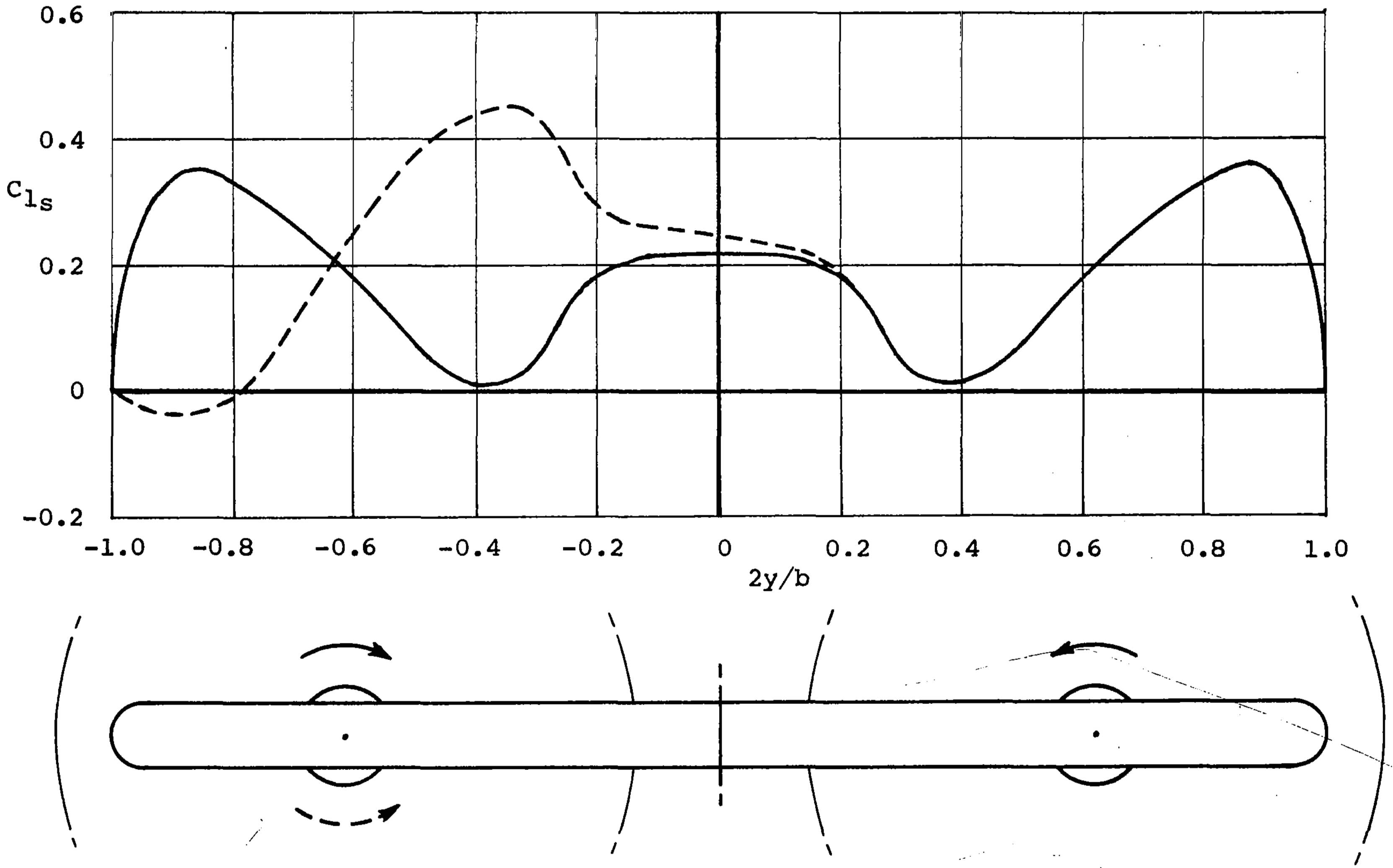


Figure 28. Effect of Propeller Rotation on Span Loading for the Configuration of Reference 42; AR = 3.0, C_{T_S} = 0.64, α = 10 Degrees.

5.2.5 Effect of Flap Deflection

One of the prime concerns in the design of modern general aviation type aircraft is the effect of flap deflection (part-span and full-span) or bring stalling characteristics during take-off and landing, i.e. power-on and power-off conditions respectively. This effect can be readily predicted by the computer program developed herein, however the adequacy of the analysis can not be verified because of the lack of suitable experimental data.

Figure 29 demonstrates the capability of the current computer program to predict power-on span load distributions associated with the deflection of part-span flaps. This figure presents the computed results for the twin-propeller configuration of Reference 44, with an arbitrary flap of 60 percent span. The predicted power-off span loadings, with and without flap deflection, are also shown for comparison.

Based on the correlations presented in this section it is concluded that the wing-in-slipstream theory developed herein provides an effective analytical tool for predicting the effects of propeller slipstream on wing spanwise loadings. Unfortunately, due to the lack of suitable experimental data, these correlations had to be limited to unflapped wings operating at conditions below stall. It is expected, however, that if the pressure data was available for wings at the onset of stall and with part-span deflected flaps, the present theory would also prove satisfactory for these conditions. It is therefore recommended that this part of the theory be verified by wind tunnel tests which should include pressure measurements for both the wing and the slipstream for typical wing/propeller combinations operating close to stall.

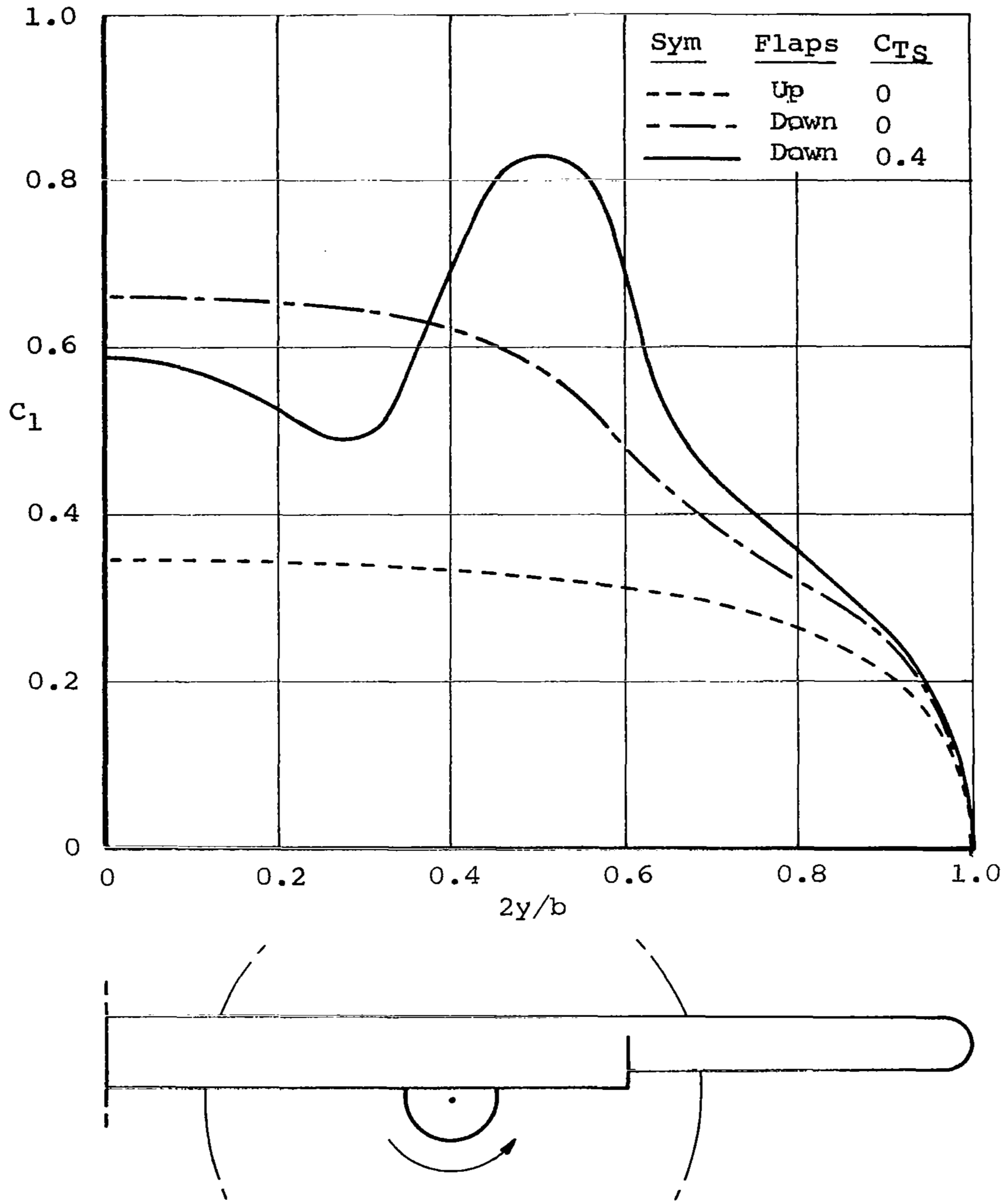


Figure 29. Predicted Spanwise Loadings for the Twin-Propeller Configuration of Reference 44 to Show the Effect of Flap Deflection; AR = 4.7, α = 10 Degrees.

SECTION 6

CONCLUSIONS AND RECOMMENDATIONS

1. This report presents analytical methods for predicting spanwise load distributions of straight-wing/propeller combinations operating up to stall, for a range of aspect ratio from about 2.0 and higher.

2. The analytical methods developed herein employ non-linear lift curves, in the form of computerized table look-up subroutines, for a variety of wing airfoils and an extensive selection of typical propeller blade sections. These methods are therefore applicable to a wide range of wing/propeller configurations and operating conditions.

3. The predicted results for both propeller slipstream velocity distributions and for wing spanwise loadings are generally in good agreement with the limited test data. However, due to the lack of suitable experimental data involving pressure measurements on wings close to stall and with part-span deflected flaps, the full capability of the analysis could not be verified.

4. Based on the correlations shown in Section 5, it is concluded that the computerized methods developed herein represent an effective analytical tool for predicting the power-on and power-off stalling characteristics of general aviation aircraft.

5. In view of the promising results obtained in this study, it is strongly recommended that a comprehensive wind tunnel program be undertaken to provide the necessary experimental data base to complete verification of the analysis.

6. It is further recommended that the wind tunnel test program must include detailed pressure measurements for both the wing and the slipstream for typical wing/propeller combinations operating throughout the entire range of angle of attack up to and including stall.

SECTION 7

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APPENDIX A

PROPELLER TIP LOSS CORRECTION TABLES

This appendix presents a computer-generated listing of the propeller tip loss correction factors utilized by the computer program described herein. These correction factors are applied as described in Section 4.1 to obtain an improvement to the approximate tip loss factor given by equation (16).

These correction factors are based directly on the tabulated values generated by Lock, as given in Reference 21. However, the original tables given by Lock have been modified and enlarged to provide for more uniform increments in the two parametric variables, $\bar{r}$ and $\sin\phi$. These changes permit a more efficient table look-up interpolation procedure and provide for an improved definition of the correction factors. The additional and intermediate values of these factors were obtained through crossplotting of the original tabulated data and by using suitably faired curves.

The following tables are presented for propellers having either 2, 3, or 4 blades. For propellers with more than 4 blades, the computer program assumes a correction factor of unity for all values of $\bar{r}$ and $\sin\phi$.

TIP LOSS CORRECTIONS - TABULATION OF F/FP FOR 2 BLADED PROPELLERS

R/RP SIN(PHI)	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1.0
0.00	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
0.05	1.000	1.000	1.000	1.000	1.000	0.998	0.994	0.990
0.10	1.000	1.000	0.999	0.998	0.996	0.990	0.983	0.976
0.15	0.999	0.998	0.997	0.994	0.987	0.978	0.966	0.955
0.20	0.995	0.994	0.990	0.984	0.972	0.959	0.940	0.923
0.25	0.987	0.984	0.979	0.965	0.949	0.929	0.906	0.877
0.30	0.978	0.974	0.965	0.945	0.923	0.894	0.865	0.827
0.35	0.969	0.963	0.949	0.924	0.894	0.858	0.820	0.785
0.40	0.961	0.952	0.931	0.902	0.863	0.822	0.781	0.746
0.45	0.956	0.941	0.913	0.879	0.831	0.786	0.746	0.711
0.50	0.954	0.929	0.894	0.852	0.801	0.757	0.717	0.681
0.55	0.953	0.918	0.876	0.827	0.777	0.734	0.694	0.656
0.60	0.955	0.907	0.856	0.806	0.759	0.715	0.674	0.635
0.65	0.963	0.900	0.843	0.791	0.746	0.699	0.656	0.616
0.70	0.975	0.897	0.835	0.780	0.735	0.685	0.639	0.598
0.75	0.992	0.899	0.833	0.777	0.726	0.671	0.622	0.581
0.80	1.015	0.911	0.840	0.777	0.718	0.658	0.607	0.564
0.85	1.056	0.942	0.855	0.780	0.710	0.646	0.592	0.547
0.90	1.128	0.990	0.877	0.784	0.704	0.634	0.577	0.531
0.95	1.240	1.060	0.906	0.791	0.698	0.623	0.563	0.515
1.00	1.512	1.170	0.940	0.798	0.692	0.612	0.550	0.500

TIP LOSS CORRECTIONS - TABULATION OF F/FP FOR 3 BLADED PROPELLERS

R/RP SIN(PHI)	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1.0
0.00	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
0.05	1.000	1.000	1.000	1.000	1.000	0.999	0.997	0.995
0.10	1.000	1.000	0.999	0.999	0.999	0.995	0.992	0.987
0.15	0.999	0.999	0.997	0.997	0.995	0.990	0.985	0.976
0.20	0.997	0.996	0.994	0.993	0.990	0.982	0.972	0.960
0.25	0.995	0.993	0.991	0.987	0.981	0.971	0.953	0.938
0.30	0.992	0.990	0.986	0.980	0.968	0.955	0.931	0.910
0.35	0.989	0.985	0.979	0.971	0.954	0.935	0.905	0.879
0.40	0.985	0.979	0.970	0.959	0.938	0.910	0.876	0.844
0.45	0.980	0.973	0.961	0.945	0.919	0.886	0.845	0.809
0.50	0.976	0.967	0.953	0.931	0.900	0.861	0.815	0.777
0.55	0.975	0.964	0.947	0.918	0.882	0.837	0.789	0.749
0.60	0.978	0.965	0.941	0.906	0.865	0.816	0.765	0.724
0.65	0.986	0.968	0.939	0.897	0.849	0.796	0.744	0.701
0.70	0.999	0.976	0.938	0.890	0.835	0.779	0.725	0.680
0.75	1.020	0.986	0.939	0.885	0.823	0.763	0.706	0.661
0.80	1.051	1.001	0.944	0.882	0.815	0.750	0.689	0.644
0.85	1.099	1.028	0.957	0.886	0.812	0.739	0.675	0.627
0.90	1.163	1.073	0.984	0.897	0.814	0.730	0.663	0.611
0.95	1.255	1.145	1.025	0.914	0.815	0.722	0.652	0.597
1.00	1.535	1.275	1.080	0.935	0.817	0.715	0.642	0.583

TIP LOSS CORRECTIONS - TABULATION OF F/FP FOR 4 BLADED PROPELLERS

R/RP SIN(PHI)	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1.0
0.00	1.000	1.000	1.000	1.000	1.000	1.000	1.000	1.000
0.05	1.000	1.000	1.000	1.000	1.000	1.000	0.998	0.997
0.10	1.000	1.000	1.000	1.000	1.000	0.997	0.995	0.992
0.15	1.000	1.000	0.999	0.998	0.998	0.995	0.990	0.985
0.20	0.999	0.999	0.998	0.996	0.995	0.991	0.983	0.975
0.25	0.998	0.998	0.996	0.994	0.991	0.983	0.972	0.959
0.30	0.996	0.995	0.993	0.991	0.984	0.973	0.956	0.937
0.35	0.994	0.992	0.989	0.985	0.974	0.959	0.936	0.909
0.40	0.991	0.988	0.984	0.977	0.962	0.940	0.912	0.878
0.45	0.987	0.984	0.979	0.967	0.948	0.920	0.887	0.850
0.50	0.985	0.981	0.974	0.958	0.933	0.901	0.863	0.824
0.55	0.985	0.980	0.970	0.949	0.919	0.882	0.841	0.800
0.60	0.987	0.981	0.966	0.940	0.905	0.864	0.820	0.777
0.65	0.993	0.985	0.965	0.932	0.892	0.848	0.801	0.756
0.70	1.004	0.991	0.965	0.926	0.880	0.833	0.783	0.737
0.75	1.022	1.004	0.969	0.922	0.870	0.818	0.765	0.719
0.80	1.049	1.021	0.976	0.921	0.862	0.806	0.748	0.701
0.85	1.090	1.048	0.989	0.923	0.857	0.795	0.733	0.684
0.90	1.155	1.090	1.009	0.929	0.855	0.786	0.720	0.667
0.95	1.250	1.156	1.043	0.943	0.857	0.779	0.709	0.652
1.00	1.493	1.267	1.095	0.965	0.862	0.773	0.699	0.637

APPENDIX B

PROPELLER AIRFOIL TABLES

This appendix presents a computer-generated listing of the propeller blade section data tables that are available for use by the computer program.

Each table contains the values of lift coefficient versus angle of attack for a range of Mach number conditions for one specified airfoil section. At the head of each table is descriptive information on the airfoil name and data source. This is follows by the airfoil series code identification and the main geometric and test parameters.

The following tables contain the selected propeller station characteristics for airfoils of the U.S.N.P.S., Clark Y, NACA 16, NACA 64 and NACA 65 families. The sets of tables for each airfoil series are arranged in the specific order of increasing Mach number, thickness/chord ratio and design lift coefficient, as necessary for proper utilization by the computer program.

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	USNPS -M04	USNPS -M06	USNPS -M08	USNPS -M10	USNPS -M12	USNPS -M14
TABLE DATA SOURCE	F E WEICK	F E WEICK	F E WEICK	F E WEICK	F E WEICK	F E WEICK
AIRFOIL SER CODE	1	1	1	1	1	1
DESIGN LIFT COEFF	0.000	0.000	0.000	0.000	0.000	0.000
THICKNESS / CHORD	0.040	0.060	0.080	0.100	0.120	0.140
MACH NUMBER	0.070	0.070	0.070	0.070	0.070	0.070
ZERO LIFT ALPHA	-1.900	-2.600	-3.350	-4.300	-5.250	-6.300
EXTRAP COEFF KCLI	0.000	0.000	0.000	0.000	0.000	0.000
ALPHA, CL VALUES	-1.900 0.000	-2.600 0.000	-3.350 0.000	-4.300 0.000	-5.250 0.000	-6.300 0.000
	0.000 0.200	2.000 0.465	4.000 0.735	4.000 0.800	2.000 0.700	-4.000 0.200
	2.000 0.420	4.000 0.660	6.000 0.925	6.000 0.985	4.000 0.880	-2.000 0.390
	4.000 0.615	6.000 0.850	8.000 1.105	8.000 1.145	6.000 1.055	2.000 0.765
	6.000 0.760	8.000 1.020	10.000 1.200	10.000 1.310	8.000 1.220	4.000 0.935
	8.000 0.860	10.000 1.075	11.000 1.215	11.000 1.395	10.000 1.370	6.000 1.090
	10.000 0.915	12.000 1.040	12.000 1.160	12.000 1.435	12.000 1.470	8.000 1.230
	12.000 0.930	14.000 1.010	14.000 1.105	14.000 1.235	13.200 1.490	10.000 1.355
	14.000 0.925	16.000 0.990	16.000 1.060	16.000 1.095	14.000 1.480	12.000 1.440
	16.000 0.885	0.000 0.000	0.000 0.000	0.000 0.000	16.000 1.240	13.000 1.430
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	14.000 1.385
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	16.000 1.230

AIRFOIL SECTION	USNPS -M16	USNPS -M18	USNPS -M20			
TABLE DATA SOURCE	F E WEICK	F E WEICK	F E WEICK			
AIRFOIL SER CODE	1	1	1	0	0	0
DESIGN LIFT COEFF	0.000	0.000	0.000	0.000	0.000	0.000
THICKNESS / CHORD	0.160	0.180	0.200	0.000	0.000	0.000
MACH NUMBER	0.070	0.070	0.070	0.000	0.000	0.000
ZERO LIFT ALPHA	-7.400	-8.700	-10.200	0.000	0.000	0.000
EXTRAP COEFF KCLI	0.000	0.000	0.000	0.000	0.000	0.000
ALPHA, CL VALUES	-7.400 0.000	-8.700 0.000	-10.200 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	-6.000 0.110	-4.000 0.320	-6.000 0.215	0.000 0.000	0.000 0.000	0.000 0.000
	-4.000 0.270	-2.000 0.490	-4.000 0.345	0.000 0.000	0.000 0.000	0.000 0.000
	-2.000 0.450	0.000 0.650	-2.000 0.490	0.000 0.000	0.000 0.000	0.000 0.000
	2.000 0.795	2.000 0.795	0.000 0.630	0.000 0.000	0.000 0.000	0.000 0.000
	4.000 0.950	4.000 0.950	2.000 0.760	0.000 0.000	0.000 0.000	0.000 0.000
	6.000 1.095	6.000 1.070	4.000 0.890	0.000 0.000	0.000 0.000	0.000 0.000
	8.000 1.210	8.000 1.165	6.000 1.010	0.000 0.000	0.000 0.000	0.000 0.000
	10.000 1.305	10.000 1.240	8.000 1.110	0.000 0.000	0.000 0.000	0.000 0.000
	12.000 1.375	12.000 1.285	10.000 1.165	0.000 0.000	0.000 0.000	0.000 0.000
	13.000 1.365	14.000 1.220	11.000 1.180	0.000 0.000	0.000 0.000	0.000 0.000
	14.000 1.305	16.000 1.135	12.000 1.170	0.000 0.000	0.000 0.000	0.000 0.000
	16.000 1.195	0.000 0.000	14.000 1.135	0.000 0.000	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	16.000 1.060	0.000 0.000	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	CLARK.Y-M06	CLARK.Y-M08	CLARK.Y-M10	CLARK.Y	CLARK.Y-M14	CLARK.Y-M18
TABLE DATA SOURCE	NACA TR-628	NACA TR-628	NACA TR-628	NACA TR-628	NACA TR-628	NACA TR-628
AIRFOIL SER CODE	2	2	2	2	2	2
DESIGN LIFT COEFF	0.000	0.000	0.000	0.000	0.000	0.000
THICKNESS / CHORD	0.060	0.080	0.100	0.117	0.140	0.180
MACH NUMBER	0.060	0.060	0.060	0.060	0.060	0.060
ZERO LIFT ALPHA	-2.950	-3.560	-4.560	-5.000	-6.200	-7.600
EXTRAP COEFF KCLI	0.000	0.000	0.000	0.000	0.000	0.000
ALPHA, CL VALUES	-6.000 -0.300	-6.000 -0.235	-6.000 -0.130	-6.000 -0.095	-6.200 0.000	-7.600 0.000
	-2.950 0.000	-3.560 0.000	-4.560 0.000	-5.000 0.000	2.000 0.800	-6.000 0.160
	4.000 0.680	6.000 0.925	2.000 0.640	-2.000 0.285	4.000 0.985	-4.000 0.350
	6.000 0.865	8.000 1.110	4.000 0.830	6.000 1.030	6.000 1.160	2.000 0.890
	8.000 0.985	10.000 1.280	6.000 1.015	8.000 1.200	8.000 1.315	4.000 1.040
	10.000 1.060	11.500 1.370	8.000 1.195	10.000 1.360	10.000 1.465	6.000 1.185
	12.000 1.070	12.000 1.290	10.000 1.365	12.000 1.485	12.000 1.590	8.000 1.315
	14.000 1.050	14.000 1.170	12.000 1.500	14.000 1.610	14.000 1.720	10.000 1.420
	16.000 1.020	16.000 1.100	14.000 1.635	15.300 1.680	16.000 1.550	12.000 1.470
	0.000 0.000	0.000 0.000	14.800 1.680	16.000 1.500	20.000 1.430	13.000 1.480
	0.000 0.000	0.000 0.000	16.000 1.420	20.000 1.300	0.000 0.000	14.000 1.470
	0.000 0.000	0.000 0.000	20.000 1.220	0.000 0.000	0.000 0.000	16.000 1.430
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	20.000 1.360

AIRFOIL SECTION	CLARK.Y-M22	NACA 16106	NACA 16106	NACA 16106	NACA 16106	NACA 16106
TABLE DATA SOURCE	NACA TR-628	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	
AIRFOIL SER CODE	2	3	3	3	3	3
DESIGN LIFT COEFF	0.000	0.100	0.100	0.100	0.100	0.100
THICKNESS / CHORD	0.220	0.060	0.060	0.060	0.060	0.060
MACH NUMBER	0.060	0.300	0.450	0.600	0.700	0.750
ZERO LIFT ALPHA	-9.290	-1.050	-1.100	-1.000	-1.000	-1.000
EXTRAP COEFF KCLI	0.000	0.760	0.760	0.760	0.760	0.760
ALPHA, CL VALUES	-9.290 0.000	-2.000 -0.085	-2.000 -0.090	-2.000 -0.105	-2.000 -0.130	-2.000 -0.145
	-6.000 0.300	-1.050 0.000	0.000 0.105	0.000 0.110	0.000 0.120	2.000 0.405
	-4.000 0.480	0.000 0.100	2.000 0.305	2.000 0.340	2.000 0.375	0.000 0.000
	0.000 0.800	2.000 0.305	4.000 0.540	4.000 0.600	3.000 0.540	0.000 0.000
	2.000 0.950	4.000 0.510	6.000 0.705	5.000 0.705	3.770 0.725	0.000 0.000
	4.000 1.085	6.000 0.670	8.000 0.820	6.000 0.780	0.000 0.000	0.000 0.000
	6.000 1.185	8.000 0.815	9.000 0.835	0.000 0.000	0.000 0.000	0.000 0.000
	8.000 1.265	9.000 0.850	10.000 0.810	0.000 0.000	0.000 0.000	0.000 0.000
	10.000 1.320	10.000 0.855	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	12.000 1.350	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	13.000 1.360	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	14.000 1.340	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	16.000 1.300	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	20.000 1.240	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16106	NACA 16109	NACA 16109	NACA 16109	NACA 16109	NACA 16109
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.100	0.100	0.100	0.100	0.100	0.100
THICKNESS / CHORD	0.060	0.090	0.090	0.090	0.090	0.090
MACH NUMBER	0.800	0.300	0.450	0.600	0.700	0.750
ZERO LIFT ALPHA	-0.950	-1.000	-1.100	-1.000	-1.000	-0.850
EXTRAP COEFF KCLI	0.760	0.730	0.730	0.730	0.730	0.730
ALPHA, CL VALUES	-2.000 -0.175	-2.000 -0.085	-2.000 -0.080	-2.000 -0.090	-2.000 -0.100	-2.000 -0.110
	-0.950 0.000	-1.000 0.000	0.000 0.095	0.000 0.100	0.000 0.115	-1.000 -0.025
	1.770 0.450	0.000 0.090	2.000 0.295	2.000 0.320	2.000 0.365	0.000 0.125
	0.000 0.000	2.000 0.295	4.000 0.470	4.000 0.500	4.000 0.520	2.000 0.420
	0.000 0.000	4.000 0.465	6.000 0.665	6.000 0.700	0.000 0.000	3.770 0.630
	0.000 0.000	6.000 0.660	8.000 0.785	7.000 0.750	0.000 0.000	0.000 0.000
	0.000 0.000	8.000 0.790	9.000 0.805	8.000 0.780	0.000 0.000	0.000 0.000
	0.000 0.000	10.000 0.835	10.000 0.795	10.000 0.790	0.000 0.000	0.000 0.000
	0.000 0.000	11.000 0.850	11.000 0.775	12.000 0.785	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	12.000 0.755	0.000 0.000	0.000 0.000	0.000 0.000

AIRFOIL SECTION	NACA 16109	NACA 16115	NACA 16115	NACA 16115	NACA 16115	NACA 16130
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.100	0.100	0.100	0.100	0.100	0.100
THICKNESS / CHORD	0.090	0.150	0.150	0.150	0.150	0.300
MACH NUMBER	0.800	0.300	0.450	0.600	0.700	0.300
ZERO LIFT ALPHA	-1.000	-0.800	-0.800	-0.800	-0.800	0.500
EXTRAP COEFF KCLI	0.730	0.600	0.600	0.600	0.600	-0.190
ALPHA, CL VALUES	-2.000 -0.130	-2.000 -0.110	-2.000 -0.110	-2.000 -0.115	-2.000 -0.130	-2.000 -0.110
	-1.000 0.000	-0.800 0.000	0.000 0.080	0.000 0.085	-0.800 0.000	0.000 -0.025
	1.770 0.360	0.000 0.080	2.000 0.260	2.000 0.280	0.000 0.085	0.500 0.000
	0.000 0.000	2.000 0.260	4.000 0.330	4.000 0.355	2.000 0.295	2.000 0.100
	0.000 0.000	4.000 0.350	5.000 0.380	5.000 0.405	3.770 0.400	4.000 0.195
	0.000 0.000	5.000 0.390	6.000 0.445	6.000 0.470	0.000 0.000	6.000 0.245
	0.000 0.000	6.000 0.445	8.000 0.620	8.000 0.620	0.000 0.000	8.000 0.260
	0.000 0.000	7.000 0.515	10.000 0.790	10.000 0.860	0.000 0.000	10.000 0.295
	0.000 0.000	8.000 0.620	11.000 0.800	11.000 0.830	0.000 0.000	11.770 0.345
	0.000 0.000	10.000 0.785	11.770 0.780	11.770 0.690	0.000 0.000	0.000 0.000
	0.000 0.000	11.770 0.855	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16130	NACA 16130	NACA 16306	NACA 16306	NACA 16306	NACA 16306
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.100	0.100	0.300	0.300	0.300	0.300
THICKNESS / CHORD	0.300	0.300	0.060	0.060	0.060	0.060
MACH NUMBER	0.450	0.600	0.300	0.450	0.600	0.700
ZERO LIFT ALPHA	0.500	-0.500	-2.400	-2.400	-2.300	-2.300
EXTRAP COEFF KCLI	-0.190	-0.190	0.760	0.760	0.760	0.760
ALPHA, CL VALUES	-2.000 -0.140	-2.000 -0.155	-4.000 -0.190	-4.000 -0.190	-4.000 -0.220	-4.000 -0.260
	0.000 -0.030	-0.500 0.000	-2.400 0.000	-2.000 0.045	-2.000 0.035	3.770 0.920
	2.000 0.080	0.000 0.030	-2.000 0.045	0.000 0.270	0.000 0.295	0.000 0.000
	4.000 0.160	2.000 0.030	0.000 0.265	2.000 0.490	1.000 0.420	0.000 0.000
	6.000 0.180	4.000 0.085	2.000 0.490	4.000 0.675	2.000 0.540	0.000 0.000
	8.000 0.185	6.000 0.105	4.000 0.640	6.000 0.865	3.000 0.650	0.000 0.000
	10.000 0.235	8.000 0.135	6.000 0.800	8.000 0.965	4.000 0.780	0.000 0.000
	11.770 0.290	9.770 0.190	8.000 0.960	9.000 0.980	5.000 0.895	0.000 0.000
	0.000 0.000	0.000 0.000	9.770 1.010	9.770 0.980	6.000 0.970	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	7.770 1.010	0.000 0.000

AIRFOIL SECTION	NACA 16306	NACA 16309	NACA 16309	NACA 16309	NACA 16309	NACA 16309
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.300	0.300	0.300	0.300	0.300	0.300
THICKNESS / CHORD	0.060	0.090	0.090	0.090	0.090	0.090
MACH NUMBER	0.750	0.300	0.450	0.600	0.700	0.750
ZERO LIFT ALPHA	-2.200	-2.450	-2.600	-2.500	-2.500	-2.350
EXTRAP COEFF KCLI	0.760	0.730	0.730	0.730	0.730	0.730
ALPHA, CL VALUES	-4.000 -0.340	-4.000 -0.155	-4.000 -0.160	-4.000 -0.190	-4.000 -0.190	-4.000 -0.285
	-2.200 0.000	-2.450 0.000	-2.000 0.055	-2.000 0.050	0.000 0.320	-2.350 0.000
	0.000 0.360	-2.000 0.045	0.000 0.250	0.000 0.270	3.770 0.890	1.770 0.650
	1.770 0.730	0.000 0.240	2.000 0.455	2.000 0.490	0.000 0.000	0.000 0.000
	0.000 0.000	2.000 0.450	4.000 0.620	4.000 0.690	0.000 0.000	0.000 0.000
	0.000 0.000	4.000 0.610	6.000 0.800	6.000 0.900	0.000 0.000	0.000 0.000
	0.000 0.000	6.000 0.750	8.000 0.930	8.000 1.080	0.000 0.000	0.000 0.000
	0.000 0.000	8.000 0.905	9.000 0.955	9.000 1.070	0.000 0.000	0.000 0.000
	0.000 0.000	9.000 0.950	10.000 0.950	9.770 0.950	0.000 0.000	0.000 0.000
	0.000 0.000	10.000 0.975	11.770 0.920	0.000 0.000	0.000 0.000	0.000 0.000
	0.000 0.000	11.000 0.980	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000
	0.000 0.000	11.770 0.960	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16312	NACA 16312	NACA 16312	NACA 16312	NACA 16315	NACA 16315
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.300	0.300	0.300	0.300	0.300	0.300
THICKNESS / CHORD	0.120	0.120	0.120	0.120	0.150	0.150
MACH NUMBER	0.300	0.450	0.600	0.700	0.300	0.450
ZERO LIFT ALPHA	-2.600	-2.800	-2.800	-3.000	-2.100	-3.000
EXTRAP COEFF KCLI	0.690	0.690	0.690	0.690	0.600	0.600
ALPHA, CL VALUES	-4.000 -0.110	-4.000 -0.100	-4.000 -0.105	-4.000 -0.080	-4.000 -0.070	-4.000 -0.045
	-2.600 0.000	-2.000 0.065	-2.000 0.080	-3.000 0.000	-3.000 -0.040	-2.000 0.050
	0.000 0.215	0.000 0.230	0.000 0.265	0.000 0.285	-2.100 0.000	-1.000 0.110
	2.000 0.410	2.000 0.420	2.000 0.490	2.000 0.505	0.000 0.195	0.000 0.200
	3.000 0.485	4.000 0.550	3.000 0.570	3.770 0.705	2.000 0.370	2.000 0.400
	4.000 0.540	5.000 0.620	4.000 0.615	0.000 0.000	4.000 0.515	4.000 0.520
	6.000 0.700	6.000 0.715	5.000 0.680	0.000 0.000	5.000 0.540	6.000 0.580
	8.000 0.835	8.000 0.900	6.000 0.770	0.000 0.000	6.000 0.570	7.000 0.650
	10.000 0.935	9.000 0.940	7.770 0.940	0.000 0.000	7.000 0.655	8.000 0.750
	11.000 0.965	9.770 0.950	0.000 0.000	0.000 0.000	8.000 0.760	10.000 0.890
	11.770 0.960	0.000 0.000	0.000 0.000	0.000 0.000	10.000 0.885	11.770 0.960
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	11.000 0.915	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	11.770 0.930	0.000 0.000

AIRFOIL SECTION	NACA 16315	NACA 16315	NACA 16321	NACA 16321	NACA 16321	NACA 16506
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.300	0.300	0.300	0.300	0.300	0.500
THICKNESS / CHORD	0.150	0.150	0.210	0.210	0.210	0.060
MACH NUMBER	0.600	0.700	0.300	0.450	0.600	0.300
ZERO LIFT ALPHA	-3.600	-3.750	-1.300	-1.300	-1.100	-3.900
EXTRAP COEFF KCLI	0.600	0.600	0.370	0.370	0.370	0.760
ALPHA, CL VALUES	-4.000 -0.020	-3.750 0.000	-4.000 -0.100	-4.000 -0.030	-4.000 0.000	-3.900 0.000
	-2.000 0.075	-2.000 0.085	-3.000 -0.090	-3.000 -0.070	-2.340 -0.100	1.000 0.525
	-1.000 0.115	-1.000 0.130	-2.000 -0.060	-2.000 -0.060	-1.100 0.000	2.000 0.625
	0.000 0.200	0.000 0.190	-1.300 0.000	2.000 0.255	3.000 0.345	4.000 0.740
	2.000 0.415	1.000 0.290	2.000 0.270	3.000 0.330	4.000 0.410	6.000 0.895
	3.000 0.515	2.000 0.440	4.000 0.425	4.000 0.400	6.000 0.430	8.000 1.050
	4.000 0.570	3.000 0.580	5.000 0.460	8.000 0.495	8.000 0.480	9.000 1.090
	6.000 0.620	4.000 0.725	6.000 0.485	11.770 0.680	10.000 0.580	0.000 0.000
	8.000 0.790	7.000 0.725	8.000 0.500	0.000 0.000	11.770 0.790	0.000 0.000
	10.000 0.940	7.770 0.740	11.770 0.720	0.000 0.000	0.000 0.000	0.000 0.000
	11.770 1.050	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16506	NACA 16506	NACA 16506	NACA 16506	NACA 16509	NACA 16509
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.500	0.500	0.500	0.500	0.500	0.500
THICKNESS / CHORD	0.060	0.060	0.060	0.060	0.090	0.090
MACH NUMBER	0.450	0.600	0.700	0.750	0.300	0.450
ZERO LIFT ALPHA	-3.900	-3.700	-3.500	-3.400	-4.200	-4.250
EXTRAP COEFF KCLI	0.760	0.760	0.760	0.760	0.730	0.730
ALPHA, CL VALUES	-4.000 -0.010	-4.000 -0.040	-4.000 -0.080	-3.400 0.000	-4.200 0.000	-4.000 0.025
	2.000 0.660	1.000 0.605	-2.000 0.230	-2.000 0.240	2.000 0.615	2.000 0.645
	3.000 0.720	2.000 0.730	1.770 0.830	0.000 0.595	4.000 0.710	4.000 0.730
	4.000 0.795	3.770 0.765	0.000 0.000	1.770 0.850	7.770 0.990	5.000 0.805
	6.000 0.970	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	6.000 0.905
	7.770 1.085	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	7.770 1.040

AIRFOIL SECTION	NACA 16509	NACA 16509	NACA 16509	NACA 16512	NACA 16512	NACA 16512
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.500	0.500	0.500	0.500	0.500	0.500
THICKNESS / CHORD	0.090	0.090	0.090	0.120	0.120	0.120
MACH NUMBER	0.600	0.700	0.750	0.300	0.450	0.600
ZERO LIFT ALPHA	-4.200	-4.200	-4.100	-4.200	-4.250	-4.200
EXTRAP COEFF KCLI	0.730	0.730	0.730	0.690	0.690	0.690
ALPHA, CL VALUES	-4.000 0.020	-5.000 -0.085	-4.100 0.000	-4.200 0.000	-4.000 0.025	-4.000 0.025
	1.000 0.605	-4.000 0.020	0.000 0.560	2.000 0.545	2.000 0.570	0.000 0.420
	2.000 0.720	1.770 0.780	1.770 0.750	4.000 0.700	3.000 0.645	2.000 0.620
	4.000 0.810	0.000 0.000	0.000 0.000	6.000 0.770	4.000 0.700	3.000 0.705
	5.770 0.930	0.000 0.000	0.000 0.000	8.000 0.905	6.000 0.790	4.000 0.775
	0.000 0.000	0.000 0.000	0.000 0.000	10.000 1.010	7.770 0.910	5.770 0.840
	0.000 0.000	0.000 0.000	0.000 0.000	11.770 1.070	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16512	NACA 16512	NACA 16515	NACA 16515	NACA 16515	NACA 16515
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.500	0.500	0.500	0.500	0.500	0.500
THICKNESS / CHORD	0.120	0.120	0.150	0.150	0.150	0.150
MACH NUMBER	0.700	0.750	0.300	0.450	0.600	0.700
ZERO LIFT ALPHA	-4.100	-3.600	-4.400	-4.500	-4.600	-4.700
EXTRAP COEFF KCLI	0.690	0.690	0.600	0.600	0.600	0.600
ALPHA, CL VALUES	-4.000 0.005	-3.600 0.000	-4.400 0.000	-4.000 0.040	-4.000 0.045	-4.000 0.065
	-2.000 0.225	-2.000 0.200	-4.000 0.025	-2.000 0.195	-2.000 0.220	-2.000 0.235
	0.000 0.440	0.000 0.380	-2.000 0.155	0.000 0.325	0.000 0.325	-1.000 0.265
	2.000 0.675	2.000 0.530	0.000 0.305	2.000 0.520	2.000 0.525	0.000 0.315
	3.000 0.780	3.770 0.680	2.000 0.495	3.000 0.600	4.000 0.720	2.000 0.530
	3.770 0.825	0.000 0.000	4.000 0.660	4.000 0.655	6.000 0.780	3.770 0.710
	0.000 0.000	0.000 0.000	6.000 0.710	6.000 0.715	7.000 0.840	0.000 0.000
	0.000 0.000	0.000 0.000	8.000 0.815	8.000 0.825	7.770 0.915	0.000 0.000
	0.000 0.000	0.000 0.000	10.000 0.920	10.000 0.940	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	12.000 1.010	12.000 1.025	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	13.770 1.030	13.770 1.010	0.000 0.000	0.000 0.000

AIRFOIL SECTION	NACA 16515	NACA 16521	NACA 16521	NACA 16521	NACA 16521	NACA 16530
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.500	0.500	0.500	0.500	0.500	0.500
THICKNESS / CHORD	0.150	0.210	0.210	0.210	0.210	0.300
MACH NUMBER	0.750	0.300	0.450	0.600	0.700	0.300
ZERO LIFT ALPHA	-1.600	-2.300	-2.000	-1.800	-0.400	1.500
EXTRAP COEFF KCLI	0.600	0.370	0.370	0.370	0.370	-0.190
ALPHA, CL VALUES	-2.000 -0.045	-2.000 0.030	-2.000 0.070	-2.000 0.120	-2.000 -0.210	0.000 -0.100
	-1.600 0.000	3.000 0.455	-1.000 0.000	-1.000 0.120	-1.000 -0.080	6.000 0.300
	0.000 0.145	4.000 0.540	0.000 0.180	0.000 0.170	-0.400 0.000	8.000 0.405
	1.000 0.210	5.000 0.605	2.000 0.360	3.000 0.440	0.000 0.035	9.770 0.470
	2.000 0.295	6.000 0.650	4.000 0.520	4.000 0.530	2.000 0.020	0.000 0.000
	3.770 0.480	8.000 0.685	6.000 0.620	5.000 0.590	3.000 0.190	0.000 0.000
	0.000 0.000	10.000 0.750	8.000 0.670	6.000 0.610	3.770 0.270	0.000 0.000
	0.000 0.000	11.770 0.860	11.770 0.840	7.000 0.660	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	8.000 0.750	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	9.770 0.910	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16530	NACA 16530	NACA 16709	NACA 16709	NACA 16709	NACA 16709
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.500	0.500	0.700	0.700	0.700	0.700
THICKNESS / CHORD	0.300	0.300	0.090	0.090	0.090	0.090
MACH NUMBER	0.450	0.600	0.300	0.450	0.600	0.700
ZERO LIFT ALPHA	2.200	3.200	-5.400	-5.500	-5.500	-5.600
EXTRAP COEFF KCLI	-0.190	-0.190	0.730	0.730	0.730	0.730
ALPHA, CL VALUES	0.000 -0.120	0.000 -0.280	-6.000 -0.065	-6.000 -0.060	-6.000 -0.050	-6.000 -0.020
	2.000 -0.015	3.200 0.000	-5.400 0.000	-4.000 0.145	-4.000 0.150	-5.000 0.040
	4.000 0.115	4.000 0.070	-4.000 0.140	0.000 0.580	-2.000 0.395	-4.000 0.150
	6.000 0.245	6.000 0.120	-2.000 0.345	1.000 0.690	0.000 0.645	-2.000 0.445
	8.000 0.355	8.000 0.280	0.000 0.550	2.000 0.780	2.000 0.855	0.000 0.750
	9.770 0.415	9.770 0.430	2.000 0.755	4.000 0.880	4.000 1.010	2.000 0.915
	0.000 0.000	0.000 0.000	3.000 0.815	6.000 1.005	5.770 1.110	0.000 0.000
	0.000 0.000	0.000 0.000	4.000 0.850	8.000 1.140	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	5.000 0.895	9.000 1.200	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	6.000 0.965	9.770 1.210	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	7.770 1.080	0.000 0.000	0.000 0.000	0.000 0.000

AIRFOIL SECTION	NACA 16709	NACA 16709	NACA 16712	NACA 16712	NACA 16712	NACA 16712
TABLE DATA SOURCE	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546	NACA TN-1546
AIRFOIL SER CODE	3	3	3	3	3	3
DESIGN LIFT COEFF	0.700	0.700	0.700	0.700	0.700	0.700
THICKNESS / CHORD	0.090	0.090	0.120	0.120	0.120	0.120
MACH NUMBER	0.750	0.775	0.300	0.450	0.600	0.700
ZERO LIFT ALPHA	-5.400	-3.500	-5.500	-5.600	-6.000	-6.000
EXTRAP COEFF KCLI	0.730	0.730	0.690	0.690	0.690	0.690
ALPHA, CL VALUES	-4.000 0.150	-4.000 -0.100	-6.000 -0.040	-6.000 -0.035	-6.000 0.000	-4.000 0.150
	-2.000 0.375	-3.500 0.000	-4.000 0.120	-4.000 0.130	-5.000 0.060	-2.000 0.365
	0.000 0.585	-2.000 0.205	0.000 0.495	-2.000 0.340	-4.000 0.150	0.000 0.575
	1.000 0.620	0.000 0.460	2.000 0.685	2.000 0.740	2.000 0.825	1.770 0.760
	2.000 0.670	1.770 0.535	4.000 0.885	4.000 0.910	3.770 1.040	0.000 0.000
	3.770 0.820	0.000 0.000	6.000 0.930	6.000 0.960	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	8.000 1.025	8.000 1.085	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	10.000 1.140	9.770 1.175	0.000 0.000	0.000 0.000
	0.000 0.000	0.000 0.000	11.770 1.210	0.000 0.000	0.000 0.000	0.000 0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 16715		NACA 16715		NACA 16715							
TABLE DATA SOURCE	NACA TN-1546		NACA TN-1546		NACA TN-1546							
AIRFOIL SER CODE	3		3		3		0		0		0	
DESIGN LIFT COEFF	0.700		0.700		0.700		0.000		0.000		0.000	
THICKNESS / CHORD	0.150		0.150		0.150		0.000		0.000		0.000	
MACH NUMBER	0.300		0.450		0.600		0.000		0.000		0.000	
ZERO LIFT ALPHA	-5.400		-5.500		-5.400		0.000		0.000		0.000	
EXTRAP COEFF KCLI	0.600		0.600		0.600		0.000		0.000		0.000	
ALPHA, CL VALUES	-6.000	-0.045	-6.000	-0.050	-6.000	-0.040	0.000	0.000	0.000	0.000	0.000	0.000
	-5.400	0.000	-4.000	0.130	-5.400	0.000	0.000	0.000	0.000	0.000	0.000	0.000
	-4.000	0.110	-2.000	0.300	-4.000	0.120	0.000	0.000	0.000	0.000	0.000	0.000
	-2.000	0.295	0.000	0.460	-2.000	0.325	0.000	0.000	0.000	0.000	0.000	0.000
	0.000	0.445	2.000	0.660	-1.000	0.390	0.000	0.000	0.000	0.000	0.000	0.000
	2.000	0.625	4.000	0.850	0.000	0.470	0.000	0.000	0.000	0.000	0.000	0.000
	4.000	0.800	5.000	0.895	2.000	0.720	0.000	0.000	0.000	0.000	0.000	0.000
	5.000	0.865	6.000	0.915	3.000	0.840	0.000	0.000	0.000	0.000	0.000	0.000
	6.000	0.905	8.000	0.975	4.000	0.930	0.000	0.000	0.000	0.000	0.000	0.000
	8.000	0.950	9.770	1.090	6.000	1.050	0.000	0.000	0.000	0.000	0.000	0.000
	10.000	1.050	0.000	0.000	7.770	1.150	0.000	0.000	0.000	0.000	0.000	0.000
	11.770	1.110	0.000	0.000	9.770	1.280	0.000	0.000	0.000	0.000	0.000	0.000

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 64-006	NACA 64-009	NACA 64-012	NACA 64-015	NACA 64-018	NACA 64-021
TABLE DATA SOURCE	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824
AIRFOIL SER CODE	4	4	4	4	4	4
DESIGN LIFT COEFF	0.000	0.000	0.000	0.000	0.000	0.000
THICKNESS / CHORD	0.060	0.090	0.120	0.150	0.180	0.210
MACH NUMBER	0.150	0.150	0.150	0.150	0.150	0.150
ZERO LIFT ALPHA	0.000	0.000	0.000	0.000	0.000	0.000
EXTRAP COEFF KCLI	0.740	0.740	0.740	0.740	0.740	0.740
ALPHA, CL VALUES	-4.000 -0.430	-6.000 -0.700	-6.000 -0.650	-6.000 -0.670	-6.000 -0.650	-6.000 -0.630
	4.000 0.430	0.000 0.000	6.000 0.650	6.000 0.670	6.000 0.650	6.000 0.630
	6.000 0.620	6.000 0.620	8.000 0.830	8.000 0.870	8.000 0.840	8.000 0.800
	8.000 0.790	8.000 0.770	10.000 0.930	10.000 1.000	10.000 0.980	10.000 0.900
	10.000 0.810	10.000 0.870	12.000 0.910	12.000 1.100	12.000 1.040	12.000 0.980
	12.000 0.770	12.000 0.890	14.000 0.850	14.000 0.900	14.000 1.070	14.000 1.010
	0.000 0.000	14.000 0.870	16.000 0.800	0.000 0.000	16.000 1.040	16.000 1.030
	0.000 0.000	16.000 0.800	0.000 0.000	0.000 0.000	17.200 0.640	17.000 1.030
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	18.000 0.900
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	19.000 0.700
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	20.000 0.680

AIRFOIL SECTION	NACA 64-206	NACA 64-209	NACA 64-212	NACA 64-215	NACA 64-218	NACA 64-221
TABLE DATA SOURCE	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824
AIRFOIL SER CODE	4	4	4	4	4	4
DESIGN LIFT COEFF	0.200	0.200	0.200	0.200	0.200	0.200
THICKNESS / CHORD	0.060	0.090	0.120	0.150	0.180	0.210
MACH NUMBER	0.150	0.150	0.150	0.150	0.150	0.150
ZERO LIFT ALPHA	-1.330	-1.400	-1.240	-1.440	-1.220	-1.260
EXTRAP COEFF KCLI	0.740	0.740	0.740	0.740	0.740	0.740
ALPHA, CL VALUES	-6.000 -0.490	-6.000 -0.490	-6.000 -0.520	-6.000 -0.490	-6.000 -0.530	-6.000 -0.540
	-1.330 0.000	-1.400 0.000	-1.240 0.000	-1.440 0.000	-1.220 0.000	-1.260 0.000
	4.000 0.560	6.000 0.790	6.000 0.790	6.000 0.800	6.000 0.800	4.000 0.600
	6.000 0.770	8.000 0.970	8.000 0.980	8.000 0.990	8.000 0.930	6.000 0.800
	8.000 0.900	10.000 1.050	10.000 1.125	10.000 1.120	10.000 1.060	8.000 0.940
	10.000 1.000	12.000 1.040	11.000 1.160	12.000 1.200	12.000 1.135	10.000 1.030
	12.000 1.010	14.000 1.010	12.000 1.130	13.000 1.200	14.000 1.180	12.000 1.090
	14.000 0.970	16.000 0.960	14.000 1.010	14.000 1.150	16.000 1.200	14.000 1.130
	16.000 0.910	18.000 0.860	0.000 0.000	16.000 1.020	17.000 1.200	16.000 1.140
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	18.000 0.790	18.000 1.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	20.000 0.760	20.000 0.920

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 64-409	NACA 64-412	NACA 64-415	NACA 64-418	NACA 64-421	
TABLE DATA SOURCE	NACA TN-1945	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	
AIRFOIL SER CODE	4	4	4	4	4	0
DESIGN LIFT COEFF	0.400	0.400	0.400	0.400	0.400	0.000
THICKNESS / CHORD	0.090	0.120	0.150	0.180	0.210	0.000
MACH NUMBER	0.150	0.150	0.150	0.150	0.150	0.000
ZERO LIFT ALPHA	-2.540	-2.860	-2.820	-2.800	-2.570	0.000
EXTRAP COEFF KCLI	0.740	0.740	0.740	0.740	0.740	0.000
ALPHA, CL VALUES	-6.000 -0.360	-6.000 -0.340	-6.000 -0.360	-6.000 -0.360	-6.000 -0.400	0.000 0.000
	-2.540 0.000	-2.860 0.000	0.000 0.320	-2.800 0.000	-2.570 0.000	0.000 0.000
	6.000 0.890	6.000 0.960	6.000 0.950	2.000 0.540	0.000 0.300	0.000 0.000
	8.000 1.040	8.000 1.150	8.000 1.110	4.000 0.740	2.000 0.530	0.000 0.000
	10.000 1.110	10.000 1.290	10.000 1.220	6.000 0.930	4.000 0.720	0.000 0.000
	11.000 1.120	11.000 1.340	12.000 1.290	8.000 1.080	6.000 0.890	0.000 0.000
	12.000 1.090	12.000 1.340	13.000 1.300	10.000 1.170	8.000 1.000	0.000 0.000
	14.000 1.020	14.000 1.220	14.000 1.290	12.000 1.230	10.000 1.070	0.000 0.000
	15.000 0.960	16.000 1.100	16.000 1.240	14.000 1.240	12.000 1.120	0.000 0.000
	0.000 0.000	0.000 0.000	18.000 1.050	16.000 1.230	14.000 1.160	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	18.000 1.170	16.000 1.180	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	18.000 1.180	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	20.000 1.000	0.000 0.000

AIRFOIL SECTION	NACA 65-006	NACA 65-009	NACA 65-012	NACA 65-015	NACA 65-018	NACA 65-021
TABLE DATA SOURCE	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824
AIRFOIL SER CODE	5	5	5	5	5	5
DESIGN LIFT COEFF	0.000	0.000	0.000	0.000	0.000	0.000
THICKNESS / CHORD	0.060	0.090	0.120	0.150	0.180	0.210
MACH NUMBER	0.150	0.150	0.150	0.150	0.150	0.150
ZERO LIFT ALPHA	0.000	0.000	0.000	0.000	0.000	0.000
EXTRAP COEFF KCLI	0.740	0.740	0.740	0.740	0.740	0.740
ALPHA, CL VALUES	-6.000 -0.640	-6.000 -0.630	-6.000 -0.650	-6.000 -0.630	-6.000 -0.600	-6.000 -0.580
	0.000 0.000	6.000 0.630	0.000 0.000	-4.000 -0.440	-4.000 -0.420	-4.000 -0.390
	4.000 0.420	8.000 0.820	6.000 0.670	6.000 0.660	0.000 0.000	0.000 0.000
	6.000 0.620	10.000 0.900	8.000 0.870	8.000 0.840	6.000 0.620	4.000 0.400
	8.000 0.770	11.000 0.920	10.000 0.970	10.000 1.000	8.000 0.800	6.000 0.590
	10.000 0.870	12.000 0.910	12.000 0.950	11.000 1.030	10.000 0.930	8.000 0.740
	12.000 0.920	14.000 0.850	14.000 0.900	12.000 1.020	12.000 1.020	10.000 0.850
	14.000 0.880	0.000 0.000	16.000 0.840	14.000 0.880	14.000 1.070	12.000 0.930
	16.000 0.790	0.000 0.000	0.000 0.000	16.000 0.790	16.000 0.900	14.000 1.020
	0.000 0.000	0.000 0.000	0.000 0.000	18.000 0.680	0.000 0.000	16.000 1.070
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	18.000 1.080
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	20.000 0.860

PROPELLER BLADE SECTION AIRFOIL TABLES

AIRFOIL SECTION	NACA 65-206	NACA 65-209	NACA 65-212	NACA 65-215	NACA 65-218	NACA 65-221
TABLE DATA SOURCE	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824
AIRFOIL SER CODE	5	5	5	5	5	5
DESIGN LIFT COEFF	0.200	0.200	0.200	0.200	0.200	0.200
THICKNESS / CHORD	0.060	0.090	0.120	0.150	0.180	0.210
MACH NUMBER	0.150	0.150	0.150	0.150	0.150	0.150
ZERO LIFT ALPHA	-1.330	-1.280	-1.180	-1.250	-1.270	-1.530
EXTRAP COEFF KCLI	0.740	0.740	0.740	0.740	0.740	0.740
ALPHA, CL VALUES	-6.000 -0.490	-6.000 -0.500	-6.000 -0.500	-6.000 -0.500	-6.000 -0.470	-6.000 -0.440
	-1.330 0.000	-1.280 0.000	-4.000 -0.310	-4.000 -0.300	-4.000 -0.280	-4.000 -0.250
	4.000 0.560	4.000 0.560	-1.180 0.000	-1.250 0.000	-1.270 0.000	-1.530 0.000
	6.000 0.760	6.000 0.770	6.000 0.790	4.000 0.570	4.000 0.540	4.000 0.560
	8.000 0.900	8.000 0.900	8.000 0.960	6.000 0.770	6.000 0.730	6.000 0.710
	10.000 0.990	10.000 0.990	10.000 1.070	8.000 0.960	8.000 0.870	10.000 0.930
	12.000 1.000	11.000 1.000	12.000 1.060	10.000 1.070	10.000 0.970	12.000 1.040
	14.000 0.960	12.000 0.980	14.000 1.000	12.000 1.130	12.000 1.060	14.000 1.100
	16.000 0.900	14.000 0.930	0.000 0.000	14.000 1.090	14.000 1.110	16.000 1.130
	0.000 0.000	0.000 0.000	0.000 0.000	16.000 0.990	15.000 1.130	17.000 1.140
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	16.000 1.120	18.000 1.130
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	18.000 0.720	20.000 0.900

AIRFOIL SECTION	NACA 65-410	NACA 65-412	NACA 65-415	NACA 65-418	NACA 65-421	
TABLE DATA SOURCE	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	NACA TR-824	
AIRFOIL SER CODE	5	5	5	5	5	0
DESIGN LIFT COEFF	0.400	0.400	0.400	0.400	0.400	0.000
THICKNESS / CHORD	0.100	0.120	0.150	0.180	0.210	0.000
MACH NUMBER	0.150	0.150	0.150	0.150	0.150	0.000
ZERO LIFT ALPHA	-2.370	-2.660	-2.640	-2.450	-2.490	0.000
EXTRAP COEFF KCLI	0.740	0.740	0.740	0.740	0.740	0.000
ALPHA, CL VALUES	-6.000 -0.390	-6.000 -0.360	-6.000 -0.360	-6.000 -0.370	-6.000 -0.360	0.000 0.000
	-2.370 0.000	4.000 0.720	4.000 0.710	4.000 0.670	-2.490 0.000	0.000 0.000
	6.000 0.900	6.000 0.920	6.000 0.890	6.000 0.820	2.000 0.460	0.000 0.000
	8.000 1.090	8.000 1.100	8.000 1.060	8.000 0.970	4.000 0.630	0.000 0.000
	10.000 1.260	10.000 1.250	10.000 1.170	10.000 1.070	6.000 0.770	0.000 0.000
	12.000 1.180	11.000 1.310	12.000 1.220	12.000 1.140	8.000 0.890	0.000 0.000
	14.000 1.100	12.000 1.300	14.000 1.240	14.000 1.200	10.000 0.990	0.000 0.000
	0.000 0.000	14.000 1.220	15.200 1.070	15.000 1.220	12.000 1.080	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	16.000 1.200	14.000 1.150	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	18.000 1.120	16.000 1.180	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	22.000 1.000	18.000 1.190	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	20.000 1.180	0.000 0.000
	0.000 0.000	0.000 0.000	0.000 0.000	0.000 0.000	22.000 0.850	0.000 0.000

APPENDIX C - PROGRAM USER INSTRUCTIONS

1.0 INTRODUCTION

This appendix contains a guide to the setting up and running of the computer program. The essential computational steps are described in Section 4, and the underlying theory in Section 3 of this report, and Section 3 of Reference 1. It should be noted that whereas this program is an extension and modification of the computer program developed in Reference 1, considerable differences exist between the two programs. Users of the old program are, therefore, cautioned against attempting modification of the former program without a careful study of the present program layout and data format requirements.

2.0 PROGRAM LANGUAGE

The program is written entirely in Fortran IV, Version 2.3 for a Scope 3.1 operating system and library tape.

3.0 MACHINE REQUIREMENTS

The program is designed to run on a CDC-6600 computer. The program makes use of the overlay capability and requires 15 disc files and a tape unit for peripheral storage.

4.0 INPUT DATA

The input data required by the program to compute a series of solutions at specified angles of attack up to stall for each wing-fuselage-propeller configuration constitutes one input case. This input data must include tabulations of applicable wing-section and propeller blade-section aerodynamic characteristics, propeller tip loss correction factors, and a range of geometric and flight condition parameters together with specification of several computational control and sequence options.

The input data for each case is in the form of punched cards arranged in sequential groups as shown in Figure 30. The principal card groups are identified as follows:

CARD GROUP	DESCRIPTION
A	Wing-fuselage geometry and control and sequencing options
B	Wing-section aerodynamic data
C	Wing geometry if not straight-tapered
D	Fuselage angles of attack
E	Propeller tip loss factors
F	Propeller blade-section aerodynamic data
G	Propeller geometry and operating conditions

In general, the first case requires a full specification of the input data contained in each group. However, for the second and subsequent cases, card groups specifying tabulated data (Groups B, C, E, and F) may be omitted where there is no change in the input data requirements. See Figure 30.

4.1 Wing Fuselage Geometry (Card Group A)

Wing-fuselage geometry is entered on the first three cards as follows:

CARD	COL.LOC	VARIABLE TYPE	PROGRAM NAME	VARIABLE	DESCRIPTION
1	1-10	Real	ASPEC	AR	Wing aspect ratio
	11-20	Real	TAUT	$(t/c)_T$	Wing tip thickness/ chord ratio
	21-30	Real	TAUR	$(t/c)_R$	Wing root thickness/ chord ratio

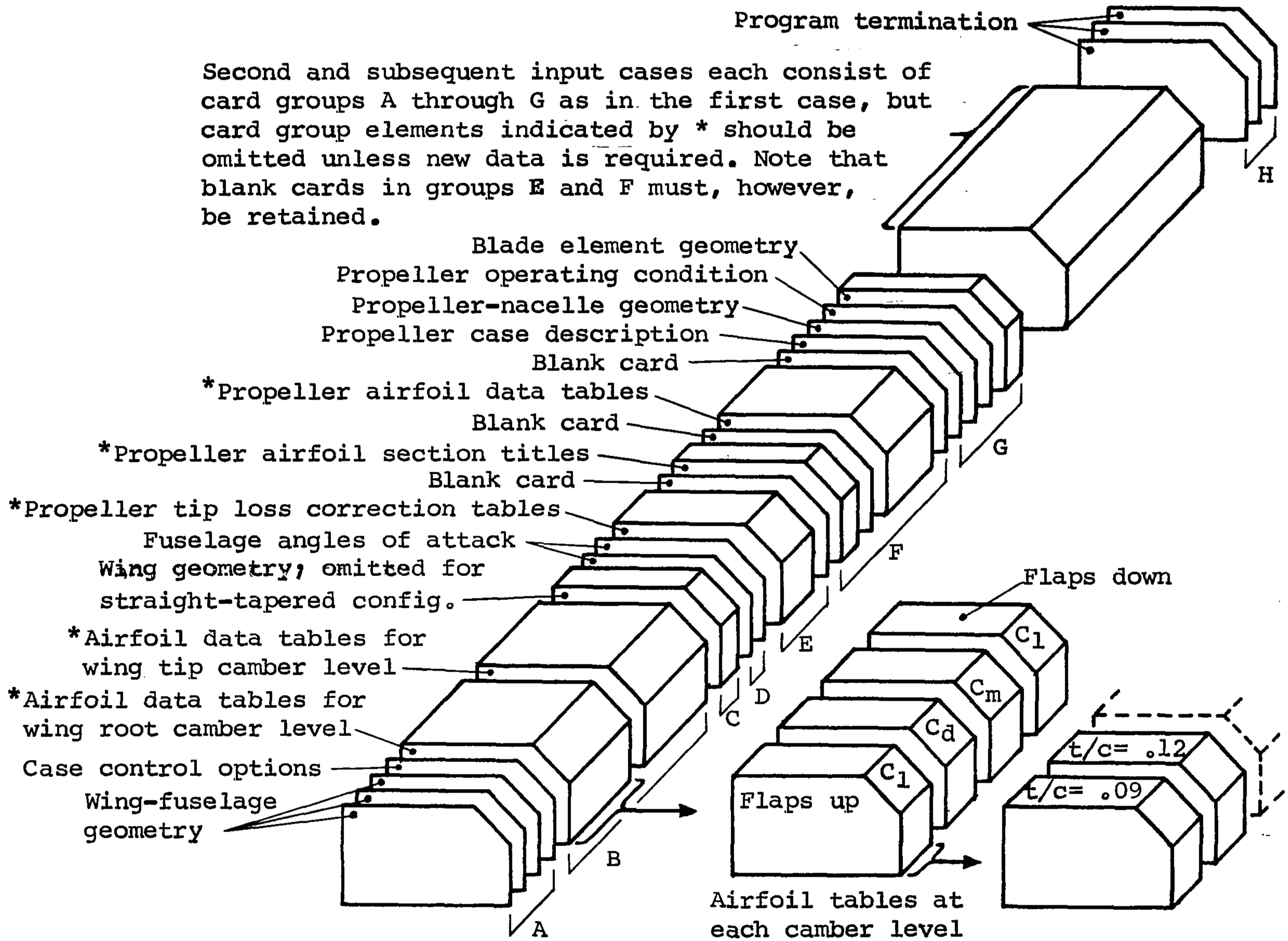

Figure 30. Assembly of Computer Program Input Data Card Deck

CARD	COL.LOC	VARIABLE TYPE	PROGRAM NAME	VARIABLE	DESCRIPTION
1	31-40	Real	TAPER	λ	Wing taper ratio
	41-50	Real	TWIST	ϵ_g	Wing geometric twist (If geometric twist is specified, TWISA on card 2 must be set to 100.0)
	51-60	Real	R	r	Number of spanwise stations
	61-70	Real	BF	b_f/b	Flap span/wing span
	71-80	Real	REYND	R_e'	Reynolds number in millions based on wing mean aerodynamic chord
2	1-10	Real	DISCR	Δ	Criterion for convergence of iteration loop
	11-20	Real	A	A	Fuselage semi-height/ wing semi-span
	21-30	Real	B	B	Fuselage semi-width/ wing semi-span
	31-40	Real	H	H	Height of wing above fuselage centerline/ wing semi-span
	41-50	Real	ALPHR	α_R	Wing/body incidence - degrees
	60	Integer	NFLAP	--	Flap indicator: if NFLAP = 1, Flap is deflected; if NFLAP = 0, Flap is undeflected

CARD	COL.LOC	VARIABLE TYPE	PROGRAM NAME	VARIABLE	DESCRIPTION
2	61-65	Real	FLAP	δ_f	Flap setting - degrees: If δ_f is zero, ie, flaps not deflected, then BF on card 1 should be set to 1.0
	66-70	Real	X	X	X-coordinate of moment reference point
	71-75	Real	Z	Z	Z-coordinate of moment reference point
	76-80	Real	TWISA	ϵ_a	Aerodynamic twist - degrees (If aerodynamic twist is specified, TWIST on card 1 must be set to 100.0)
3	1-10	Real	CAMBT	K_T	Tip airfoil series camber level
	11-20	Real	CAMBR	K_R	Root airfoil series camber level
	21-30	Integer	NSLIP	--	NSLIP = 0, power off case, no propeller data required; NSLIP = 1, power on case, propeller data required
	31-40	Real	CDNAC	C_{D_N}	Total drag coefficient of all nacelles based on wing area, set to zero if no nacelles

4.2 Control and Sequencing Options (Card Group A)

Card 4 of Group A controls data read-in and optional printout features, in addition to standard output. The card layout is shown below.

CARD	COL.LOC	VARIABLE TYPE	PROGRAM NAME	VARIABLE	DESCRIPTION
4	1	Integer	NLVL	--	Number of values of thickness chord ratio (limit 5)
	2	Integer	ISWIT(1)	--	Option for reading in wing geometric parameters
	3	Integer	ISWIT(2)	--	Option to print out intermediate calculations as they are performed
	4	Integer	ISWIT(3)	--	Option to print out matrices
	5	Integer	IG	--	Switch used to set up data tables
	6-55	Alphanumeric	--	--	Run number, date, etc.

The options are as follows:

ISWIT(1)

This control option is used for wings having nonlinear taper. The local values of Reynolds number, geometric twist, and ratios of thickness/chord and local chord/root chord must be key-punched for each spanwise station together with the values of edge velocity factor and the ratios of mean aerodynamic chord/root chord and root chord/span. By setting ISWIT(1) = 1, these values (Card Group C) are read in the following order using Format (16F5.0):

PROGRAM VARIABLE NAME	DESCRIPTION	TYPE	NUMBERS OF VALUES
TAU	Thickness/chord ratios	array	R-1
REY	Reynolds numbers	array	R-1
C	Chord/root chord ratios	array	R-1
EPS	Geometric twist	array	R-1
EDGE	Edge velocity factor	single value	1
CRB	Root chord/span ratio	single value	1
ACC	Mean aerodynamic chord	single value	1

If ISWIT(1) = 0, the program assumes a straight-tapered wing and calculates the values. Normally this is the case.

ISWIT(2)

This option allows for different types of printout required in the computation.

Setting ISWIT(2) = 1, causes the program to print out intermediate calculations as they are performed. This printout is very lengthy and should be used only when absolutely necessary to aid in debugging.

If ISWIT(2) = 0, intermediate calculations are not printed.

ISWIT(3)

If this control option is set to 1, the two major matrices of the program will be listed. First, β_{mk} (BETA), the matrix of multipliers used to obtain induced angles of attack is printed out. Then this is followed by the matrix K_{ij} (TRIX) which is used in the iteration cycle.

If ISWIT(3) is set to zero, no printout of the matrices will be obtained.

IG

This control option is used to set up the wing airfoil data tables. For the case of a wing having the root airfoil series the same as the tip airfoil series, or for a wing with a deflected part-span flap, set:

IG = 3, if airfoil data is being read for the first time from cards

or IG = 1, if the airfoil data has already been read in and stored on tape

If the root airfoil series is different from the tip series then set:

IG = 4, if airfoil data is being read for the first time from cards

or IG = 2, if the airfoil data has already been read and stored on tape

Note that for a wing with a deflected part-span flap, the airfoil series from root to tip must be the same. The value of IG causes the computer to perform the following operations.

VALUE OF IG	OPERATION
1	Read airfoil data from peripheral storage device (PSD) to cube 1 on disc then copy cube 1 to cube 2 (See Figure 7).
2	Read root series airfoil data from PSD to cube 1 on disc, then read tip series data from PSD to cube 2 on disc.
3	Read airfoil data from cards , load to PSD, then read from PSD to cube 1 on disc. Copy cube 1 to cube 2.
4	Read root series airfoil data from cards,load to PSD, then to cube 1 on disc. Read tip series airfoil data from cards, then load to PSD, then to cube 2 on disc.

4.3 Wing Section Aerodynamic Data (Card Group B)

The aerodynamic section characteristics of the airfoil are read into the computer in the form of tables of lift coefficient (C_l) versus angle of attack (α), drag coefficient (C_d) versus C_l, pitching moment coefficient (C_m) versus C_l and lift coefficient with flap deflected versus α. The tables must be selected to cover the range of values of thickness/chord ratio, Reynolds number, and camber associated with the wing under consideration.

For each value of thickness/chord ratio, the data tables are arranged as indicated in Table V. The first card (punched in columns 1 through 7) indicates the number of rows in the table (columns 1 and 2), the number of columns in the table (columns 3 and 4) and the airfoil thickness/chord ratio (columns 5, 6, and 7). The second card (in alphanumeric format) indicates the airfoil type and the type of data e.g. NACA 230XX, C_l.

All cards in the table have format (F7.3,9F8.3). The first card contains the values of Reynolds number (in millions) and begins with a blank in columns 1 through 7. If the table is to contain C_l values, the next card reads - 90.0 in columns 1 through 7, and -9.0 in columns 8 through 80. If the table is for values of C_d, the card reads -10.0 in columns 1 through 7, and 2.0 in columns 8 through 80. If the table is to contain C_m values, the card reads -10.0 in columns 1 through 7, and zeros in columns 8 through 80.

The remaining cards contain either: (1) a value of angle of attack (columns 1 through 7) followed by the values of C_l corresponding to each Reynolds number, or (2) a value of C_l (columns 1 through 7) followed by the values of C_d, or (3) a value of C_l (columns 1 through 7) followed by the values of C_m, depending on whether the table contains C_l, C_d, or C_m data. The last three cards in each table are:

Table V. Card Format for Wing Section Airfoil Tables

Col. Loc.	1	8	16
Header Cards	1311-12		
	NACA 23012 Lift Coefficient CL		
Reynolds Numbers		0.5	1.0
	-90.0	-9.0	-9.0
	-14.0	-0.5	-0.45
	α Values		C_l Values
	⋮		⋮
	+90.0	4.0	9.0
C_l Max Values		1.51	1.54
α Max Values		17.2	17.0
	1511.12		
	NACA 23012 Drag Coefficient CD		
		0.5	1.0
	-10	2.0	2.0
	-0.3	0.004	0.0042
	C_l Values		C_d Values
	⋮		⋮
	10.0	2.0	2.0
		Blank card	
		Blank card	
	1611.12		
	NACA 23012 Pitching Moment Coefficient CM1/4 Chord		
	-10.0	0.0	0.0
	-0.36	-0.40	-0.41
	C_l Values		C_m Values
	⋮		⋮
	10.0	0.0	0.0
		Blank card	
		Blank card	

C_l Table

Card No.	Columns 1 through 7	Remaining Fields
1	90.0	9.0 9.0 etc.
2*	blank	Values of C_l max
3*	blank	Values of α at C_l max i.e. α_{max}

* In the C_l tables the values of C_{lmax} and α_{max} appearing on cards 2 and 3 respectively, must also appear in the main body of the table.

C_d Table

1	10.0	2.0 2.0 etc.
2	blank card	
3	blank card	

C_m Table

1	10.0	0.0 0.0 etc.
2	blank card	
3	blank card	

Up to 5 "levels" of thickness/chord ratio may be used. Within each level the table size is limited to 8 columns (7 values of Reynolds number) and 25 rows (22 different values of α or C_l).

The airfoil data cards are assembled as shown in Figure 30.

4.4 Fuselage Angles of Attack (Card Group D)

The fuselage angles-of-attack at which calculations are to be made are read in on 2 cards, Format (10F8.0). These are used sequentially as punched until either 99.0 is encountered or stall is reached. If the former condition is encountered, the program will automatically proceed to the next case; if stall is reached, the program will search for an angle-of-attack close to the value of angle-of-attack at which stall just occurs. The accuracy of this search depends on how closely the angles-of-attack are chosen near stall.

Note that all the lift tables are artificially extended beyond the $C_{l\ max}$ point with a positive slope. For example, in all tables C_l at $\alpha = \pm 90$ is set to ± 9.0. This is done to ensure convergence. The effect of this is that the overall wing C_L versus α curve predicted by the program will be correct up to the angle-of-attack at which stall first occurs on the wing. Thereafter, the predicted C_L is incorrect. This is consistent with the purpose of the program which is to predict the point of stall onset only.

4.5 Propeller Tip Loss Correction Factors (Card Group E)

The propeller tip loss correction tables are defined by a three-dimensional array of size (3, 21,8). The data values are read in on a total of 63 data cards, each card containing 8 values and having the following format:

COL.LOC	VARIABLE TYPE	PROGRAM SYMBOL	ALGEBRAIC SYMBOL	DESCRIPTION
1	Integer	IB	(B-1)	Array index identifying number of blades per propeller, less one
2-3	Integer	IP	$(1+20 \sin\phi)$	Array index identifying value of $\sin \phi$
4-32	--	--	--	Blank
33-38	Real	TLOSS(1)	F/FP	Data value at r/R = 0.3
39-44	Real	TLOSS(2)	F/FP	Data value at r/R = 0.4

COL. LOC.	VARIABLE TYPE	PROGRAM SYMBOL	ALGEBRAIC SYMBOL	DESCRIPTION
⋮	⋮	⋮	⋮	⋮
75-80	Real	TLOSS(8)	F/FP	Data value at r/R = 1.0

These 63 data cards may be assembled in any order.

4.6 Propeller Blade Section Aerodynamic Data (Card Group F)

Up to 150 propeller blade-section data tables*can be accepted and stored by the computer program. Each table contains an array of up to 20 pairs of α and C_l values for one airfoil section at one Mach number condition. Up to 25 airfoil sections may be specified for each airfoil family. A maximum of 9 families can be stored, each being assigned an arbitrary single-digit airfoil series code between 1 and 9 inclusive.

The standard blade-section data tables available with the program use preassigned airfoil series codes (as given below) for which the computer program stores the following titles and values of two constants, k_1 and k_2.

Code	Airfoil Series Name	k_1	k_2
1	USNPS	0.	-40.0
2	Clark, Y	0.	-40.0
3	NACA 16XXX	-7.3	0.
4	NACA 64-XXX	-6.9	0.
5	NACA 65-XXX	-6.9	0.
6-9	Blank	0.	0.

*See section 4.1.3 of main report for order of assembly.

The constants k_1, k_2 are empirical values that permit an initial value for α_0, angle-of-attack at zero lift used in equation 104, to be obtained from a linearized approximation to the airfoil characteristics given by the expression

$$\alpha_0 = k_1 \times (\text{design lift coefficient}) + k_2 \times (\text{thickness/chord ratio}) \tag{146}$$

The airfoil series title card is used to reassign airfoil series names and k_1, k_2 values as required. One card is used for each reassignment and has the following format:

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
1	Integer	I	--	Airfoil series code
11-18	Alphanumeric	AFSER(I)	--	Airfoil series name
21-30	Real	AK(I,1)	k_1	Empirical constants defined in preceeding text
31-40	Real	AK(I,2)	k_2	

Each airfoil table is defined on a sequence of from 2 to 5 cards. The first card in the sequence contains header information specifying the airfoil section and other pertinent parameters as follows:

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
1-2	Integer	IHEDD(1)	--	Number of pairs of α, C_l values in table
5	Integer	IHEDD(2)	--	Airfoil series code
9-16	Real	THEDD(1)	C_{li}	Design lift coefficient
17-24	Real	THEDD(2)	t/c	Thickness/chord ratio

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
25-32	Real	THEDD(3)	Mo	Mach number
33-40	Real	THEDD(4)	α_0	Angle-of-attack for zero lift
41-48	Real	THEDD(5)	k_{Cli}	Extrapolation coefficient given by equation (126)
57-68*	Alphanumeric	--	--	Table data source
69-80*	Alphanumeric	--	--	Airfoil section name

* For reference only; these columns not read by program.

The second and subsequent cards in each airfoil table sequence contain the pairs of α and C_l values. These values must be arranged in order of α increasing. The format is as follows:

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
1-8	Real	TALFA(1)	α	1st pair of values
9-16	Real	TLIFT(1)	C_l	
17-24	Real	TALFA(2)	α	2nd pair of values
25-32	Real	TLIFT(2)	C_l	
⋮	⋮	⋮	⋮	⋮
65-72	Real	TALFA(5)	α	5th pair of values
73-80	Real	TLIFT(5)	C_l	

4.7 Propeller Geometry and Operating Conditions (Card Group G)

Information to specify the propeller geometric and operating parameters is provided on a series of cards of four types. These cards must be arranged in the order described below. Each card type is assigned a numerical code value in column 80. The value is read by the program to ensure that the cards are in proper sequence.

The first card provides for the inclusion of arbitrary title information as follows:

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	DESCRIPTION
1-76	Alphanumeric	TITLE	Propeller identification title
80	Integer	IDENT	Card identification code = 1

The second card provides information specifying the propeller and nacelle geometry as follows:

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
1-2	Integer	NP	--	Number of propellers
11-12	Integer	NB	B	Number of blades per propeller
21-30	Real	DPB	D/b	Propeller diameter/ wing span
31-40	Real	RHBR	r_{hub}/R	Hub radius/tip radius
41-50	Real	RNBR	$r_{nacelle}/R$	Nacelle radius/tip radius
80	Integer	IDENT	--	Card identification code = 2

The third card provides information specifying the operating conditions for the propeller(s) as follows:

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
1-2	Integer	NROT(1)	--	L.H. propeller rotation index
11-12	Integer	NROT(2)	--	R.H. propeller rotation index
21-30	Real	AJ	J	Propeller advance ratio
31-40	Real	AMCHU	M_O	Flight Mach number
80	Integer	IDENT	--	Card identification code = 3

The positive value (01) for the propeller rotation index is used to specify a right-hand rotation for either propeller. The negative value (-1) is used to specify a left-hand rotation. Where a single-propeller configuration is considered, the rotation index for the L.H. propeller must be set to zero (00) while the R.H. propeller index has the appropriate value for the single propeller rotation sense.

The fourth-through-last cards of the series provide information specifying the blade section geometry. One card is required for each selected blade station between hub and tip, up to a maximum of 12 cards. These cards must be assembled in order of increasing blade station radius. The first selected station need not be at the hub, but the last card must specify the station at the tip with a value of 1.0 in columns 1-10.

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
1-10	Real	RPBR	r/R	Blade station radius/ tip radius
11-20	Real	CPBR	C/R	Blade-section chord/ tip radius

COL.LOC.	VARIABLE TYPE	PROGRAM NAME	ALGEBRAIC SYMBOL	DESCRIPTION
21-30	Real	BETA	β	Blade section pitch angle, degrees
31	Integer	NA	--	Blade section airfoil series code
41-50	Real	CLI	C_{l_i}	Blade section design lift coefficient
51-60	Real	TOC	t/c	Blade section thickness/chord ratio
80	Integer	IDENT	--	Card identification code = 4

4.8 Program Termination (Card Group H)

The program operation is terminated by three input data cards located at the end of the input data deck. The first card uses the format of card number 1 of Group A with the value of ASPEC set to 99.0. The second and third cards must be blank.

4.9 Data Restrictions

The following is a list of input quantities together with the restrictions and normal range of values.

QUANTITY	SIGN-RESTRICTIONS-NORMAL RANGE
ASPEC	+, $\geq$ 2.0
TAUT, TAUR, TAPER	+, $\leq$ 1.0
TWIST, TWISA	between $+15^{o}$ and -15^{o}
R	+20 only*
BF	+, $\leq$ 1.0

QUANTITY	SIGN-RESTRICTIONS-NORMAL RANGE
REYND	+
DISCR	+, suggested value .001
A, B, H	+, $\leq$ 1.0
ALPHR	between $+10^{\circ}$ and -10°
NFLAP	0 or 1
FLAP	+, between 0° and 90°
X, Z	+ or -
NSLIP	0 or 1
CDNAC	+, 0 $\leq$ 1.0

* R, which must be an even integer, may be changed to allow calculation at a greater number of spanwise stations. This requires changing the DIMENSION statements.

5.0 OUTPUT

5.1 Printout Options

All output is from a standard 120-characters-per-line printer. The amount and type of data output depends on the options exercised and on whether the computation is for a wing with or without a deflected flap/ with or without slipstream. For the standard run (without the debug print-out), the output data is self-explanatory. When the option for printing intermediate calculations is exercised, the output contains the following additional parameters whose meaning is given below.

QUANTITY	TYPE	DESCRIPTION (see also Reference 1)
A1	single value	Angle of attack corresponding to C_l at flap end, from flapped section data
A2	single value	Angle of attack corresponding to C_l at flap end, from unflapped section data
A3	single value	Zero-lift angle at flap end - flapped section data
A4	single value	Zero-lift angle at flap end - unflapped section data
ALPC	array	Angle of attack corrections for flap effect
ALPH	array	Angles of attack corrected for downwash, flap effects and body upwash
ALPHE	array	Effective angles of attack
ALPG	array	Wing section geometric angles of attack
ALPHU	array	Section downwash angles corrected for fuselage effects
ALPHZ	array	Section zero-lift angles
ALFPR	single value	Propeller shaft angle of attack in radians
ASBAR	single value	Average slipstream angle
CBC	array	Calculated values of C_{l_c}/b
CBG	array	Approximate values of C_{l_c}/b
CDOC	array	Values of $C_d c/b$
CL	single value	Integrated lift coefficient used to normalize CLADD

QUANTITY	TYPE	DESCRIPTION (see also Reference 1)
CLADD	array	Additional lift coefficient distribution
CLAD1	array	Modified distribution of additional lift coefficients from tip to flap end
CLAD2	array	Modified distribution of additional lift coefficients from flap end to wing/fuselage junction
CL2CB	array	Distribution of lift associated with equation (56)
CLDEL	array	Distribution of lift coefficient due to flap deflection only
CLMAX	array	Values of section maximum lift coefficients
CLSTA	single value	Section lift coefficient at flap end
CLSTU	single value	Uncorrected section lift coefficient at flap end
CVAL	array	Lift coefficients corresponding to ALPG
DELTA	array	Differences between guessed and calculated lift distributions
DDCLMA	single value	Increment in section maximum lift coefficient at flap end due to flap deflection
EDGE	single value	Edge velocity factor
F	array	Factors used in altering two-dimensional section data to three-dimensional data

QUANTITY	TYPE	DESCRIPTION (see also Reference 1)
FF	single value	Factors used at flapside of flap end to alter 2-dimensional data
F1	single value	Factor used to scale additional lift distribution CLADD
F2	single value	Factor used to scale CLDEL
FUNC	single value	Wing-on-propeller upwash function, equation (128)
GENE	array	Values of equation (88) for $\theta^* = \pi - \theta^*$
HOPP	array	Values of α_{c_K}/δ - see Reference 1 equation (38)
RDUBAR/DU	single value	Real part of the derivative of the conformal transformation function, equation (9) of Ref. 1
SGENE	array	Function associated with equation (76)
STONY	array	Function associated with equation (76)
SIGMA	array	Function associated with HOPP
SV	array	Slipstream crossflow distribution
TONY	array	Values of equation (88)
VW	single value	Wing-induced upwash in propeller disc plane

5.2 Error Messages

In developing the program it was found that the most common source of error was the airfoil data tables. In particular, the tables of C_l versus α are most critical. The variation of C_l with increasing α should be smooth without sudden breaks, especially for values higher than α_{max}. A sharp break in the slope of C_l versus α after α_{max} may cause the iteration procedure to diverge. If this occurs a message is printed as follows:

UNABLE TO CONVERGE AFTER 30 ITERATIONS ABORTED

The last values of DELTA and the values of lift coefficient are then listed together with a dump of the airfoil tables in core at that time.

A second error message associated with table interpolation is as follows:

ERROR CODE (N)

IF 1 CVAL GTR THAN MAX VALUE LISTED

IF 2 CVAL GTR THAN TABLE VALUE

IF 3 ALPHA VALUE GTR THAN TABLE VALUE

IF 6 THICKNESS CHORD RATIO VALUE CANNOT BE FOUND

If these errors occur, the tables should be examined for mistakes in key punching, etc.

Several error messages are associated with the propeller slipstream analysis subroutine and operate as follows:

(a) If an invalid number of tip loss correction cards are read, i.e. other than 0 or 63, this is indicated by the message:

XX TIP LOSS CORRECTION TABLE DATA CARDS READ IS INVALID - SLIPSTREAM COMPUTATIONS ABORTED

(b) If the propeller geometry and operating condition data cards are read out of sequence, this is indicated by the message:

CARD IDENT XX HAS BEEN READ OUT OF SEQUENCE, SLIPSTREAM COMPUTATIONS ABORTED

(c) If propeller airfoil tables are not stored for the airfoil series code specified on a blade station data card (IDENT 4), this is indicated by the following message:

AIRFOIL TABLES NOT STORED FOR AIRFOIL SERIES XX
SPECIFIED AT RB/RP = XX•XXXX - THIS ELEMENT IS
DELETED FROM THE ANALYSIS

(d) If the propeller-slipstream analysis fails to converge within 9 iterations, then the following message appears after printing out the solution for the last iteration:

SOLUTION FOR PRECEEDING ELEMENT FAILED TO CONVERGE
IN 9 ITERATIONS AND IS DELETED FROM THE SLIPSTREAM
ANALYSIS

6.0 PROGRAM STRUCTURE

The program is written to operate in OVERLAY mode. The central controlling portion of the overall program is called STALL. The remainder of the program is split into 3 parts, ONE, TWO, THREE, which are overlaid. STALL calls ONE which then calls either TWO or THREE depending on whether the calculation is for a flapped wing or not. The major subroutines called by each overlay are as follows:

ONE	calls	MAIN, MAINA, MAIN1
TWO	calls	MAIN2, MAN2A, MAIN4
THREE	calls	MAIN2, MAIN3, MAIN5

These major subroutines call the remaining subroutines in the program as follows:

MAIN	calls	MAINA, SETSW, AERDA, DATSW, ZZZ
MAINA	calls	SLIP, AAA, DATSW, ZZZ, BRIDG, MAIN1
MAIN1	calls	DATSW, AAA, MINV, GRIDG, SSS
MAN2A	calls	DATSW, AAA, BRIDG, ZZZ, MAIN4
MAIN3	calls	DAGET, DATSW, AAA, ARC, ZZZ, MAIN5
MAIN4	calls	BRIDG, AAA, DATSW, MAN2A
MAIN5	calls	DAGET, ARC, DATSW, AAA, ZZZ
BRIDG	calls	DAGET, ARC
ARC	calls	LOOK

7.0 OPERATING PROCEDURE

Logical TAPE5 is named as the working (scratch) tape. The program deck and data deck are loaded in the following sequence: job card, system control cards, end-of-record card, program deck, end-of-record card, data deck, end-of-file card.

8.0 PROGRAM TIMING

Central processor unit time for an average run of six (6) angles-of-attack is approximately 60 seconds.

APPENDIX D

INTERNAL LISTING OF THE COMPUTER PROGRAM

Presented in this appendix is an internal listing of the computer program developed under the present contract.

```
      OVERLAY(BLINCA,0,0)
      PROGRAM STALL(INPUT=201,OUTPUT=1001,TAPE8=INPUT,TAPE6=OUTPUT,TAPE1
     1=201,TAPE2=201,TAPE3=201,TAPE4=201,TAPE5=201,TAPE10=201,TAPE15=201
     2,TAPE20=201,TAPE44=201,TAPE7=1001,TAPE9=201)
      DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(1
     19),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19),
     2MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),TRIX
     3(19,19),CM(19),DIST(19),YDA(19),YDAX(19),YX(19),TONY(19),ALPG(19),
     4XHERE(2,6),MWCCL(6),MAWW(6)
      DIMENSION CVAL(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(19),
     1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KCR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNCW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,Q,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCOL(6),STONY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LCCER
      BLINCA=6LBLINCA
      RECALL=6HRECALL
30    IR=8
50    CALL OVERLAY(BLINCA,1,0,RECALL)
      IF(NFLAP.NE.0) GO TO 10
      GO TO 20
10    CALL OVERLAY(BLINCA,3,0,RECALL)
      GO TO 40
20    CALL OVERLAY(BLINCA,2,0,RECALL)
40    IF(IR.EQ.0) GO TO 50
      GO TO 30
      END
      OVERLAY(BLINCA,1,0)
      PROGRAM ONE
      DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(1
     19),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19),
     2MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),TRIX
     3(19,19),CM(19),DIST(19),YDA(19),YDAX(19),YX(19),TONY(19),ALPG(19),
     4XHERE(2,6),MWCCL(6),MAWW(6)
      DIMENSION CVAL(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(19),
     1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KCR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNCW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,Q,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCOL(6),STONY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LCCER
      IF(IR.EQ.0) GO TO 20
10    CALL MAIN
20    CALL MAINA
      IF(ALPHB.EQ.99.) GO TO 10
      IF(IPRAB.NE.1) GO TO 30
      CALL MAIN1
30    RETURN
      END
      OVERLAY(BLINCA,2,0)
      PROGRAM TWO
      DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(1
     19),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19),
     2MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),TRIX
     3(19,19),CM(19),DIST(19),YDA(19),YDAX(19),YX(19),TONY(19),ALPG(19),
     4XHERE(2,6),MWCCL(6),MAWW(6)
      DIMENSION CVAL(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(19),
     1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KCR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
```

```
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CLL,REYON,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STONY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LOOER
      CALL MAIN2
      CALL MAN2A
      RETURN
      END
      OVERLAY(BLINDA,3,0)
      PROGRAM THREE
      DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(1
     19),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19),
     2MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),TRIX
     3(19,19),CM(19),DIST(19),YDA(19),YDAX(19),YX(19),TONY(19),ALPG(19),
     4XHERE(2,6),MWCCL(6),MAWW(6)
      DIMENSION CVAL(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(19),
I    1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CLL,REYON,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STONY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LOOER
      CALL MAIN2
      CALL MAIN3
      RETURN
      END
      SUBROUTINE MINV(A,N,D,L,M)
C
C                                    MATRIX INVERSION SUBROUTINE
C
      DIMENSION A(1),L(1),M(1)
C
C                                    SEARCH FOR LARGEST ELEMENT
C
C
      D=1.0
      NK=-N
      DO 180 K=1,N
      NK=NK+N
      L(K)=K
      M(K)=K
      KK=NK+K
      BIGA=A(KK)
      DO 20 J=K,N
      IZ=N*(J-1)
      DO 20 I=K,N
      IJ=IZ+I
      IF(ABS(BIGA)-ABS(A(IJ))) 10,20,20
   10 BIGA=A(IJ)
      L(K)=I
      M(K)=J
   20 CONTINUE
C
C                                    INTERCHANGE ROWS
C
      J=L(K)
      IF(J-K) 50,50,30
   30 KI=K-N
      DO 40 I=1,N
      KI=KI+N
      HOLD=-A(KI)
      JI=KI-K+J
      A(KI)=A(JI)
   40 A(JI)=HOLD
C
```

```
C                                        INTERCHANGE COLUMNS
C
   50 I=M(K)
      IF(I-K) 80,80,60
   60 JP=N*(I-1)
      DO 70 J=1,N
      JK=NK+J
      JI=JP+J
      HOLD=-A(JK)
      A(JK)=A(JI)
   70 A(JI)=HOLD
C
C                                        DIVIDE COLUMN BY MINUS PIVOT (VALUE OF
C                                        PIVOT ELEMENT IS CONTAINED IN BIGA)
C
   80 IF(ABS(BIGA)-1.E-20) 90,90,100
   90 D=0.0
      RETURN
  100 DO 120 I=1,N
      IF(I-K) 110,120,110
  110 IK=NK+I
      A(IK)=A(IK)/(-BIGA)
  120 CONTINUE
C
C                                        REDUCE MATRIX
C
      DO 150 I=1,N
      IK=NK+I
      HOLD=A(IK)
      IJ=I-N
      DO 150 J=1,N
      IJ=IJ+N
      IF(I-K) 130,150,130
  130 IF(J-K) 140,150,140
  140 KJ=IJ-I+K
      A(IJ)=HOLD*A(KJ)+A(IJ)
  150 CONTINUE
C
C                                        DIVIDE ROW BY PIVOT
C
      KJ=K-N
      DO 170 J=1,N
      KJ=KJ+N
      IF(J-K) 160,170,160
  160 A(KJ)=A(KJ)/BIGA
  170 CONTINUE
C
C                                        PRODUCT OF PIVOTS
C
      D=D*BIGA
C
C                                        REPLACE PIVOT BY RECIPROCAL
C
      A(KK)=1.0/BIGA
  180 CONTINUE
C
C                                        FINAL ROW AND COLUMN INTERCHANGE
C
      K=N
  190 K=K-1
      IF(K) 260,260,200
  200 I=L(K)
      IF(I-K) 230,230,210
  210 JQ=N*(K-1)
      JR=N*(I-1)
      DO 220 J=1,N
      JK=JQ+J
      HOLD=A(JK)
      JI=JR+J
      A(JK)=-A(JI)
  220 A(JI)=HOLD
  230 J=M(K)
      IF(J-K) 190,190,240
  240 KI=K-N
      DO 250 I=1,N
      KI=KI+N
      HOLD=A(KI)
      JI=KI-K+J
      A(KI)=-A(JI)
```

```
  250 A(JI)=HCLD
      GO TO 190
  260 RETURN
      END
      SUBROUTINE CAGET(ARRAY,IFILE,II)
C
C                                   SUBROUTINE TO GET TABLE FROM DISK AND PUT
C                                   IT INTO CORE
C
      DIMENSION ARRAY(5,25,8),CRRAY(5,25,8)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPH3,KSTAL,ISTAL,KCUNT,AB(3)
      COMMON NSLIP,VR(19),CA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX(19
     1),C(19),EPS(19),TRANS(19),RLY(19),ETA(19),HCPP(19),ZHERE(2,6),
     2WHERE(2,6),CRRAY,Y(19),BETA(19,19),TFAC,TRIX(19,19),TAU(19),MAXX(6
     3),MAZZ(6),MXCOL(6),MZCOL(6),ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,Q,
     4TSTAX,EDGE,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,ISIS,ISTAR,A,B,H,
     5TAUT,TAUR,TWIST,R,BRX,YHERE(2,6),MYCOL(6),MAYY(6),FLAP,TCNY(19),
     6TWISA,X,Z,CM(19),ACC,XHERE(2,6),MWCOL(6),MAWW(6),CAMB(19),CAMER,
     7CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),MAAA(6),MACOL(6)
C
C                                   TABLE PRESENTLY IN CORE IS INNOW
C
C                                   IF TABLE IS ALREADY IN CORE THEN RETURN
C                                   IF NOT THEN GET TABLE FROM DISK
C
      IF(IFILE-INNOW) 10,90,10
   10 II=1
      REWIND IFILE
      READ(IFILE) ARRAY
      REWIND IFILE
C
C                                   RESET TABLE PRESENTLY IN CORE INDICATOR
C
      INNOW=IFILE
      IF(IFILE-5) 50,30,90
   30 DO 40 JFOX=1,NLVL
      KROW=MAAA(JFOX)-2
      KCOL=MACOL(JFOX)
      DO 40 KBOB=2,KCOL
      ARRAY(JFOX,2,KBOB)=-9.0
   40 ARRAY(JFOX,KROW,KBOB)=9.0
      GO TO 90
   50 IF(IFILE-1) 90,60,90
   60 DO 80 JFOX=1,NLVL
      KROW=MAXX(JFOX)-2
      KCOL=MXCOL(JFOX)
      DO 70 KBOB=2,KCOL
      ARRAY(JFOX,2,KBOB)=-9.0
   70 ARRAY(JFOX,KROW,KBOB)=9.0
   80 CONTINUE
   90 RETURN
      END
      SUBROUTINE AAA(CLIST,NV)
C
C                                   SUBROUTINE TO OUTPUT A ONE DIMENSIONAL
C                                   ARRAY
C
      DIMENSION CLIST(1)
      IP=6
      WRITE(IP,10) (J,CLIST(J),J=1,NV)
      WRITE(IP,20)
      RETURN
C
   10 FORMAT(5(1X,1H(,I2,2H) ,E16.6))
   20 FORMAT(2(1H ))
      END
      SUBROUTINE ZZZ(VALUE)
C
C                                   SUBROUTINE TO OUTPUT A SINGLE VALUE
C
C                                   OUTPUT VALUE IN BOTH F AND E FORMAT
C
      IP=6
      WRITE(IP,10) VALUE,VALUE
      RETURN
C
   10 FORMAT(1X,F15.8,E16.8/1X)
```

```
      ENC
      SUBRCUTINE SSS(A,NP)
C
C                                SUBRCUTINE TC CUTPUT A SCUARE TWC
C                                CIMENSICNAL ARRAY
C
      DIMENSICN A(19,19)
      IP=6
      WRITE(IP,10) ((J,K,A(J,K),K=1,NP),J=1,NP)
      WRITE(IP,20)
      RETURN
C
10    FCRMAT(5(1X,1H(,I2,1H,,I2,2H) ,E13.6))
   20 FCRMAT(3(1HC/))
      ENC
      FUNCTICN TERP(R1,R2,R3,C1,C3)
C
C                                INTERPOLATICN FUNCTION
C
      IF(R1-R3) 20,10,20
   10 TERP=C1
      RETURN
   20 TERP=C1+((C3-C1)*((R2-R1)/(R3-R1)))
      RETURN
      ENC
      SUBRCUTINE SETSW
C
C                                SUBRCUTINE CHECKS SWITCH SETTINGS ON DATA
C                                CARC ANC LISTS OPTIONS SELECTED
C
      CCMMCN KOR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      CCMMCN NSLIP,VR(19),CA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      CCMMCN INNCW,ISWIT(3)
      K=0
      IP=6
      WRITE(IP,90)
      IF(ISWIT(1)) 20,20,10
   10 K=K+1
      WRITE(IP,100)
   20 IF(ISWIT(2)) 40,40,30
   30 K=K+1
      WRITE(IP,110)
   40 IF(ISWIT(3)) 60,60,50
   50 K=K+1
      WRITE(IP,120)
   60 IF(K) 80,70,80
   70 WRITE(IP,130)
   80 RETURN
C
   90 FCRMAT(1H1/1HC/1HC,50X,29HSPECIAL FEATURES FOR THIS RUN/1HO)
  100 FCRMAT(51X,14HREAC IN VALUES/1HC)
  110 FCRMAT(51X,34HTRACE (CUTPUT CF KEY CALCULATIONS)/1HO)
  120 FCRMAT(51X,28HPRINT MATRICES BETA AND TRIX/1HO)
  130 FCRMAT(51X,31HNC SWITCH SETTINGS  REGULAR RUN)
      ENC
      SUBRCUTINE CATSW(N,ICCNC)
C                 SUBRCUTINE MAKES CECISION BASED CN SETTING CF THE SWITCH,
C                 SET=1,NCT SET=C. ICCNC=1 IF SWITCH IS SET,ICOND=2 IF
C                 SWITCH IS NOT SET
      CCMMCN INNCW,ISWIT(3)
      IF(N=3) 10,20,30
10    IF(ISWIT(1)) 40,40,50
10    IF(ISWIT(2)) 40,40,50
10    IF(ISWIT(3)) 40,40,50
40    ICCNC=2
      RETURN
50    ICCNC=1
      RETURN
      ENC
      SUBRCUTINE AERCA(ARRAY,NLVL,WHERE,MXCOL,MAXX,IFILE,KEEP)
C
C                                SUBRCUTINE TC REAC AERODYNAMIC TABLES FRCM
C                                CARCS ANC PUT TABLES ON DISK
C
      CIMENSICN NAME(40),MAXX(6),MXCCL(6),WHERE(2,6),ARRAY(5,25,8)
      CCMMCN KOR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      CCMMCN NSLIP,VR(19),CA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
```

```
      1ALPHV(16)
       COMMON INNCH
       INTEGER*2 KEEP
C
C                                   MAXX(6) CONTAINS NUMBER OF ROWS IN A TABLE
C                                   WHERE(2,6) CONTAINS TABLE NUMBER AND TAU
C                                   VALUE
C                                   INNCH IS THE TABLE PRESENTLY IN CORE
C
       IR=8
       IP=6
C
C                                   INITIALIZE LEVELS IN WHERE
C
       DO 10 J=1,NLVL
    10 WHERE(1,J)=J
C
C                                   LOOP TO READ IN LEVELS OF A SET OF EITHER
C                                   LIFT, DRAG, OR PITCHING MOMENT
C
       DO 40 LVL=1,NLVL
C
C                                   READ AND PRINT NUMBER OF ROWS, COLUMNS, TAU
C                                   VALUES FOR A GIVEN LEVEL
C
       READ(IR,50) NC,NCOL,WHERE(2,LVL)
       WRITE(IP,60) NC,NCOL,WHERE(2,LVL)
C
C                                   STORE NUMBER OF COLUMNS
C
       MXCOL(LVL)=NCOL
C
C                                   STORE NUMBER OF ROWS
C
       MAXX(LVL)=NC
C
C                                   READ AND WRITE TITLE OF TABLE
C
       READ(IR,70) NAME
       WRITE(IP,80) NAME,LVL
C
C                                   READ VALUES FOR TABLE
C
       DO 20 I=1,NC
20     READ(IR,90) (ARRAY(LVL,I,J),J=1,8)
       DO 30 JJ=1,NC
    30 WRITE(IP,100) (ARRAY(LVL,JJ,II),II=1,NCOL)
    40 CONTINUE
C
C                                   WRITE COMPLETE TABLE ON DISK
C
       RETURN
C
100    FORMAT(8F10.5)
    50 FORMAT(2I2,F10.0)
    60 FORMAT(1X,2I2,F5.2)
    70 FORMAT(40A2)
    80 FORMAT(11X,40A2,2X,I2)
90     FORMAT(F7.3,7F8.3)
       END
       SUBROUTINE BRIDG(CCZ1,NS,NP,LDM,IZXY,IE,EY,AU,AMB,XX,IS,IP,YY)
C
C                                   SUBROUTINE TO INTERPOLATE BETWEEN TABLES
C
       DIMENSION XX(1),EY(1),AU(1),CCZ1(1),CCZ2(19),CCLR1(19),CALR1(19),
      1CCLR2(19),CALR2(19),ATMP(19),REY(19),TAU(19),ARRAY(5,25,8 ),C(19),
      2EPS(19),TRANS(19),ETA(19),HCPP(19),CLMAX(19),ZHERE(2,6),WHERE(2,6)
      3,Y(19),BETA(19,19),TRIX(19,19),MAZZ(6),MZCOL(6),MAXX(6),MXCOL(6),
      4YHERE(2,6),MYCOL(6),MAYY(6),CM(19),XHERE(2,6),MAWW(6),MWCOL(6)
       COMMON KOR,KIR,KIL,KCL,VSA(19),SW(19),VSCAR,EYETL,CTS,XPB,YPB,JP,
      1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
       COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
      1ALPHV(16)
       COMMON INNCH,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
      1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
      2,MAZZ,MXCOL,MZCOL,ASPEC,TAPER,BF,REYNO,DISCR,PIER,CRB,Q,TSTAX,EDGE
      3,SIG,ALPHR,NFLAP,NLVL,NC,IY,IZ,IR,IO,IT,ISTAR,A,B,H,TAUT,TAUR,
      4TWIST,R,BWX,YHERE,MYCOL,MAYY,FLAP,TCNY(19),TWISA,X,Z,CM,ACC,XHERE,
      5MWCOL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
      6MAAA(6),MACOL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
```

```
      7MCCCL(6),CHERE(2,6),MACC(6),MCCOL(6)
C
C                                  INITIALIZATION
C                                  IF REYCN=999 THEN SPECIAL CASE
C
      DUMY2=0.0
      DUMY1=0.0
      IF(REYCN-999.) 20,10,20
   10 KEY=1
      GO TO 30
   20 KEY=2
C
C                                  STORE C VALUE
C
   30 CLT=CLL
C
C                                  NEW IS THE FILE NUMBER OF THE OTHER CUBE
C                                  PRIMARY CUBES ARE NUMBERED 1,2,3,4
C                                  SECONDARY CUBES ARE NUMBERED 5,10,15,20
C
      NEW=LCM*5
C
C                                  STORE XMAX
C
      XMX=XMAX
C
C                                  DETERMINE IF SINGLE VALUE IS TO BE USED OR
C                                  LIST OF VALUES IS TO BE USED IN LOOK UP
C
      IF(IS-IP) 50,40,50
   40 KGC=1
C
C                                  SET UP FOR CONSTANT VALUE OF X
C
      ATMP(1)=XX(1)
      GO TO 70
C
C                                  SET UP FOR VARIABLE VALUE OF X
C
   50 KGC=2
      DO 60 J=NS,NP
   60 ATMP(J)=XX(J)
C
C                                  PUT PROPER TABLE IN CORE
C
   70 CALL CACET(ARRAY,NEW,IZXY)
C
C                                  SET UP ALPHA EITHER VARIABLE OR CONSTANT
C
      DO 190 K=NS,NP
      GO TO (80,90), KGC
   80 ALPHA=ATMP(IS)
      GO TO 100
   90 ALPHA=ATMP(K)
C
C                                  GET TAU VALUE FROM ARRAY
C
  100 TAUX=TAU(K)
      GO TO (120,110), KEY
C
C                                  GET PROPER REYNOLDS NUMBER EITHER REGULAR
C                                  OR MAX NUMBER
C
  110 REYCN=REY(K)
      REYN=999.
      GO TO 130
  120 REYN=REY(K)
  130 CLL=CLT
      XMAX=XMX
C
C                                  PERFORM LOOK FOR LIFT, DRAG, PITCHING
C                                  MOMENT OR FLAP CASE
C
      GO TO (140,150,160,170), LCM
  140 CALL ARC(ARRAY,TAUX,MAAA,MACCL,IE,AHERE,NLVL)
      GO TO 180
  150 CALL ARC(ARRAY,TAUX,MABB,MBCCL,IE,BHERE,NLVL)
      GO TO 180
  160 CALL ARC(ARRAY,TAUX,MACC,MCCCL,IE,CHERE,NLVL)
      GO TO 180
```

```
  170 CALL ARC(ARRAY,TAUX,MADD,MDCCL,IE,DHERE,NLVL)
C
C                               STORE VALUES FOUND IN LOOK UP
C
  180 CCZ2(K)=YY
      CCLR2(K)=DUMY1
      CALR2(K)=DUMY2
C
C                               REPEAT FOR NUMBER OF VALUES TO BE FOUND
C
  190 CONTINUE
C
C                               GET NEXT TABLE IN CORE
C
      CALL DAGET(ARRAY,LCM,IY)
C
C                               SET UP ALPHA VALUE
C
      DO 310 K=NS,NP
      GO TO (200,210), KGO
  200 ALPHA=ATMP(IS)
      GO TO 220
  210 ALPHA=ATMP(K)
C
C                               SET UP REYNOLDS NUMBER
C
  220 GO TO (240,230), KEY
  230 REYDN=REY(K)
      REYN=999.
      GO TO 250
  240 REYN=REY(K)
C
C                               SET UP CVAL, TAU, AND XMAX VALUE
C
  250 CLL=CLT
      TAUX=TAU(K)
      XMAX=XMX
C
C                               LOOK UP FOR SECOND LIFT, DRAG, PITCHING
C                               MOMENT, OR FLAP CASE
C
      GO TO (260,270,280,290), LCM
  260 CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      GO TO 300
  270 CALL ARC(ARRAY,TAUX,MAZZ,MZCCL,IE,ZHERE,NLVL)
      GO TO 300
  280 CALL ARC(ARRAY,TAUX,MAYY,MYCCL,IE,YHERE,NLVL)
      GO TO 300
  290 CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
C
C                               STORE VALUES FOUND IN LOOK UP
C
  300 CCZ1(K)=YY
      CCLR1(K)=DUMY1
      CALR1(K)=DUMY2
  310 CONTINUE
C
C                               SPECIAL CASE FOR LIFT LOOK UP
C
      GO TO (320,400,400,400), LCM
C
C                               REPEAT FOR ALL VALUES OF TAU
C
  320 DO 390 K=NS,NP
      ALPHA=ATMP(K)
      CCLR=TERP(CAMBR,CAMB(K),CAMBT,CCLR1(K),CCLR2(K))
      CALR=TERP(CAMBR,CAMB(K),CAMBT,CALR1(K),CALR2(K))
      IF((ALPHA-CALR1(K))*(ALPHA-CALR2(K))) 340,330,330
  330 CCZ1(K)=TERP(CAMBR,CAMB(K),CAMBT,CCZ1(K),CCZ2(K))
      GO TO 390
  340 IF(CALR2(K)-CALR1(K)) 350,350,360
  350 IF(ALPHA-CALR) 370,370,380
  360 IF(ALPHA-CALR) 380,380,370
  370 CLA=TERP(CALR1(K),CALR2(K),ALPHA,CCLR1(K),CCZ1(K))
      CLC=TERP(CAMBR,CAMB(K),CAMBT,CLA,CCLR2(K))
      CCZ1(K)=TERP(CALR2(K),ALPHA,CALR,CLC,CCLR)
      GO TO 390
  380 CLB=TERP(CALR2(K),CALR1(K),ALPHA,CCLR2(K),CCZ2(K))
      CLC=TERP(CAMBR,CAMB(K),CAMBT,CCLR1(K),CLB)
      CCZ1(K)=TERP(CALR1(K),ALPHA,CALR,CLC,CCLR)
```

```
  390 CONTINUE
      RETURN
C
C                                   REGULAR INTERPOLATION BETWEEN TABLES
C
  400 DO 410 K=NS,NP
  410 CCZ1(K)=TERP(CAMBR,CAMB(K),CAMBT,CCZ1(K),CCZ2(K))
      RETURN
      END
      SUBROUTINE ARC(RRAY,TAU,NROWS,NCOLS,IE,WHERE,NLVLS)
C
C                                   SUBROUTINE TO INTERPOLATE BETWEEN LEVELS OF
C                                   A GIVEN TABLE
C
      DIMENSION ARRAY(5,25,8 ),NROWS(6),NCOLS(6),CLR(2),ALR(2),TONY(19),
     1WHERE(2,6),YHERE(2,6),MYCOL(6),MAYY(6),CM(19),XHERE(2,6),MAWW(6),
     2MWCOL(6),C(19),EPS(19),TRANS(19),REY(19),ETA(19),HOPP(19),CLMAX(19
     3),ZHERE(2,6),HERE(2,6),Y(19),UAU(19),BETA(19,19),TRIX(19,19),MAZZ(
     46),MZCOL(6),MAXX(6),MXCOL(6)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CVAL,REYON,XMAX,ALMAX(19),CLMAX,C
     1,EPS,TRANS,REY,ETA,HOPP,ZHERE,HERE,ARRAY,Y,BETA,IFAC,TRIX,UAU,MAXX
     2,MAZZ,MXCOL,MZCOL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IV,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCOL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCOL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACOL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCOL(6),DHERE(2,6),MADD(6),MDCOL(6)
      IE=0
      XTRAL=0.
      XTRAU=0.
C
C                                   IS TAU LESS THAN OR EQUAL TO LOWEST LEVEL
C                                   TAU VALUE
C
      IF(TAU-WHERE(2,1)) 10,20,20
   10 J=2
      XTRAL=1.
      GO TO 80
   20 IF(TAU-WHERE(2,NLVLS)) 40,30,30
C
C                                   YES, SET J EQUAL TO MAX LEVEL
C
   30 J=NLVLS
      XTRAU=1.
      GO TO 80
C
C                                   NO, SEARCH LIST OF LEVELS (TAU VALUES)
C                                   UNTIL ONE GIVEN IS GREATER THAN ONE TO BE
C                                   FOUND
C
   40 DO 50 J=2,NLVLS
      IF(TAU-WHERE(2,J)) 90,90,50
   50 CONTINUE
C
C                                   ERROR IF NONE CAN BE FOUND
C
      IE=6
      IF(IE) 70,70,60
   60 WRITE(IP,540) IE,INNOW,CVAL,ALPHA,REYN,REYON,XMAX
   70 RETURN
   80 CONTINUE
   90 LVL=WHERE(1,J)
      MAXR=NROWS(J)
      MAXC=NCOLS(J)
      IF(ALPHA-999.) 100,130,100
  100 IF(REYN-999.) 110,120,110
  110 IF(CVAL-999.) 260,250,260
  120 IF(CVAL-999.) 240,260,240
  130 IF(CVAL-999.) 140,150,140
  140 IF(REYN-999.) 230,260,230
  150 IF(REYN-999.) 260,160,260
  160 IF(REYON-999.) 180,170,180
  170 IF(XMAX) 220,260,220
  180 IF(XMAX) 200,190,200
  190 IFLP=1
```

```
      GC TC 210
  20C IFLP=2
  210 IS=5
      GC TC 270
  220 IS=4
      GC TC 270
  230 IS=1
      GC TC 270
  24C IS=2
      GC TC 270
  25C IS=3
      GC TC 270
  26C IE=6
      RETLRN
  27C CALL LCCK(ARRAY,LVL,MAXR,MAXC,IE)
      CLR(2)=CUMY2
      ALR(2)=CUMY1
      GC TC (28C,29C,30C,31C,32C), IS
  280 FINC=ALPHA
      ALPHA=999.
      GC TC 350
  290 FINC=REYN
      REYN=999.
      GC TC 350
  300 FINC=CVAL
      CVAL=999.
      GC TC 350
  310 FINC=REYON
      REYCN=999.
      GC TC 350
  320 FINC=XMAX
      GC TC (33C,34C), IFLP
  330 XMAX=C.
      GC TC 350
  340 XMAX=1CC.
  35C LVL=WHERE(1,J-1)
      MAXR=NRCWS(J-1)
      MAXC=NCCLS(J-1)
C
C                             SPECIAL PROCEDURE FOR INTERPCLATICN IN THE
C                             NEIGHBCRHCOD OF CL MAX
C
      CALL LCCK(ARRAY,LVL,MAXR,MAXC,IE)
      CLR(1)=CUMY2
      ALR(1)=CUMY1
      IF(IE) 360,360,60
  36C R1=WHERE(2,J-1)
      R2=TAL
      R3=WHERE(2,J)
      C3=FINC
      GC TC (49C,5CC,37C,52C,53C), IS
  370 IF(INNCW-1) 38C,39C,38C
  380 IF(INNCW-5) 51C,39C,51C
  39C DUMY1=TERP(R1,R2,R3,CLR(1),CLR(2))
      DUMY2=TERP(R1,R2,R3,ALR(1),ALR(2))
      IF((ALPHA-ALR(1))*(ALPHA-ALR(2))) 400,51C,510
  4C0 IF(XTRAL-1.) 41C,420,41C
  41C IF(XTRAL-1.) 44C,430,44C
  42C CVAL=CUMY1-CLR(1)+TERP(ALPHA,CUMY2,ALR(1),CLR(1),CVAL)
      RETLRN
  430 CVAL=DUMY1-CLR(2)+TERP(ALPHA,DUMY2,ALR(2),CLR(2),C3)
      RETLRN
  440 IF(ALR(2)-ALR(1)) 45C,45C,460
  45C IF(ALPHA-DUMY2) 470,470,48C
  460 IF(ALPHA-DUMY2) 480,480,47C
  47C CLA=TERP(ALR(1),ALR(2),ALPHA,CLR(1),CVAL)
      CLC=TERP(R1,R2,R3,CLA,CLR(2))
      CVAL=TERP(ALR(2),ALPHA,DUMY2,CLC,DUMY1)
      RETLRN
  480 CLB=TERP(ALR(2),ALR(1),ALPHA,CLR(2),C3)
      CLC=TERP(R1,R2,R3,CLR(1),CLB)
      CVAL=TERP(ALR(1),ALPHA,DUMY2,CLC,DUMY1)
      RETLRN
  490 C1=ALPHA
      ALPHA=TERP(R1,R2,R3,C1,C3)
      RETLRN
  50C C1=REYN
      REYN=TERP(R1,R2,R3,C1,C3)
      RETLRN
  510 C1=CVAL
```

```
      CVAL=TERP(R1,R2,R3,C1,C3)
      RETURN
  520 C1=REYCN
      REYCN=TERP(R1,R2,R3,C1,C3)
      RETURN
  530 C1=XMAX
      XMAX=TERP(R1,R2,R3,C1,C3)
      RETURN
C
  540 FORMAT(20X,12HERROR CODE (,I2,42H ) IF 1 CVAL GREATER THAN MAX VAL
     1UE LISTED/37X,34HIF 2 CVAL GREATER THAN TABLE VALUE/37X,41HIF 3 AL
     2PHA VALUE GREATER THAN TABLE VALUE/37X,29HIF 6 TAU VALUE CANNOT BE
     3FOUND/37X,13HTABLE NUMBER ,I2,8H IN CORE/10X,8HC-VALUE=,F10.2,5X,
     46HALPHA=,F10.2,5X,13HREYNOLDS NO.=,F10.2,5X,6HREYCN=,F10.2,5X,5HXM
     5AX=,F10.2/120(1H/))
      END
      SUBROUTINE LOOK(AR,LVL,MAXR,MAXC,IE)
C
C                                 SUBROUTINE FOR TWO DIMENSIONAL TABLE LOOKUP
C
      DIMENSION A(5,25,8),TONY(19),YHERE(2,6),MYCOL(6),MAYY(6),CM(19),
     1XHERE(2,6),MAWW(6),MWCOL(6),C(19),EPS(19),TRANS(19),REY(19),ETA(19
     2),HCPP(19),CLMAX(19),ZHERE(2,6),HERE(2,6),Y(19),UAU(19),BETA(19,19
     3),TRIX(19,19),MAZZ(6),MZCOL(6),MAXX(6),MXCOL(6)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CVAL,REYCN,XMAX,ALMAX(19),CLMAX,C
     1,EPS,TRANS,REY,ETA,HCPP,ZHERE,HERE,A,Y,BETA,TFAC,TRIX,UAU,MAXX,
     2MAZZ,MXCOL,MZCOL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,O,TSTAX,EDGE,
     3SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,AA,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCOL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCOL,MAWW,CAM3(19),CAMBR,CAMBT,DUMY1,DUMY2
C
C                                 SECTION TO LOCATE REYNOLDS NUMBER IN TABLE
C
      XTRAL=0.
      XTRAU=0.
C
C                                 LROW IS THE LAST ROW CONTAINING INFORMATION
C                                 IN ANY TABLE. THE LAST TWO ROWS OF ANY
C                                 TABLE CONTAIN MAX DATA.
C
      LROW=MAXR-2
      IS=1
C
C                                 IS REYCN EQUAL TO 999.
C                                 IF YES THEN NORMAL AND WE WILL ALREADY HAVE
C                                 SET THE SWITCH IS TO 1 AND WE WILL USE THE
C                                 VALUE OF REYN
C                                 IF NO THEN SET REYN TO REYON AND SET SWITCH
C                                 IS TO 2
C
      IF(REYCN-999.) 10,20,10
   10 REYN=REYON
      IS=2
C
C                                 IS THE REYNOLDS NUMBER WE ARE LOOKING FOR
C                                 GREATER THAN OR EQUAL TO THE REYNOLDS
C                                 NUMBER GIVEN ALONG THE FIRST ROW OF THE
C                                 TABLE
C
   20 IF(REYN-A(LVL,1,MAXC)) 40,30,30
   30 LOCR=MAXC
      XTRAL=1.
      GO TO 70
C
C                                 NORMAL CASE. IF REYNOLDS NUMBER IS LESS
C                                 THAN MAX REYN NUMBER IN TABLE THEN WE
C                                 CHECK TO SEE IF IT IS GREATER THAN THE
C                                 MINIMUM REYNOLDS NUMBER LISTED IN THE TABLE
C
   40 IF(REYN-A(LVL,1,2)) 50,60,60
C
C                                 IF LESS THAN MINIMUM WE WANT TO USE THE
C                                 MINIMUM VALUE IN THE TABLE
C
   50 LOCR=3
      XTRAL=1.
```

```
      GC TC 7C
C
C                                         IF LESS THAN MAXIMUM AND GREATER THAN
C                                         MINIMUM WE WANT TC SEARCH TABLE UNTIL WE
C                                         FIND A REYNGLDS NUMBER GREATER THAN THE
C                                         GIVEN REYNCLCS NUMBER
C                                         IF REYN LESS THAN KEEP LCCKING
C                                         IF REYN GREATER THAN SET UP R1,R3 FOR
C                                         INTERPCLATICN
C
   60 DC 61C LOCR=3,MAXC
      IF(REYN-A(LVL,1,LCCR)) 7C,7C,610
   70 R1=A(LVL,1,LCCR-1)
      R3=A(LVL,1,LCCR)
      GC TC (80,150), IS
C
C                                         SECTICN FOR NORMAL LOOK UP CF ALPHA AND
C                                         CVAL. A VALUE CF 999 SPECIFIES THAT THIS
C                                         VARIABLE IS THE ONE WHOSE VALUE IS TC BE
C                                         FCUND IN THE TABLE.
C
   80 IF(ALPHA-999.) 90,110,90
   90 IF(REYN-999.) 1C0,600,1CC
  100 IF(CVAL-999.) 6C0,190,6CC
  110 IF(CVAL-999.) 12C,130,12C
  120 IF(REYN-999.) 160,6CC,16C
  130 IF(REYN-999.) 6C0,140,6CC
  140 IF(REYCN-999.) 15C,6CC,15C
C
C                                         SECTICN FCR LOOKING UP MAX VALUES
C
  150 IF(XMAX) 58C,590,580
C
C                                         SECTICN TO LCOK UP ALPHA FCR GIVEN CVAL BY
C                                         INTERPCLATING BETWEEN CCLUMNS WHERE
C                                         REYNCLCS NUMBERS BRACKET GIVEN REYN
C
  160 DC 17C J=3,LRCW
      CCC=TERP(R1,REYN,R3,A(LVL,J,LCCR-1),A(LVL,J,LOCR))
      IF(CVAL-CCC) 18C,180,170
  170 CCNTINUE
      IE=2
      GC TC 6C0
  180 CC2=TERP(R1,REYN,R3,A(LVL,J-1,LOCR-1),A(LVL,J-1,LOCR))
      ALPHA=TERP(CC2,CVAL,CCC,A(LVL,J-1,1),A(LVL,J,1))
      GC TC 6C0
C
C                                         LCCK UP CVAL FOR GIVEN ALPHA. CHECK TO SEE
C                                         IF WE HAVE A LIFT CR DRAG TABLE IN AT THIS
C                                         TIME. DRAG TABLES HAVE ZERO VALUES FCR ALL
C                                         MAXIMUM VALUES (LAST TWO RCWS).
C
  190 IF(A(LVL,MAXR,LCCR-1)) 210,2C0,210
  200 IF(A(LVL,MAXR,LCCR)) 21C,550,21C
C
C                                         FCR LIFT TABLE. DUMY1 AND DUMY2 ARE THE
C                                         MAXIMUM INTERPCLATED VALUES FOR GIVEN
C                                         REYNCLCS NUMBERS CF ALPHA MAX AND CVAL MAX
C
  210 CCNTINUE
      DUMY1=TERP(R1,REYN,R3,A(LVL,MAXR,LCCR-1),A(LVL,MAXR,LOCR))
      DUMY2=TERP(R1,REYN,R3,A(LVL,MAXR-1,LOCR-1),A(LVL,MAXR-1,LOCR))
      IF(XTRAL-1.) 22C,24C,22C
  220 IF(XTRAL-1.) 23C,24C,23C
  230 CCNTINUE
      IF((ALPHA-A(LVL,MAXR,LCCR-1))*(ALPHA-A(LVL,MAXR,LCCR)))240,250,250
  240 IF(A(LVL,MAXR,LCCR)-A(LVL,MAXR,LOCR-1)) 270,260,26C
  25C GC TC 55C
  26C ALPH1=A(LVL,MAXR,LCCR-1)
      ALPH2=A(LVL,MAXR,LCCR)
      GC TC 28C
  270 ALPH1=A(LVL,MAXR,LCCR)
      ALPH2=A(LVL,MAXR,LCCR-1)
  28C IF(XTRAL-1.) 290,3CC,290
  290 IF(XTRAL-1.) 48C,390,48C
  300 IF(ALPH2-DUMY1) 35C,550,31C
  310 IF((ALPHA-DUMY1)*(ALPHA-ALPH2)) 32C,550,550
  320 DC 33C J=3,LRCW
      IF(ALPH2-A(LVL,J,1)) 34C,340,330
  330 CCNTINUE
```

```
      IE=3
      GO TO 600
  340 C1=A(LVL,J,LOCR-1)
      CVAL=DUMY2+(A(LVL,MAXR-1,LOCR-1)-C1)/(ALPH1-ALPH2)*(ALPHA-DUMY1)
      GO TO 600
  350 IF((ALPHA-DUMY1)*(ALPHA-ALPH1)) 360,550,550
  360 DO  370 J=3,LROW
      IF(ALPH1-A(LVL,J,1)) 380,380,370
  370 CONTINUE
      IE=3
      GO TO 600
  380 C1=A(LVL,J,LOCR-1)
      CVAL=DUMY2+(A(LVL,MAXR-1,LOCR-1)-C1)/(ALPH2-ALPH1)*(ALPHA-DUMY1)
      GO TO 600
  390 IF(ALPH2-DUMY1) 400,550,440
  400 IF((ALPHA-DUMY1)*(ALPHA-ALPH1)) 410,550,550
  410 DO 420 J=3,LROW
      IF(ALPH1-A(LVL,J,1)) 430,430,420
  420 CONTINUE
      IE=3
      GO TO 600
  430 C1=A(LVL,J,LOCR)
      CVAL=DUMY2+(A(LVL,MAXR-1,LOCR)-C1)/(ALPH2-ALPH1)*(ALPHA-DUMY1)
      GO TO 600
  440 IF((ALPHA-DUMY1)*(ALPHA-ALPH2)) 450,550,550
  450 DO 460 J=3,LROW
      IF(ALPH2-A(LVL,J,1)) 470,470,460
  460 CONTINUE
      IE=3
      GO TO 600
  470 C1=A(LVL,J,LOCR)
      CVAL=DUMY2+(A(LVL,MAXR-1,LOCR)-C1)/(ALPH2-ALPH1)*(ALPHA-DUMY1)
      GO TO 600
  480 IF(ALPHA-DUMY1) 490,490,520
  490 DO 500 J=3,LROW
      IF(ALPH1-A(LVL,J,1)) 510,510,500
  500 CONTINUE
      IE=3
      GO TO 600
  510 C1=A(LVL,J,LOCR-1)
      C3=A(LVL,J,LOCR)
      C1=TERP(R1,REYN,R3,C1,C3)
      CVAL=TERP(ALPH1,ALPHA,DUMY1,C1,DUMY2)
      GO TO 600
  520 DO 530 J=3,LROW
      IF(ALPH2-A(LVL,J,1)) 540,540,530
  530 CONTINUE
      IE=3
      GO TO 600
  540 C1=A(LVL,J,LOCR-1)
      C3=A(LVL,J,LOCR)
      C3=TERP(R1,REYN,R3,C1,C3)
      CVAL=TERP(DUMY1,ALPHA,ALPH2,DUMY2,C3)
      GO TO 600
  550 DO 560 J=3,LROW
C
C                          SEARCH FOR ALPHA
C
      IF(ALPHA-A(LVL,J,1)) 570,570,560
  560 CONTINUE
      IE=3
      GO TO 600
  570 CCC=TERP(R1,REYN,R3,A(LVL,J-1,LOCR-1),A(LVL,J-1,LOCR))
      CC2=TERP(R1,REYN,R3,A(LVL,J,LOCR-1),A(LVL,J,LOCR))
      CVAL=TERP(A(LVL,J-1,1),ALPHA,A(LVL,J,1),CCC,CC2)
      GO TO 600
  580 XMAX=TERP(R1,REYN,R3,A(LVL,MAXR,LOCR-1),A(LVL,MAXR,LOCR))
      GO TO (600,620), IS
  590 XMAX=TERP(R1,REYN,R3,A(LVL,MAXR-1,LOCR-1),A(LVL,MAXR-1,LOCR))
      GO TO (600,620), IS
600   RETURN
  510 CONTINUE
      IE=1
      GO TO 600
  620 REYN=999.
      GO TO 600
      END
      SUBROUTINE FDISK(ITNO,TLOSS)
C
C
```

```
C
C                                     SUBROUTINE TO STORE TABULATED PROPELLER
C                                     TIP LOSS CORRECTION DATA ON DISK
C
      REAL TLOSS(8)
      READ(11'ITNC) TLOSS
      RETURN
      END
      SUBROUTINE GDISK(ITNC,TALFA,TLIFT)
C
C
C                                     SUBROUTINE TO STORE TABULATED PROPELLER
C                                     AIRFOIL SECTION DATA ON DISK -
C                                     ALPHA AND CL VALUES ONLY
C
C
      REAL TALFA(20),TLIFT(20)
      READ(12'ITNC) TALFA
      READ(13'ITNC) TLIFT
      RETURN
      END
      SUBROUTINE SLIP
C
C
C                                     SUBROUTINE TO CALCULATE PROPELLER
C                                     SLIPSTREAM VELOCITY DISTRIBUTION USING
C                                     NON - LINEAR BLADE AIRFOIL SECTION
C                                     LIFT CHARACTERISTICS
C
C
      DIMENSION NS(19),YBW(19),RSBW(19),RSBA(19),VST(19),
     1RSBR(12),USABR(12),USTBR(12),BDATA(12,5),NDATA(12),TITLE(19),
     2IHEAD(150,2),THEAD(150,5),TALFA(20),TLIFT(20),TLOSS(8),
     3AFSER(9,2),AK(5,2),NG(9),ITA(9,2),ITN(9,25),ITC(4,2),ITM(2),FR(2),
     4TCLI(4),TTCC(4),TXCL(4),CLG(4),CLM(2),CLC(2),NRCT(2),ROT(2),
     5IHECC(2),THECC(5)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB
C
C     ********************            INITIALIZE DATA ARRAYS
      DATA AK/10*0.0/
      DATA NG/9*0/
      DATA ITA/18*0/
      DATA AFSER/18*4H     /
      DATA AFSER(1,1),AFSER(1,2)/4H  US,4HNPS /
      DATA AFSER(2,1),AFSER(2,2)/4H CLA,4HRK.Y/
      DATA AFSER(3,1),AFSER(3,2)/4H NAC,4HA 16/
      DATA AFSER(4,1),AFSER(4,2)/4H NAC,4HA 64/
      DATA AFSER(5,1),AFSER(5,2)/4H NAC,4HA 65/
      DATA ARCTL,ARCTC,ARCTR/3H LH,3H   ,3H RH/
C
C                                     CHECK VALUE OF IPRAB
C
C                                     EQ 1   -   NEW CASE, READ CARD INPUT
C                                     NE 1   -   NEW ALPHB ONLY, SKIP CARD INPUT
C
      IF (IPRAB-1) 2350,2000,2350
2000  CONTINUE
C
C                                     SET FIXED CONSTANTS
      KP=6
      KR=8
      PI=3.1415927
      RTD=57.29578
C                                     SET AIRFOIL CONSTANTS FOR INITIAL SOLUTION
      AK(1,2)=-40.0
      AK(2,2)=-40.0
      AK(3,1)=-7.3
      AK(4,1)=-6.9
      AK(5,1)=-6.9
C     ..................              READ, STORE AND INDEX TIP LOSS CORR TABLES
C
C                                     **                                          **
C                                     **    DATA CARDS ACCEPTED IN ANY ORDER,     **
C                                     **    BUT MUST NUMBER EITHER 0 OR 63        **
C                                     **                                          **
C                                     **    NONE REQUIRED IF TIP LOSS TABLES      **
C                                     **    ARE ALREADY STORED ON DISK FILE       **
C                                     **                                          **
C                                     **    FOLLOWING CARD MUST BE BLANK          **
```

```
C                                                     **                                                       **
C
      IC=0
 2100 READ(KR,2901) IB,IP,(TLOSS(IR),IR=1,8)
      IF (IB) 2110,2120,2110
 2110 IA=21*(IB-1)+IP
      WRITE(11'IA) TLOSS
      IC=IC+1
      GO TO 2100
 2120 WRITE(KP,2981) IC
      IF (IC) 2130,2200,2130
 2130 IF (IC-63) 2140,2200,2140
 2140 WRITE(KP,2982)
      GO TO 2899
C     ......................  READ, STORE AND INDEX AIRFOIL TABLES
C
C                             **                                      **
C                             **   TITLE CARDS MUST BE READ FIRST -   **
C                             **   NOT REQUIRED FOR STANDARD TABLES   **
C                             **                                      **
C                             **   FOLLOWING CARD MUST BE BLANK       **
C                             **                                      **
C                             **   DATA CARDS FOR EACH AIRFOIL SET    **
C                             **   MUST BE ASSEMBLED IN DESCENDING    **
C                             **   ORDER OF  -  CLI INCREASING        **
C                             **            -  T/C INCREASING         **
C                             **            - MACH INCREASING         **
C                             **                                      **
C                             **   NONE REQUIRED FOR SECOND CASE      **
C                             **                                      **
C                             **   FOLLOWING CARD MUST BE BLANK       **
C                             **                                      **
C
C                             READ AIRFOIL SERIES TITLE CARDS
C
 2200 READ(KR,2911) I,AFSR1,AFSR2,AK1,AK2
      IF (I) 2201,2202,2201
 2201 AFSER(I,1)=AFSR1
      AFSER(I,2)=AFSR2
      AK(I,1)=AK1
      AK(I,2)=AK2
      GO TO 2200
 2202 CONTINUE
C                             SET TABLE INDEX AND READ FIRST TABLE
      IT=1
      READ(KR,2912) IHEDD(1),IHEDD(2),(THEDD(I),I=1,5)
      IF (IHEDD(1)) 2204,2203,2204
 2203 WRITE(KP,2988) NT
      GO TO 2296
C
C                             INITIALIZE AIRFOIL SECTION INDEX
 2204 DO 2205 I=1,9
 2205 ITN(I,1)=0
      IHEAD(IT,1)=IHEDD(1)
      IHEAD(IT,2)=IHEDD(2)
      DO 2208 I=1,5
 2208 THEAD(IT,I)=THEDD(I)
      IB=IHEAD(IT,1)
      READ(KR,2913) (TALFA(I),TLIFT(I),I=1,IB)
      WRITE(12'IT) TALFA
      WRITE(13'IT) TLIFT
C                             SET LAST VALUES OF AIRFOIL CODE,
C                             CLI, T/C, AND AIRFOIL SECTION INDEX
 2210 IA=IHEAD(IT,2)
      THED1=THEAD(IT,1)
      THED2=THEAD(IT,2)
      IG=1
      ITN(IA,IG)=IT
      ITA(IA,1)=IT
C                             SET TABLE INDEX, READ HEADER FOR NEXT TABLE
 2220 IT=IT+1
      READ(KR,2912) IHEAD(IT,1),IHEAD(IT,2),(THEAD(IT,I),I=1,5)
C
C                             CHECK FOR LAST AIRFOIL TABLE
C
      IF (IHEAD(IT,2)) 2225,2230,2225
C
C                             CONTINUE, READ DATA CARDS FOR NEXT TABLE
 2225 IB=IHEAD(IT,1)
      READ(KR,2913) (TALFA(I),TLIFT(I),I=1,IB)
```

```
      WRITE(12'IT) TALFA
      WRITE(13'IT) TLIFT
      IF (IHEAD(IT,2)-IA) 2230,2250,2230
2230  NG(IA)=IG
      IG=IG+1
      ITN(IA,IG)=IT
      ITA(IA,2)=IT-1
      IF (IHEAD(IT,2)) 2210,2240,2210
2240  NT=IT-1
      GO TO 2295
2250  IF(THEAD(IT,1)-THED1) 2270,2260,2270
2260  IF(THEAD(IT,2)-THED2) 2280,2220,2280
C
C                                   RESET LAST VALUE OF CLI
2270  THED1=THEAD(IT,1)
      IF(THEAD(IT,2)-THED2) 2280,2290,2280
C                                   RESET LAST VALUE OF T/C
2280  THED2=THEAD(IT,2)
C                                   SET AIRFOIL SECTION INDEX
2290  IG=IG+1
      ITN(IA,IG)=IT
      GO TO 2220
2295  CONTINUE
C                                   PRINT SUMMARY OF AIRFOIL TABLES READ IN
      WRITE(KP,2983) NT
2296  WRITE(KP,2984) (I,I=1,9)
      WRITE(KP,2985) (AFSER(I,1),AFSER(I,2),I=1,9)
      WRITE(KP,2986) (NG(I),I=1,9)
      WRITE(KP,2987) (ITA(I,1),ITA(I,2),I=1,9)
C
C     ..................            READ CASE INPUT DATA CARDS IDENT 1 - 4
C
C                                   READ TITLE CARD
C
      READ(KR,2921) TITLE,IDENT
C
      IF (IDENT-1) 2897,2330,2897
C
C                                   READ PROPELLER-NACELLE GEOMETRY CARD
C
2330  READ(KR,2923) NP,NB,DPB,RHBR,RNBR,IDENT
      IF (IDENT-2) 2897,2340,2897
C
C                                   READ PROPELLER OPERATING CONDITION CARD
C
2340  READ(KR,2924) NROT(1),NROT(2),AJ,AMCHU,IDENT
      IF (IDENT-3) 2897,2341,2897
2341  DO 2345 I=1,2
      IF (NROT(I)) 2342,2343,2344
2342  ROT(I)=AROTL
      GO TO 2345
2343  ROT(I)=AROTO
      GO TO 2345
2344  ROT(I)=AROTR
2345  CONTINUE
C
C     ..................            CALCULATE BASIC CASE PARAMETERS
2350  ALFP=ALPHB+EYETL
      ALFPR=ALFP/RTD
      CA=COS(ALFPR)
      AMUT=AJ*CA/PI
      AMCHT=PI*AMCHU/AJ
      DNP=DPB*RNBR
C     ..................            PRINT MAIN HEADER TITLES AND INPUT VALUES
      WRITE(KP,2951) TITLE
      WRITE(KP,2952)
      WRITE(KP,2953) NP,NB,ROT(1),ROT(2),XPB,RHBR,AJ
      WRITE(KP,2954) YPB,RNBR,AMCHU,DPB,EYETL,ALFP
      WRITE(KP,2956)
C     ..................            INITIALIZE SLIPSTREAM VALUES
C                                   AT HUB AND NACELLE
      IS=0
      RPBRX=RHBR
      RSBRX=RNBR
      USBRX=1.0
      DCTX=0.0
      DCQX=0.0
      CT=0.0
      CQ=0.0
C     ..................            READ BLADE STATION DATA CARD IDENT 4
```

```
C
C                                             **                                      **
C                                             **    LAST DATA CARD MUST HAVE RPBR=1.0 **
C                                             **                                      **
C
2360  IS=IS+1
      IF (IPRAB-1) 2370,2365,2370
2365  READ(KR,2925) (BDATA(IS,I),I=1,3),NDATA(IS),
     1(BDATA(IS,I),I=4,5),IDENT
      IF (IDENT-4) 2897,2370,2897
2370  RPBR=BDATA(IS,1)
      CPBR=BDATA(IS,2)
      BETA=BDATA(IS,3)
      CLI=BDATA(IS,4)
      TCC=BDATA(IS,5)
      NA=NDATA(IS)
C                                   CHECK FOR LAST BLADE STATION DATA CARD
C
      IF (RPBR-1.0) 2400,2380,2380
C
C     ...................           CALCULATE BOUNDARY VALUES FOR SLIPSTREAM
2380  NIT=0
      F=0.0
      AMACH=AMCHT*SQRT(1.0+AMUT**2)
      ALPHA=0.0
      CL=0.0
      CD=0.0
      RSBR(IS)=SQRT(RSBRX**2+(1.0-RPBRX**2)*(0.5+1.0/(1.0+USBRX)))
      RSTBR=RSBR(IS)
      USABR(IS)=1.0
      USTBR(IS)=0.0
      PHIS=0.0
      CT=CT+0.5*DCTX*(1.0-RPBRX)
      CQ=CQ+0.5*DCQX*(1.0-RPBRX)
      GO TO 2750
2400  CONTINUE
C     ...................           LOCATE AND INDEX AIRFOIL TABLES
C                                   AND GEOMETRIC DATA AS REQUIRED
C                                   TO LOOK UP CL FOR SPECIFIED
C                                   AIRFOIL SECTION GEOMETRY
      IT=ITN(NA,1)
      IF (IT) 2410,2405,2410
2405  WRITE(KP,2994) NA,RPBR
      IS=IS-1
      GO TO 2360
2410  THED1=THEAD(IT,1)
      DO 2411 I=1,4
      ITG(I,1)=0
2411  ITG(I,2)=0
      IA=1
      IB=2
      IC=1
      ID=0
      IE=0
      IG=0
2415  IG=IG+1
      IT=ITN(NA,IG)
      IF (THEAD(IT,1)-CLI) 2420,2440,2445
2420  IF (THEAD(IT,1)-THED1) 2425,2430,2425
2425  IC=IG
      THED1=THEAD(IT,1)
2430  IF (IG-NG(NA)) 2415,2435,2415
2435  IG=IC
      IT=ITN(NA,IG)
      GO TO 2455
2440  THED1=THEAD(IT,1)
      GO TO 2455
2445  IF (IG-1) 2450,2455,2450
2450  IC=IG
      IG=IC
2455  IF (THEAD(IT,2)-TCC) 2460,2475,2470
2460  IE=1
      ITG(IA,1)=IT
      TCLI(IA)=THEAD(IT,1)
      TTCC(IA)=THEAD(IT,2)
      TXCL(IA)=THEAD(IT,5)
      IG=IG+1
      IT=ITN(NA,IG)
      ITG(IA,2)=IT-1
      IF (IG-NG(NA)) 2465,2465,2485
```

```
2465   IF (THEAD(IT,1)-THED1) 2485,2455,2485
2470   IF (IE) 2480,2475,2480
2475   ITG(IA,1)=IT
       TCLI(IA)=THEAD(IT,1)
       TTCC(IA)=THEAD(IT,2)
       TXCL(IA)=THEAD(IT,5)
       IG=IG+1
       IT=ITN(NA,IG)
       ITG(IA,2)=IT-1
       GO TO 2485
2480   ITG(IB,1)=IT
       TTCC(IB)=THEAD(IT,2)
       TXCL(IB)=THEAD(IT,5)
       IG=IG+1
       IT=ITN(NA,IG)
       ITG(IB,2)=IT-1
2485   IF (ID) 2490,2495,2490
2490   IG=ID
       IT=ITN(NA,IG)
       THED1=THEAD(IT,1)
       IA=3
       IB=4
       ID=0
       IE=0
       GO TO 2455
2495   CONTINUE
C      ...................       CALCULATE INITIAL VALUES OF PHIR AND CX FOR
C                                BASIC ITERATION ROUTINE
C
       SCL=NB*CPBR/(2.0*PI*RPBR)
       AMU=AMUT/RPBR
       PHIOR=ATAN(AMU)
       PHIO=PHIOR*RTD
       AMACH=AMCHT*RPBR/COS(PHIOR)
       AC=0.1/SQRT(1.0-AMACH**2)
       ALPHO=CLI*AK(NA,1)+TCC*AK(NA,2)
       X=-0.5*NB*(1.0-RPBR)*SQRT(1.0+(1.0/AMUT)**2)
       Y=EXP(X)
       FP=(2.0/PI)*ATAN((SQRT(1.0-Y**2))/Y)
       X=SCL*AC*RTD*SQRT(1.0+AMU**2)/(4.0*FP)
       Y=(BETA-PHIO-ALPHO)/RTD
       UXOZR=0.5*(SQRT((AMU+X)**2+4.0*X*Y)-(AMU+X))
       PHIR=PHIOR+UXOZR
       PHI=PHIR*RTD
       X=UXOZR*SIN(PHIR)/COS(PHIR)
       CX=X/(1.0-X)
       NIT=1
2500   CONTINUE
C      ...................       BASIC ITERATION ROUTINE
       CP=COS(PHIR)
       SP=SIN(PHIR)
C                                LOOK UP TIP LOSS CORRECTION FACTOR
C
       X=-0.5*NB*(1.0-RPBR)*SQRT(1.0+(1.0/((RPBR*SP/CP)**2)))
       Y=EXP(X)
       FP=(2.0/PI)*ATAN((SQRT(1.0-Y**2))/Y)
C
C                                CHECK IF NB GT 4  -  SET FCFP = 1.0
C
       IF (NB-4) 2520,2520,2510
2510   FCFP=1.0
       GO TO 2560
2520   IB=NB-1
C                                CHECK IF RPBR LT 0.3  -  ASSUME RPBR = 0.3
       AR=10.0*RPBR
       IR=AR
       IF (IR-3) 2530,2540,2540
2530   IR=3
       AR=3.0
C                                CONTINUE - COMPUTE INTERPOLATION FRACTIONS
C                                AND ARRAY INDICES FOR TIP LOSS CORRECTION
C                                VALUES TO BE INTERPOLATED
2540   DR=AR-IR
       IC=IR-2
       ID=IC+1
       AP=20.0*SP
       IP=AP
       DP=AP-IP
C                                INTERPOLATE FOR RPBR AT EACH SIN(PHI)
       DO 2550 I=1,2
```

```
      IP=IP+1
      IA=21*(IB-1)+IP
      CALL FCISK(IA,TLCSS)
2550  FR(I)=TLCSS(IC)+(TLCSS(IC)-TLCSS(IC))*DR
C
C                                   INTERPCLATE FOR SIN(PHI)
C
      FCFP=FR(1)+(FR(2)-FR(1))*CP
2560  F=FP*FCFP
      ALPHA=BETA-PHI
      AMACH=AMCHT*RPBR/((1.C+CX)*CP)
C
C     ...................           LCCK UP CL FCR EACH AIRFCIL TABLE
C                                   AS RECUIRED, THEN INTERPCLATE CL
C                                   FCR ALPHA AND AMACH
      DC 2695 IA=1,4
      IT=ITG(IA,1)
      IF (IT) 26C5,2695,26C5
2605  ALPHC=ALPHA
      ITM(2)=C
      IB=C
261C  IF (THEAD(IT,3)-AMACH) 2615,262C,2625
2615  IB=1
      ITM(1)=IT
      TMCH1=THEAD(IT,3)
      IT=IT+1
      IF (IT-ITG(IA,2)) 261C,261C,2640
262C  IB=1
      ITM(1)=IT
      TMCH1=THEAD(IT,3)
      GC TC 2645
2625  IF (IB) 2635,263C,2635
2630  IB=1
      ITM(1)=IT
      TMCH1=THEAD(IT,3)
      GC TC 264C
2635  IB=2
      ITM(2)=IT
      TMCH2=THEAD(IT,3)
      GC TC 2645
264C  IT=ITM(1)
      ALPHC=THEAD(IT,4)+(ALPHA-THEAD(IT,4))*SQRT((1.0-THEAD(IT,3)**2)/
     1(1.C-AMACH**2))
2645  DC 269C IC=1,2
      IT=ITM(IC)
      IF (IT) 265C,2690,2650
2650  IC=1
      IE=C
      CALL CDISK(IT,TALFA,TLIFT)
2655  IF (TALFA(IC)-ALPHC) 266C,2665,267C
266C  IE=1
      TALF1=TALFA(IC)
      TLFT1=TLIFT(IC)
      IC=IC+1
      IF (IC-IHEAD(IT,1)) 2655,2655,2685
2665  IE=1
      TALF1=TALFA(IC)
      TLFT1=TLIFT(IC)
      GC TC 2685
267C  IF (IC) 268C,2675,268C
2675  IE=1
      TALF1=TALFA(IC)
      TLFT1=TLIFT(IC)
      GC TC 2685
268C  IE=2
      TALF2=TALFA(IC)
      TLFT2=TLIFT(IC)
2685  GC TC (268C,2687),IE
C
C                                   EXTRAPCLATE FOR ALPHA
2686  CLM(IC)=TLFT1
      GC TC 269C
C                                   INTERPCLATE FOR ALPHA
C
2687  CLM(IC)=TLFT1+(TLFT2-TLFT1)*(ALPHC-TALF1)/(TALF2-TALF1)
269C  CCNTINUE
      GC TC (2691,2692),IB
C
C                                   EXTRAPCLATE FOR AMACH
2691  CLG(IA)=CLM(1)
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      GC TC 2695
C                                   INTERPCLATE FOR AMACH
C
2692  CLG(IA)=CLM(1)+(CLM(2)-CLM(1))*(AMACH-TMCH1)/(TMCH2-TMCH1)
2695  CCNTINUE
C                                   INTERPCLATE CL FOR TOC AND CLI
      DC 272C IC=1,2
      IA=2*IC-1
      IB=2*IC
      CLC(IC)=0
      IF (ITG(IA,1)) 27C5,272C,2705
27C5  IF (ITG(IB,1)) 271C,2715,2710
C
C                                   INTERPCLATE FOR TCC
C
2710  CLC(IC)=CLG(IA)+(CLG(IB)-CLG(IA))*
     1(TCC-TTCC(IA))/(TTCC(IB)-TTCC(IA))
      TXCL(IC)=TXCL(IA)+(TXCL(IB)-TXCL(IA))*
     1(TCC-TTCC(IA))/(TTCC(IB)-TTCC(IA))
      TCLI(IC)=TCLI(IA)
      GC TC 2720
C                                   EXTRAPCLATE FOR TCC
2715  CLC(IC)=CLG(IA)
      TCLI(IC)=TCLI(IA)
      TXCL(IC)=TXCL(IA)
2720  CCNTINUE
      IF (CLC(2)) 2725,2730,2725
C
C                                   INTERPCLATE FOR CLI
C
2725  CL=CLC(1)+(CLC(2)-CLC(1))*(CLI-TCLI(1))/(TCLI(2)-TCLI(1))
      GC TC 2735
C                                   EXTRAPCLATE FOR CLI OUTSIDE TABLE LIMIT
C
2730  CL=CLC(1)+TXCL(1)*(CLI-TCLI(1))/SCRT(1.0-AMACH**2)
2735  CCNTINUE
C                                   SET BLADE SECTION DRAG  CD = 0.01C
      CD=C.C1
      X=SCL/(4.C*F)
      CX=X*(CL/CP+CD/SP)
      CY=X*(CL/SP-CD/CP)
      CZ=AMU*(1.C+CX)+CY
      PHIN=ATAN(CZ)*RTC
      IF (ABS(PHIN-PHI)-C.1) 2745,2745,2740
2740  X=AMU*CX*(AC*RTC/CL-SP/CP)
      Y=CY*(AC*RTC/CL+CP/SP)
      CC=1.C+(X+Y)/(1.0+CZ**2)
      PHI=PHI+(PHIN-PHI)/CC
      PHIR=PHI/RTC
      NIT=NIT+1
      IF (NIT-9) 25CC,25CC,2745
C
C                                   CALCULATE VALUES FOR DISK PLANE ELEMENT
C
2745  UCZR=C.5*(SCRT(AMU**2+4.C*F*CY*CZ/(1.0+CX)**2)-AMU)
      UCUA=UCZR/AMU
      WC2Z=UCZR*CX/CY
C                                   CALCULATE VALUES FCR SLIPSTREAM ELEMENT
C
      USABR(IS)=1.C+2.C*UCUA
      RSBR(IS)=SCRT(RSBRX**2+(RPBR**2-RPBRX**2)*
     1(C.5+1.C/(USABR(IS)+USBRX)))
      USTBR(IS)=2.C*WC2Z*RPBR/(RSBR(IS)*AMU)
      PHIS=RTC*ATAN(USTBR(IS)/USABR(IS))
      DCT=(PI*RPBR)**3*F*CY*CZ/(1.C+CX)**2
      DCQ=DCT*(RPBR/2.C)*CX/CY
C
C     ..................            PRINT BLADE ELEMENT SOLUTION
C
2750  WRITE(KP,2957) RPBR,CPBR,BETA,AFSER(NA,1),AFSER(NA,2),CLI,TOC,
     1F,AMACH,ALPHA,CL,CD,RSBR(IS),USABR(IS),USTBR(IS),PHIS
      IF (RPBR-1.C) 2755,281C,281C
2755  IF (NIT-9) 28CC,28CC,276C
2760  WRITE(KP,2995)
      IS=IS-1
      GC TC 2360
2800  CCNTINUE
C                                   SUM INTEGRAL TERMS FOR SLIPSTREAM ELEMENT
C
      CT=CT+C.5*(DCT+DCTX)*(RPBR-RPBRX)
```

```
      CQ=CQ+0.5*(CCC+DCCX)*(RPBR-RPBRX)
C
C                               RESET LAST VALUES OF INTEGRAL TERMS
      RPBRX=RPBR
      RSBRX=RSBR(IS)
      USBRX=USABR(IS)
      DCTX=DCT
      DCCX=DCC
      GO TO 2360
2810  CONTINUE
C
C     ...................       CALCULATE FINAL SLIPSTREAM INTEGRALS
C
      VSBAR=CA*SQRT(1.0+8.0*CT/(PI*(PI*AMUT)**2))
      CTS=1.0/(1.0+(PI*(PI*AMUT)**2)/(8.0*CT))
C
C     ...................       PRINT FINAL SLIPSTREAM INTEGRALS
C
      WRITE(KP,2958) CTS,CT,CQ,VSBAR
C
C     ...................       CALCULATE SLIPSTREAM VALUES FOR
C                               INPUT TO WING ANALYSIS
C
      IF (NRCT(1)) 2812,2811,2812
2811  NR=-1
      GO TO 2813
2812  NR=-1+IABS(NRCT(2)-NROT(1))
2813  KCR=0
      NW=2*JP
      DO 2860 K=1,JP
      YBW(K)=COS(K*PI/NW)
      RSBW(K)=(YBW(K)-YPB)/DPB
      RSBA(K)=ABS(RSBW(K))
      IF (RSBA(K)-RSTBR) 2815,2850,2850
2815  NS(K)=1
      IF (KCR) 2817,2816,2817
2816  KCR=K
2817  KIR=K
      ISIGN=-NRCT(2)*RSBW(K)/RSBA(K)
      IA=0
      IB=1
2820  IF (RSBR(IB)-RSBA(K)) 2825,2830,2835
2825  IA=IB
      IB=IB+1
      GO TO 2820
2830  VSA(K)=CA*USABR(IB)
      VST(K)=CA*USTBR(IB)*ISIGN
      SW(K)=VST(K)/VSA(K)
      GO TO 2855
2835  IF (IA) 2845,2840,2845
2840  VSA(K)=CA*USABR(IB)
      VST(K)=CA*USTBR(IB)*ISIGN*RSBA(K)/RSBR(IB)
      SW(K)=VST(K)/VSA(K)
      GO TO 2855
2845  X=(RSBA(K)-RSBR(IA))/(RSBR(IB)-RSBR(IA))
      VSA(K)=CA*(USABR(IA)+(USABR(IB)-USABR(IA))*X)
      VST(K)=CA*(USTBR(IA)+(USTBR(IB)-USTBR(IA))*X)*ISIGN
      SW(K)=VST(K)/VSA(K)
      GO TO 2855
2850  NS(K)=0
      VSA(K)=1.0
      VST(K)=0.0
      SW(K)=0.0
2855  L=NW-K
      NS(L)=NS(K)
      YBW(L)=YBW(K)*(-1.0)
      RSBA(L)=RSBA(K)
      VSA(L)=VSA(K)
      VST(L)=VST(K)*NR
      SW(L)=VST(L)/VSA(L)
2860  CONTINUE
      IF (KIR-JP) 2862,2861,2862
2861  KIR=NW-KCR
2862  KIL=NW-KIR
      KCL=NW-KCR
C     ...................       PRINT SLIPSTREAM VALUES AT WING STATIONS
      WRITE(KP,2961)
      IC=NW-1
      DO 2870 K=1,IC
      IF (NS(K)) 2870,2870,2865
2865  WRITE(KP,2962) K,YBW(K),RSBA(K),VSA(K),VST(K),SW(K)
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 2870  CCNTINUE
 2897  WRITE(KP,2991) IDENT
 2899  CCNTINUE
C
C      ....................     READ FORMATS
C
 2901  FCRMAT(I1,I2,29X,8F6.0)
 2911  FCRMAT(I1,9X,2A4,2X,2F10.0)
 2912  FCRMAT(I2,2X,I1,3X,5F8.0)
 2913  FCRMAT(10F8.0)
 2921  FCRMAT(19A4,3X,I1)
 2923  FCRMAT(I2,8X,I2,8X,3F10.0,29X,I1)
 2924  FCRMAT(2(I2,8X),2F10.0,39X,I1)
 2925  FCRMAT(3F10.0,I1,9X,2F10.0,19X,I1)
C
C      ....................     PRINT FCRMATS
C
 2951  FCRMAT(1H1///8X,32HPRCPELLER SLIPSTREAM ANALYSIS  -,2X,19A4/1X)
 2952  FCRMAT(1H0,7X,25HPROPELLER - WING GEOMETRY,14X,
      A              28HPROPELLER - NACELLE GEOMETRY,12X,
      B              29HPRCPELLER CPERATING CONDITION/1X)
 2953  FCRMAT(1H ,4X,25HNUMBER CF PRCPS          =,I2,13X,
      A              25HNC CF BLADES PER PROP    =,I2,13X,
      B              25HLEFT / RIGHT PROP ROTN   =,A3,1H,,A3/
      C         5X,25HPROP FWC CCCRD 2.XP/B    =,F7.4,8X,
      D              25HHUB DIA / PRCP DIA       =,F7.4,8X,
      E              25HPRCP ADVANCE RATIO       =,F7.4)
 2954  FCRMAT(1H ,4X,25HPROP SPAN CCCRD 2.YP/B   =,F7.4,8X,
      A              25HNACELLE DIA / PRCP DIA   =,F7.4,8X,
      B              25HFLIGHT MACH NUMBER       =,F7.4/
      C         5X,25HPRCP DIA / WING SPAN     =,F7.4,8X,
      D              25HPROP AXIS REL BODY AXIS =,F7.3,4H DEG,4X,
      E              25HPRCP ANGLE CF ATTACK     =,F7.3,4H DEG)
 2956  FCRMAT(1H-,12X,22HBLADE ELEMENT GEOMETRY,22X,
      A22HBLADE ELEMENT SCLUTION,14X,27HSLIPSTREAM ELEMENT SOLUTION//
      B2X,5HRB/RP,3X,5HCB/RP,3X,5HPITCH,2X,7HA/F SER,
      C3X,3HCLI,4X,3HT/C,9X,1HF,5X,4HMACH,2X,5HALPHA,4X,2HCL,
      D5X,2HCD,8X,5HRS/RP,2X,6HUSA/UA,2X,6HUST/UA,4X,4HPHIS/1X)
 2957  FCRMAT(1H ,F6.4,F8.4,F8.3,1X,2A4,2F7.3,4X,5F7.3,4X,3F8.4,F8.3)
 2958  FCRMAT(1H-,    4X,36HPRCPELLER THRUST COEFFICIENT, CT'' =,F7.4/
      A              5X,36HPRCPELLER THRUST CCEFFICIENT, CT   =,F7.4/
      B              5X,36HPRCPELLER TORQUE COEFFICIENT, CQ   =,F7.4/
      C              5X,36HMOMENTUM WGTD SLIPSTREAM VEL RATIO =,F7.4)
 2961  FCRMAT(1H-,4X,42HSLIPSTREAM VALUES AT WING CONTROL STATIONS//6X,
      A1HK,7X,4H2Y/B,7X,5HRS/RP,7X,6HUSA/U0,6X,6HUST/U0,5X,7HUST/USA/1X)
 2962  FCRMAT(1H ,4X,I2,5(5X,F7.4))
 2981  FCRMAT(1H1,9X,I3,42H TIP LCSS CORRECTION TABLE DATA CARDS READ/1X)
 2982  FCRMAT(1H ,13X,44HIS INVALID - SLIPSTREAM CCMPUTATICNS ABORTED/1X)
 2983  FCRMAT(1H-,9X,I3,41H PRCPELLER AIRFCIL TABLES READ AS FOLLOWS/1X)
 2984  FCRMAT(1H0,14HAIRFCIL CODES ,9(5X,I3,3X))
 2985  FCRMAT(1H0,14HAIRFCIL SERIES,9(3X,2A4))
 2986  FCRMAT(1H0,14HNC CF SECTICNS,9(5X,I3,3X))
 2987  FCRMAT(1H0,14HTABLE NUMBERS ,9(3X,I3,2H -,I3))
 2988  FCRMAT(1H-,11X,32H0 PRCPELLER AIRFCIL TABLES READ,,I3,47H PREVIOUS
      A TABLES READ AND STORED ARE AS FOLLOWS)
 2991  FCRMAT(1H-,10X,10HCARD IDENT,I2,
      A62H HAS BEEN READ CUT CF SEQUENCE, SLIPSTREAM COMPUTATION ABORTED)
 2994  FCRMAT(1H-,44HAIRFCIL TABLES NCT STORED FCR AIRFOIL SERIES,I2,
      A21H SPECIFIED AT RB/RP =,F7.4,
      B46H  -  THIS ELEMENT IS DELETED FROM THE ANALYSIS/1X)
 2995  FCRMAT(1H0,109HSCLUTION FCR PRECEDING ELEMENT FAILED TO CONVERGE I
      AN 9 ITERATIONS AND IS DELETED FROM THE SLIPSTREAM ANALYSIS/1X)
       RETURN
       END
       SUBROUTINE MAIN
       DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(1
      19),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19),
      2MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),TRIX
      3(19,19),CY(19),DIST(19),YDA(19),YDAX(19),YX(19),TCNY(19),ALPG(19),
      4XHERE(2,6),MWCCL(6),MAWW(6)
       DIMENSION CVAL(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(19),
      1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
       CCMMCN KCR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
      1IPRAH,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
       CCMMCN NSLIP,VR(19),CA(19),AS(19),TL(19),SV(19),FUNC,YO,CONAC,
      1ALPHV(16)
       CCMMCN INACH,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
      1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
      2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,Q,TSTAX,EDGE
      3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,F,TAUT,TAUR,
```

```
      4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TCNY,TWISA,X,Z,CM,ACC,XHERE,
      5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
      6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCCL(6),CHERE(2,6),MACC(6),
      7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STONY(19),SGENE(19),CVAL,
      8ALPHO,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
      9F,IRI,FF,LCCER
       ACCS(X)=ATAN(SQRT(1.-X*X)/X)
C
C                                     READER (IR) AND PRINTER (IP) LOGICAL UNIT
C                                     NUMBERS
C
       IR=8
       IP=6
       REWIND 1
       REWIND 2
       REWIND 3
       REWIND 4
       REWIND 5
       REWIND 10
       REWIND 15
       REWIND 20
       REWIND 7
       KEEP=1
C
C                                     INPUT DATA SECTION
C
       IS=1
       READ(IR,650) ASPEC,TALT,TALR,TAPER,TWIST,R,BF,REYNC,DISCR,A,B,H,
      1ALPHR,NFLAP,FLAP,X,Z,TWISA,CAMBT,CAMBR,NSLIP,CONAC
       IF(ASPEC-99.) 20,10,10
   10  CALL EXIT
   20  CONTINUE
C
C                                     LAYOUT OF FOURTH DATA CARD
C
C                                     FIELD 1   I1   NUMBER OF TAU VALUES PER TABLE
C
C                                     FIELD 2   I1   1 FOR READ IN OF TAU, REY,ETC.
C                                                    0 FOR NO READ IN
C
C                                     FIELD 3   I1   1 FOR DUMP OF COMPUTED ARRAYS
C                                                    0 FOR NO DUMP
C
C                                     FIELD 4   I1   1 FOR DUMP OF BETA ARRAY
C                                                    0 FOR NO DUMP
C
C                                     FIELD 5   I1   1 READ CUBE 1 FROM TAPE, LOAD
C                                                      TO DISK, COPY CUBE 1 TO CUBE
C                                                      2 ON DISK.
C                                                    2 READ CUBE 1 FROM TAPE, LOAD
C                                                      TO DISK, READ CUBE 2 FROM
C                                                      TAPE,LOAD TO DISK.
C                                                    3 READ CUBE 1 FROM CARDS, LOAD
C                                                      TO TAPE, LOAD TO DISK, COPY
C                                                      CUBE 1 TO CUBE 2 ON DISK
C                                                    4 READ CUBE 1 FROM CARDS, LOAD
C                                                      TO TAPE, LOAD TO DISK, READ
C                                                      CUBE 2 FROM CARDS, LOAD TO
C                                                      TAPE, LOAD TO DISK
C
C                                     FIELD 6   25A2   FIFTY COLUMNS OF IDENTIFING
C                                                      INFORMATION. THIS IS PRINTED
C                                                      AT THE TOP OF EACH PAGE OF
C                                                      OUTPUT
C
       READ (IR,660) NLVL,ISWIT,IG,NAME
C
C                                     SWITCH ZERO ON WILL READ IN VALUES FROM
C                                     CARDS FOR TAU, REY, C, EPS, EDGE, CRB, AND
C                                     ACC. FORMAT IS 16F5.0. ARRAYS ARE READ IN
C                                     ROWWISE
C
       CALL SETSW
       DO 30 J=1,6
       WHERE(1,J)=0.
       WHERE(2,J)=0.
       MAWW(J)=0
   30  MACCL(J)=0
       IY=0
       IF(IG-3) 80,40,40
```

```
   40 DO 50 J=1,4
      CALL AERDA(ARRAY,NLVL,WHERE,MACCL,MAWW,7,KEEP)
      WRITE(7) ARRAY
   50 WRITE(7) WHERE,MACCL,MAWW
      IF(IG-4) 80,60,80
   60 DO 70 J=1,4
      CALL AERDA(ARRAY,NLVL,WHERE,MACCL,MAWW,7,KEEP)
      WRITE(7) ARRAY
   70 WRITE(7) WHERE,MACCL,MAWW
   80 IK=1
      REWIND 7
      READ(7) ARRAY
      READ(7) WHERE,MXCCL,MAXX
      WRITE(7) ARRAY
      IK=IK+1
      READ(7) ARRAY
      READ(7) ZHERE,MZCCL,MAZZ
      WRITE(7) ARRAY
      IK=IK+1
      READ(7) ARRAY
      READ(7) YHERE,MYCCL,MAYY
      WRITE(7) ARRAY
      IK=IK+1
      READ(7) ARRAY
      READ(7) XHERE,MWCCL,MAWW
      WRITE(7) ARRAY
      REWIND 1
      REWIND 2
      REWIND 3
      REWIND 4
      IF(IG-1) 100,100,90
   90 IF(IG-3) 130,100,130
  100 IK=1
      DO 110 J=1,4
      READ(7) ARRAY
      KK=IK*5
      IK=IK+1
  110 WRITE(KK) ARRAY
      DO 120 J=1,6
      AHERE(1,J)=WHERE(1,J)
      AHERE(2,J)=WHERE(2,J)
      BHERE(1,J)=ZHERE(1,J)
      BHERE(2,J)=ZHERE(2,J)
      CHERE(1,J)=YHERE(1,J)
      CHERE(2,J)=YHERE(2,J)
      DHERE(1,J)=XHERE(1,J)
      DHERE(2,J)=XHERE(2,J)
      MACCL(J)=MXCCL(J)
      MBCCL(J)=MZCCL(J)
      MCCCL(J)=MYCCL(J)
      MDCCL(J)=MWCCL(J)
      MAAA(J)=MAXX(J)
      MABB(J)=MAZZ(J)
      MACC(J)=MAYY(J)
  120 MADD(J)=MAWW(J)
      GO TO 140
  130 IK=5
      READ(7) ARRAY
      READ(7) AHERE,MACCL,MAAA
      WRITE(7) ARRAY
      IK=IK+5
      READ(7) ARRAY
      READ(7) BHERE,MBCCL,MABB
      WRITE(7) ARRAY
      IK=IK+5
      READ(7) ARRAY
      READ(7) CHERE,MCCCL,MACC
      WRITE(7) ARRAY
      IK=IK+5
      READ(7) ARRAY
      READ(7) DHERE,MDCCL,MADD
      WRITE(7) ARRAY
140   ALPO(1)=0.0
      REWIND 5
      REWIND 10
      REWIND 15
      REWIND 20
      REWIND 1
      REWIND 2
      REWIND 3
```

```
      REWIND 4
      NP=R-1.
      JP=R/2.
      PIER=3.14159/R
      CALL DATSW(C,I)
      GO TO (150,160), I
  150 READ(IR,670) (TAU(I),I=1,NP)
      READ(IR,670) (REY(I),I=1,NP)
      READ(IR,670) (C(I),I=1,NP)
      READ(IR,670) (EPS(I),I=1,NP)
      READ(IR,670) EDGE
      READ(IR,670) CRB
      READ(IR,670) ACC
  160 CONTINUE
C
C                                   IF NO FUSE (FUSELAGE)
C
      IF(A) 410,410,170
  170 YO=B*SQRT(1.-H**2/A**2)
      ECC=SQRT(A**2-B**2)
      CALL DATSW(C,I)
      GO TO (190,180), I
  180 CRB=2.*(1.-YO)/(ASPEC*(1.+TAPER-2.*YO*TAPER))
  190 TFAC=1.-(YO*TAUR*CRB/(3.14159*A*B))*4.
      JPP=JP+1
C
C     YA(I)    (SEE CR1646 FOR EXPLANATIONS)
C
      DO 200 I=1,JPP
      XII=I-1
  200 YDA(I)=YO+(1.-YO)*COS(XII*PIER)
C
C                                   CHECK IF FUSE ELLIPTICAL OR CIRCULAR
C
      IF(A-B) 230,230,210
C
C                                   ELLIPTICAL FUSE
C
  210 CONTINUE
      DISTX=0.5*(SQRT(1.+(H-ECC)**2)+SQRT(1.+(H+ECC)**2))
      BWX=1./(A-B)*(A-B*DISTX/SQRT(DISTX**2-ECC**2))
      DO 220 I=1,JPP
      DIST(I)=0.5*(SQRT(YDA(I)**2+(H-ECC)**2)+SQRT(YDA(I)**2+(H+ECC)**2
     1))
C
C                                   Y BAR PRIME (I)
C
  220 YDAX(I)=(YDA(I)/(A-B)*(A-B*DIST(I)/SQRT(DIST(I)**2-ECC**2)))/BWX
      GO TO 250
C
C                                   Y BAR PRIME (I) FOR CIRCULAR FUSE
C
  230 BWX=1.-A**2/(1.+H**2)
      DO 240 I=1,JPP
  240 YDAX(I)=YDA(I)*(1.-A**2/(YDA(I)**2+H**2))/BWX
C
C                                   COMMON TO ELLIPTIC AND CIRCULAR FUSE
C
  250 DO 260 I=1,JPP
C
C                                   Y BAR (I)
C
      AI=I-1
  260 YX(I)=COS(AI*PIER)
C
C                                   Y (I)
C
      DO 270 I=2,JPP
  270 Y(I)=TERP(YDAX(I-1),YX(I),YDAX(I),YDA(I-1),YDA(I))
      DO 280 I=1,JP
      Y(I)=Y(I+1)
  280 YX(I)=YX(I+1)
      M=JP-1
      DO 290 I=1,M
      IRI=R
      IRI=IRI-I
      YX(IRI)=-YX(I)
  290 Y(IRI)=-Y(I)
      IF(A-B) 360,360,300
C
```

```
C                                   FUSE IS ELLIPTICAL
C
  300 DO 310 I=1,JP
      DIST(I)=0.5*(SQRT(Y(I)**2+(H-ECC)**2)+SQRT(Y(I)**2+(H+ECC)**2))
      Q=DIST(I)/SQRT(DIST(I)**2-ECC**2)
      S=1.+(ECC*Y(I)/(DIST(I)**2-ECC**2))**2
      TRANS(I)=1./(A-B)*(A-B*Q/S)
      IRI=R
      IRI=IRI-I
  310 TRANS(IRI)=TRANS(I)
C
C                                   WING ON TOP OR BOTTOM OF FUSELAGE
C
      IF(A-H) 320,320,330
  320 TRANS(JP)=1.
  330 IF(NFLAP) 520,520,340
  340 IF(BF-1.) 350,520,520
  350 DISF=0.5*(SQRT(BF**2+(H-ECC)**2)+SQRT(BF**2+(H+ECC)**2))
      BFX=(BF/(A-B)*(A-B*DISF/SQRT(DISF**2-ECC**2)))/BWX
      GO TO 460
  360 DO 370 I=1,JP
      TRANS(I)=1.+A**2*(Y(I)**2-H**2)/((Y(I)**2+H**2)**2)
      IRI=R
      IRI=IRI-I
  370 TRANS(IRI)=TRANS(I)
      IF(A-H) 380,380,391
391   IF(NFLAP) 520,520,390
  380 TRANS(JP)=1.0
  390 IF(BF-1.) 400,520,520
  400 BFX=(BF*(1.-A**2/(BF**2+H**2)))/BWX
      GO TO 460
  410 DO 420 I=1,NP
      AI=I
      Y(I)=COS(AI*PIER)
      IF(AI.EQ.JP) Y(I)=0.0
      YY(I)=Y(I)
  420 TRANS(I)=1.
      YO=0.
C
C                                   SPECIAL CASE IF VALUES HAVE BEEN READ IN
C                                   DO NOT WANT TO COMPUTE CRB.
C                                   WILL NOT COMPUTE VALUES ALREADY READ IN
C
      CALL DATSW(C,I)
      GO TO (440,430), I
  430 CRB=2.*(1.-YO)/(ASPEC*(1.+TAPER-2.*YO*TAPER))
  440 TFAC=1.
      BWX=1.
      IF(BF-1.) 450,520,520
  450 BFX=BF
  460 TSTAX=ACOS(BFX/BWX)
      DO 510 I=1,JP
      AI=I
      C3=AI*PIER
      TSTX=TSTAX
      IF(C3-TSTX) 510,470,480
  470 TSTAX=C3
      ISTAR=I
      GO TO 520
  480 AM=C3-TSTAX
      C1=(AI-1.)*PIER
      AK=TSTAX-C1
      IF(AM-AK) 490,490,500
  490 ISTAR=I
      TSTAX=C3
      GO TO 520
  500 ISTAR=I-1
      TSTAX=C1
      GO TO 520
  510 CONTINUE
  520 CALL DATSW(C,I)
      GO TO (550,530), I
  530 EDGE=SQRT(1.+4./(ASPEC**2))
      EN=1.-0.5/SQRT(EDGE)
      PI=3.14159
      EDGE=0.5*(1./EN-PI*COS(PI*EN)/SIN(PI*EN))
      CALL DATSW(3,I)
      GO TO (541,542),I
 541  WRITE(IP,1011)
      CALL ZZZ(EDGE)
```

```
  542 DO 540 I=1,NP
      C(I)=1.-(1.-TAPER)*(ABS(Y(I))-YO)/(1.-YO)
      TAU(I)=TAUR/C(I)*(1.-(1.-TAPER*TAUT/TAUR)*(ABS(Y(I))-YO)/(1.-YO))
      CAMB(I)=CAMBR+(CAMBT-CAMBR)*(ABS(Y(I))-YO)/(1.-YO)
      ACC=0.6666*(1.+TAPER*(1.+TAPER))/(1.+TAPER)
  540 REY(I)=REYNO*C(I)/ACC
 550  CALL MAINA
      RETURN
1011  FORMAT(10X,5HEDGE=)
 650  FORMAT(8F10.0/5F10.0,I10,4F5.0/2F10.0,I10,1F10.0)
  660 FORMAT(5I1,25A2)
  670 FORMAT(16F5.0)
      END
      SUBROUTINE MAINA
C     ************** MAINA-CONTINUATION OF SUBROUTINE MAIN *************
      DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(1
     19),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19),
     2MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),TRIX
     3(19,19),CM(19),TONY(19),
     4ALPHZ(19),XHERE(2,6),MWCCL(6),MAWW(6)
      DIMENSION CVAL(19),ALPHL(19),CBC(19),CBG(19),DELTA(19),
     1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,BYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CLL,REYON,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYNO,DISCK,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,RWX,YHERE,MYCCL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAML(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STONY(19),SGENE(19),CVAL,
     8ALPHL,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LOCER
      IF(IR-8) 20,10,10
   10 IPRAB=1
      KSTAL=0
      ISTAL=0
      KOUNT=0
      READ(IR,320) ALPHV
      GO TO 30
   20 IPRAB=IPRAB+1
   30 ALPHB=ALPHV(IPRAB)
      IF(ALPHB.EQ.99.) GO TO 641
      IF(IPRAB.NE.1) GO TO 61
      DO 549 K=1,NP
      VSA(K)=1.
      VR(K)=1.0
      SW(K)=0.0
      DA(K)=0.0
      AS(K)=0.0
      TL(K)=0.0
 549  SV(K)=0.0
  61  IF(NSLIP.EQ.0) GO TO 552
      CALL SLIP
      IF(IPRAB.NE.1) GO TO 81
C     READ(IR,320) (DA(K),TL(K),K=1,NP)
      SUM=0.7854+YPB
      DIFF=0.7854-YPB
      SUM2=SUM**2
      DIFF2=DIFF**2
      RTSUM=SQRT(SUM2+XPB**2)
      RTDIF=SQRT(DIFF2+XPB**2)
      FUNC=DIFF/(XPB*RTDIF)+SUM/(XPB*RTSUM)-(RTSUM-XPB)/(SUM*RTSUM)
      IF(ABS(YPB-0.7854)-.01) 71,71,72
   71 FUNC1=0.0
      GO TO 73
   72 FUNC1=(RTDIF-XPB)/(DIFF*RTDIF)
   73 FUNC=FUNC-FUNC1
      CALL DATSW(3,JUNK)
      GO TO (80,81),JUNK
   80 WRITE(IP,5491)
5491  FORMAT(10X,4HFUNC)
      CALL ZZZ(FUNC)
   81 DO 553 J=1,NP
 553  REY(J)=REY(J)*SQRT(VSA(J)**2+SW(J)**2)
 552  IF(IPRAB.NE.1) GO TO 641
      LOCER=3
```

```
      ALPHA=999.
      CLL=0.
      REYCN=999.
      XMAX=0.
C
C                                LOOK UP ZERO LIFT ANGLE FOR TIP SECTION
C
      CALL BRIDG(ALPHZ,1,1,1,IY,IE,REY,TAU,CAMB,ALPHA,1,1,ALPHA)
      IF(IE) 560,560,640
C
C                                LOOK UP ZERO LIFT ANGLE FOR ROOT SECTION
C
560   ALPHA=999.
      CALL BRIDG(ALPHZ,JP,JP,1,IY,IE,REY,TAU,CAMB,ALPHA,1,1,ALPHA)
      IF(IE) 570,570,640
C
C                                CALCULATE AERODYNAMIC AND GEOMETRIC TWIST
C
  570 IF(TWISA-100.) 580,590,590
  580 TWIST=TWISA-(ALPHZ(JP)-ALPHZ(1))
      GO TO 600
  590 TWISA=TWIST+(ALPHZ(JP)-ALPHZ(1))
  600 CALL CATSW(C,I)
      GO TO (630,610), I
  610 DO 620 I=1,NP
  620 EPS(I)=TWIST*TAPER*(ABS(Y(I))-YO)/(C(I)*(1.-YO))
      IF(NFLAP.NE.0) GO TO 641
  630 LOCER=1
      CLL=999.
      ALPHA=999.
      REYN=999.
      REYCN=0.
C
C                                FOR XMAX=100. ARC WILL LOOK UP ALPHA MAX
C                                FOR XMAX=0. ARC WILL LOOK UP CL MAX
C
      XMAX=100.
      CALL BRIDG(ALMAX,1,NP,1,IY,IE,REY,TAU,CAMB,ALPHA,1,1,XMAX)
  640 WRITE(IP,680) IE,LOCER
      CALL EXIT
641   RETURN
C
  320 FORMAT(8F10.0)
  680 FORMAT(1X,11HERROR CODE ,I2,13H -IN SECTION ,I3,9H  ABORTED)
      END
      SUBROUTINE MAIN1
C
C                        MAIN1----CONTINUATION OF SUBROUTINE MAIN
C
      DIMENSION C(19),EPS(19),TRANS(19),REY(19),ETA(19),HCPP(19),CLMAX(
     119),ZHERE(2,6),WHERE(2,6),ARRAY(5,25,8 ),Y(19),TAU(19),BETA(19,19)
     2,MAZZ(6),MZCCL(6),MAXX(6),MXCCL(6),YHERE(2,6),MYCCL(6),MAYY(6),
     3TRIX(19,19),TONY(19),GENE(19),SIGMA(19),LL(19),MM(19),CM(
     419),XHERE(2,6),MWCCL(6),MAWW(6)
      DIMENSION CVAL(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(19),
     1ALPH(19),ALPHE(19),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BH,REYNO,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TONY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCCL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STONY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LOCER
C
C                               COMPUTATION OF BETA
C
      OMEGA=2.-1./((1.+4./(ASPEC**2))**0.25)
      DO 800 M=1,NP
      DO 800 K=1,NP
      AK=K
      AM=M
      IF(K-M) 20,10,20
   10 BETA(M,K)=180.*R/(8.*3.14159*SIN(AK*PIER))*OMEGA
```

```
      GO TO 50
   20 KK=IABS(K-M)-2*((IABS(K-M))/2)
      IF(KK-1) 40,30,40
   30 BETA(M,K)=180./(4.*3.14159*R*SIN(AK*PIER))*(1./(1.-COS((AK+AM)*
     13.14159/R))-1./(1.-COS((AK-AM)*3.14159/R)))*OMEGA
      GO TO 50
   40 BETA(M,K)=0.
   50 IF(M-K) 70,60,70
   60 TRIX(M,K)=1.+0.1*C(K)*CRB/BWX*BETA(M,K)*TRANS(K)/EDGE
      GO TO 80
   70 TRIX(M,K)=0.1*C(K)*CRB/BWX*BETA(M,K)*TRANS(K)/EDGE
   80 I=NP+1-M
      J=NP+1-K
      BETA(I,J)=BETA(M,K)
  800 TRIX(I,J)=TRIX(M,K)
      CALL DATSW(4,I)
      GO TO (90,100), I
   90 WRITE(IP,430)
      CALL SSS(BETA,NP)
  100 DO 110 I=1,NP
      AI=I
  110 ETA(I)=0.523598/R*(3.-(-1.)**I)*SIN(AI*PIER)
C
C                               CHECK IF FLAP CASE
C
      IF(NFLAP) 310,310,120
C
C                               IS THERE A PART-SPAN FLAP
C
  120 IF(BF-1.) 130,310,310
  130 DO 170 I=1,NP
      AI=I
      THE=AI*PIER
      IF(THE-TSTAX) 150,140,150
  140 TONY(I)=1./45.*TSTAX*SIN(THE)
      GO TO 160
  150 TONY(I)=1./90.*((COS(THE)-COS(TSTAX))*(ALOG(1.-COS(THE+TSTAX
     1))-ALOG (1.-COS(THE-TSTAX)))+2.*TSTAX*SIN(THE))
  160 GENE(I)=1./90.*((COS(THE)+COS(TSTAX))*(ALOG(1.+COS(THE-TSTAX
     1))-ALOG(1.+COS(THE+TSTAX)))+2.*(3.14159-TSTAX)*SIN(THE))
      STONY(I)=2.*SIN(THE)-90./3.14159*TONY(I)
      SGENE(I)=2.*SIN(THE)-90./3.14159*GENE(I)
      TONY(I)=TONY(I)/OMEGA
      GENE(I)=GENE(I)/OMEGA
  170 CONTINUE
C
C                               CHECK FOR DUMP
C
      CALL DATSW(3,I)
      GO TO (180,190), I
  180 WRITE(IP,440)
      CALL AAA(GENE,NP)
      WRITE(IP,450)
      CALL AAA(TONY,NP)
  190 DO 230 K=1,NP
      SUM=0.
      DO 200 M=1,NP
  200 SUM=TONY(M)*BETA(M,K)+SUM
      IF(K-ISTAR) 210,210,220
  210 HOPP(K)=1.-SUM
      GO TO 230
  220 HOPP(K)=-SUM
  230 CONTINUE
      DO 270 K=1,NP
      SUM=0.
      DO 240 M=1,NP
  240 SUM=GENE(M)*BETA(M,K)+SUM
      IRI=R
      IF(K-IRI+ISTAR) 250,260,260
  250 SIGMA(K)=1.-SUM
      GO TO 270
  260 SIGMA(K)=-SUM
  270 CONTINUE
      DO 280 K=1,NP
      HOPP(K)=SIGMA(K)-HOPP(K)
  280 TONY(K)=GENE(K)-TONY(K)
C
C                               CHECK FOR DUMP
C
      CALL DATSW(3,I)
```

```
      GC TC (29C,3CC), I
  290 WRITE(IP,46C)
      CALL AAA(HCPP,NP)
      WRITE(IP,47C)
      CALL AAA(SIGMA,NP)
      WRITE(IP,44C)
      CALL AAA(GENE,NP)
      WRITE(IP,45C)
      CALL AAA(TCNY,NP)
      WRITE(IP,59C)
      CALL AAA(STCNY,NP)
      WRITE(IP,6CC)
      CALL AAA(SGENE,NP)
  300 CCNTINUE
  310 IBTA=1
C
C                               STCRE BETA TEMPORARILY ON DISK SO WE CAN
C                               CCMPUTE THE TRANSPCSE OF TRIX
C
      REWINC 44
      WRITE(44) BETA
      REWINC 44
C
C                               STCRE TRIX IN BETA
C
      DC 320 M=1,NP
      DC 320 K=1,NP
  320 BETA(M,K)=TRIX(M,K)
C
C                               NCW TRANSPOSE BETA (CLD TRIX)
C
      DC 33C M=1,NP
      DC 33C K=1,NP
  330 TRIX(M,K)=BETA(K,M)
      IBTA=1
C
C                               RESTCRE BETA
C
      REAC(44) BETA
C
C                               INVERT TRIX
C
      CALL MINV(TRIX,NP,AK,LL,NM)
      IF(NFLAP-1) 36C,42C,42C
  360 WRITE(IP,49C) NAME
      WRITE(IP,5CC)
      CALL AAA(Y,NP)
      WRITE(IP,51C)
      CALL AAA(ALMAX,NP)
C
C                               SET UP FOR CL MAX LOOK UP
C
      IY=1
      REYCN=C.
      ALPHA=999.
      CLL=999.
      REYN=999.
C
C                               LCCK UP CL MAX VALUES
C
      XMAX=0.
      CALL BRIDG(CLMAX,1,NP,1,IY,IE,REY,TAU,CAMB,ALPHA,1,1,XMAX)
      WRITE(IP,52C)
      CALL AAA(CLMAX,NP)
      WRITE(IP,53C)
      CALL AAA(TAU,NP)
      WRITE(IP,54C)
      CALL AAA(REY,NP)
      WRITE(IP,55C)
      CALL AAA(C,NP)
      WRITE(IP,56C)
      CALL AAA(EPS,NP)
      WRITE(IP,57C)
      CALL AAA(CAMB,NP)
420   RETURN
C
  430 FCRMAT(10X,11HMATRIX BETA)
  440 FCRMAT(10X,4HGENE)
  450 FCRMAT(10X,4HTCNY)
  460 FCRMAT(10X,4HHCPP)
```

```
  470 FCRMAT(10X,5HSIGMA)
  490 FCRMAT(1H1/1H0/35X,25A2/1X)
500   FCRMAT(1X/10X,22HSPANWISE STATIONS-2Y/B)
  510 FCRMAT(10X,9HALPHA MAX)
  520 FCRMAT(10X,7H CL MAX)
  530 FCRMAT(10X,30HTHICKNESS / CHCRD DISTRIBUTION)
540   FCRMAT(10X,33HSECTICN REYNCLDS NUMBERS,MILLIONS)
  550 FCRMAT(10X,18HCHCRD DISTRIBUTICN)
  560 FCRMAT(10X,18HTWIST DISTRIBUTION)
  570 FCRMAT(10X,19HCAMBER DISTRIBUTION)
590   FCRMAT(10X,5HSTCNY)
600   FCRMAT(10X,5HSGENE)
      END
      SUBRCUTINE MAIN2
C
C                    MAIN2-----GENERAL PRINT SUBRCUTINE---------
C
      DIMENSICN ARRAY(5,25,8 ),C(19),EPS(19),TRANS(19),REY(19),ETA(19),
     1HCPP(19),CLMAX(19),ZHERE(2,6),WHERE(2,6),Y(19),TAU(19),BETA(19,19)
     2,TRIX(19,19),MAZZ(6),MZCCL(6),MAXX(6),MXCLL(6),CBG(19),CVAL(19),
     3ALPG(19),CBC(19),ALPHL(19),ALPH(19),ALPHZ(19),ALPHE(19),DELTA(19),
     4YHERE(2,6),MYCCL(6),MAYY(6),CM(19),XHERE(2,6),MAWW(6),MWCOL(6)
      DIMENSICN CLACC(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      CCMMCN KOR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      CCMMCN NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      CCMMCN INNCW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IG,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TCNY(19),TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STCNY(19),SGENE(19),CVAL,
     8ALPHL,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLACC,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LCCER
      ALPG(1)=ALPG(1)
      WRITE(IP,330) NAME
      WRITE(IP,340) ALPHB,CTS,A,B,ASPEC,H,ALPHR,TAUT
      WRITE(IP,350) TAUR,TWIST,R,TWISA,BF,TAPER,FLAP,REYND,X,Z
      WRITE(IP,360)
      RETURN
  330 FCRMAT(1H1/1H0/35X,25A2/1X)
  340 FCRMAT(1X,40(3H./.)/1H0/15X,30HBCDY ANGLE OF ATTACK, DEG. . =,F10.
     12,10X,30HPRCPELLER THRUST COEFFICIENT =,F11.3/15X,30HBODY HEIGHT /
     2 SPAN . . . . . =,F10.2,10X,30HBODY WIDTH / SPAN. . . . . =,F10.
     32/15X,30HASPECT RATIC . . . . . . . . =,F10.2,10X,30HWING HEIGHT /
     4 SPAN . . . . . =,F10.2/15X,30HWING BODY INCIDENCE, DEG . . =,F10.
     52,10X,30HTIP THICKNESS CHCRD. . . . . =,F10.2)
  350 FCRMAT(15X,30HRCCT THICKNESS CHORD . . . . =,F10.2,10X,30HGECMETRI
     1C TWIST, DEG . . . . =,F10.2/15X,30HNUMBER CF SPANWISE STATIONS. =
     2,F10.2,10X,30HAERCDYNAMIC TWIST, DEG . . . =,F10.2/15X,30HFLAP SPA
     3N / WING SPAN. . . . =,F10.2,10X,30HTAPER RATIO. . . . . . . . . =
     4,F10.2/15X,30HFLAP SETTING, DEG. . . . . . =,F10.2,10X,30HREYNCLDS
     5 NUMBER,MILLICNS . . =,F10.2/15X,30HX-COCRDINATE CF MOMENT REF . =
     6,F10.2,10X,30HZ-CCCRDINATE OF MOMENT REF . =,F10.2)
  360 FCRMAT(1H0/1X,40(3H./.)/1X)
      END
      SUBRCUTINE MAN2A
C
C  *****************MAN2A--CCNTINUATION CF MAIN2 *********************
C
      DIMENSICN TCNY(19)
      DIMENSICN ARRAY(5,25,8 ),C(19),EPS(19),TRANS(19),REY(19),ETA(19),
     1HCPP(19),CLMAX(19),ZHERE(2,6),WHERE(2,6),Y(19),TAU(19),BETA(19,19)
     2,TRIX(19,19),MAZZ(6),MZCCL(6),MAXX(6),MXCLL(6),CBG(19),CVAL(19),
     3ALPG(19),CBC(19),ALPHL(19),ALPH(19),ALPHZ(19),ALPHE(19),DELTA(19),
     4YHERE(2,6),MYCCL(6),MAYY(6),CM(19),XHERE(2,6),MAWW(6),MWCOL(6)
      DIMENSICN CLACC(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      CCMMCN KCR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
      CCMMCN NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      CCMMCN INNCW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,BF,REYND,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,IS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TCNY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
```

```
      6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
      7MCCCL(6),CHERE(2,6),MACD(6),MDCCL(6),STCNY(19),SGENE(19),CVAL,
      8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
      9F,IRI,FF,LCCER
       ITR=C
       IY=1
       DC 7C K=1,NP
70     ALPG(K)=ALPHB+ALPHR+EPS(K)+ALPHB*TFAC*(TRANS(K)-1.)
C
C                                    SWITCH NUMBER 3 IS USED FCR AN INTERNAL
C                                    DUMP CF ARRAYS COMPUTED DURING  ITERATION
C                                    PRCCESS
C
C
       CALL DATSW(3,JUNK)
       GC TC (80,9C), JUNK
   80  WRITE(IP,37C)
       CALL AAA(ALPG,NP)
90     LCCER=3
       ALPHA=999.
       CLL=C.C
       REYCN=999.
       XMAX=C.C
       CALL BRIDG(ALPHZ,1,NP,1,IY,IE,REY,TAU,CAMB,ALPHA,1,1,ALPHA)
       CALL DATSW(3,JUNK)
       GC TC (66,67),JUNK
66     WRITE(IP,68)
       CALL AAA(ALPHZ,NP)
67     IF(IE) 101,101,310
101    IF(NSLIP-1) 135,1C2,1C2
135    DC 136 K=1,NP
136    TCNY(K)=0.0
       GC TC 130
102    DC 103 K=1,NP
       ALPG(K)=ALPG(K)+TL(K)
103    ALPHZ(K)=ALPHZ(K)+DA(K)
       IF(ITR) 104,1C4,1C5
104    DUM=C.1*(ALPHB+ALPHR-C.4*ALPHZ(1)-C.6*ALPHZ(JP))/(1.+1.82/ASPEC)
       GC TC 109
105    DUM=C.0
       DC 106 K=1,NP
106    DUM=CBC(K)*ETA(K)+DUM
       DUM=DUM*ASPEC*BWX**2
109    VW=DUM*FUNC/(9.87*ASPEC)
       ALFPR=(ALPHB+EYETL)/57.293
       ASBAR=ATAN((SIN(ALFPR)+VW)/VSBAR)-ALFPR
       CALL DATSW(3,JUNK)
       GC TC (50,51),JUNK
   50  WRITE(IP,1009)
1009   FCRMAT(10X,2HVW)
       CALL ZZZ(VW)
       WRITE(IP,1C1C)
1010   FCRMAT(10X,5HALFPR)
       CALL ZZZ(ALFPR)
       WRITE(IP,1C11)
1011   FCRMAT(10X,5HASBAR)
       CALL ZZZ(ASBAR)
51     DC 1031 K=1,NP
1031   HCPP(K)=ALPG(K)-ALPHZ(K)
       DC 1C7 K=KCR,KIR
       AS(K)=ASBAR
       VR(K)=VSA(K)/CCS(ALFPR+ASBAR)
       ANG=AS(K)+HCPP(K)/57.293
       SYN=SIN(ANG)
       CCZ=CCS(ANG)
107    SV(K)=VR(K)*SYN+VR(K)*SW(K)*CCZ-SIN(HCPP(K)/57.293)
       DC 1C8 K=KIL,KCL
       AS(K)=ASBAR
       VR(K)=VSA(K)/CCS(ALFPR+ASBAR)
       ANG=AS(K)+HCPP(K)/57.293
       SYN=SIN(ANG)
       CCZ=CCS(ANG)
108    SV(K)=VR(K)*SYN+VR(K)*SW(K)*CCZ-SIN(HCPP(K)/57.293)
       CALL DATSW(3,JUNK)
       GC TC (52,53),JUNK
   52  WRITE(IP,1071)
1071   FCRMAT(10X,2HSV)
       CALL AAA(SV,NP)
       WRITE(IP,1072)
1072   FCRMAT(10X,2HVR)
       CALL AAA(VR,NP)
```

```
   53 DO 126 K=1,NP
      AK=K
      SUM1=0.0
      DO 121 N=1,NP
      AN=N
      SUM2=0.0
      DO 122 M=1,NP
      AM=M
  122 SUM2=SUM2+SV(M)*SIN(AM*PIER)*SIN(AN*AM*PIER)
  121 SUM1=SUM1+SUM2*SIN(AN*AK*PIER)/AN
  126 TCNY(K)=VR(K)*SUM1*4./R
      CALL DATSW(3,JUNK)
      GO TO (54,130),JUNK
   54 WRITE(IP,1301)
 1301 FORMAT(10X,5HCL2CB)
      CALL AAA(TCNY,NP)
  130 IF(ITR) 131,131,140
  131 LOOER=2
      DO 1311 K=1,NP
 1311 HOPP(K)=(ALPG(K)-ALPHZ(K)*(1.-EDGE))/EDGE
      CLL=999.
      REYCN=999.
      XMAX=0.0
      CALL BRIDG(CVAL,1,NP,1,IY,IE,REY,TAU,CAMB,HOPP,1,NP,CLL)
      IF(IE) 100,100,310
  100 DO 110 K=1,NP
      AK=K
  110 CBG(K)=CVAL(K)*(ASPEC/(ASPEC+1.8))*C(K)*CRB*(0.5+
     1(1.+TAPER)*SIN(AK*PIER)/(3.14159*C(K)))
      CALL DATSW(3,JUNK)
      GO TO (120,140),JUNK
  120 WRITE(IP,380)
      CALL AAA(CVAL,NP)
  140 DO 160 K=1,NP
      SIG=0.0
      DO 150 M=1,NP
  150 SIG=(CBC(M)-TCNY(M))*BETA(M,K)+SIG
      ALPHU(K)=SIG*(1.+TFAC*(TRANS(K)-1.))
      ALPH(K)=ALPG(K)-ALPHU(K)
  160 ALPHE(K)=(ALPH(K)-ALPHZ(K)*(1.-EDGE))/EDGE
C
C                                 LOOK UP CL VALUES FOR A LIST OF ALPHA
C                                 VALUES
C
      LOOER=4
      CLL=999.
      REYCN=999.
      XMAX=0.
      CALL BRIDG(CVAL,1,NP,1,IY,IE,REY,TAU,CAMB,ALPHE,1,NP,CLL)
      IF(IE) 170,170,310
  170 DO 180 K=1,NP
      CBC(K)=CVAL(K)*C(K)*CRB/BWX
  180 DELTA(K)=CBC(K)-CBG(K)
C
C                                 CHECK FOR DUMP
C
      CALL DATSW(3,JUNK)
      GO TO (190,200), JUNK
  190 WRITE(IP,390)
      CALL AAA(CBG,NP)
      WRITE(IP,400)
      CALL AAA(ALPHU,NP)
      WRITE(IP,410)
      CALL AAA(ALPHE,NP)
      WRITE(IP,380)
      CALL AAA(CVAL,NP)
      WRITE(IP,420)
      CALL AAA(CBC,NP)
      WRITE(IP,430)
      CALL AAA(DELTA,NP)
C
C                                 CHECK TOLERANCE
C
  200 DO 220 K=1,NP
      IF(ABS(DELTA(K))-DISCR) 210,210,230
  210 IF(K-NP) 220,300,300
  220 CONTINUE
  230 DO 250 I=1,NP
      SUM=0.
      DO 240 J=1,NP
```

```
  240 SUM=TRIX(I,J)*DELTA(J)+SUM
  250 CBG(I)=CBG(I)+SUM
C
C                                    CHECK FOR DUMP
C
      CALL DATSW(3,JUNK)
      GO TO (260,270), JUNK
  260 WRITE(IP,390)
      CALL AAA(CBG,NP)
C
C                                    REPEAT CYCLE
C
  270 ITR=ITR+1
      IF(ITR-30) 141,141,280
141   IF(NSLIP-1) 140,105,105
C
C                                    IF UNABLE TO CONVERGE AFTER 30 ITERATIONS
C                                    DUMP DELTA VALUES, C VALUES, AND TABLE
C                                    PRESENTLY IN CORE BEING USED FOR LOOK UP
C
  280 WRITE(IP,440)
      WRITE(IP,430)
      CALL AAA(DELTA,NP)
      WRITE(IP,380)
      CALL AAA(CVAL,NP)
      CALL EXIT
  300 WRITE(IP,470) ITR,ALPHB
      CALL MAIN4
  310 WRITE(IP,480) IE,LOCER
      CALL EXIT
C
68    FORMAT(10X,16HZERO-LIFT ANGLES)
  370 FORMAT(10X,4HALPG)
  380 FORMAT(10X,4HCVAL)
  390 FORMAT(10X,3HCBG)
  400 FORMAT(10X,5HALPHU)
  410 FORMAT(10X,5HALPHE)
  420 FORMAT(10X,3HCBC)
  430 FORMAT(10X,5HDELTA)
  440 FORMAT(1X,48HUNABLE TO CONVERGE AFTER 30 ITERATIONS   ABORTED)
  470 FORMAT(10X,I2,63H  ITERATIONS REQUIRED TO CONVERGE FOR ANGLE OF AT
     1TACK EQUAL TO ,F8.2/1X)
  480 FORMAT(1X,10HERROR CODE,I2,1X,10HAT SECTION,I3,1X,32HIN PROGRAM, E
     1XECUTION TERMINATED)
      END
      SUBROUTINE MAIN4
C
C                              MAIN4-----CONTINUATION OF SUBROUTINE MAIN2
C
      DIMENSION ARRAY(5,25,8 ),C(19),EPS(19),TRANS(19),REY(19),ETA(19),
     1HCPP(19),CLMAX(19),ZHERE(2,6),WHERE(2,6),Y(19),TAU(19),BETA(19,19)
     2,TRIX(19,19),MAZZ(6),MZCOL(6),MAXX(6),MXCOL(6),ALPG(19),ALPHE(19),
     3CL(19),CDC(19),CDCC(19),ALPHU(19),CBC(19),CBG(19),DELTA(19),ALPHZ(
     419),ALPH(19),CVAL(19),YHERE(2,6),MYCOL(6),MAYY(6),CM(19),
     5XHERE(2,6),MAWW(6),MWCOL(6)
      DIMENSION CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19)
      COMMON KOR,KIR,KIL,KOL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),DA(19),AS(19),TL(19),SV(19),FUNC,YO,CONAC,
     1ALPHV(16)
      COMMON INNCW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCOL,MZCOL,ASPEC,TAPER,BF,REYNO,DISCR,PIER,CRB,Q,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IK,IP,IG,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCOL,MAYY,FLAP,TONY(19),TWISA,X,Z,CM,ACC,XHERE,
     5MWCOL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACOL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCOL(6),CHERE(2,6),MADD(6),MDCOL(6),STONY(19),SCENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LOCER
      DO 10 K=1,NP
   10 CL(K)=CVAL(K)
      IZ=1
      LOCER=5
C
C                                    COMPUTE PROFILE DRAG COEFFICIENTS
C
      CLL=999.
      REYCN=999.
```

```
      XMAX=0.
      CALL BRIDG(CDC,1,NP,2,IZ,IE,REY,TAU,CAMB,CL,1,NP,CLL)
      DO 20 K=1,NP
   20 CL(K)=CVAL(K)
C
C                                 COMPUTE QUARTER CHORD PITCHING MOMENT
C                                 COEFFICIENTS
C
      CLL=999.
      REYCN=999.
      XMAX=0.
      CALL BRIDG(CM,1,NP,3,IW,IE,REY,TAU,CAMB,CL,1,NP,CLL)
      DO 50 K=1,NP
C
C                                 CHECK IF SECTION STALLED
C
      IF(ALPHE(K)-ALMAX(K)) 40,30,30
   30 WRITE(IP,260) K,ALPHE(K),ALMAX(K)
      KSTAL=1
C
C                                 COMPUTE SECTION PITCHING MOMENT
C
   40 SUM1=COS(3.14159/180.*(ALPHB-ALPHU(K)))
      SUM2=SIN(3.14159/180.*(ALPHB-ALPHU(K)))
      CM(K)=CM(K)-X/C(K)*(CL(K)*SUM1+CDO(K)*SUM2)-Z/C(K)*(CL(K)*SUM2-CDO
     1(K)*SUM1)
   50 CDCC(K)=CDC(K)*C(K)*CRB/BWX
      WRITE(IP,390)
      CALL AAA(Y,NP)
      WRITE(IP,270)
      CALL AAA(CM,NP)
C
C                                 CHECK FOR DUMP
C
      CALL DATSW(3,JUNK)
      GO TO (60,70), JUNK
   60 WRITE(IP,280)
      CALL AAA(CLMAX,NP)
      WRITE(IP,290)
      CALL AAA(CDCC,NP)
   70 WRITE(IP,300)
      CALL AAA(ALPHE,NP)
      WRITE(IP,310)
      CALL AAA(CDC,NP)
      DO 80 K=1,NP
   80 ALPG(K)=ALPHU(K)*CL(K)*3.14159/180.
      WRITE(IP,320)
      CALL AAA(ALPG,NP)
      WRITE(IP,330)
      CALL AAA(CL,NP)
      IF(NSLIP.EQ.0) CTS=0.0
      DO 801 K=1,NP
  801 CDC(K)=CL(K)*(1.-CTS)
      WRITE(IP,220)
      CALL AAA(CDC,NP)
C
C                                 COMPUTE OVERALL LIFT, DRAG, AND PITCHING
C                                 MOMENT COEFFICIENTS
C
      SUM1=0.
      SUM2=0.
      SUM3=0.
      SUM4=0.
      DO 90 I=1,NP
      SUM1=CBC(I)*ETA(I)+SUM1
      SUM2=CBC(I)*ALPHU(I)*ETA(I)+SUM2
      SUM3=CDCC(I)*ETA(I)+SUM3
   90 SUM4=CM(I)*C(I)**2*ETA(I)+SUM4
      Q=ASPEC*BWX**2
      CLIFT=Q*SUM1
      CLPP=CLIFT*(1.-CTS)
      CDI=0.017453*Q*SUM2
      CDP=Q*SUM3
      CD=CDI+CDP+CDNAC
      CDPP=CD*(1.-CTS)
      ZM=ASPEC*BWX*CRB/ACC*SUM4
      CMPP=ZM*(1.-CTS)
      WRITE (IP,340)
      WRITE (IP,341) ALPHB,CDI,CLIFT,CDP,CLPP,CDNAC,ZM,CD,CMPP,CDPP
      WRITE (IP,340)
```

```
      IF((KSTAL.EQ.0).AND.(KCUNT.EQ.0)) GO TO 250
C
C                                     DEFINE EXACT STALL ANGLE OF ATTACK
C
      IF((IPRAB.EQ.1).OR.(KCUNT.EQ.3)) GO TO 9900
      KCUNT=KCUNT+1
      GO TO (1000,2000,3000),KCUNT
 1000 AB(1)=(ALPHB+ALPHV(IPRAB-1))/2.
      ALPHB=AB(1)
      GO TO 9900
 2000 IF(KSTAL.EQ.0) GO TO 2100
      ISTAL=1
      AB(2)=(AB(1)+ALPHV(IPRAB-1))/2.
      ALPHB=AB(2)
      GO TO 9900
 2100 ISTAL=0
      AB(2)=(AB(1)+ALPHV(IPRAB))/2.
      ALPHB=AB(2)
      GO TO 9900
 3000 IF(KSTAL+ISTAL-1) 3100,3200,3300
 3100 AB(3)=(AB(2)+ALPHV(IPRAB))/2.
      ALPHB=AB(3)
      GO TO 9900
 3200 AB(3)=(AB(2)+AB(1))/2.
      ALPHB=AB(3)
      GO TO 9900
 3300 AB(3)=(AB(2)+ALPHV(IPRAB-1))/2.
      ALPHB=AB(3)
 9900 KSTAL=0
      DO 170 K=1,NP
  170 CDC(K)=CLMAX(K)-CL(K)
      WRITE(IP,350)
      CALL AAA(CDC,NP)
      IF((IPRAB.EQ.1).OR.(KCUNT.EQ.3)) GO TO 171
      WRITE(IP,370) NAME
      WRITE(IP,360) ALPHB
      IR=100
      CALL MAN2A
  250 IR=0
      RETURN
171   IR=101
      RETURN
C
260   FORMAT(10X,19HSTALLED AT STATION ,I2,18H ANGLE OF ATTACK =,F7.3,
     110X,21HSECTION STALL ANGLE =,F7.3)
  270 FORMAT(10X,35HSECTION PITCHING MOMENT COEFFICIENT)
  280 FORMAT(10X,5HCLMAX)
  290 FORMAT(10X,4HCDCC)
300   FORMAT(10X,33HEFFECTIVE SECTION ANGLE OF ATTACK)
  310 FORMAT(10X,32HSECTION PROFILE DRAG COEFFICIENT)
  320 FORMAT(10X,32HSECTION INDUCED DRAG COEFFICIENT)
  330 FORMAT(10X,43HDISTRIBUTION OF SECTION LIFT COEFFICIENT-CL)
  340 FORMAT(1X,4(3H./.)/1H0/)
341   FORMAT(10X,36HFUSELAGE ANGLE OF ATTACK,DEGREES. .=,F9.5,
     110X,32HINDUCED DRAG COEFFICIENT,CDI   = ,F9.5/10X,
     236HLIFT COEFFICIENT,CL . . . . . . . .=,F9.5,
     310X,32HPROFILE DRAG COEFFICIENT,CD    = ,F9.5/10X,
     436HLIFT COEFFICIENT,CLS. . . . . . . .=,F9.5,
     510X,32HNACELLE DRAG COEFFICIENT,CDN   = ,F9.5/10X,
     636HPITCHING MOMENT COEFFICIENT,CM. . .=,F9.5,
     710X,32HTOTAL   DRAG COEFFICIENT,CD      ,F9.5/10X,
     836HPITCHING MOMENT COEFFICIENT,CMS . .=,F9.5,
     910X,32HTOTAL   DRAG COEFFICIENT,CDS   = ,F9.5/1H0/)
350   FORMAT(10X,34HSTALL MARGIN DISTRIBUTION,CLMAX-CL,/1X)
  360 FORMAT(1X,4(3H./.)/1X/10X,28HADJUSTED ANGLE OF ATTACK TO,F8.3,32H
     1,SEARCHING FOR EXACT STALL POINT,/1X/1X,4(3H./.)/1X)
  370 FORMAT(1H1/1H0/35X,25A2/1X)
380   FORMAT(10X,44HDISTRIBUTION OF SECTION LIFT COEFFICIENT-CLS)
390   FORMAT(10X,22HSPANWISE STATIONS-2Y/B)
      END
      SUBROUTINE MAIN3
C
C-----MAIN3--SUBROUTINE FOR THE CASE WITH PART-SPAN FLAPS--------------
C
      DIMENSION ARRAY(5,25,8 ),C(19),EPS(19),TRANS(19),REY(19),ETA(19),
     1HCPP(19),CLMAX(19),ZHERE(2,6),WHERE(2,6),Y(19),TAU(19),BETA(19,19)
     2,TRIX(19,19),MAZZ(6),MZCOL(6),MAXX(6),MXCOL(6),XHERE(2,6),MWCOL(6)
     3,MAWW(6),CM(19),CBG(19),CVAL(19),ALPG(19),CBC(19),ALPHU(19),ALPH(1
     49),ALPHZ(19),ALPHE(19),DELTA(19),CL(19),YHERE(2,6),MYCOL
     5(6),MAYY(6),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19),ALPC(19)
```

```
     6,ICNY(19)
      COMMON KOR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KOUNT,AB(3)
      COMMON NSLIP,VR(19),CA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
      COMMON INNOW,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCCL,MZCCL,ASPEC,TAPER,DF,REYNO,DISCR,PIER,CRB,Q,TSTAX,EDGE
     3,SIG,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,ISIS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCCL,MAYY,FLAP,TCNY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCCL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCOL(6),STONY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LOCER
      IF(IPRAB.NE.1)GO TO 690
      ALPB=0.
      CL(1)=0.
      IRI=R
      ITR=0
      IY=1
C
C                        PUT FLAPS-UP CL DATA INTO CORE
C
      CALL DAGET(ARRAY,1,IY)
      LOCER=2
      DO 60 K=1,NP
      ALPG(K)=3.
C
C                               LOOK UP CL VALUES
C
      CLL=999.
      ALPHA=ALPG(K)
      REYN=REY(K)
      TAUX=TAU(K)
      REYCN=999.
      XMAX=0.
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
C
C                               CHECK FOR ERROR STOP
C
      IF(IE) 50,50,40
   40 WRITE(IP,720) IE,LOCER
      CALL EXIT
   50 CVAL(K)=CLL
      AK=K
   60 CBG(K)=CVAL(K)*(ASPEC/(ASPEC+1.8))*C(K)*CRB*(0.5+(1.+TAPER)*SIN(AK
     1*PIER)/(3.14159*C(K)))
      CALL DATSW(3,JUNK)
      GO TO (70,80), JUNK
   70 WRITE(IP,730)
      CALL AAA(CVAL,NP)
   80 DO 120 K=1,NP
      SIG=0.
      DO 90 M=1,NP
   90 SIG=CBG(M)*BETA(M,K)+SIG
      ALPHU(K)=SIG*(1.+TFAC*(TRANS(K)-1.))
      ALPH(K)=ALPG(K)-ALPHU(K)
C
C                               LOOK UP ALPHA FOR ZERO LIFT
C
      LOCER=3
      ALPHA=999.
      CLL=0.0
      REYN=REY(K)
      TAUX=TAU(K)
      REYCN=999.
      XMAX=0.
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      IF(IE) 100,100,95
   95 WRITE(IP,720) IE,LOCER
      CALL EXIT
  100 ALPHZ(K)=ALPHA
      ALPHE(K)=(ALPH(K)-ALPHZ(K)*(1.-EDGE))/EDGE
C
C                               LOOK UP CL VALUES
C
      LOCER=4
      CLL=999.
      ALPHA=ALPHE(K)
```

```
      REYN=REY(K)
      TAUX=TAU(K)
      REYCN=999.
      XMAX=0.
      CALL ARC(ARRAY,TAUX,MAXX,MXCOL,IE,WHERE,NLVL)
      IF(IE) 110,110,95
  110 CVAL(K)=CLL
      CBC(K)=CVAL(K)*C(K)*CRB/BWX
  120 DELTA(K)=CBC(K)-CBG(K)
      CALL DATSW(3,JUNK)
      GO TO (130,140), JUNK
  130 WRITE(IP,740)
      CALL AAA(CBG,NP)
      WRITE(IP,750)
      CALL AAA(ALPHL,NP)
      WRITE(IP,760)
      CALL AAA(ALPHE,NP)
      WRITE(IP,730)
      CALL AAA(CVAL,NP)
      WRITE(IP,770)
      CALL AAA(CBC,NP)
      WRITE(IP,780)
      CALL AAA(DELTA,NP)
  140 DO 160 K=1,NP
      IF(ABS(DELTA(K))-DISCR) 150,150,170
  150 IF(K-NP) 160,240,240
  160 CONTINUE
  170 DO 190 I=1,NP
      SUM=0.
      DO 180 J=1,NP
  180 SUM=TRIX(I,J)*DELTA(J)+SUM
  190 CBG(I)=CBG(I)+SUM
      CALL DATSW(3,JUNK)
      GO TO (200,210), JUNK
  200 WRITE(IP,740)
      CALL AAA(CBG,NP)
  210 ITR=ITR+1
C
C                                ITERATION CONTROL
C
      IF(ITR-30) 80,80,220
C
C                                IF UNABLE TO CONVERGE, DUMP ALL
C
  220 WRITE(IP,790)
      WRITE(IP,780)
      CALL AAA(DELTA,NP)
      WRITE(IP,730)
      CALL AAA(CVAL,NP)
      DO 230 J=1,NLVL
      NR=MAXX(J)
      NC=MXCOL(J)
      TAUX=WHERE(2,J)
      WRITE(IP,800) J,J,NR,NC,TAUX
      DO 230 K=1,NR
  230 WRITE(IP,810) (ARRAY(J,K,L),L=1,NC)
      CALL EXIT
C
C                                CONVERGENCE
C
  240 WRITE(IP,820) NAME
      WRITE(IP,830) ITR,ALPB
      SUM=0.
      DO 250 I=1,NP
  250 SUM=CBC(I)*ETA(I)+SUM
      G=ASPEC*BWX**2
C
C         CALCULATE ADDITIONAL LIFT DISTRIBUTION,CL1
C
      CCLIF=G*SUM
      DO 260 K=1,NP
  260 CLADD(K)=CVAL(K)/CCLIF
      CLL=0.
      ALPHA=999.
      REYN=REY(ISTAR)
      TAUX=TAU(ISTAR)
C           CALCULATE ZERO-LIFT ANGLES OUTBOARD AND INBOARD OF
C           FLAP END
      REYCN=999.
      CALL ARC(ARRAY,TAUX,MAXX,MXCOL,IE,WHERE,NLVL)
```

```
      A1=ALPHA
      ALPHA=999.
C                  LOAD FLAP CL TABLES INTO CORE
      CALL DAGET(ARRAY,4,IW)
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      A2=ALPHA
      SDELT=A1-A2
      ALPG(JP)=0.
      ALPHA=ALPG(JP)
      CLL=999.
      REYN=REY(JP)
      TAUX=TAU(JP)
C
C         CALCULATE APPROXIMATE SPAN LOADING WITH FLAP
C
      REYCN=999.
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      CVAL(JP)=CLL
      DO 290 K=1,JP
      AK=K
      STAR=ISTAR
      IF(K-ISTAR) 280,280,270
  270 CBG(K)=0.5*CVAL(JP)*C(K)*CRB/BWX*(1.+SQRT(1.-(COS(AK*PIER)/COS(
     1STAR*PIER))**2))
      GO TO 290
  280 CBG(K)=0.5*CVAL(JP)*C(K)*CRB/BWX*(1.-SQRT(1.-((1.-COS(AK*PIER))/
     1(1.-COS(STAR*PIER)))**2))
  290 CONTINUE
      DO 300 K=1,JP
      IRI=R
      IRI=IRI-K
  300 CBG(IRI)=CBG(K)
      IRI=R
      DO 310 K=1,NP
  310 ALPG(K)=0.
      CALL DATSW(3,I)
      GO TO (320,330), I
  320 WRITE(IP,840)
      CALL AAA(CLADD,NP)
      WRITE(IP,850)
      CALL ZZZ(CLL)
      WRITE(IP,740)
      CALL AAA(CBG,NP)
C       CALCULATE LIFT DISTRIBUTION WITH FLAP-UNCORRECTED FOR FLAP
C        END EFFECTS ON SECTION DATA
  330 ITR=0
  340 DO 400 K=1,NP
      SIG=0.
      DO 350 M=1,NP
  350 SIG=CBG(M)*BETA(M,K)+SIG
      ALPHU(K)=SIG*(1.+TFAC*(TRANS(K)-1.))
      ALPH(K)=ALPG(K)-ALPHU(K)
      ALPC(K)=SDELT*HLPP(K)
      ALPH(K)=ALPH(K)-ALPC(K)
      CLL=0.
      ALPHA=999.
      REYN=REY(K)
      REYCN=999.
      TAUX=TAU(K)
      IF(K-ISTAR) 380,380,360
  360 IF(K-IRI+ISTAR) 370,380,380
  370 CALL DAGET(ARRAY,4,IW)
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      ALPHZ(K)=ALPHA
      ALPHA=(ALPH(K)-ALPHZ(K)*(1.-EDGE))/EDGE
      CLL=999.
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      CVAL(K)=CLL
      GO TO 390
  380 CALL DAGET(ARRAY,1,IY)
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      ALPHZ(K)=ALPHA
      ALPHA=(ALPH(K)-ALPHZ(K)*(1.-EDGE))/EDGE
      CLL=999.
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      CVAL(K)=CLL
  390 CBC(K)=CVAL(K)*C(K)*CRB/BWX
  400 DELTA(K)=CBC(K)-CBG(K)
      CALL DATSW(3,JUNK)
      GO TO (410,420), JUNK
```

```
410   WRITE(IP,75C)
      CALL AAA(ALFHL,NP)
      WRITE(IP,88C)
      CALL AAA(ALPC,NP)
      WRITE(IP,89C)
      CALL AAA(ALPH,NP)
      WRITE(IP,9CC)
      CALL AAA(ALPHZ,NP)
      WRITE(IP,73C)
      CALL AAA(CVAL,NP)
      WRITE(IP,92C)
      CALL AAA(CBC,NP)
      WRITE(IP,78C)
      CALL AAA(CELTA,NP)
  42C DC  44C K=1,NP
      IF(ABS(CELTA(K))-CISCR) 43C,43C,45C
  43C IF(K-NP) 44C,5CC,5CO
  44C CCNTINUE
  450 DC 47C I=1,NP
      SUM=C.
      DC 46C J=1,NP
  46C SUM=TRIX(I,J)*CELTA(J)+SUM
  47C CBG(I)=CBG(I)+SUM
      CALL CATSW(3,JUNK)
      GC TC (480,49C), JUNK
480   WRITE(IP,74C)
      CALL AAA(CBG,NP)
  490 ITR=ITR+1
      IF(ITR-3C) 34C,34C,22C
  5CC DC 51C K=1,NP
  51C CLCEL(K)=CVAL(K)
      F1=CLCEL(JP)/CLACC(JP)
      F2=CLCEL(1)/CLACC(1)
      DC 520 K=1,ISTAR
  52C CLAC2(K)=F2*CLACC(K)
      DC 53C K=ISTAR,JP
  53C CLAC1(K)=F1*CLACC(K)
      BDELT=CLAC1(ISTAR)-CLAC2(ISTAR)
      DC 540 K=1,ISTAR
  54C F(K)=(CLCEL(K)-CLAC2(K))/BDELT
      FF=(CLCEL(ISTAR)-CLAC1(ISTAR))/BDELT
      IST=ISTAR+1
      DC 55C K=IST,JP
  55C F(K)=(CLCEL(K)-CLAC1(K))/BCELT
      DC 560 K=1,JP
      IRI=R
      IRI=IRI-K
  560 F(IRI)=F(K)
      CALL CATSW(3,JUNK)
      GC TC (57C,58C), JUNK
570   WRITE(IP,84C)
      CALL AAA(CLACC,NP)
      WRITE(IP,96C)
      CALL AAA(CLCEL,NP)
      WRITE(IP,97C)
      CALL ZZZ (F1)
      WRITE(IP,98C)
      CALL ZZZ (F2)
      WRITE(IP,99C)
      CALL AAA(CLAC2,NP)
      WRITE(IP,1CCC)
      CALL AAA(CLAC1,NP)
      WRITE(IP,1C1C)
      CALL AAA(F,NP)
      WRITE(IP,1020)
      CALL ZZZ(FF)
C
C     LCCK UP CL MAX FOR SECTICN CN UNFLAPPED SIDE OF FLAP END
C
  580 CALL CAGET(ARRAY,1,IY)
      REYN=999.
      XMAX=C.
      CLL=999.
      ALPHA=999.
      REYCN=REY(ISTAR)
      TAUX=TAU(ISTAR)
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      CLMNF=XMAX
      XMAX=C.
C
```

```
C      LOOK UP CL MAX FOR SECTION ON   FLAPPED SIDE OF FLAP END
C
       CALL CAGET(ARRAY,4,IW)
       CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
       CLMF=XMAX
       DCLMA=CLMF-CLMNF
       CLL=999.
       ALPHA=999.
       REYN=999.
       XMAX=0.
       DO 630 K=1,NP
       IRI=R
       IF(K-ISTAR) 600,600,590
  590  IF(K-IRI+ISTAR) 610,600,600
  600  XMAX=0.
       REYCN=REY(K)
       TAUX=TAU(K)
C
C          LOOK UP CLMAX FOR UNFLAPPED WING SECTIONS
C
       CALL CAGET(ARRAY,1,IY)
       CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
       GO TO 620
  610  XMAX=0.
       REYCN=REY(K)
       TAUX=TAU(K)
C
C          LOOK UP CLMAX FOR   FLAPPED WING SECTIONS
C
       CALL CAGET(ARRAY,4,IW)
       CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
  620  F(K)=1.+F(K)*DCLMA/XMAX
  630  CLMAX(K)=F(K)*XMAX
       FF=1.+FF*DCLMA/CLMF
       DO 670 K=1,NP
       IF(K-ISTAR) 650,650,640
  640  IF(K-IRI+ISTAR) 660,650,650
C
C                    LOOK UP ALPHA AT CLMAX FOR UNFLAPPED WING SECTIONS
C
  650  CALL CAGET(ARRAY,1,IY)
       ALPHA=999.
       REYN=999.
       CLL=999.
       REYCN=REY(K)
       TAUX=TAU(K)
       XMAX=100.
       CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
       GO TO 670
C
C                    LOOK UP ALPHA AT CLMAX FOR   FLAPPED WING SECTIONS
C
  660  CALL CAGET(ARRAY,4,IW)
       XMAX=100.
       REYCN=REY(K)
       TAUX=TAU(K)
       REYN=999.
       ALPHA=999.
       CLL=999.
       CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
  670  ALMAX(K)=ALPHZ(K)+(XMAX-ALPHZ(K))*EDGE*F(K)
       WRITE(IP,820) NAME
       WRITE(IP,1030)
       CALL AAA(Y,NP)
       WRITE(IP,1040)
       CALL AAA(ALMAX,NP)
       WRITE(IP,1050)
       CALL AAA(CLMAX,NP)
       WRITE(IP,1060)
       CALL AAA(TAU,NP)
       WRITE(IP,1070)
       CALL AAA(REY,NP)
       WRITE(IP,1080)
       CALL AAA(C,NP)
       WRITE(IP,1090)
       CALL AAA(EPS,NP)
       CALL DATSW(3,JUNK)
       GO TO (680,690), JUNK
  680  WRITE(IP,1100)
       CALL ZZZ(DCLMA)
```

```
      WRITE(IP,1010)
      CALL AAA(F,NP)
      WRITE(IP,1020)
      CALL ZZZ(FF)
690   CALL MAIN5
C
  720 FORMAT(1X,10HERROR CODE,I2,1X,10HAT SECTION,I3,1X,32HIN PROGRAM, E
     1XECUTION TERMINATED)
  730 FORMAT(10X,4HCVAL)
  740 FORMAT(10X,3HCBG)
  750 FORMAT(10X,5HALPHU)
  760 FORMAT(10X,5HALPHE)
  770 FORMAT(10X,3HCBC)
  780 FORMAT(10X,5HDELTA)
  790 FORMAT(2X,48HUNABLE TO CONVERGE AFTER 30 ITERATIONS   ABORTED)
  800 FORMAT(1H0/1H0,7H ARRAY(,I1,11H 1, 1) TO (,I1,1H,,I2,1H,,I2,1H),F
     110.5/1H0)
  810 FORMAT(12F10.4)
  820 FORMAT(1H1/1H0/35X,25A2/1X)
  830 FORMAT(1X,33H ITERATIONS REQUIRED TO CONVERGE ,I2,19H FOR ALPHB EQ
     1UAL TO,F10.1/1H0)
  840 FORMAT(10X,5HCLADD)
  850 FORMAT(10X,2HCL)
  880 FORMAT(10X,4HALPC)
  890 FORMAT(10X,4HALPH)
  900 FORMAT(10X,5HALPHZ)
  920 FORMAT(10X,3HCBC)
  960 FORMAT(10X,5HCLDEL)
  970 FORMAT(10X,2HF1)
  980 FORMAT(10X,2HF2)
  990 FORMAT(10X,5HCLAD2)
 1000 FORMAT(10X,5HCLAD1)
 1010 FORMAT(10X,1HF)
 1020 FORMAT(10X,2HFF)
 1030 FORMAT(1X/10X,'SPANWISE STATIONS-2Y/B')
 1040 FORMAT(10X,10H ALPHA MAX)
 1050 FORMAT(10X,7H CL MAX)
 1060 FORMAT(10X,31H THICKNESS / CHORD DISTRIBUTION)
 1070 FORMAT(10X,'SECTION REYNOLDS NUMBERS,MILLIONS')
 1080 FORMAT(10X,19H CHORD DISTRIBUTION)
 1090 FORMAT(10X,16H GEOMETRIC TWIST)
 1100 FORMAT(10X,6HDDCLMA)
      END
      SUBROUTINE MAIN5
C
C    ******MAIN5--CONTINUATION OF SUBROUTINE MAIN3     ********
C
       DIMENSION ARRAY(5,25,8 ),C(19),EPS(19),TRANS(19),REY(19),ETA(19),
     1HCPP(19),CLMAX(19),ZHERE(2,6),WHERE(2,6),Y(19),TAU(19),BETA(19,19)
     2,TRIX(19,19),MAZZ(6),MZCOL(6),MAXX(6),MXCOL(6),XHERE(2,6),MWCOL(6)
     3,MAWW(6),CM(19),CBG(19),CVAL(19),ALPG(19),CBC(19),ALPHU(19),ALPH(1
     49),ALPHZ(19),ALPHE(19),DELTA(19),CL(19),YHERE(2,6),MYCOL
     5(6),MAYY(6),CLADD(19),CLDEL(19),CLAD2(19),CLAD1(19),F(19),ALPC(19)
     6,TCNY(19)
       COMMON KOR,KIR,KIL,KCL,VSA(19),SW(19),VSBAR,EYETL,CTS,XPB,YPB,JP,
     1IPRAB,ALPHB,KSTAL,ISTAL,KCUNT,AB(3)
       COMMON NSLIP,VR(19),CA(19),AS(19),TL(19),SV(19),FUNC,YO,CDNAC,
     1ALPHV(16)
       COMMON INNCH,ISWIT(3),ALPHA,REYN,CLL,REYCN,XMAX,ALMAX(19),CLMAX,C,
     1EPS,TRANS,REY,ETA,HCPP,ZHERE,WHERE,ARRAY,Y,BETA,TFAC,TRIX,TAU,MAXX
     2,MAZZ,MXCOL,MZCOL,ASPEC,TAPER,BF,REYNO,DISCR,PIER,CRB,G,TSTAX,EDGE
     3,SIC,ALPHR,NFLAP,NLVL,NP,IY,IZ,IR,IP,ISIS,ISTAR,A,B,H,TAUT,TAUR,
     4TWIST,R,BWX,YHERE,MYCOL,MAYY,FLAP,TCNY,TWISA,X,Z,CM,ACC,XHERE,
     5MWCOL,MAWW,CAMB(19),CAMBR,CAMBT,DUMY1,DUMY2,NAME(25),AHERE(2,6),
     6MAAA(6),MACCL(6),BHERE(2,6),MABB(6),MBCOL(6),CHERE(2,6),MACC(6),
     7MCCCL(6),DHERE(2,6),MADD(6),MDCCL(6),STUNY(19),SGENE(19),CVAL,
     8ALPHU,CBC,CBG,DELTA,ALPHZ,ALPH,ALPHE,CLADD,CLDEL,CLAD2,CLAD1,
     9F,IRI,FF,LCCER
3030   DO 60 K=1,NP
       AK=K
       ALPG(K)=ALPHB+ALPHR+EPS(K)+ALPHB*TFAC*(TRANS(K)-1.)
       IF(K-ISTAR) 40,40,30
   30  IF(K-IRI+ISTAR) 50,40,40
C
C                   FIND CL FROM FLAPS-UP DATA FOR UNFLAPPED SPAN STATIONS
C
   40 ALPHA=ALPG(K)
      CLL=999.
      REYN=REY(K)
      TAUX=TAU(K)
```

```
      REYCN=999.
      CALL CAGET(ARRAY,1,IY)
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      CVAL(K)=CLL
      GO TO 60
C
C                FIND CL FROM FLAP     DATA FOR UNFLAPPED SPAN STATIONS
C
   50 ALPHA=ALPG(K)
      CLL=999.
      REYN=REY(K)
      TAUX=TAU(K)
      REYCN=999.
      CALL CAGET(ARRAY,4,IW)
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      CVAL(K)=CLL
   60 CBG(K)=CVAL(K)*(ASPEC/(ASPEC+1.8))*C(K)*CRB*(0.5+(1.+TAPER)*SIN(AK
     1*PIER)/(3.14159*C(K)))
C
C                                CHECK FOR DUMP
C
      CALL DATSW(3,JUNK)
      GO TO (70,81),JUNK
   70 WRITE(IP,610)
      CALL AAA(TRANS,NP)
      WRITE(IP,600)
      CALL ZZZ(TFAC)
      WRITE(IP,620)
      CALL AAA(ALFG,NP)
      WRITE(IP,630)
      CALL AAA(CBG,NP)
81    ITR=0
      IF(NSLIP-1) 135,80,80
135   DO 136 K=1,NP
136   TCNY(K)=0.0
   80 CLSTA=CBG(ISTAR)*BWX/(C(ISTAR)*CRB)
C
C                                CHECK FOR DUMP
C
      CALL DATSW(3,JUNK)
      GO TO (90,100), JUNK
   90 WRITE(IP,640)
      CALL ZZZ(CLSTA)
C
C           LOOK UP ZERO LIFT ANGLES AT FLAP END-FLAPSIDE
C
100   IF(ITR.NE.0) GO TO 120
      CALL CAGET(ARRAY,4,IW)
      ALPHA=999.
      CLL=0.
      REYN=REY(ISTAR)
      TAUX=TAU(ISTAR)
      REYCN=999.
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      A3=ALPHA
      CALL DATSW(3,JUNK)
      GO TO (110,120), JUNK
  110 WRITE(IP,650)
      CALL ZZZ(A3)
120   A4=ALPHZ(ISTAR)
      CLSTU=CLSTA/FF
      CALL DATSW(3,JUNK)
      GO TO (130,140), JUNK
  130 WRITE(IP,660)
      CALL ZZZ(CLSTU)
      WRITE(IP,670)
      CALL ZZZ(A4)
  140 CALL CAGET(ARRAY,4,IW)
      ALPHA=999.
      CLL=CLSTU
      REYN=REY(ISTAR)
      TAUX=TAU(ISTAR)
      REYCN=999.
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      ALPHX=ALPHA
      A1=EDGE*FF*(ALPHX-A3)+A3
      CALL DATSW(3,JUNK)
      GO TO (150,160), JUNK
  150 WRITE(IP,680)
      CALL ZZZ(A1)
```

```
  160 CLSTU=CLSTA/F(ISTAR)
C
C                                   LOOK UP ALPHA CURVE 1
C
      CALL CAGET(ARRAY,1,IY)
      ALPHA=999.
      CLL=CLSTU
      REYN=REY(ISTAR)
      TAUX=TAU(ISTAR)
      REYCN=999.
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      ALPHX=ALPHA
      A2=EDGE*F(ISTAR)*(ALPHX-A4)+A4
      CALL DATSW(3,JUNK)
      GO TO(170,180), JUNK
  170 WRITE(IP,690)
      CALL ZZZ(A2)
  180 SDELT=A2-A1
      IF(NSLIP.EQ.0) GO TO 1802
      CCLIFT=0.
      DO 1801 K=1,NP
      SV(K)=0.0
 1801 CCLIFT=CBG(K)*ETA(K)+CCLIFT
      CCLIFT=ASPEC*BWX**2*CCLIFT
      VW=CCLIFT*FUNC/(9.87*ASPEC)
      ALFPR=(ALPHB+EYETL)/57.293
      ASBAR=ATAN((SIN(ALFPR)+VW)/VSBAR)-ALFPR
      CALL DATSW(3,JUNK)
      GO TO (57,51),JUNK
 57   WRITE(IP,1009)
 1009 FORMAT(10X,2HVW)
      CALL ZZZ(VW)
      WRITE(IP,1010)
 1010 FORMAT(10X,5HALFPR)
      CALL ZZZ(ALFPR)
      WRITE(IP,1011)
 1011 FORMAT(10X,5HASBAR)
      CALL ZZZ(ASBAR)
  1   IF(ITR.NE.0) GO TO 519
      DO 103 K=1,NP
      ALPG(K)=ALPC(K)+TL(K)
      ALPHZ(K)=ALPHZ(K)+DA(K)
 103  ALPHE(K)=ALPG(K)-ALPHZ(K)
 519  DO 107 K=KOR,KIR
      AS(K)=ASBAR
      VR(K)=VSA(K)/COS(ALFPR+ASBAR)
      ANG=AS(K)+ALPHE(K)/57.293
      SYN=SIN(ANG)
      COZ=COS(ANG)
 107  SV(K)=VR(K)*SYN+VR(K)*SW(K)*COZ-SIN(ALPHE(K)/57.293)
      DO 108 K=KIL,KOL
      AS(K)=ASBAR
      VR(K)=VSA(K)/COS(ALFPR+ASBAR)
      ANG=AS(K)+ALPHE(K)/57.293
      SYN=SIN(ANG)
      COZ=COS(ANG)
 108  SV(K)=VR(K)*SYN+VR(K)*SW(K)*COZ-SIN(ALPHE(K)/57.293)
      CALL DATSW(3,JUNK)
      GO TO(52,53),JUNK
 52   WRITE(IP,1071)
 1071 FORMAT(10X,2HSV)
      CALL AAA(SV,NP)
      WRITE(IP,1072)
 1072 FORMAT(10X,2HVR)
      CALL AAA(VR,NP)
 53   ANG=AS(ISTAR)+(ALPG(ISTAR)-A3)/57.293
      SYN=SIN(ANG)
      COZ=COS(ANG)
      SVSRN=VSA(ISTAR)*SYN+VR(ISTAR)*SW(ISTAR)*COZ-SIN(ANG-AS(ISTAR))
      DELVR=SVSRN-SV(ISTAR)
      KSTAR=IRI-ISTAR
      SVSLP=VSA(KSTAR)*SYN+VR(KSTAR)*SW(KSTAR)*COZ-SIN(ANG-AS(KSTAR))
      DELVL=SVSLP-SV(KSTAR)
      KBGIN=ISTAR+1
      KEND=KSTAR-1
      DO 1700 K=KBGIN,KEND
 1700 SV(K)=SV(K)-DELVR
      KBGIN=KSTAR+1
      DO 1710 K=KBGIN,NP
 1710 SV(K)=SV(K)-DELVR+DELVL
```

```
      DO 126 K=1,NP
      AK=K
      SUM1=0.0
      DO 121 N=1,NP
      AN=N
      SUM2=0.0
      DO 122 M=1,NP
      AM=M
122   SUM2=SUM2+SV(M)*SIN(AM*PIER)*SIN(AN*AM*PIER)
121   SUM1=SUM1+SUM2*SIN(AN*AK*PIER)/AN
126   TONY(K)=VR(K)*(4.*SUM1/R+DELVR*STONY(K)-DELVL*SGENE(K))
1802  DO 240 K=1,NP
      SIG=0.
      DO 190 M=1,NP
190   SIG=(CBG(M)-TONY(M))*BETA(M,K)+SIG
      ALPHU(K)=SIG*(1.+TFAC*(TRANS(K)-1.))
      ALPH(K)=ALPG(K)-ALPHU(K)
      ALPC(K)=SDELT*HCPP(K)
      ALPH(K)=ALPH(K)-ALPC(K)
      REYN=REY(K)
      TAUX=TAU(K)
      REYCN=999.
      IF(K-ISTAR) 220,220,200
  200 IF(K-IRI+ISTAR) 210,220,220
  210 CALL DAGET(ARRAY,4,IZ)
      ALPN=(ALPH(K)-ALPHZ(K)*(1.-F(K)*EDGE))/(EDGE*F(K))
      CLL=999.
      ALPHA=ALPN
      CALL ARC(ARRAY,TAUX,MAWW,MWCCL,IE,XHERE,NLVL)
      CVAL(K)=CLL*F(K)
      GO TO 230
  220 CALL DAGET(ARRAY,1,IY)
      ALPN=(ALPH(K)-ALPHZ(K)*(1.-F(K)*EDGE))/(EDGE*F(K))
      ALPHA=ALPN
      CLL=999.
      CALL ARC(ARRAY,TAUX,MAXX,MXCCL,IE,WHERE,NLVL)
      CVAL(K)=CLL*F(K)
  230 CBC(K)=CVAL(K)*C(K)*CRB/BWX
  240 DELTA(K)=CBC(K)-CBG(K)
      CALL DATSW(3,JUNK)
      GO TO (250,260), JUNK
  250 WRITE(IP,720)
      CALL AAA(ALPHU,NP)
      WRITE(IP,700)
      CALL AAA(ALPH,NP)
      WRITE(IP,710)
      CALL AAA(ALPC,NP)
      WRITE(IP,730)
      CALL AAA(CVAL,NP)
      WRITE(IP,740)
      CALL AAA(DELTA,NP)
  260 DO 280 K=1,NP
      IF(ABS(DELTA(K))-DISCR) 270,270,290
  270 IF(K-NP) 280,360,360
  280 CONTINUE
  290 DO 310 I=1,NP
      SUM=0.
      DO 300 J=1,NP
  300 SUM=TRIX(I,J)*DELTA(J)+SUM
  310 CBG(I)=CBG(I)+SUM
      CALL DATSW(3,JUNK)
      GO TO (320,330), JUNK
  320 WRITE(IP,630)
      CALL AAA(CBG,NP)
  330 ITR=ITR+1
      IF(ITR-30) 80,80,340
C
C                                      NON CONVERGENCE DUMP
C
  340 WRITE(IP,750)
      WRITE(IP,740)
      CALL AAA(DELTA,NP)
      WRITE(IP,730)
      CALL AAA(CVAL,NP)
      DO 350 J=1,NLVL
      NR=MAXX(J)
      NC=MXCCL(J)
      TAUX=WHERE(2,J)
      WRITE(IP,760) J,J,NR,NC,TAUX
      DO 350 K=1,NR
```

```
  350 WRITE(IP,770) (ARRAY(J,K,L),L=1,NC)
      CALL EXIT
  360 DO 380 K=1,NP
      IF(ALPH(K)-ALMAX(K)) 380,370,370
  370 WRITE(IP,780) K,ALPH(K),ALMAX(K)
      KSTAL=1
380   CONTINUE
      WRITE(IP,3901)
3901  FORMAT(10X,22'SPANWISE STATIONS,2Y/B)
      CALL AAA(Y,NP)
      CALL DATSW(3,JUNK)
      GO TO (390,400),JUNK
390   WRITE(IP,790)
      CALL AAA(CLMAX,NP)
400   WRITE(IP,800)
      CALL AAA(CVAL,NP)
800   FORMAT(10X,43HDISTRIBUTION OF SECTION LIFT COEFFICIENT-CL)
      IF(NSLIP.EQ.0) CTS=0.0
      DO 801 K=1,NP
801   CL(K)=CVAL(K)*(1.-CTS)
      WRITE(IP,3801)
      CALL AAA(CL,NP)
3801  FORMAT(10X,44HDISTRIBUTION OF SECTION LIFT COEFFICIENT-CLS)
      SUM1=0.0
      DO 900 I=1,NP
900   SUM1=CBC(I)*ETA(I)+SUM1
      C=ASPEC*BWX**2
      CLIFT=C*SUM1
      CLPP=CLIFT*(1.-CTS)
      WRITE(IP,3401)
3401  FORMAT(1X,40(3H./.)/1H0/)
      WRITE(IP,341) ALPHB,CLIFT,CLPP
341   FORMAT(10X,35HFUSELAGE ANGLE OF ATTACK,DEGREES..=,F9.5
     1/10X,35HLIFT COEFFICIENT,CL...............=,F9.5
     2/10X,35HLIFT COEFFICIENT,CLS..............=,F9.5)
      WRITE(IP,3401)
      IF((KSTAL.EQ.0).AND.(KOUNT.EQ.0)) GO TO 2501
C
C                      DEFINE EXACT SYALL ANGLE OF ATTACK
      IF((IPRAB.EQ.1).OR.(KOUNT.EQ.3)) GO TO 9900
      KOUNT=KOUNT+1
      GO TO (1000,2000,3000),KOUNT
1000  AB(1)=(ALPHB+ALPHV(IPRAB-1))/2.
      ALPHB=AB(1)
      GO TO 9900
2000  IF(KSTAL.EQ.0) GO TO 2100
      ISTAL=1
      AB(2)=(AB(1)+ALPHV(IPRAB-1))/2.
      ALPHB=AB(2)
      GO TO 9900
2100  ISTAL=0
      AB(2)=(AB(1)+ALPHV(IPRAB))/2.
      ALPHB=AB(2)
      GO TO 9900
3000  IF(KSTAL+ISTAL-1) 3100,3200,3300
3100  AB(3)=(AB(2)+ALPHV(IPRAB))/2.
      ALPHB=AB(3)
      GO TO 9900
3200  AB(3)=(AB(2)+AB(1))/2.
      ALPHB=AB(3)
      GO TO 9900
3300  AB(3)=(AB(2)+ALPHV(IPRAB-1))/2.
      ALPHB=AB(3)
9900  KSTAL=0
      DO 1701 K=1,NP
1701  CL(K)=CLMAX(K)-CVAL(K)
      WRITE(IP,3501)
3501  FORMAT(10X,25HSTALL MARGIN DISTRIBUTION /1X)
      CALL AAA(CL,NP)
      IF((IPRAB.EQ.1).OR.(KOUNT.EQ.3)) GO TO 171
      WRITE(IP,3701) NAME
      WRITE(IP,3601) ALPHB
3701  FORMAT(1H1/1H0/35X,25A2/1X)
3601  FORMAT(1X,40(3H./.)/1X/10X,28HADJUSTED ANGLE OF ATTACK TO,
     1F8.3,32H,SEARCHING FOR EXACT STALLPOINT,/1X/1X/,40(3H./.)/1X)
      IR=100
      GO TO 3030
2501  IR=0
      RETURN
171   IR=101
```

```
      RETURN
  600 FORMAT(10X,16HTHICKNESS FACTOR)
  610 FORMAT(10X,13HR CU BAR / CU)
  620 FORMAT(10X,4HALPG)
  630 FORMAT(10X,3HDEG)
  640 FORMAT(10X,5HCLSTA)
  650 FORMAT(10X,2HA3)
  660 FORMAT(10X,5HCLSTU)
  670 FORMAT(10X,2HA4)
  680 FORMAT(10X,2HA1)
  690 FORMAT(10X,2HA2)
  700 FORMAT(10X,4HALPH)
  710 FORMAT(10X,4HALPC)
  720 FORMAT(10X,5HALPHU)
  730 FORMAT(10X,4HCVAL)
  740 FORMAT(10X,5HDELTA)
  750 FORMAT(1X,40HUNABLE TO CONVERGE AFTER 30 ITERATIONS   ABORTED)
  760 FORMAT(1H0/1H0,7H ARRAY(,I1,11H 1,  1) TO (,I1,1H,,I2,1H,,I2,1H),F
     110.5/1H0)
  770 FORMAT(12F10.4)
780    FORMAT(10X,19HSTALLED AT STATION ,I2,18H ANGLE OF ATTACK =,F7.3,
     110X,21HSECTION STALL ANGLE =,F7.3)
  790 FORMAT(10X,5HCLMAX)
      END
```